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The Guidance and Control of Helicopters and V/STOL Aircraft at Night and in Poor Visibility

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PREFACE

In May 1974, the Guidance and Control Panel held a symposium at Stuttgart, Germany on "The Guidance and Control of V/STOL Aircraft and Helicopters at Night and in Poor Visibility." At that conference it was apparent that a lot of technological developments were at an early stage to meet the operational requirements and also the optimisation of the whole vehicle/systems/pilot needed a lot more attention.

The desire of operators of all three services of the NATO nations to extend the use of helicopters and V/STOL aircraft into night and conditions of poor visibility has caused an expansion of activity in various fields, but most particularly in the electro-optical and radar sensors of various kinds. The integration of these new sensors into modern navigation, flight control and display systems is an important aspect – particularly in helicopters where space and weight are at a premium. Also, the pilot workload is already high in helicopters and V/STOL aircraft near the ground and any new technology must be introduced in a manner to keep the workload within bounds so as not to lose the advantages of the new sensors.

This conference was planned with a view to assessing where we have reached in trying to meet the stringent operational requirements and what new technologies are available to make up total systems to meet these requirements.

The Round Table Discussion at the end of the Conference is intended to address the question of whether the new technologies, such as Electro-Optic sensors have led to significant improvements in operational capability and should be a valuable way of having a general discussion on the theme of the Symposium.

G.C.HOWELL
Program Chairman

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ADDING THE CHALLENGE OF NAP-OF-THE-EARTH

by
Joseph C. Tirre, Jr., Lt Colonel
Headquarters United States Army Training and Doctrine Command
HQ USATADC (ATCD-AM)
Ft Monroe, Virginia 23651

Good morning, gentlemen. I am Lt Colonel Jay Tirre of the US Army Training and Doctrine Command's Airmobility Systems Directorate. As announced earlier, Brig General Canedy was to make this presentation, but he moved to another assignment. Brig General McNair, the new Director of Army Aviation, expressed his desire to be here but also is unable to attend. His pressing duties assumed such a short time ago kept him in Washington. He asks, though, that I extend his best wishes to all of you. As I said, I am from Headquarters Training and Doctrine Command, Ft Monroe, Virginia, and as such represent the user of Army aviation. My job as a soldier-aviator is to operate the fleet of Army aircraft in harmony with combat and combat support units to accomplish the Army's overall mission of defense of the USA and its allies.

I will discuss with you today one small portion of the US Army aviation mission --- a particular mission segment with a disproportionally large number of problems. In addition to the difficulties of effectively employing attack and scout helicopters, at night, and in adverse weather, a restrictive flight regime is now imposed. We call it Nap-of-the-Earth---a third problematic dimension. Nap-of-the-Earth, or NOE, is an attempt to defy the law of physics as it pertains to having two or more bodies occupy the same space at the same time (see figure 1). Whereas in the past we have opted for increasingly higher aircraft altitudes and airspeeds, NOE places the helicopter weapons system and its crew in the same environment as the ground soldier. Altitudes and obstruction clearances are measured at 2-5 meters rather than hundreds or thousands of meters. Terrain becomes both friend and foe alike. Beneficially it is used to mask movement and conceal aircraft firing positions. However, unintentional contact with the earth or anything attached to it is likely to be fatal. NOE has literally been forced on us by the capabilities of the Warsaw Pact Air Defense Threat.

As you may well know, components of this air defense capability were vividly and devastatingly demonstrated in the 1973 mid-east war. Tactics and weaponry of the Warsaw Pact nations placed a steel umbrella over the Arab ground forces, and this umbrella virtually prohibited all but few Israeli incursions into Arab airspace. A complete description of that threat is readily available to you. What this means now for the US Army helicopters can be summed up in the phrase, "lower and slower." To enhance your understanding of these night, weather and NOE problems from an operational point of view, I will describe a perceived mission of an attack helicopter platoon in what General Donn Starry, my Commanding General, calls the "central battle." As we progress through the scenario, I will highlight problems of the helicopter crews and their commanders. Shown here now is a US division positioned on a 60 kilometer front. In the first few hours of hostilities, the enemy has driven back our screen of covering forces and is massing 20-25 maneuver battalions and 60 or more artillery tubes per kilometer to punch through the depicted US forces in a narrow sector in order to drive deep into our rear area. (See figure 2.)

As the location of the breakthrough attack becomes clear (itself a distinct problem set), the division commander begins his race against time to mass forces to defeat the enemy's main attack. The division's organic scout and attack helicopters will be committed to the decisive point in the battle area to defeat the massed enemy armor and to buy time for the ground battalions to move into position. The scouts and attack helicopters will use low level and contour flight profiles from their staging areas until reaching the point where enemy air defense weapon ranges may become effective. This is envisioned to be at, or near, brigade rear boundaries. Transition to NOE altitude is then accomplished and the helicopter force proceeds at a slower pace. The scouts will move in front to begin acquiring targets while the attack helicopters stay slightly to the rear awaiting instructions from the scouts.

To this point, the aircraft crews and commanders have encountered two significant problem areas: Navigation and communication. If visibility is reasonable and the crews are familiar with the terrain, navigation should not be a limiting factor. If, on the otherhand, visibility (darkness and/or weather) is poor and the aircrews are unfamiliar with the terrain, low level navigation becomes an acute problem. The aviation company commander will be coordinating the actions of the scouts and attack ships by radio, and the scouts will be directing the attack ships also by radio. This assumes that radio communications are possible. When operating at NOE altitudes, loss of line-of-sight between cooperating elements (the attack and scout helicopters) severely degrades frequency modulated radio transmissions. Additionally, the Warsaw Pact nations are known to have sophisticated jamming and other electronic warfare techniques, capable of disrupting our unimpeded use of radio communications.

In the next phase of our scenario, the scouts will attempt to detect enemy air defense weapons and tanks. Specific attack ships will be assigned individual targets to be acquired and taken under engagement. Ideally, acquisition and engagement would be at the maximum effective range of the helicopter weapon systems being employed.

Attack ships will unmask from masked positions and hover high enough to visually acquire scout designated targets and undertake engagement. At night, flight control of the helicopter becomes a problem of great magnitude for the pilot. Without outside cues to guide him, he cannot adequately maintain a precise position over the ground and may be unable to hold the aircraft in an attitude stable enough to fire on-board weapons systems mindful that he himself has now become a priority target of the enemy. Also, darkness and limited visibility hamper the acquisition and engagement process. This problem does have an impact on aircrew and aircraft survivability. Should close-in engagement be required, enemy air defense weapons, primarily the radar controlled ZSU 23-4 will be able to effectively engage our aircraft.

As the battle progresses, the aviation unit (see figure 4) commander will coordinate the next wave of scouts and attack ships in movement to the battle area in order to relieve the first wave. The initial attack aircraft will require refueling and re-arming. This, again, requires extensive radio coordination to insure that the second wave arrives in the correct place in time to permit continuous target engagement by the attack helicopters.

Another problem occurs at this point. As wire-guided missiles are used in the engagement, their control wires become strewn on the tree canopy about the battlefield and present an increasing obstacle problem for helicopters in the NOE flight regime. Other previously undetected or uncharted wires and cables present this same problem. The aviation unit commander will continue to rotate his forces in this manner until committed to other missions or until the breakthrough attempt is contained. This broad overview of a very complex scenario has attempted to highlight these major problems as I see them. (See figure 5.)

I've portrayed a rather depressing picture of accomplishing the mission, but it's not quite that hopeless. For communications, high power (40 watts) frequency modulated radios will be installed in all aircraft and high frequency single band radios for selected aircraft will be purchased in the near future. These radios will improve radio communications capability in the NOE regime substantially. Navigation and positioning will be assisted by an internally referenced lightweight doppler navigation system currently in production. An add-on to this capability will come from the introduction of a projected map display. In tests conducted at the Aviation Center at Ft Rucker, Alabama, the doppler drive projected map display produced quantum jumps in navigation accuracy when compared to accuracies using hand-held paper maps. Other systems, based on external referencing are under development and evaluation for potential use. Included are a global positioning satellite system and a position locating and reporting system which uses time allocation and triangulation techniques. Aircraft flight control at night has been enhanced by the introduction of night vision goggles, which are light intensification devices.

This is the second generation of this device. Early models were heavy, lacked depth perception and were difficult to focus. The third generation of these goggles is being evaluated now and should overcome most of the problems of earlier models.

In the area of aircraft control, stabilization augmentation systems have overcome some of the inherent instability of helicopters. Some other programs underway include: computer generated command symbols which provide cues to the pilot for maintaining precise position during the hover, unmask, and remask maneuvers during night and poor visibility conditions. Two systems, a target acquisition designator system (i.e., a combination of optics, FLIR and TV for acquisition with a laser for designation) and a pilot's night vision system use common module, forward-looking infrared to improve the night vision of the pilot and the co-pilot gunner. These systems are in competitive development and will enter production next year. In later presentations there will be discussions of some efforts underway to use laser technology for wire detection. While the devices I have briefly described here will assist in overcoming some of our problems, much work is needed to optimize our capabilities at night with poor visibility at NOE. Such an item as a very lightweight mast-mounted forward-looking infrared would be a welcome addition to the repertoire of night vision devices. Solutions are also needed which minimize space, weight, and power requirements of existing systems.

I appreciate this opportunity to describe our operational problems and some of the near-term fixes expected and proposed. I sincerely enlist your assistance in furthering our capabilities. Gentlemen, now I would like to entertain any questions you may have.

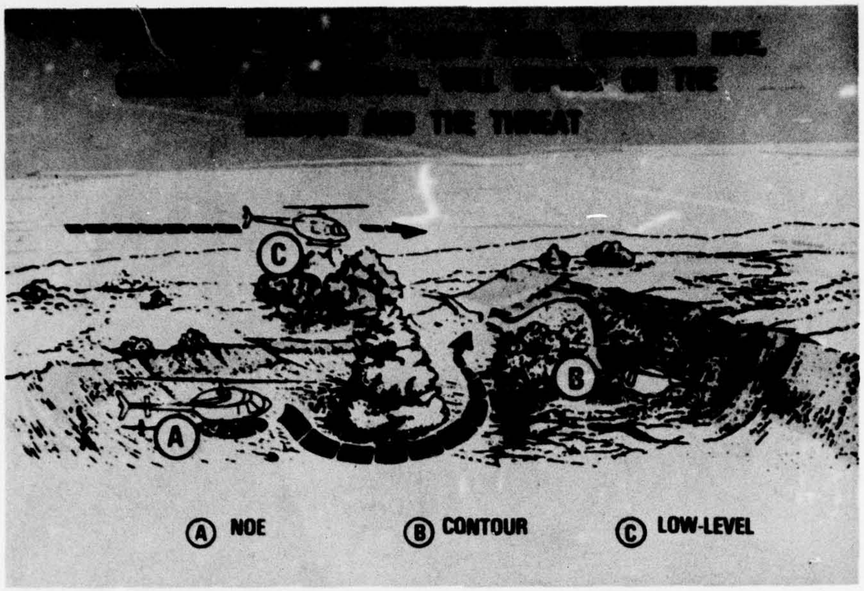


FIGURE 1.

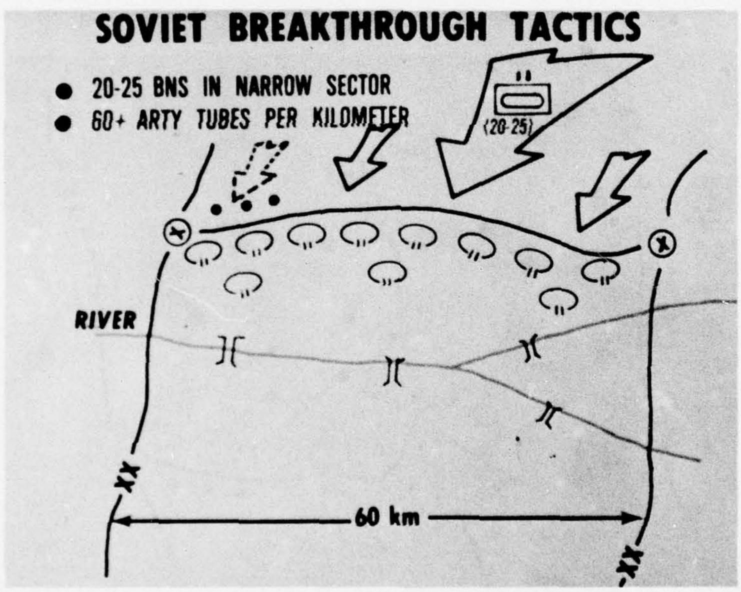


FIGURE 2.

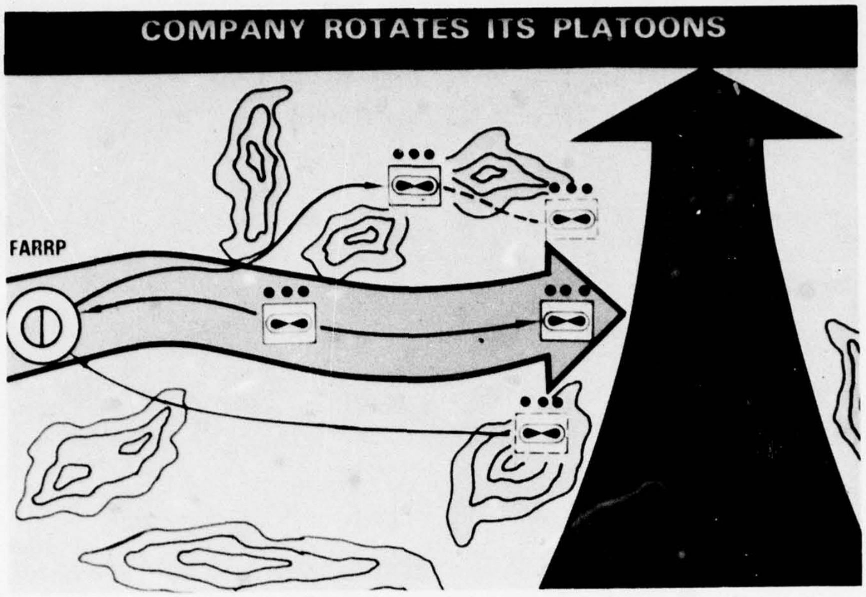


FIGURE 3.

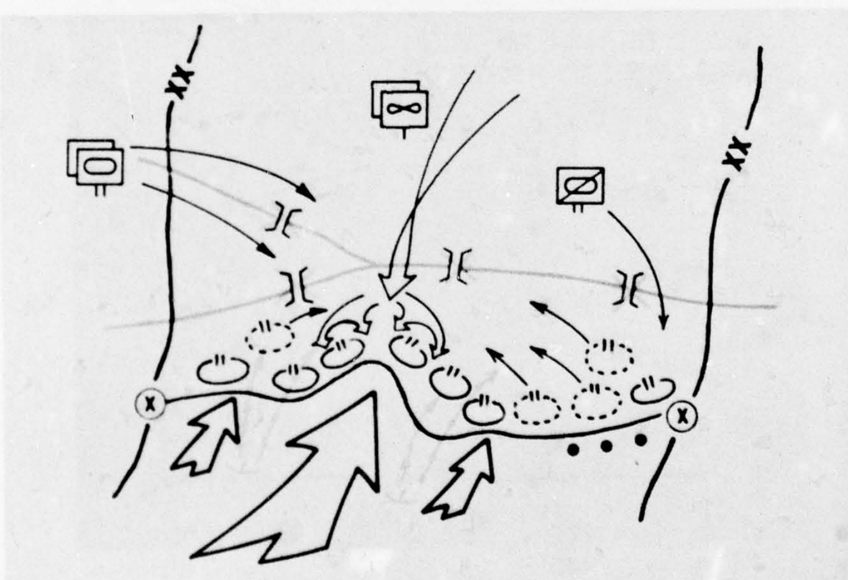


FIGURE 4.

PROBLEMS OF NIGHT, WEATHER, AND NOE FLYING:

- NAVIGATION
- COMMUNICATION
- AIRCRAFT CONTROL
- WIRE DETECTION
- TARGET ACQUISITION/ENGAGEMENT

FIGURE 5.

THE DEVELOPMENT AND IN-FLIGHT EVALUATION OF A TRIPLEX DIGITAL AUTOSTABILISATION SYSTEM FOR A HELICOPTER

by

J. Meadows
Smiths Industries Limited
Bishops Cleeve, Cheltenham, Glos.

P. Robinson
Royal Aircraft Establishment
Farnborough, Hants.

SUMMARY

At the previous AGARD Conference on "The Guidance and Control of V/STOL Aircraft and Helicopters at Night and in Poor Visibility", a paper was presented describing a triplex autostabilisation system then being developed by Smiths Industries Limited on behalf of MOD. This system was scheduled for installation and flight evaluation by the Royal Aircraft Establishment at Farnborough in a Sea King helicopter. This research programme was successfully completed during 1977 following 200 hours flying with the system being installed and operational throughout the programme.

Following a review of the programme objectives and a brief description of the system and its installation in the aircraft, the present paper discusses the final stages of development, with particular emphasis on the major contribution afforded by a comprehensive systems rig, both to the hardware and software development programmes. The paper continues with a presentation and discussion of the results of the in-flight evaluation programme including performance under both fault-free and simulated runaway conditions.

Complete achievement of the failure survival objectives was demonstrated with an insignificant incidence of nuisance warning or cutouts which bedevil the majority of analogue implementations of flight control systems having comparable integrity objectives. The smooth flight development programme was significantly aided by the excellent installation, pre-flight and in-flight testing facilities afforded by the comprehensive logic capabilities of the digital implementation, which provided the means for the identification of faulty units with a high success probability.

A number of problem areas encountered during the initial phases of the flight trials and the means adopted to overcome these are discussed in the paper which concludes with an assessment of achievements in relation to the initial objectives.

1 INTRODUCTION

A continuing programme of research into avionic equipment to meet the total requirement for night and poor visibility operations in helicopters is being carried on at the Royal Aircraft Establishment. A Sea King Mk 1 helicopter is being used as a vehicle for this work and as a part of the overall programme, a specification was issued for the development and manufacture of a defect survival autostabilisation system (DSAS) for this aircraft, with a preference for a digital solution, encouraging development in this area and to provide flight experience of digital techniques applied to flight control systems in the helicopter environment.

An autostabilisation system using triplex redundancy concepts to achieve the failure survival requirement, with digital processing of the sensor signals, was proposed by Smiths Industries Limited and selected by the Ministry of Defence (MOD) for development.

At the previous AGARD conference on the Guidance and Control of Helicopters in Poor Visibility a paper was presented discussing in detail the programme objectives and describing the system then under development (Ref. 1). This system was installed in the aircraft during the first half of 1976 and the flight evaluation programme was successfully concluded in September 1977.

Following a brief review of the salient features of the system and adopting the theme of the present symposium, this paper discusses the overall achievements of this programme in relation to the initial objectives, detailing the extensive ground test and in-flight facilities, and presenting some of the more important ground and in-flight test results.

The promised advantages of this new system, namely the provision of a failure survival autostabilisation system, relatively immune to nuisance disconnect problems by virtue of the use of digital rather than analogue computing techniques and the promise of enhanced built-in test facilities with a high probability of successful faulty LRU identification have been demonstrated.

By virtue of the rapid advances in digital technology during the period of this programme, application of these concepts using lighter and cheaper computing equipment is now practical; however, a number of related problem areas remain to be investigated. These include

the need to minimise the number and type of sensors employed

the interrelationships between digital autostabilisation and autopilot computing

the impact of the higher dynamic response requirements imposed by semi-rigid rotor configurations

the need to restructure the pre-flight test routines in order to reduce the test duration below the 1½ to 2 minutes which was assessed as operationally undesirable, whilst at the same time maintaining the levels of integrity afforded by these tests.

2 PROGRAMME OBJECTIVES

Recognising the increasing dependence being placed on artificial stabilisation systems, particularly in helicopters using advanced rotor technology, and the increasing emphasis on night time and low level poor visibility operations, with their attendant problems in terms of pilot workload and safety, the RAE formulated a research programme aimed at demonstrating in-flight, a high integrity autostabilisation system for a helicopter. The trends towards the digital implementation of avionic systems and an increase of the need to ease the maintenance problems by providing good built-in test facilities led to a preference for a digital solution which would afford the opportunity to assess both the merits and problems of such a system.

A specification defining the MOD requirements for a defect survival, three axis autostabilisation system (DSAS) to be supplied for evaluation in the RAE Sea King was issued to UK specialist avionic equipment manufactures in 1971.

Fault-free performance requirements, including long term attitude hold capabilities comparable to those being achieved with the existing simplex analogue autostabilisation system in the Sea King were specified. The primary challenge was the requirement that any single defect should have negligible effect upon the system performance, with transient deviations during trimmed "hands-off" flight, limited in pitch and roll to 1 deg of attitude and 1 deg/s rate, and in yaw to 2 deg of attitude and 1 deg/s rate. In the event of a complete system failure, the pilot should be able to assume manual control after an intervention time of 1.5 seconds.

A GO/NO indication for the pilot prior to take-off signifying correct system functionability and availability was specified, together with a self testing capability for the detection and location of defective line replaceable units.

A mean time between defects (MTBD) for the stabilisation system of 400 flying hours and an overall mean time between failure (MTBF) in excess of 10^5 flying hours was specified.

3 SYSTEM DESCRIPTION

3.1 Test Aircraft

The Sea King HAS Mk 1 is an anti-submarine helicopter powered by two Gnome engines. It has a single 5-bladed main rotor and a 6-bladed tail rotor. The aircraft is normally fitted with a Mk.31 AFCS system providing simplex autostabilisation and autopilot facilities.

3.2 Flying Controls

A block schematic of the aircraft's flying controls is shown in Fig. 1. The pilot and co-pilot's controls are mechanically connected to the input stages of a four axes auxiliary servo system, providing servo assistance for cyclic pitch, cyclic roll, collective and yaw control movements. The yaw axis is the only direct source of servo assistance for the tail rotor. The other three jacks form part of the control linkage via a mechanical mixing unit to primary jacks located at the main rotor head.

AFCS demands are electrically introduced into the flying controls in series with the pilot's inputs via electro-hydraulic servo valves incorporated in the auxiliary servo stage which also provides cyclic trim and force-feel facilities. AFCS inputs via the servo valves are limited to ±10% of full authority in the cyclic channels and ±5% in the yaw and collective channels.

3.3 System Mechanisation

A detailed description of the defect survival autostabilisation system is presented in Ref. 1. The following sections provide a brief outline of some of the more important features.

The defect survival capability is achieved by adopting triplex redundancy, using three "independent" full functional lanes with a lane or unit being identified as defective on a two out of three majority vote. This development programme had been preceded by extensive research studies into triplex digital AFCS systems during which a number of basic concepts had successfully been implemented. A schematic of the triplex configuration is given in Fig. 2. Given identical digital input data, then three independent but functionally identical digital computers will generate identical digital output data on completion of their full program cycle, avoiding interlane errors associated with computing which normally constitute a significant adverse factor in multilane analogue systems. In this implementation, each lane computer is identified with its own lane sensor and, following the sensor signal conversion to digital format within the computer unit, the data is exchanged between computers, each of which, in normal triplex operation, then forms the mean of the three input signals for subsequent processing. This amalgamation process also enables a faulty sensor unit to be identified when its digitised signal differs from the other two sensor signals by a predefined level. When this occurs it is automatically rejected from the amalgamation process with the system subsequently using the mean of the remaining two good sensor signals whilst retaining triplex computing.

Since cost and timescales precluded the development of a new triplex actuation system for this installation, it was necessary to interface with the existing limited authority simplex auxiliary servo system. The servo valves of the auxiliary power units are fitted with two coils, normally driven in parallel by the Mk 31 AFCS. For the DSAS these coils are driven independently by two out of the three computer units, each normally contributing 50% to the total output. In the third computer, defined by its location on the mounting rack in the aircraft, a simple resistive model of the coil is used. In the event of one of the two active lane computers being cut out, the gain in the remaining two computers is doubled with the actuation system then operating in a monitored simplex mode.

The primary sensors selected for the system were high reliability Series 700 gas bearing rate gyroscopes which had recently been developed by Smiths Industries Limited. Short term attitude hold is provided by pseudo integration of the rate gyro signals. Three identical gyroscope units, each containing three rate gyroscopes with their axes aligned were produced using 45 deg/s maximum rate gyroscopes. With two mounting faces on the unit, it was possible by appropriate mounting onto a horizontal reference plane to measure pitch, roll and yaw rates respectively with the three units.

It was recognised that the long term attitude hold requirements could only be achieved by using less reliable vertical gyroscopes and it was initially proposed that these should be provided at a lower level of redundancy. In the event it was ultimately decided to use three vertical gyroscopes, the signals from which were amalgamated in the same manner as all other triplex sensor signals. Pitch and roll attitude demand signals proportional to the cyclic control column displacement are obtained from two triplex position sensor units.

The three digital computer units, each incorporating a CPU, memory, input and output peripherals, engage/disconnect logic circuits and the serial data transmission system are housed in a ½ ATR short case. Small and medium scale integrated circuits based on bipolar transistor technology was used with components mounted onto multi-layer printed circuit boards. In order to increase the overall life of the computer units in high ambient temperature conditions, a fan cooling system draws air through external ducting over the sides of the power supply section and over the top of the computer card section of the unit.

Each computer incorporates its own basic 8 MHz clock and synchronisation between them is required. This is achieved through the use of software controlled synchronisation pulses transmitted between computers on dedicated synchronisation status lines.

The interlane digital data highways and the engage-disconnect logic circuits represent some of the most critical integrity regions of the design. Consequently duplicated circuits within each computer have been used for both of these functional areas, with the individual boards being designed to meet segregation rules formulated during the initial stages of design.

The triplex redundancy extends into the pilot's control unit where lane segregation has again been carefully controlled.

Two complete sets of equipment (Fig. 3) were manufactured, the second set being used to back up the airborne set.

4 SYSTEM OPERATION

4.1 Failed Lane Identification

In each of the three lanes the digital output signals are crossfed between computers and compared for exact identity. In the event of a discrepancy an amber integrity warning is lit on the pilot's control unit. The coil current generated in each lane is fed back, through a resistive network via the A/D converter to provide a digital measure of achieved output. These are also crossfed between lanes and compared within each computer. If the difference level exceeds 10% of the series actuator authority in two successive iteration cycles, that is for a maximum period of 100 msec, the appropriate lane, identified on a two out of three vote in triplex operation is cut out. When operating in the duplex mode, both computers are cut out when this difference persists for two successive cycles.

4.2 Engagement/Disengagement

Engagement of the system is achieved by means of the ENGAGE lever in the pilot's control unit (Fig. 4). Movement of this switch from the OFF to the ON position applies power to each of the three computers, which commence operation from the first programme step as soon as the internal power supplies reach their normal operating levels, indicated by a power supply monitoring system. In this state, the three lane indicators on the PCU change from OFF to OUT, indicating that power is being supplied to the computers but the lanes remain disengaged. Actual lane engagement occurs when the pilot momentarily pushes the ENGAGE lever to its forward position. The three lane indicators now change to give an IN indication. Manual disengagement of the system is achieved either by returning the ENGAGE lever to the OFF position or by operating the pilot's or co-pilot's cut out buttons on the cyclic sticks.

4.3 Interlane Data Transmission System

Each computer contains duplicated transmitter-receiver systems and the three computers are interconnected by duplicated data highways, all data is sequentially transmitted twice using these two highways. In the receiving computer the results of the two transmissions are compared for identity and, if satisfactory, the data is used for subsequent processing. If a difference of data is detected, the receiving computer will assume this data to be invalid and will operate the amber warning lamp on the PCU, and withdraw its validity discriminant to the first computer. If the third computer also detects a difference in received data on the two highways, it will also withdraw its validity discriminant and lane 1 computer will be rejected. If however its received data is in agreement, then it will retransmit this, via its own dual data link, to the second computer so that the latter receives input data necessary to continue computing for triplex operation.

Whilst the amount of time required for data transmission remains a very small proportion of each computing cycle, the doubling up of the amalgamation routines, etc. for the direct and indirect data inputs is expensive in computing time and this of course increases directly with the number of sensor signals involved.

The relatively complex interlane system was implemented rather than a single system in order to avoid the possibility of a complete system failure due to a single fault. It was postulated that a slowly deteriorating computer could initially transmit correct data which, due to receiver tolerances, might be differently interpreted by the two receiving computers, one of which could therefore erroneously be rejected as faulty to be followed by a relatively rapid disagreement between the remaining good computer and the failing unit, leading to a complete cut out. Whilst an extensive study of transmitter/receiver characteristics and reliabilities might establish that the probability of such an event would be acceptably small, it was decided that, in the absence of such an investigation, the more complex dual system should be implemented.

4.4 Computer Synchronisation

Each computer operates at its own independent clock frequency, the three computers being held in synchronisation by the transmission of software controlled synch stat pulses along dedicated interconnecting wires. Each computer sequentially takes it in turn, once per computing cycle, to send synchronisation pulses simultaneously to the other two computers. In each case the receiving computers on seeing the pulse jumps to a predefined step in the programme, knowing that the sending computer will also be at the corresponding step in its programme. If a computer is too far out of synchronisation and is not at the correct point at the correct time, an amber warning is generated.

To accommodate relatively wide variations in the start up time of the three computers when power is switched on, a coarse synchronisation programme is initially used to reduce the timing difference down to 1 word cycle. If only two out of the three computers achieve synchronisation, within a time slot of 200 msecs, these two proceed to operate as a duplex pair. If all three computers fail to synchronise a dynamic halt is executed.

5 SYSTEM DEVELOPMENT

5.1 Initial Assessment

The development of new or improved control laws was not an objective of the present programme; consequently it was decided in general to implement identical autostabilisation control laws to those provided in the existing analogue system. Using general purpose hybrid computing facilities, the fundamental parameters for a digital implementation such as iteration rates, converter quantisation levels etc were established and an assessment of the required program size was made.

5.2 System Test Rig

To perform both hardware and software development testing, computer commissioning and release testing and as a facility for faulty component location during the flight trials programme, a systems rig was built at Cheltenham including the following interconnected facilities

Three engineers test panels
A systems control panel
A breadboard computer unit
A TREND interface terminal

The three engineers test panels, each incorporating 8K of core store, are connected to the digital computer units via commissioning connectors at the rear of the units. With the computer cover removed, the unit can function using the core store rather than its own in-built program (PROM) store. Each test panel provides facilities for halting, stepping and inching the computer, for inspecting the state of individual registers and loading individual core store locations with data from "Data" register switches. A store protection switch prevents corruption of store except when it is required to load these stores with program or data. Switches with extension leads allows simultaneous initiation of operations of all three computers and their test panels.

The systems control panel provides a housing for the pilot's control unit, the interseat null indicator and a junction box for all unit to unit interconnections provide access to every pin connection in the system and includes switches for open circuit failure simulations. Facilities are also provided allowing the connection to real or simulated sensors and for interfacing to a hybrid computing facility so that closed loop operations can be carried out. The nominal 115 volts, three phase AC supplies can be adjusted within the range of 85 to 135 volts.

Preceding the final design of the airborne computer unit a breadboard computer was built for rig evaluation. This unit was functionally identical to the airborne unit in all respects including the input and output facilities, dual engage/disconnect circuits, dual interlane data transmission system as well as the basic central processing unit. In addition to providing an invaluable means of checking the correct functioning of the basic design, this breadboard formed an important feature of the systems rig for development testing and re-release testing.

The TREND terminal, adapted to interface with the three engineers racks, incorporates keyboard and paper tape input/output facilities for program insertion and modification.

5.3 Rig Testing

The systems rig facility was initially used for system development and to functionally check the operation of the breadboard computer unit. Any design modifications introduced during the development programme were incorporated and evaluated on this rig. A full triplex system using two airborne computer units and the breadboard unit was available in May 1975, subsequently being in virtually continuous use on development, unit release and flight trials support until the flying programme was completed in July 1977.

Apart from the normal system and software development work, the rig was also used to carry out noise susceptibility testing which involved the injection of spurious voltages over a range of frequencies between specific regions of the system. In a high proportion of cases noise signals were injected into only one out of three lanes in order to increase the severity of the tests. From this work the general system immunity to injected RF was established. However the need for electrical isolation between the 28 volt DC return line and 400 Hz neutral line was established and this involved the introduction of small isolating transformers in each of the three 26 volt supply lines to the rate gyroscopes. The computer interlane data transmission highways, manufactured from screened and twisted pairs of wires, showed a high degree of immunity to injected RF noise.

The effects of transient power supply variations and interruptions were also investigated using the systems rig, from which it was established that transient power supply variations to BS3G 100 specifications had no effect on system performance. The in-built self-monitoring system on the computer power supply unit resulted in system disengagement for interrupt durations lasting between 1 cycle at nominal 115 volts level down to 1/4 cycle at 105 volts level.

5.4 Servo Interface Testing

It was necessary to establish the ability to successfully drive the auxiliary servo system using either one or both coils of the servo valve, and this was done during a short test programme using an auxiliary servo test rig at Westland Helicopters Limited (WHL). This series of tests also attempted to foresee problems likely to result from driving the servo valves with quantised rather than continuous input signals. To achieve this, sinusoidal input signals, sampled at a rate of 20 per second, corresponding to the predicted rate necessary for stabilisation control, were injected into the system and no problems were foreseen at this stage.

Following the initial flight of the system it was observed during ground testing that rapid movements of the cyclic stick produced a loud noise from the auxiliary servo system responding to quantised outputs from the digital computers. A detailed study to examine the nature and consequences of this effect was carried out before the aircraft was allowed to fly again. High speed film of the auxiliary and primary actuator responses confirmed the step-wise output response at the iteration frequency. The possibilities of such motion leading to hydraulic seal wear, fatigue in the mechanical control circuits, the excitation of rotor blade structural modes and possible mechanical damage to the auxiliary and primary servo systems were investigated. A strain gauge, fitted to the lateral cyclic servo output rod was used to establish that in-flight loads did not exceed 20% of the fatigue endurance limit, and only 8% of the loads exceeded 4% of the limit. To ensure that seal damage would not occur, it was, however, decided to fit filter circuits to the outputs of the DSAS system on all three control axes.

5.5 Software Developments

A top down structured software development procedure was adopted, starting from a definition of the overall system tasks and progressing sequentially through sub-tasks and sub-routines to the production of line machine instructions via assembler language. At each stage of development, the interrelationship between software and hardware was checked using the systems rig facility. A flow diagram for the overall executive computer program is shown in Fig. 5. This essentially sub-divides into three regions, namely a pre-engage section, including the pre-flight testing routines, a multiplex mode path and a simplex mode path.

6 AIRCRAFT INSTALLATION

The DSAS equipment was installed in the Sea King Helicopter by RAE personnel during the first quarter of 1976. In order to retain autopilot facilities not provided by the DSAS system which were considered necessary for some phases of the overall Sea King research programme, it was decided to retain the existing Mk 31 analogue AFCS system in the aircraft and to provide a changeover facility which allowed reversion from one system to the other in one working day.

A schematic of the aircraft installation is shown in Fig. 6. For convenience the system unit, sensor and actuator interconnections, were routed through a junction box. The output signals from the computers were routed through a channel isolation switch box to the servo actuator coils. This enabled individual channel outputs to be isolated during initial flight trials.

An indication of the location of the system units is shown in Fig. 7. The three digital computers, the control and display test unit and the three rate gyroscope units, were located on the starboard side of the cabin. The two triplex position pick off units connected to the flying control runs presented the most difficult installation problem due to confined conditions. The pilot's control unit was mounted in the inter-seat console in the position normally occupied by the Mk 31 AFCS runaway selector box.

7 INITIAL FLIGHT ASSESSMENT

7.1 Sensor Tracking

Because the three vertical gyroscopes were not co-located in the aircraft, it was not possible to align them for output tracking until they were installed. Static alignment was carried out to achieve a maximum inter-lane error of 0.25 deg in both pitch and roll. Assessment of the vertical gyroscope tracking during flights designed to produce maximum errors established that both pitch and roll inter-lane errors did not exceed 1.0 deg.

The tracking of the rate gyroscope outputs had been measured earlier in the programme, prior to the installation of the complete system. Maximum inter-lane errors of 0.2, 0.27 and 0.3 deg/s were recorded in pitch, roll and yaw axes respectively, at the peak transient rotations for which the aircraft was cleared. When severe airframe vibration was induced, significant 17 Hz components were measured on the output of each rate gyroscope, resulting from the predominant 5R vibration mode at this frequency. This had been anticipated and the decision previously made to sample the rate gyroscope outputs more rapidly than other sensor inputs to prevent aliassing of this (and other noise sources) to lower frequencies.

At the conclusion of the flight trials programme, the rate gyroscope tracking was re-checked and found not to have deteriorated in the 200 hours of flying since the trials described above.

7.2 Control Law Modifications

During the first flights with the DSAS engaged, two unsatisfactory features became apparent, which were investigated before further evaluation took place.

When the aircraft was deliberately held light on the ground, virtually airborne, a resonance in yaw developed. This resonance was either convergent, mildly divergent, or neutrally stable, under different conditions of AUW, torque setting and possible wind speed but no correlation was found between these parameters and the stability of the mode. Sometimes it was necessary to excite the mode by a yaw pedal input. When the oscillation reached an undesirable amplitude, the pilot would disconnect the DSAS and the oscillation would slowly subside, as illustrated in Fig. 8. It was felt that this subsidence was a manifestation of an unanticipated lightly damped yaw mode of the aircraft and that when the loop was closed by the DSAS, the time delay of the digital system produced sufficient phase lag at the relatively high frequency involved, to destabilise the mode.

At this stage it was not practical to reduce the pure time delay in the system, so an alternative solution was evaluated in flight. This consisted of reducing the gain of the yaw rate term and making a slight reduction in phase lag by decreasing the time constant of the analogue smoothing filter on the yaw channel output. The combined effect of these two modifications was to stabilise this yaw oscillation and achieve a relative damping comparable to that of the open loop mode, as shown in Fig. 8.

The other initial problem was concerned with the pitch axis performance, where a higher than expected general level of pitch servo activity was experienced. It was considered that this over-activity was a sign of a lightly damped short period mode. Pilot opinion was that the pitch axis was somewhat 'livelier' than normal, but not necessarily unsatisfactory. More detailed examination of the pitch axis confirmed that the short period mode was in fact under-damped, as shown in Fig. 9(a).

A series of tests was conducted, aimed at improving the short period pitch stability, without adversely affecting pitch attitude holding or turbulence response. This was achieved by varying both the pitch attitude and pitch rate gains until the performance was optimum by the above criteria. As was expected, this occurred at reduced values of both gains. For comparison, the response to a longitudinal cyclic stick pulse with the modified pitch axis gains is shown in Fig. 9(b).

In resolving both the features described above, the Control and Display Test Unit was used to effect parameter changes whilst in flight.

7.3 PCU Ergonomics

Several valid comments of the Pilots Control Unit facilities were made by the evaluation pilots, the front panel of which is illustrated in Fig. 4. The most significant point is that the amber and red integrity warnings were found to have low attention value. Clearly in a more developed installation it would be necessary to interface the amber/red signals with central caution/warning panels and possible other warning systems. This is particularly so of the amber warning, since this is not normally accompanied by a discernible motion transient or stabilisation degradation. The fixed intensity of these integrity warnings (and other legends) was too high at night and inadequate in direct sunlight and a dimming facility should be introduced on a production system.

7.4 Pre-Flight Testing

The pre-flight testing facilities have been described previously. The sequence of tests was designed to be as natural and instinctive as possible and was generally agreed to be so by all evaluation pilots. After about three operations, pilots carried out the tests from memory, without reference to the check list. The flashing "TESTING" indication provided adequate warning that pilot action was required or that the wrong action had been taken (in the case of a pilot not following the sequence correctly). When pilots were familiar with the procedure, the complete sequence took slightly less than two minutes to perform. However, it was noted that correct operation of the sequence could be more difficult when the pilot was subject to the distractions encountered in operational flying. Furthermore the two minutes test duration may not be operationally acceptable. A possible solution would be to generate two levels of testing: a rapid pre-flight test, with minimal pilot involvement which could be supplemented by a complete test at scheduled intervals.

No insurmountable problems were discovered using the pre-flight test on the ground. Although operation of the system from ships was not specified in the system requirements, the opportunity arose to carry out deck landings for other reasons. Thus it was possible to carry out pre-flight tests on a moving deck and confirm that under certain conditions, angular motion of the ship could interfere with pitch or roll rate gyro and heading and result in a test failure. No attempt was made to devise modifications to overcome these problems during the trials programme, although software changes could be made to permit satisfactory tests to be conducted under these conditions.

8 ASSESSMENT OF STABILISATION QUALITIES

8.1 Standard System

Having resolved all known deficiencies in the autostabiliser control laws (albeit on the basis of tests made over a limited range of speed and altitude) it was possible to conduct a more thorough assessment of the DSAS as a pure autostabiliser, i.e. without consideration of the multiplex aspects. The objectives of this assessment were,

(i) to confirm that the modifications already made were satisfactory over the complete flight envelope

(ii) to check that no additional problems existed

(iii) to make a general qualitative evaluation of the autostabiliser performance

The assessment conducted was far less rigorous than would be the case if totally different control laws were used or a new aircraft type was under evaluation. In particular, turbulence levels were not included as an experimental variable.

Table 1 lists the tests carried out and the altitude/airspeed conditions covered. Variations in the position of the aircraft c.g. were not studied, with the single exception of an extreme aft c.g. case, for which some pitch axis tests were conducted.

The results obtained from the assessment showed that the above objectives (i) and (ii) had been met, i.e. the control law modifications were satisfactory and no further problems were revealed over the flight envelope covered. With regard to the evaluation outlined in (iii) above, the results obtained were satisfactory in all respects, there being only a few points of interest listed below.

there was a marginal decrease in pitch axis damping at altitudes above 5000 feet, but this was not considered to be significant.

a slight oscillatory tendency was revealed on the roll axis computer output and on the roll rate signal, but this appeared to be a low amplitude residual oscillation rather than a symptom of a lack of damping in the dominant mode.

attitude holding in both calm and gusty conditions was measured and found to be within the limits set out in the Requirements Specification. However there was a subjective impression that response to heading trim demands was somewhat slower and less precise than normal, which could be a result of the gain reduction made in the yaw axis.

8.2 Rate-based System

By virtue of the digital implementation and by using the control and display test unit, it was possible to carry out a comparative evaluation of alternative system configurations utilising rate sensor inputs only. The following four configurations were investigated:

(i) standard DSAS pitch control

(ii) rate stabilisation only, in pitch

(iii) rate plus pure integrated rate

(iv) rate plus 'leaky' integrated rate

Roll control in all cases (except (i)) was as obtained with vertical rate gyros rejected.

All variants were found to be acceptable in the context of a simple 'cruise' autopilot, intended to provide a reduction in workload together with a limited hands-off capability. Variants (iii) and (iv) gave a performance comparable in the short term to that of the standard system (i). Workload with the rate-only variant (ii) was noticeably higher and a significant aft stick movement was required during turns.

Further evaluation of variant (iv) confirmed that such a system had value as a piloting aid, with a workload mid-way between the standard system and the unstabilised aircraft.

9 EVALUATION OF PERFORMANCE FOLLOWING IN-FLIGHT DEFECTS

A group of system defects was selected, to be simulated in flight. Of the many conceivable defects, the choice was made on the basis of those giving rise to the most severe consequent aircraft motion. The Control and Display Test Unit was used to introduce the simulated defects into the system.

Two types of defect were simulated, single axis output failures and offsets on the rate gyro inputs to the system. The tests carried out, together with the results, are described below together with a limited comparison of simulated defects in systems of lower redundancy levels.

9.1 Simulated Output Failures

Instructions were devised, which modified the programme to simulate the following conditions, when inserted into a system computer

(i) an output hardover situation, in either sense

(ii) an output fail to zero, from an initial servo out-of-trim condition

Thus a total of nine output failures were simulated i.e. pitch up hardover, pitch down hardover, pitch output fail to zero and the corresponding failures in the roll and yaw axes. Each failure was injected in the hover and in forward flight at 70 and 100 knots.

In flight, each injected fault produced the expected result, i.e. amber integrity warning, disconnect of the 'faulty' lane, lane OUT indication and appropriate indication of the 'failed' LRU on the computer display. It is not possible to present in this paper all the results obtained, hence representative cases are discussed below.

Examination of a typical simulated hardover, shown in Fig. 10, illustrates the basic sequence. The output of Lane 2 (pitch) can be seen increasing rapidly to 8 mA (off the scale) as the hardover is injected, followed rapidly by a reversal to zero as the lane is disconnected by lanes 1 and 3 acting on the comparison of output feedback voltages. Simultaneously the outputs of lane 3 and lane 1 (not shown) are automatically doubled, to preserve the gain and authority of the system. This is shown in Fig. 10 as a step increase in lane 3 output at the instant lane 2 disconnects. Later on, having removed the hardover from lane 2, the system is re-engaged in triplex and lane 3 output is seen to decrease by a factor of two as the nominal (triplex) output gains are re-established. Examination of the pitch rate and attitude responses on Figure 10 shows that it is difficult to detect any transient motion resulting from the fault injection which could not be ascribed to the random motion due to turbulence. Even if all the motion were assumed to result from the failure transient, then the maximum is approximately 0.8 deg of attitude and 0.6 deg/s of pitch rate.

A typical fail to zero test is shown in Figure 11 and again the difficulty of correlating the aircraft motion with the fault injection is seen. The above comments apply also to the representative roll axis test shown in Fig. 12.

Summarising the results of these many tests, it can be said that the system completely meets the Requirements Specification in terms of transient motion in the pitch axis due to the difficulties described above, but it cannot be proved that this is so in the roll and yaw axes. However, for all tests the transients are within the bounds of the normal deviations of attitude and rate due to turbulence. Indeed it was considered that a detectable aircraft motion about the trimmed state would be a valuable clue to the change of system status.

9.2 Simulated rate gyro errors

Again using the control and Display Test Unit steady offsets on a single rate gyro signal were simulated, in the hover and at 70 and 100 kts. Table 2 shows the results obtained at 100 kts, being representative of all speed conditions.

It can be seen (Table 2) that all offsets above the rate gyro comparator setting of 2.0 deg/s were recognised as defects and the appropriate gyro input rejected with no detectable attitude or heading change. When the offset was just below the comparator setting (i.e. 1.93 deg/s), sufficient inter-lane error was present in some cases to augment the offset value and resulted in the comparator setting being exceeded. In all other cases the offset was not detected and therefore amalgamated as a valid signal. When this happened, a detectable attitude/heading change generally resulted, which in some cases was higher than the value defined in the Requirements Specification for a single defect.

9.3 Failure characteristics of other systems

Comparison was made between simplex, duplex and triplex configurations fitted in the same aircraft type. The failure responses of the equivalent simplex system are well documented, so it was not necessary to conduct failure tests with the DSAS in this mode. Using the Control and Display Test Unit, the DSAS was re-configured as a duplex system with no automatic disconnect following a lane failure. In this condition, pitch, roll and yaw hardovers were injected (in forward flight), for a range of initial servo trim states. Because general information only on the failure responses was required, the pilot intervened at relatively low attitudes and rates, when intervention was required.

Figure 13 illustrates the response to nose up hardovers for various initial servo trims. As expected, for a central or nose up initial trim, no pilot intervention was necessary in the short term. With a nose down initial trim, however, the "good" lane lacked sufficient authority to counteract the hardover and the pilot intervened after 2.7 s, at a nose up attitude of 10 deg.

In Fig. 14, the attitude responses to nose down hardovers for simplex, duplex and triplex systems are superimposed for comparison purposes. Apart from the higher speed in the simplex response, all flight conditions are similar. With the simplex system, intervention occurs at 1.25 s whilst with the duplex system, it is delayed until 3.5 s and could probably have been delayed somewhat longer. No intervention is, of course, necessary with the triplex system.

10 CONCLUDING REMARKS

In general terms and in most specific instances, the objectives established at the start of this development and trials programme have been met. Where this is not the case, it is maintained that this is due to the inherent limitations of a programme of this nature.

Considerable confidence has been gained, during the course of the trials, in the triplex redundancy philosophy and its implementation in the system flown. During this time there has been no evidence that the particular design features (e.g. software synchronisation) have not proved effective.

Following certain modifications to the control laws originally programmed, the digitised control laws provided satisfactory stabilisation and handling qualities.

Two particular problems encountered during the flight trials have highlighted the need with digital systems to consider very carefully, at the earliest opportunity the interface with the mechanical/hydraulic elements of the aircraft flying controls and the interaction between fundamental system parameters and aircraft structural modes.

Transient aircraft motion following specific simulated system defects has been determined in flight and in no case was short term pilot intervention required. In many cases the transient was within the general level of motion due to turbulence and could only be estimated. Certain defects resulted in transients in excess of the values quoted in the original Requirements Specification but it was felt that the response was still acceptable and that it may be advantageous to increase these motion limits.

Within the context of the flying tasks carried out, the pre-flight test routine is satisfactory, no fundamental problems have been exposed. However the routine in its present form is likely to be incompatible with operational flying requirements. Possible modifications have been suggested to cure this problem.

A reasonable level of EMC testing and experience has been accumulated, with the equipment both in and out of the aircraft. Corruption of the installed system by one externally radiated frequency occurred which requires further testing to assess the nature of the problem.

11 REFERENCES

1. P. Robinson, J.L. Hollington, et al.
"Helicopter Automatic Flight Control System for poor Visibility Operations" AGARD Conference Proceedings, No. 148, Paper 24.

2. P. Robinson.
Flight assessment of a digital, defect survival helicopter auto-stabiliser. RAE Technical Report 78121.

TABLE 1 - STABILISATION PERFORMANCE TESTS

Test No.	Test Item	HT.	500 ft			5000 ft		10000 ft	500 ft		
		Speed	Hover	70	100	50	80	55	Hover	70	100
1	Pitch stick pulses		✓	✓	✓	✓	✓	✓	✓	✓	✓
2	Pitch stick steps		✓	✓	✓	✓	✓	✓	✓	✓	✓
3	Pitch stick steps DSAS out		✓	✓	✓	✓	✓	✓	✓	✓	✓
4	Trimmed, hands off flight		✓	✓	✓	✓	✓	✓	✓	✓	✓
5	As 4 VGs out		✓	✓	✓	✓	✓	✓	✓	✓	✓
6	Roll stick pulses		✓	✓	✓	✓	✓	✓	c.g.near aft limit (3)		
7	Roll stick steps		✓	✓	✓	✓	✓	✓			
8	Roll stick steps DSAS out		✓	✓	✓	✓	✓	✓			
9	Trimmed, hands off flight		✓	✓	✓	✓	✓	✓			
10	As 9, VGs out		✓	✓	✓	✓	✓	✓			
11	Yaw Pedal kicks		✓	✓	✓	✓	✓	✓			
12	Co-ordinated turns		-	✓	✓	✓	✓	✓			
13	Flat turns using heading trim		✓	✓	✓	✓	✓	✓			
14	Torque changes		✓	✓	✓	✓	✓	✓			
15	Torque changes (1)		✓	✓	✓	✓	✓	✓			
16	Jump take-off		✓	-	-	-	-	-			
17	Jump take off (1)		✓	-	-	-	-	-			
18	Trimmed Hands off Flight		✓	✓	✓	✓	✓	✓			
			Central c.g. (2)								

(1) Heading hold disengaged for these tests

(2) AUW in the range 17 000-19 750 lb c.g approx -0.5 in (fwd of datum)

(3) AUW in the range 16 700-17 050 lb c.g in the range +4.6 in to +6.06 (aft of datum)

TABLE 2 - RATE GYRO OFFSET TESTS

Rate Gyro offset DEG/S	Pitch Axis		Roll Axis		Yaw Axis	
	Amber Warning	Attitude Changes	Amber Warning	Attitude Changes	Amber Warning	Heading Changes
± 11.0 ± 5.6 ± 2.8	Immediately	*	Immediately	*	Immediately	*
- 1.4	No	2 Deg NU	No	1 deg LWD	No	3 deg stbd
+ 1.4	No	1 deg ND	No	*	No	4 deg port
- 1.93	After 1.5s	*	After 2s	*	After 1s	3 deg stbd
+ 1.93	No	3 deg ND	After 2s	*	After 4s	2 deg port

* denotes no detectable attitude/heading change

All tests made with DSAS in triplex
Speed 100 kts IAS
Altitude 1000 ft
AUW in the range 17 000 - 19 750 lb
c.g approximately 0.5 in forward of datum

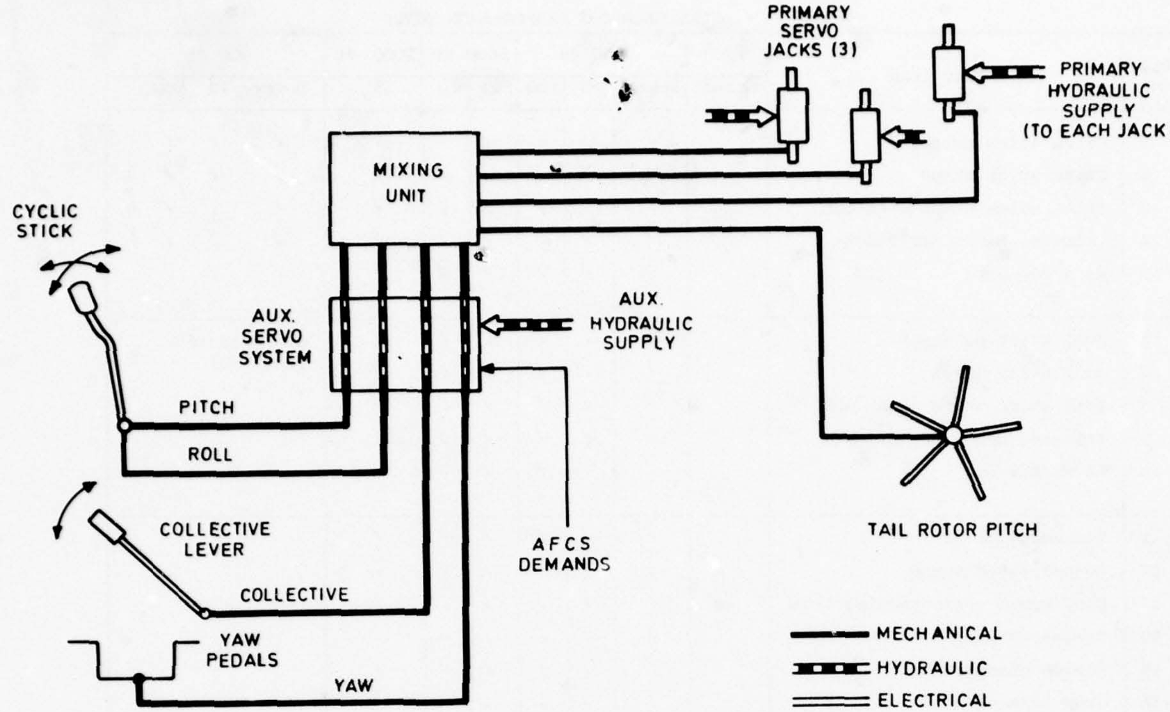


Fig.1 Schematic of flying controls

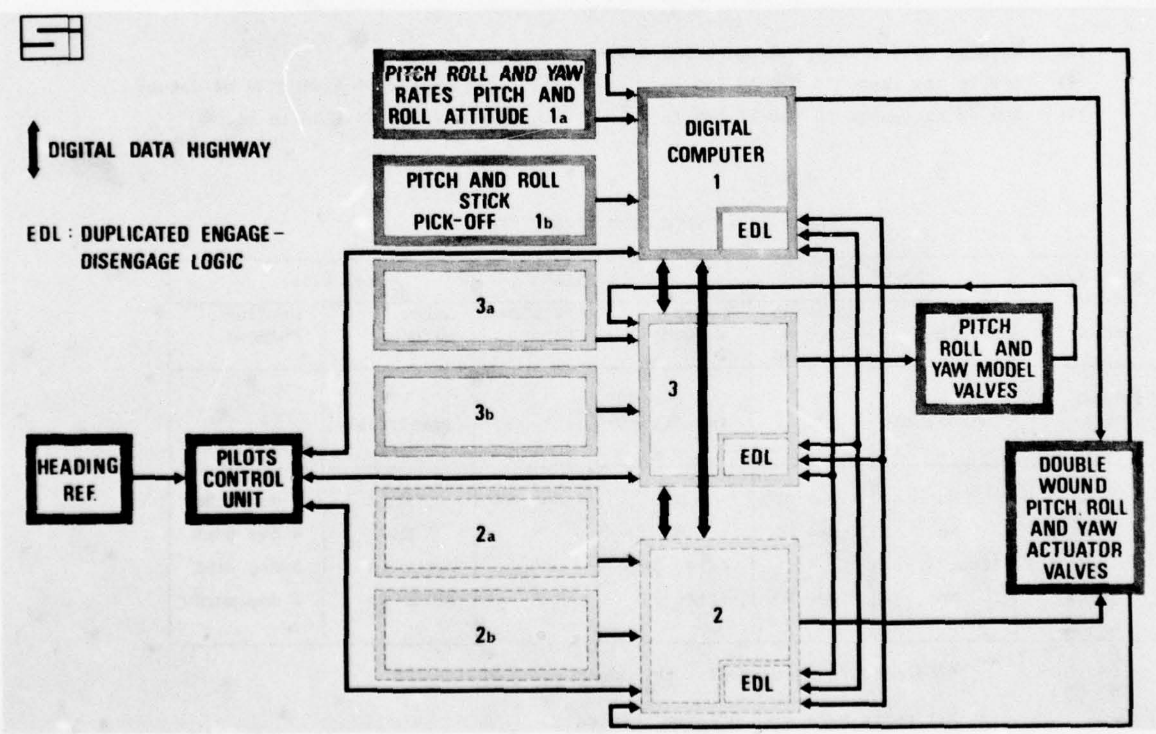


Fig.2 Schematic of triplex configuration

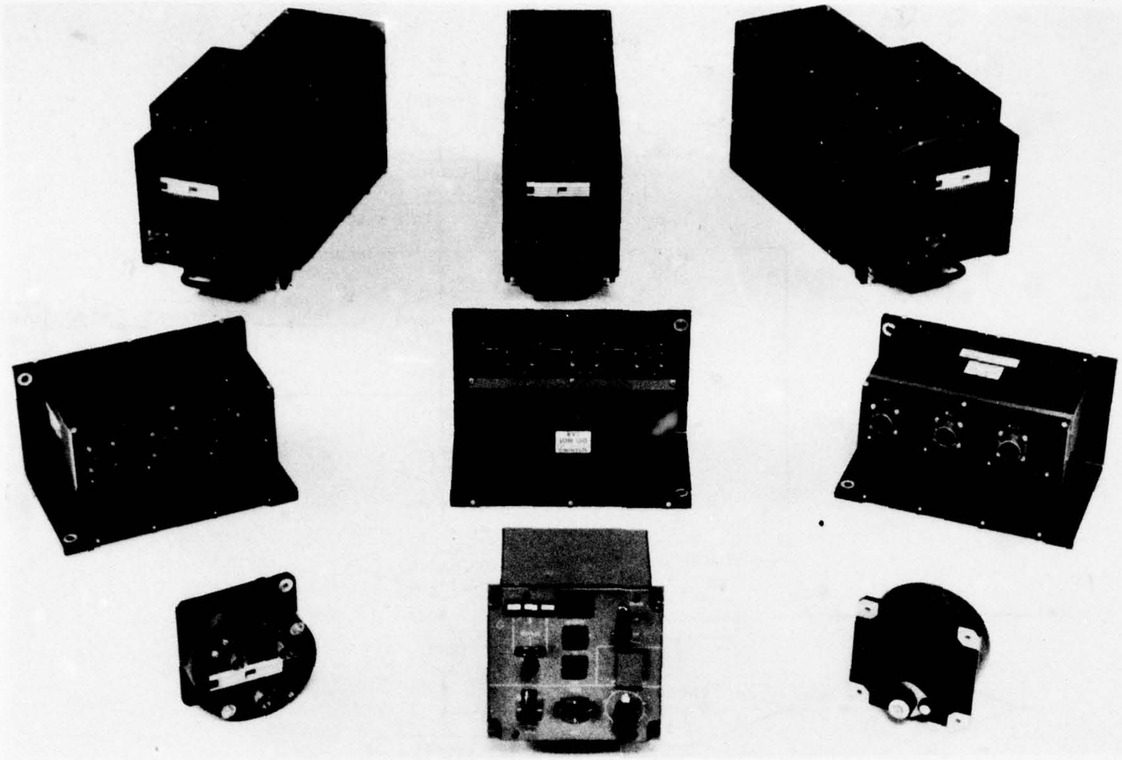


Fig.3 Triplex digital helicopter autostabiliser system

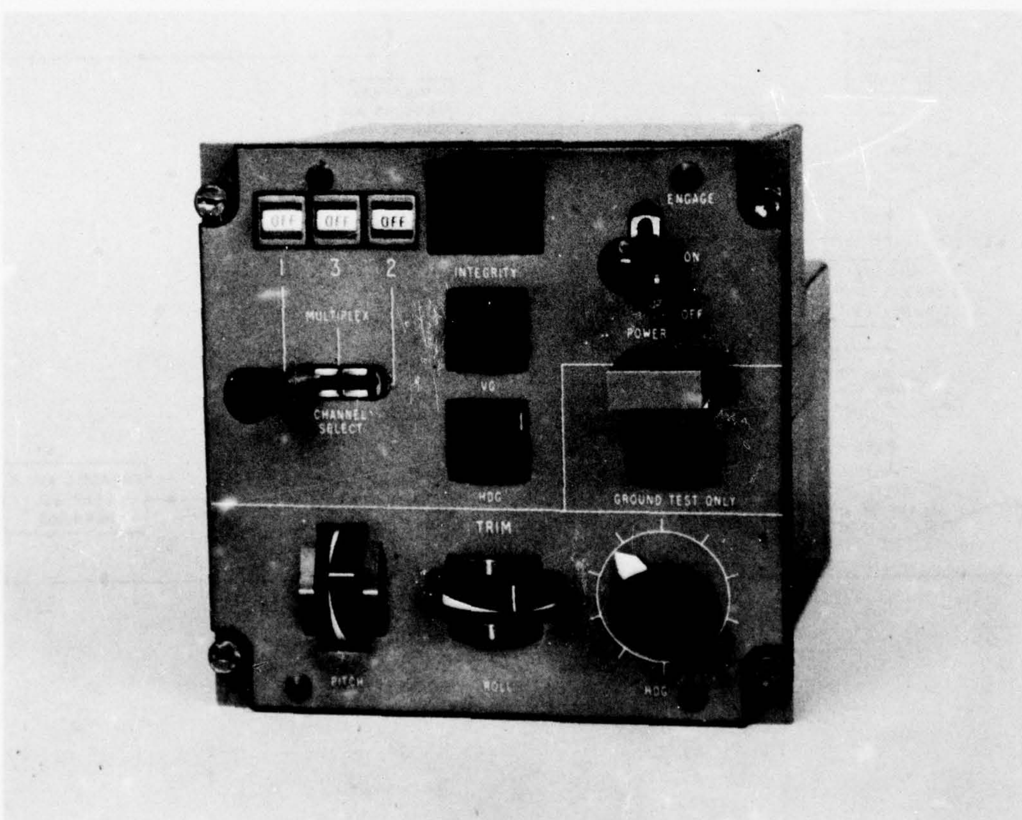


Fig.4 Pilots control panel

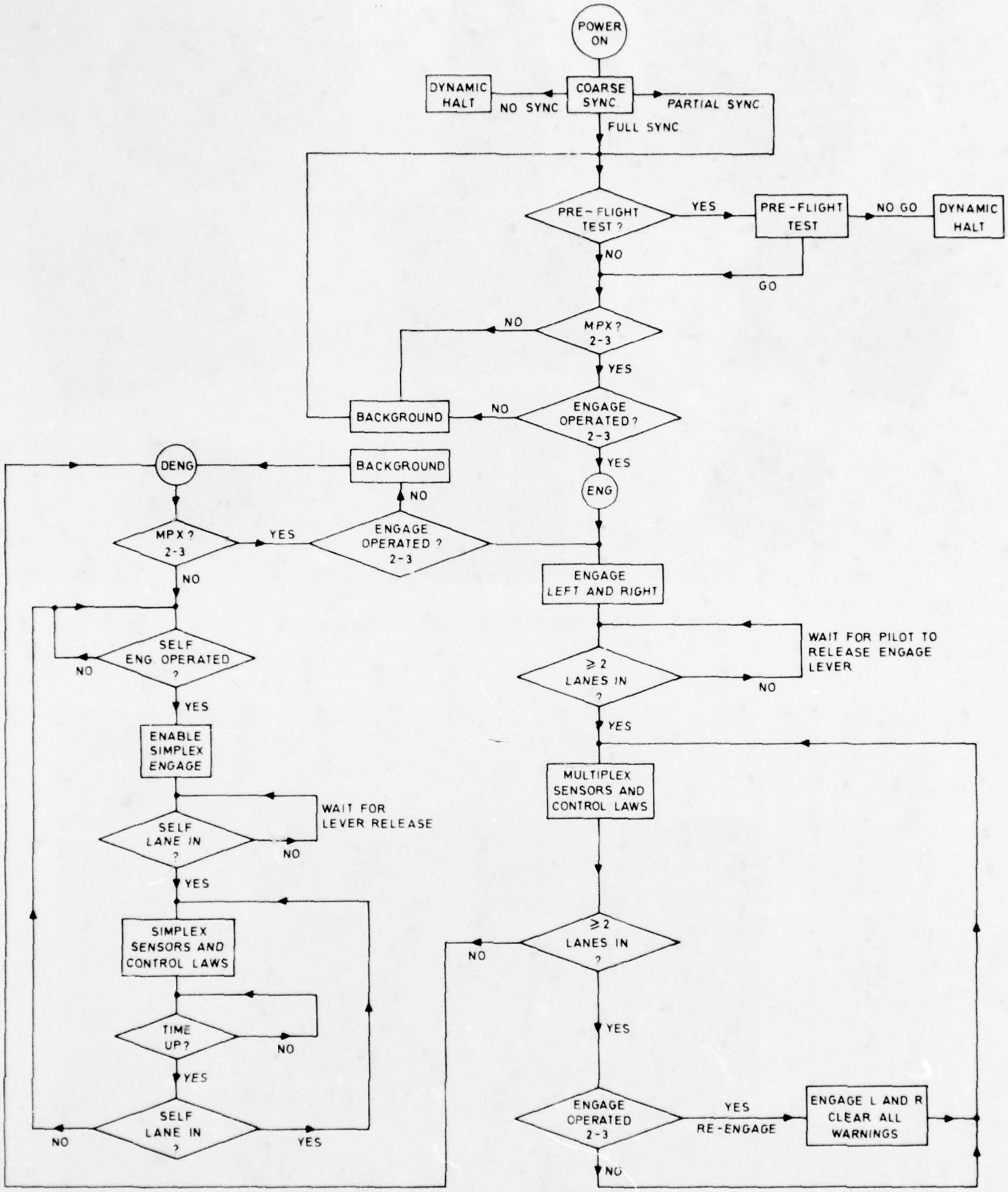


Fig.5 Executive program

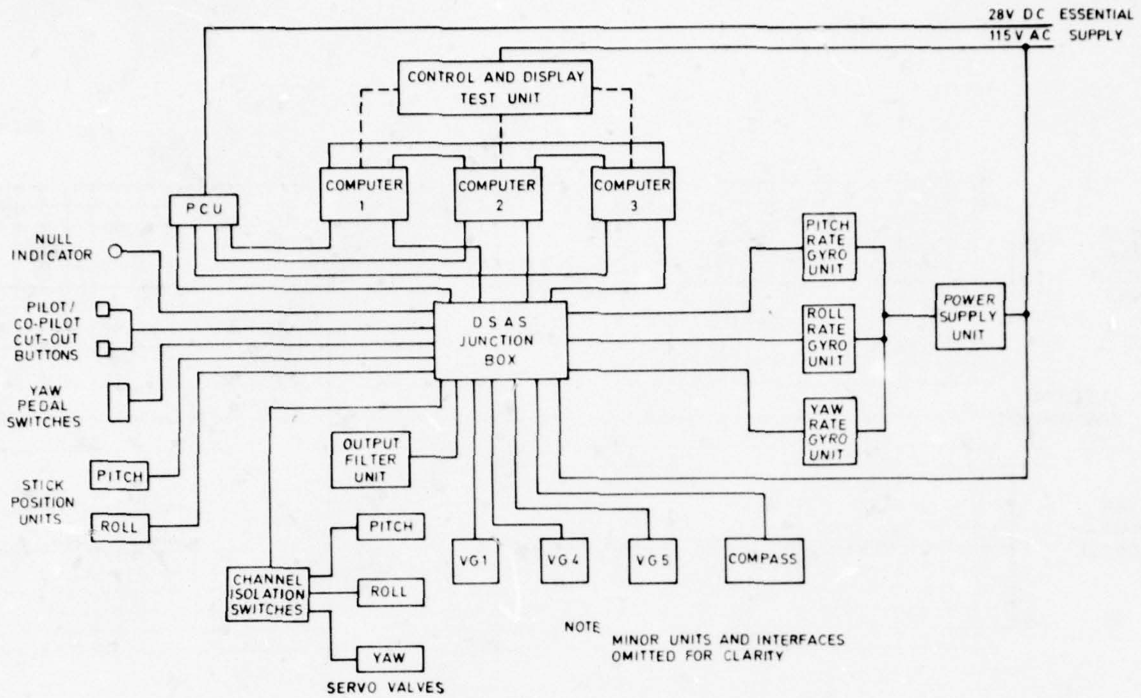


Fig.6 Schematic of aircraft installation

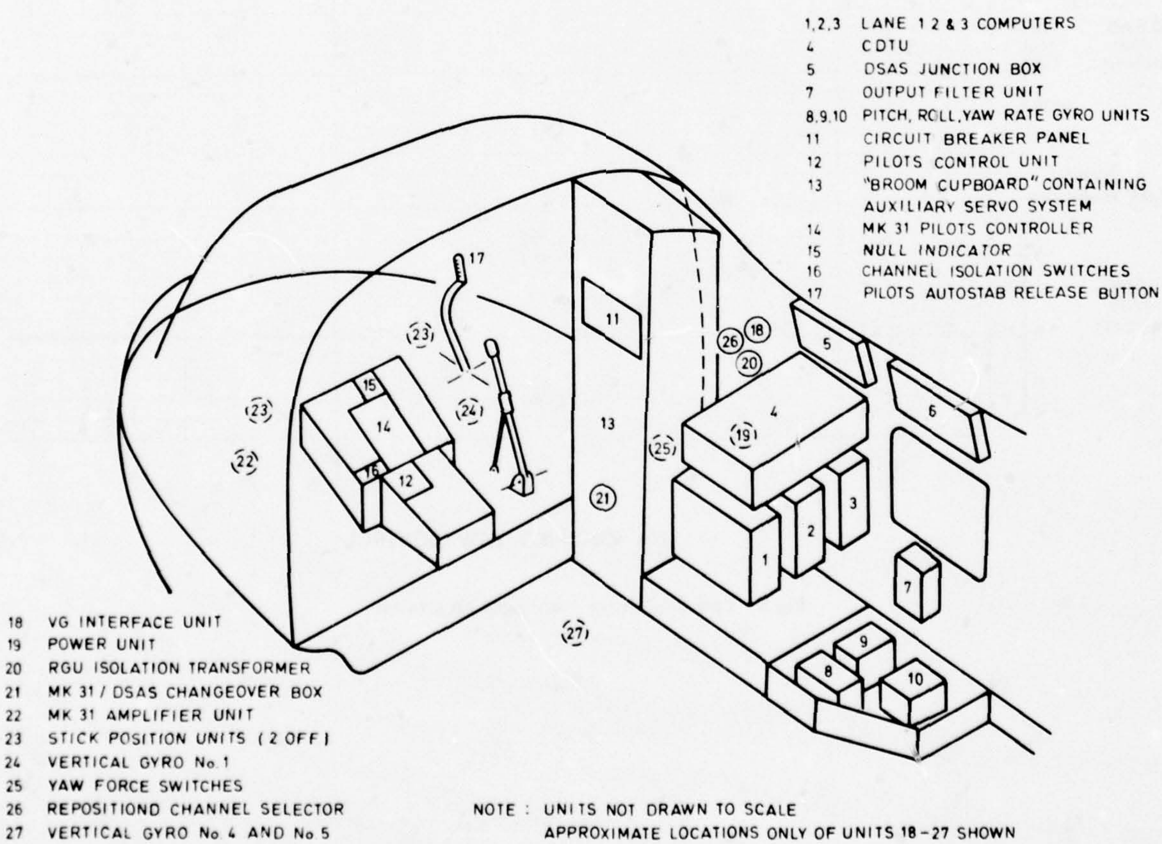
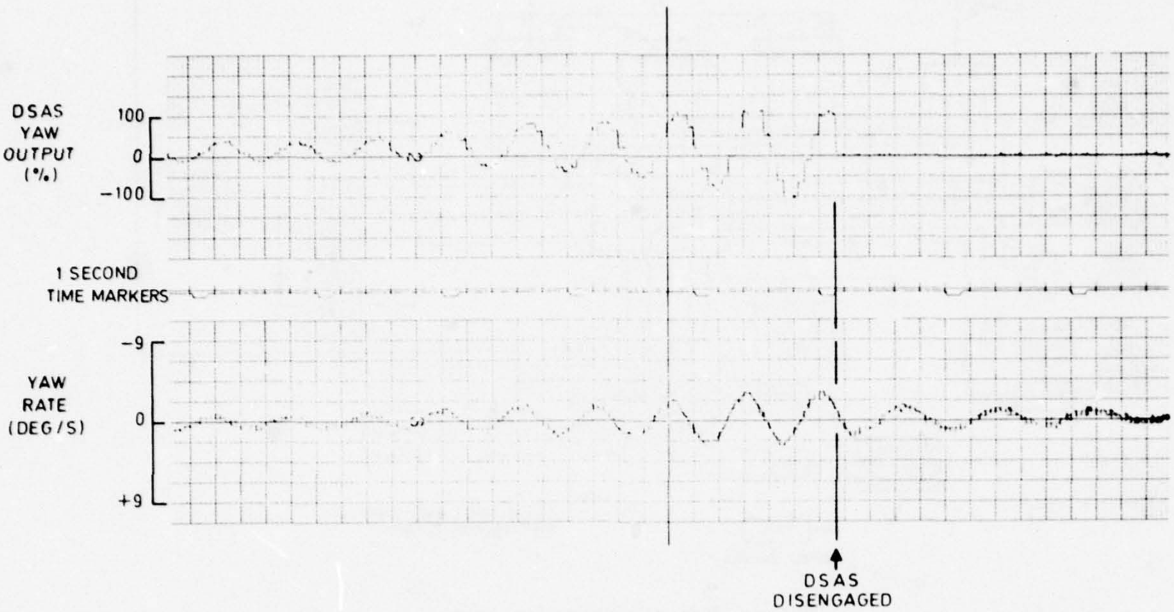
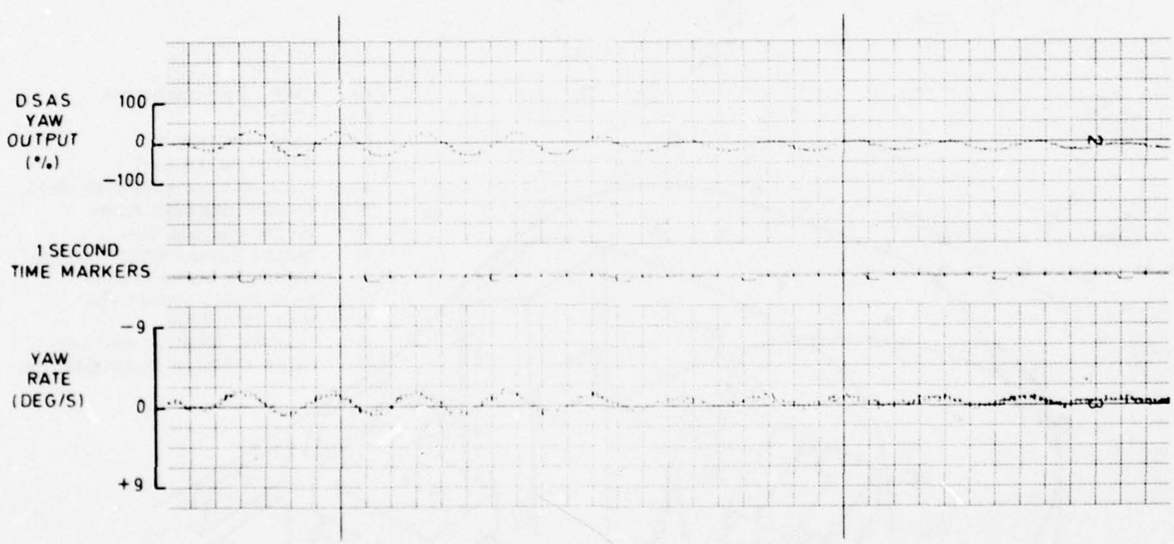


Fig.7 Location of units in the aircraft



(a) ORIGINAL YAW CONTROL



(b) MODIFIED YAW CONTROL

Fig.8 Yaw oscillation – a/c light on ground

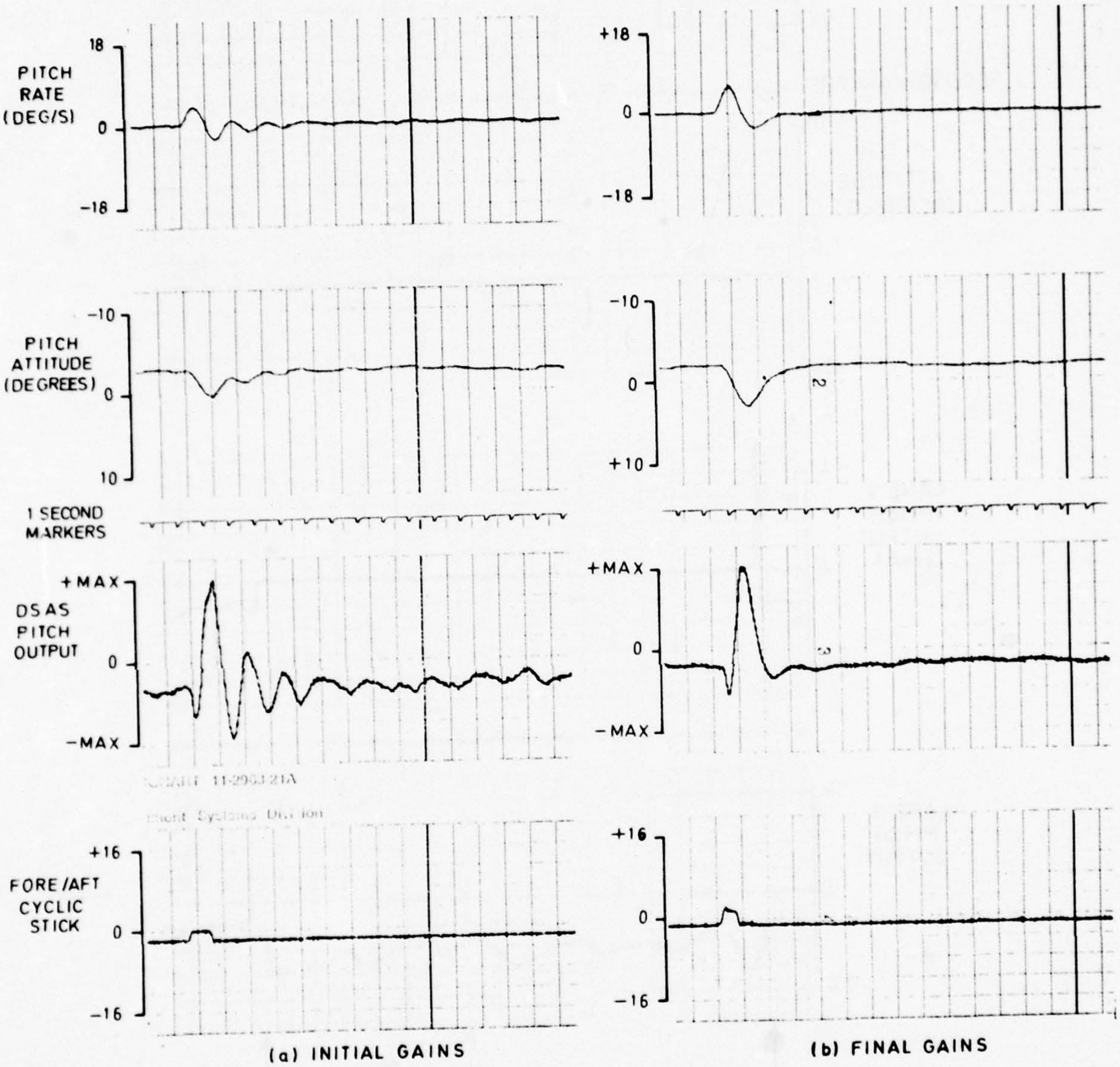


Fig.9 Pitch axis nose up pulse response at 100 kts

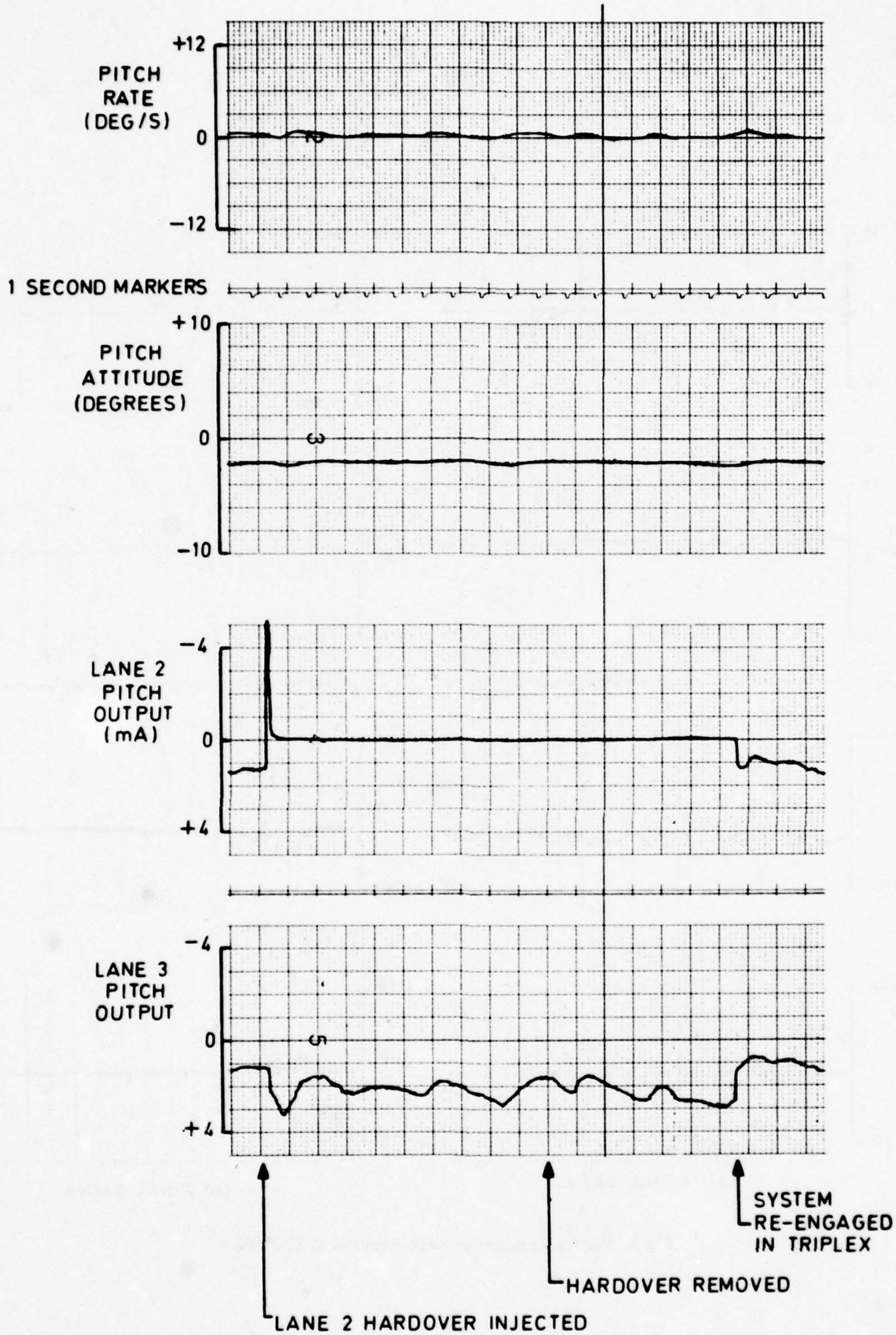


Fig.10 Nose down hardover

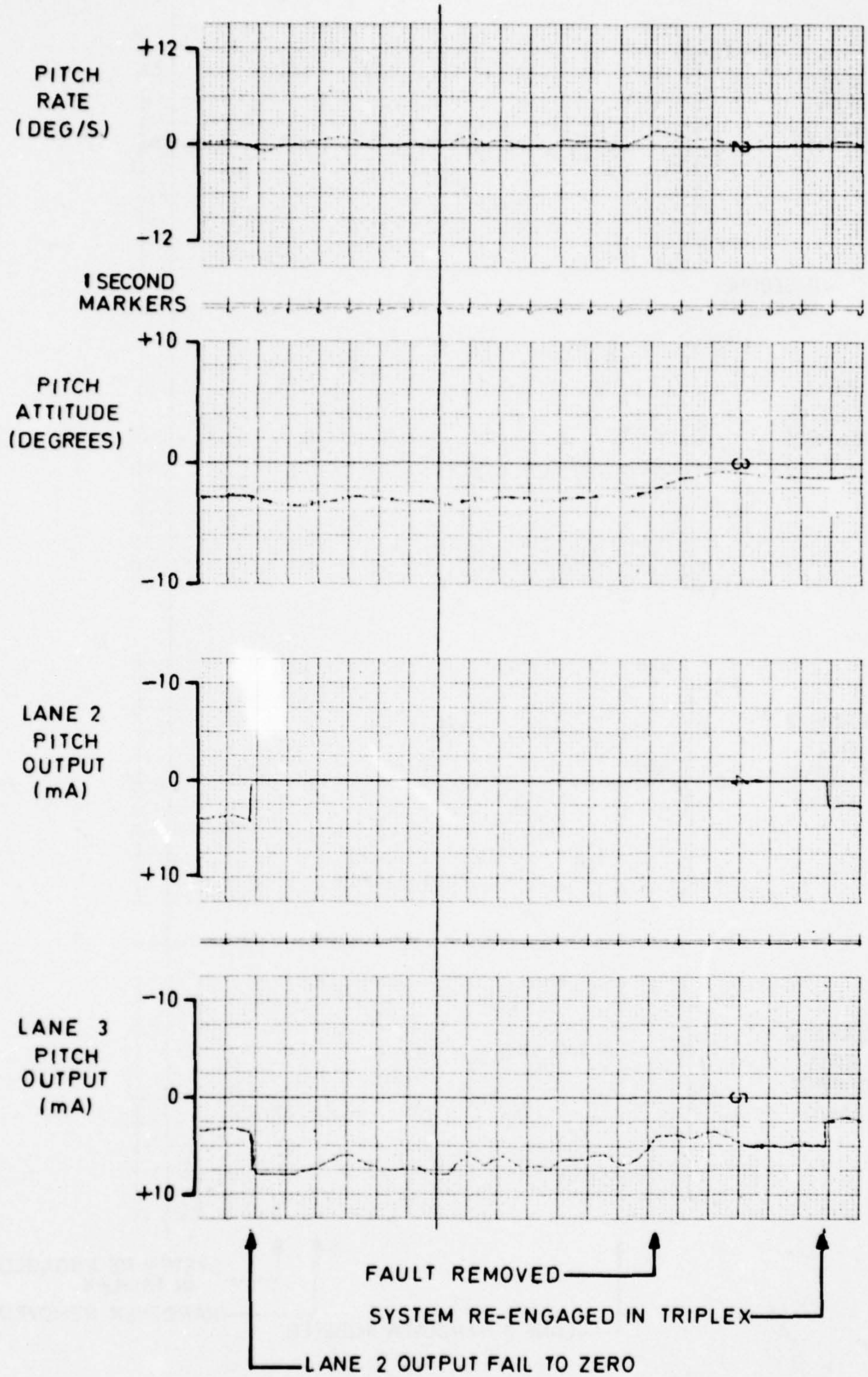


Fig.11 Pitch output fail to zero

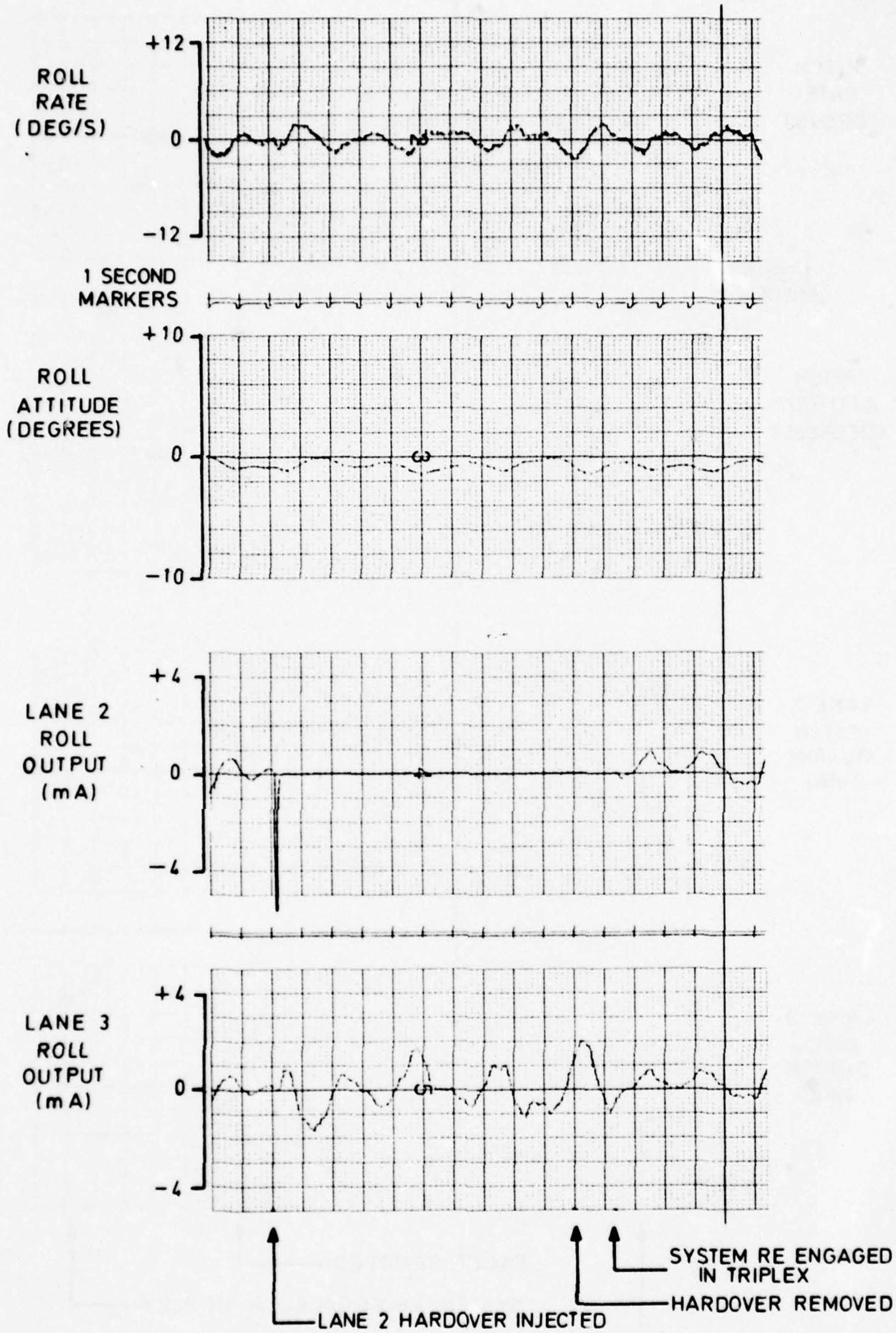


Fig.12 Left wing down hardover

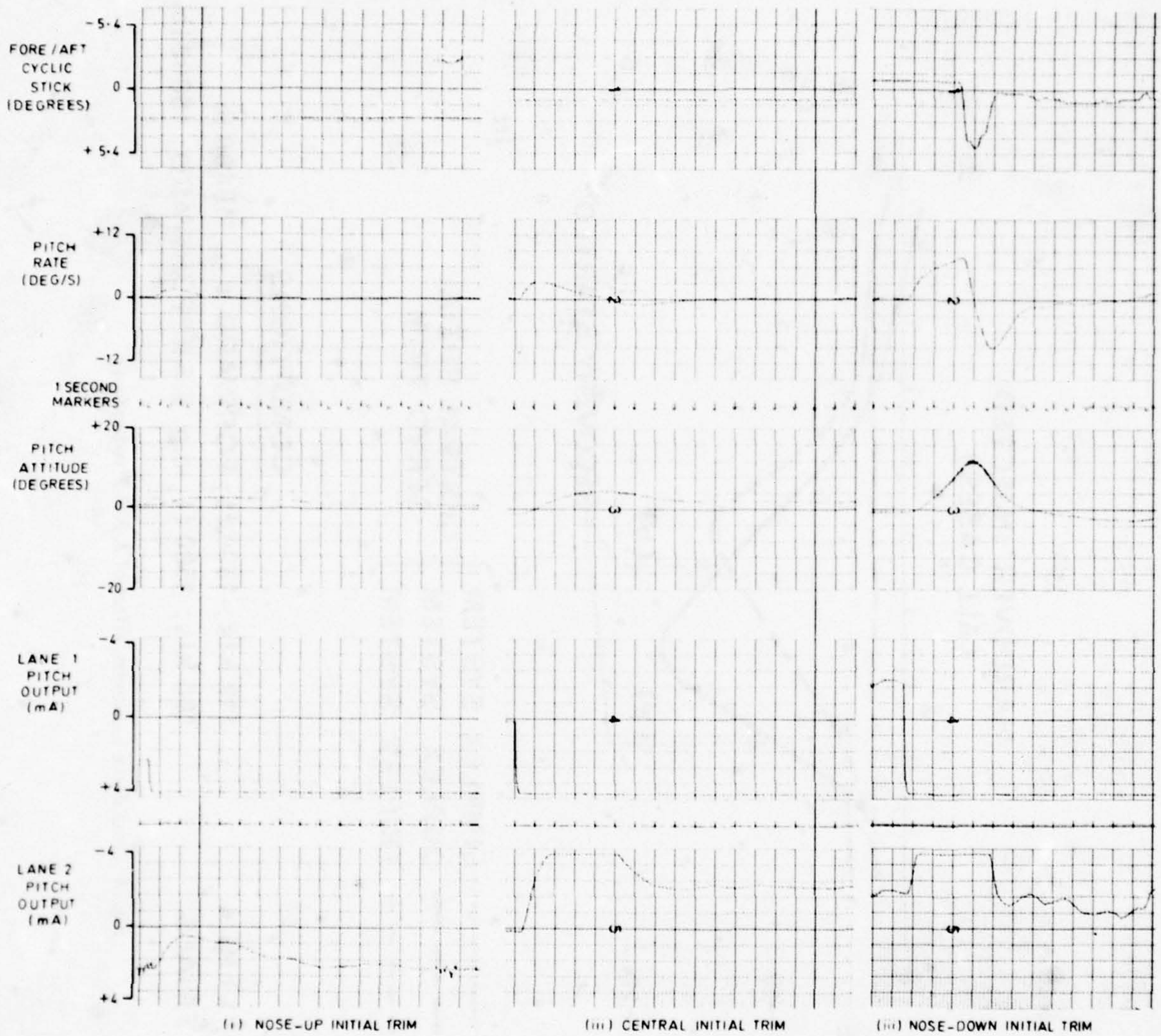


Fig.13 Duplex configuration, nose-up hardovers

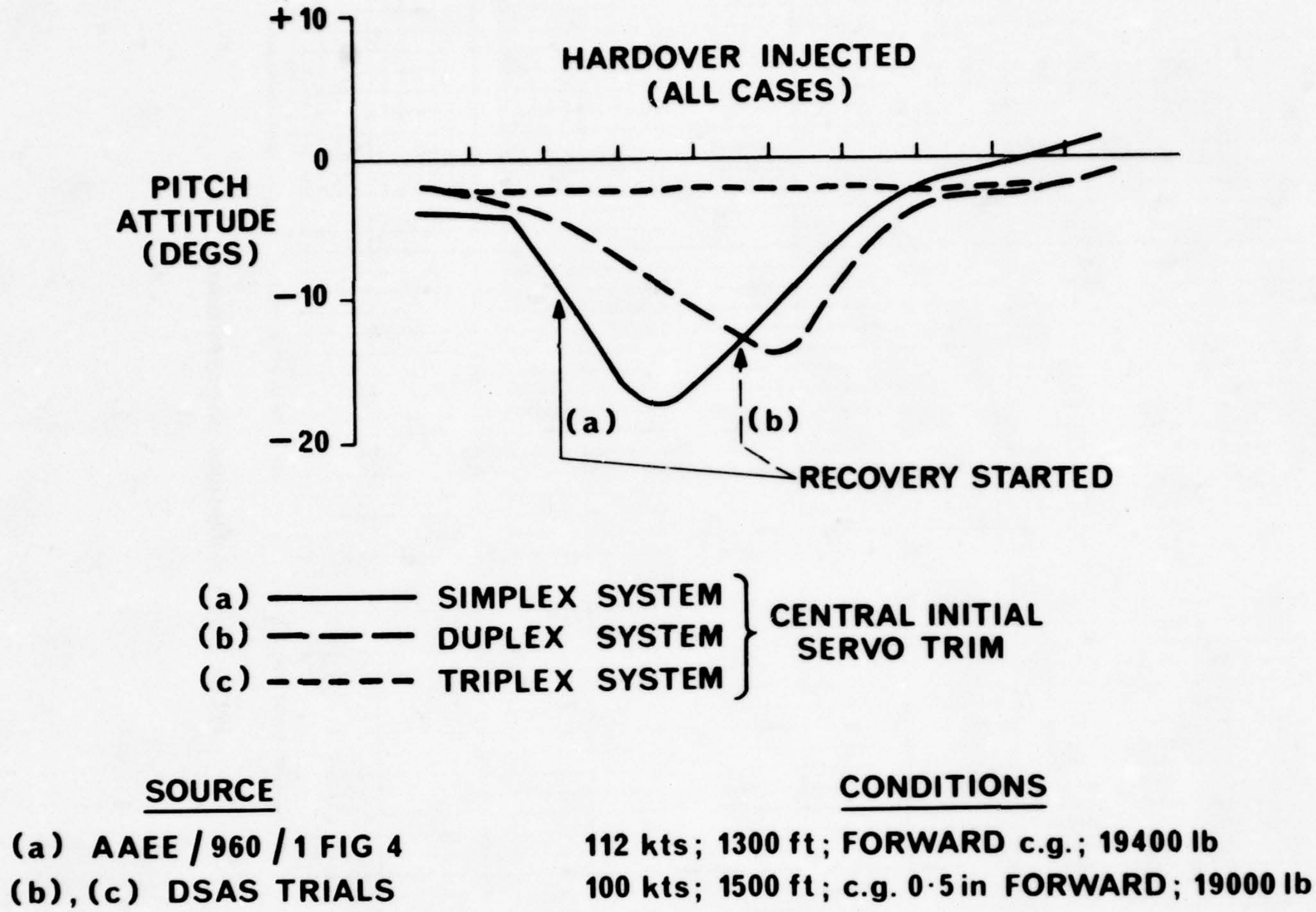


Fig.14 Nose down hardovers, effect of A.F.C.S. redundancy

SOME ASPECTS OF THE DESIGN AND DEVELOPMENT OF THE MARITIME AUTOPILOT MODES FOR THE WESTLAND LYNX HELICOPTER

K. S. SNELLING
Project Manager
Flight Controls Division
Marconi Avionics Limited
Airport Works
Rochester
Kent
England

and

M. V. COOK
Lecturer-Flight Mechanics
College of Aeronautics
Cranfield Institute of Technology
Cranfield
Bedford
England

SUMMARY

This paper describes the design and development of the Automatic Transition and Dunking Sonar Deployment Modes of the Autopilot fitted to some Naval versions of the Westland Lynx helicopter. Although development of these facilities has been in progress for several years and helicopters fitted with these autopilot modes are now entering service, the relevant design and implementation technology represent a significant step forward in helicopter systems. These automatic manoeuvring modes were developed for specific naval operational requirements, but applications for this type of autopilot facility are envisaged and are actively being investigated, for more general offshore helicopter operations including approach aids, air sea rescue activities and survey applications.

After a brief review of the basic stability augmentation system fitted to the Lynx Helicopter, the paper describes the Synchronised Transition autopilot mode. This leads to the requirement for Sea State Filtering of the height control signal and the development of the filter presently used in the Lynx autopilot is described.

The paper concludes with a review of the design and development of the Cable Angle and Cable Height hold autopilot modes that together provide the Sonar modes for stabilising the position of a dunking sonar.

1. STABILITY AUGMENTATION SYSTEM

To fully appreciate the way in which the Maritime Autopilot modes interface with the flight control system it is useful to review the stability augmentation system (SAS) fitted to the Lynx helicopter. This review is an outline description only of the key features of the SAS; for a more detailed discussion the reader is referred to the paper 'Some Design Aspects of the WG13 Rigid Rotor Helicopter' (Ref. 1).

The Lynx is fitted with a full four axis stability augmentation system in which the computing and series actuation are duplicated. In each control axis, the outputs of both lane series actuators are mechanically force summed to provide a single output to the power control unit and mechanical flying controls. This output is mechanically limited to nominally 10% of the total available blade angle travel in each control axis, and this authority limitation is a significant factor which greatly influenced the initial design of the SAS.

The flight control system is implemented using analogue computing technology. Although using the most up to date analogue electronic components, reliability, size and cost restraints have required that the SAS, and indeed the autopilot control laws remain relatively simple: a further constraint on the design process. Consequently maximum use has been made of fixed gains and fixed time constant control filters. Long term storage of a parameter datum always presents a problem in analogue systems. The Lynx automatic flight control system (AFCS), utilizes an electro-mechanical store for the heading datum and a digital store for height. Future systems based on digital processors will remove many of the design constraints since scheduled filters and gains and datum stores can be incorporated with minimal hardware penalties.

1.1 Pitch Axis Autostabiliser

One lane of the pitch SAS and its interface with the mechanical flying controls is shown in Figure 1. The control system shown is much simplified and shows only the key elements.

The mechanical controls reflect standard helicopter practice and incorporate an artificial feel unit, a trim motor for manual adjustment of the datum and a mechanical summing device such that the output from the SAS series actuators is incorporated into the drive to the power control unit. As the SAS

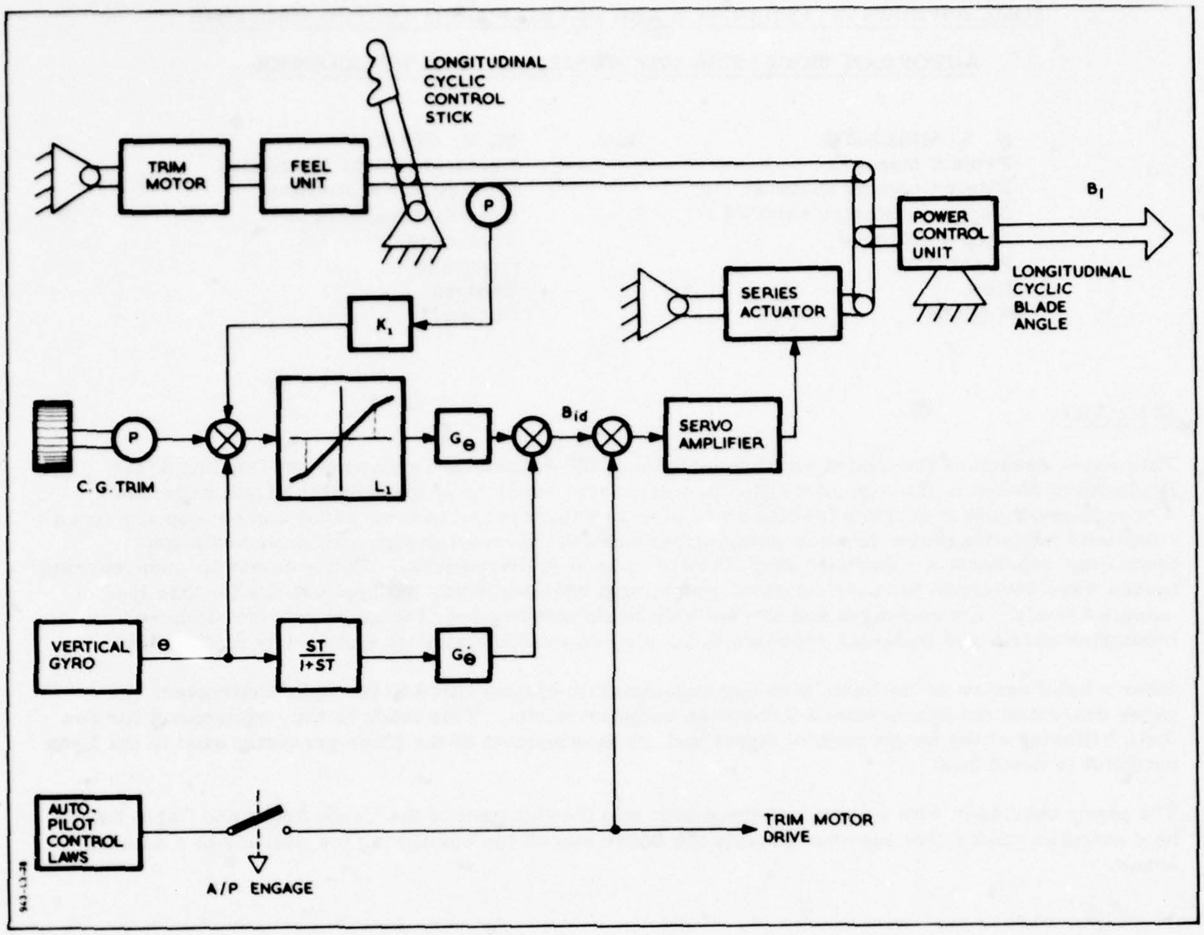


Figure 1 Pitch Autostabiliser

feedback loop tends to oppose any pilot manoeuvre inputs, the pilot demand is electrically sensed and a cancelling signal via the gain K_1 is fed into the SAS control law.

Stability augmentation is provided by sensing pitch attitude and feeding back the attitude and derived pitch rate. In view of the characteristics of the basic helicopter it was found necessary to utilize a gain range on G_θ which would contravene the control authority limitations. It was therefore necessary to incorporate the limit function L_1 which has a gain of 1 for values of θ less than $\pm 5°$, hence allowing full stabilization in this attitude regime and, having a reduced gain for larger attitude excursions. Therefore stability augmentation is somewhat impaired during manoeuvring flight but, as this normally occurs under pilot control, it is acceptable.

The C. G. trimmer is simply a means which allows the pilot to trim the SAS electrically such that the series actuators operate about a null position. This facility is necessary in view of the limited authority available to the series actuators for control.

When the transition autopilot mode is engaged the pitch axis control law demands a continually changing trim condition as the speed is wound off. This would cause series actuator saturation if some means were not included to prevent it. Although the circuitry is not shown on Figure 1, the autopilot output demand also drives the trim motor directly via a hysteresis switch function. The characteristics of this function are described in greater detail in the yaw axis SAS description. By this means the basic trim of the pitch control axis is updated at intervals which prevents saturation of the series actuator.

1.2 Roll Axis Autostabiliser

One lane of the roll SAS and its interface with the mechanical flying controls is shown on Figure 2. The control system is shown in its simplest form and in this respect is very similar to that described for the pitch axis.

Other than the form of the feedback control law the pitch SAS and the roll SAS are very similar. Stability augmentation is again provided by feeding back roll attitude and derived roll rate. In view of the requirement for stabilization over large attitude excursions and the incompatible limit on control authority the attitude term is limited to small excursions by the limit L_2. Thus at large bank angles

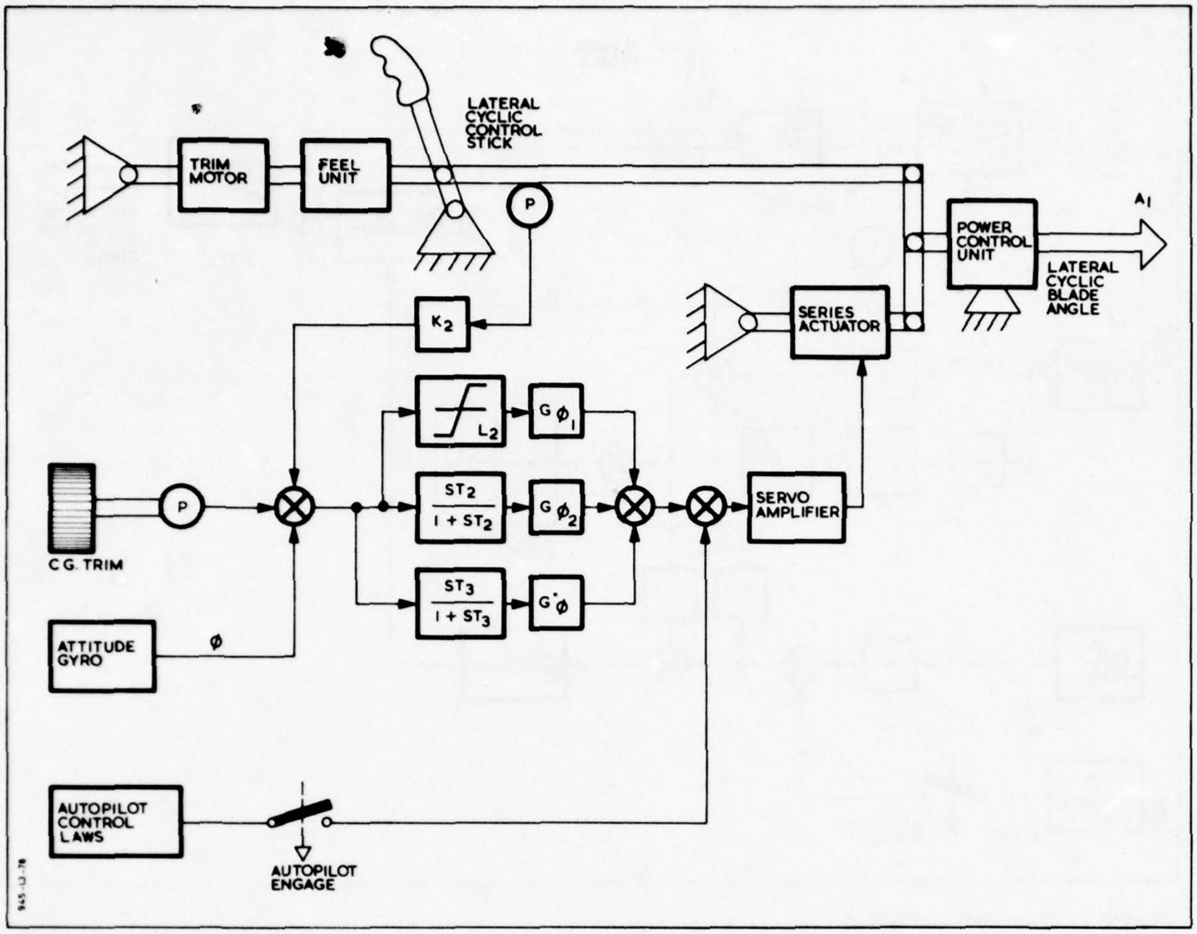


Figure 2 Roll Autostabiliser

stability is provided mainly by the derived rate term with gain $G_{\dot\phi}$. Additional stability is required at the low speed end of the flight envelope and in the hover, this is provided by the washed out attitude feedback term with gain G_{ϕ_2}. In this case the time constant T_2 is very much longer than the time constant T_3 in the derived rate feedback path. In order to restore stability to an acceptable level at higher speeds the time constant T_2 is reduced by means of a simple speed switched filter arrangement.

1.3 Yaw Axis Autostabiliser

The yaw axis autostabiliser incorporates a duplex yaw damper and a simplex heading hold autopilot. One lane of the yaw damper and its interface with the heading hold autopilot and the mechanical flying controls is shown in Figure 3.

The mechanical flying controls in the yaw axis are essentially similar to those in the pitch and roll axes, but the yaw axis has an automatically controlled parallel trim motor. Small amplitude demand signals are fed to the series actuator but, because of the very large yaw amplitudes possible, it is necessary to transfer most of the demand to the parallel trim motor in order not to saturate the series actuator. Whenever the series actuator demand exceeds about 50% of its total authority a hysteresis switch operates to transfer the demand to the parallel trim motor, thereby leaving the series actuator with authority to cope with the higher frequency stability augmentation function.

The yaw damper utilizes yaw rate feedback which is fed via the simple gain G_r to the series actuator. The heading hold autopilot control law receives a heading error signal from an electro-mechanical store, the error then feeds through a proportional plus integral function to provide the demand signal. The heading demand signal feeds to the parallel trim motor drive where it is summed with a component of the yaw damper demand before feeding to the hysteresis switch. Similarly a component of the heading demand signal is summed with the yaw damper demand to form the series actuator drive signal.

To facilitate manual manoeuvring in yaw it is necessary to disengage the heading hold autopilot and also reduce the gain G_r. This function is controlled by pressure switches on the rudder pedals.

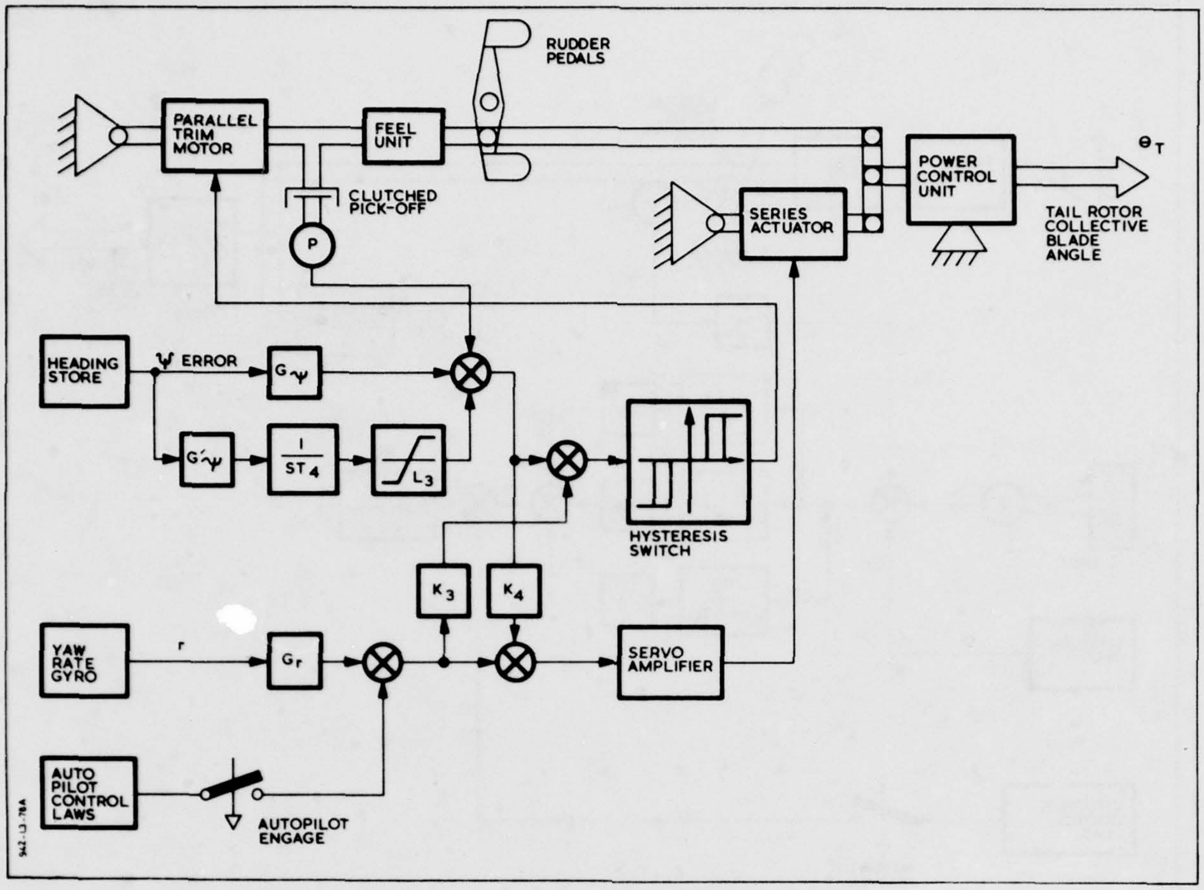


Figure 3 Yaw Autostabiliser

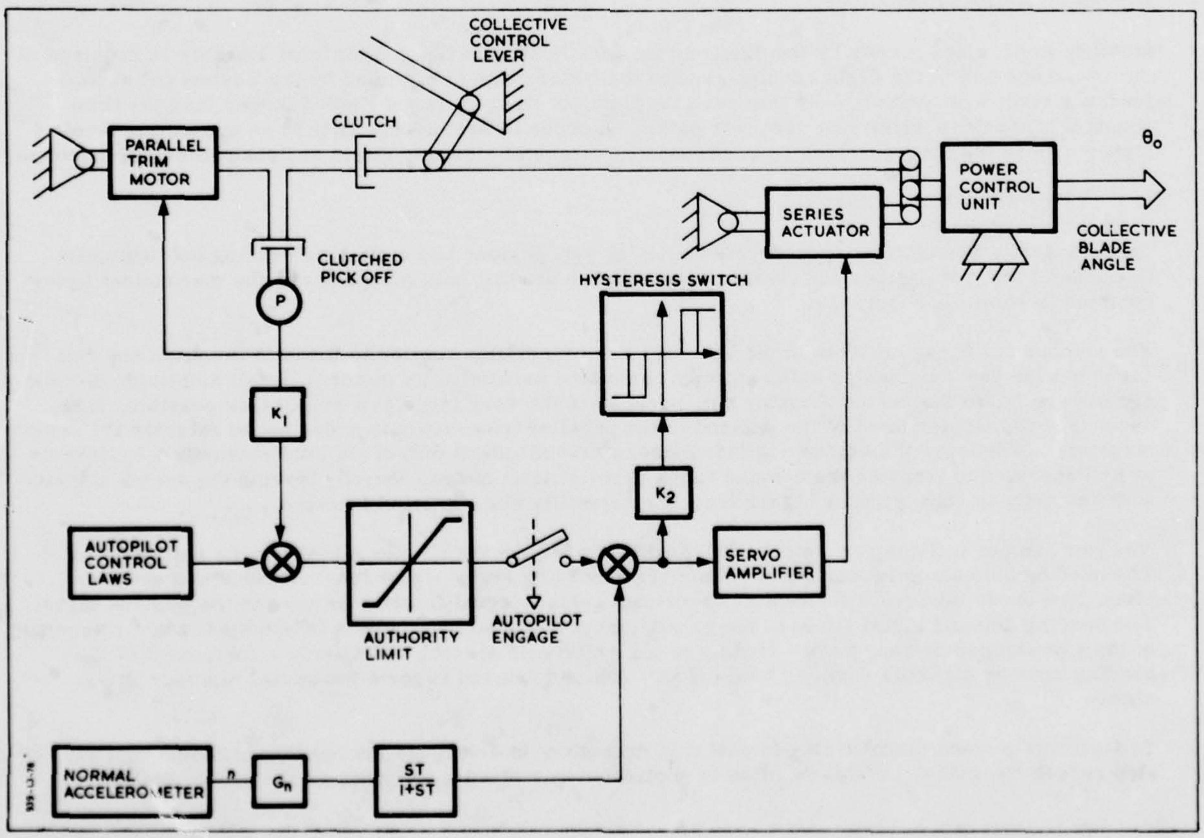


Figure 4 Collective Autostabiliser

1.4 Collective Axis Autostabiliser

At high forward speeds the Lynx helicopter has a tendency to instability in the pitch axis. The control of this characteristic solely in the pitch axis necessitates increased control authority which would lead to an unacceptable runaway behaviour. Thus it was necessary to provide additional independent stability augmentation. This was incorporated in the collective axis control, one lane and its interface with the mechanical flying controls is shown in Figure 4.

Rapid change in pitch attitude gives rise to a normal acceleration change which is sensed, fed back through the gain G_n, and the washout filter with time constant T , to provide the drive to the collective axis series actuators. The washout filter is necessary to remove long term acceleration signals which would be experienced in manoeuvring flight.

The mechanical flying controls incorporate a parallel trim motor which is only clutched to the mechanical output during automatic manoeuvring involving collective control. The operation of the parallel trim function is otherwise similar to that in the yaw axis and is used to remove large trim offsets from the series actuator which would otherwise result from autopilot control.

2. AUTOMATIC TRANSITION AUTOPILOT

The automatic transition autopilot provides the facility for a controlled precision translation down from an initial height and speed datum to the hover at a very much lower height. The manoeuvre entry and exit conditions are variable within defined ranges. To achieve the required linear glideslope profile it is necessary to control height and speed synchronously, whilst heading and lateral speed control ensure that the helicopter maintains constant heading throughout the manoeuvre. A summary of the manoeuvre is shown in Figure 5.

The Transition provides a linear descent profile by synchronously maintaining constant deceleration in both the horizontal and vertical axes. At engagement, the time to complete the manoeuvre is defined by the forward ground speed and the fixed deceleration in the speed control law; the deceleration is chosen to give a suitable nominal execution time for the range of entry speeds. The

Figure 5 The Transition Manoeuvre

vertical manoeuvre is therefore programmed at engagement to reduce height with constant vertical deceleration, which is a function of both initial speed and height errors. Both the pitch axis and collective axis demand signals are controlled to produce a smooth entry into the manoeuvre which is essential to prevent a large build up of pitch attitude. The control laws are configured in such a way as to ensure that the height transition finishes at the same time or before the speed transition to avoid the undesirable condition of vertical descent at low altitude. The flare out of the manoeuvre is approximately exponential and is controlled in such a way as to prevent undershoot and overshoot of the final height and speed values. On completion of the manoeuvre the autopilot automatically switches into the hover hold mode.

2.1 Pitch Axis Control

Control of forward speed during the transition manoeuvre is effected by means of the pitch axis control system shown in Figure 6.

The transition speed error U_e is continuously computed by comparing the preset exit speed U_H, which is nominally zero, with the doppler derived ground speed. The speed error passes through the gain G_u and amplitude limit L_2 to produce an acceleration demand signal $\dot{U}_d$. During the linear descent period of transition the limit L_4 is saturated to ensure a constant value of acceleration demand. As the speed approaches the final value the speed error U_e diminishes to a level such that the limit L_4 is no longer saturated. At this point

$$\dot{U}_d = G_u U_e \qquad (1)$$

and the exponential flare to the final value of U_H commences with G_u effectively controlling the time constant of the flare.

The acceleration error signal provides the demand to the longitudinal cyclic control system and is computed by comparing the demand $\dot{U}_d$ with derived forward acceleration feedback $\dot{U}$. Forward acceleration feedback is derived from doppler ground speed U by means of a washout filter with time constant T_6 and a suitably chosen gain $G_{\dot{u}}$. The doppler speed signal is compensated for attitude,

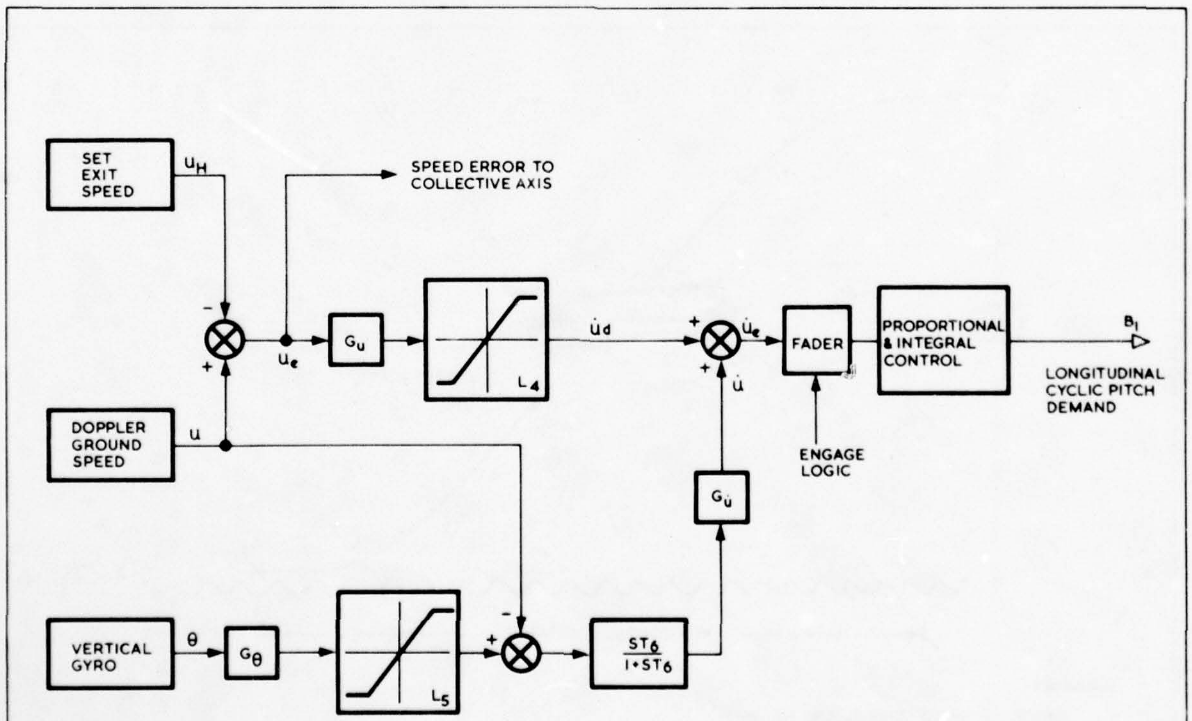


Figure 6 Longitudinal Cyclic Pitch Transition Control System

and hence speed changes, by means of the pitch attitude feedback through the gain G_θ and amplitude limit L_5. The limit L_5 is to prevent autopilot saturation for large attitude transients which could occur at commencement of the manoeuvre and would modify the transition profile undesirably. The acceleration error signal $\dot{U}_e$ passes through the fader circuit and thence a standard proportional plus integral autopilot control law.

The resultant longitudinal cyclic pitch demand signal then feeds into the pitch axis autostabilization computing. The fader circuit is designed to provide a smooth entry to the manoeuvre at engagement to prevent very large pitch attitude transients developing.

G_u was selected by experimental analysis to provide an exponential exit flare. In practice the selected value of G_u required that both axes of the transition manoeuvre were operated simultaneously in order to obtain a satisfactory flare for all entry and exit conditions.

2.2 Collective Axis Control

Control of height during the transition manoeuvre is effected by means of the collective axis control system shown in Figure 7.

The control signal is derived from radio altitude. To obtain a linear glideslope the vertical acceleration must be constant and synchronized to the forward speed transition except during the exit flare. This is achieved by controlling height rate during the descent.

2.3 Implementation of Control Law

Referring to Figure 7, the controlling height h_f and height rate $\dot{h}_f$ signals are obtained from the Sea State Filter circuitry, which filters raw radio altitude and normal acceleration in a complementary fashion. The derivation of the filter is discussed later. The controlling height error h_e is obtained from the height signal h_f and the preset hover height h_H. Height error is then passed through the square root circuit, multiplied by the constant K and rate limited by L_6 to provide the height rate demand signal $\dot{h}_d$.

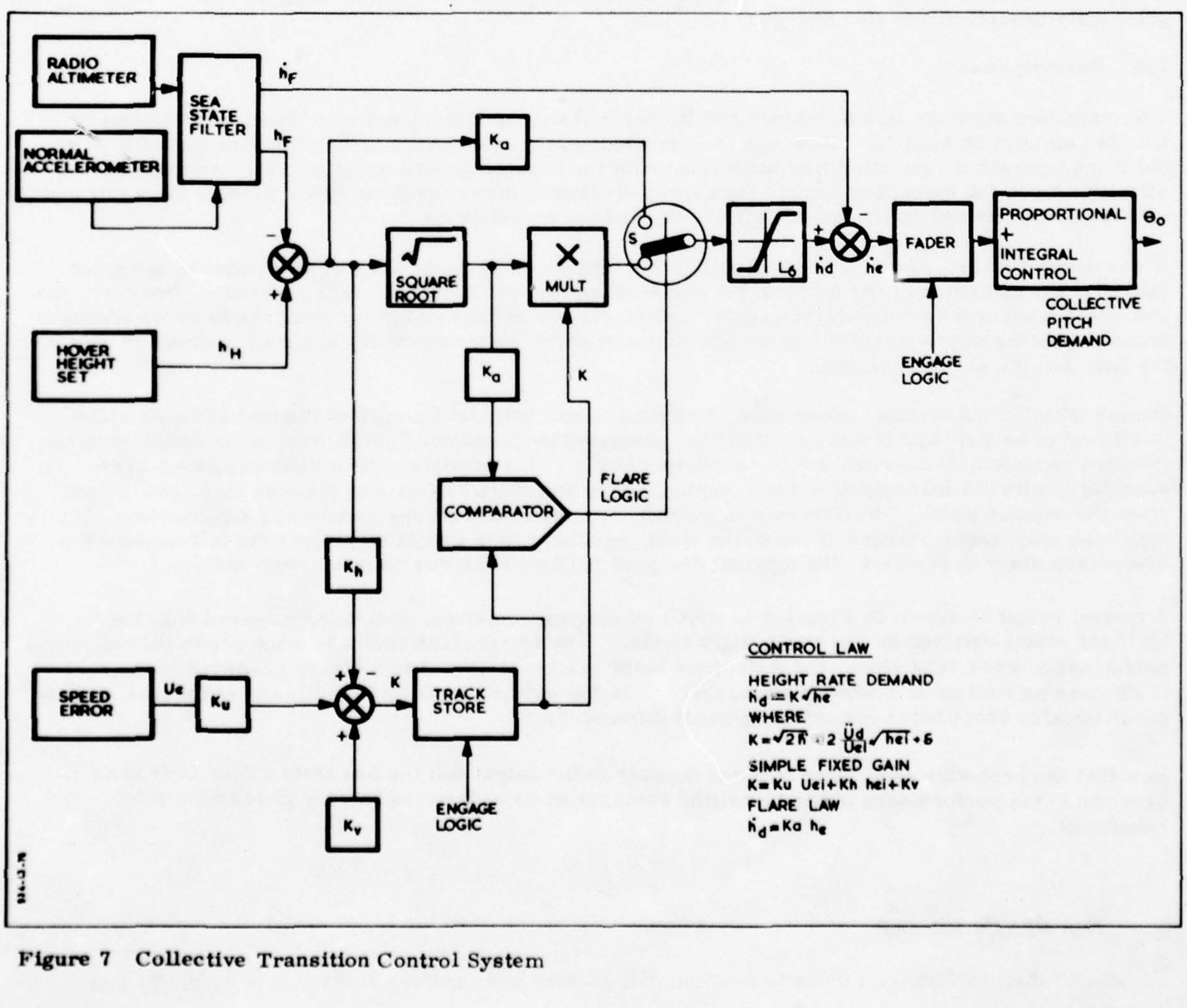


Figure 7 Collective Transition Control System

Height rate error $\dot{h}_e$ is computed from height rate demand $\dot{h}_d$ and height rate $\dot{h}_f$, which is then passed via the engagement fader to the autopilot loop.

The fader circuit ensures a smooth entry to the manoeuvre and the autopilot computing function, provides a standard proportional plus integral function suitably chosen to give good tracking characteristics during the synchronized manoeuvre.

The constant K is computed continuously by taking the speed error u_e, from the pitch control law and the height error h_e. The track-store circuit tracks K until the manoeuvre is engaged. At that instant h_e and u_e become the initial values h_{ei} and u_{ei} respectively thus, the track-store circuit outputs the stored value of K which is correct for the manoeuvre.

The exit flare in the height transition manoeuvre is arranged to take place over the last few feet of height error. The comparator continuously compares K with $K_a h_e^{\frac{1}{2}}$ and switches at the appropriate value of h_e. At this point the switch S is operated by the switching logic changing the height rate demand control law.

This control law results in an approximately exponential exit flare in height and the constant K_a is chosen to give a flare which is adequately synchronized with the flare in speed.

2.4 Lateral Axes Control-Roll Axis

When the transition autopilot mode is engaged a simple roll autopilot function is activated. The control law has two functional terms, the first provides additional roll rate damping by feeding back a pseudo rate term derived from roll attitude and the second provides a lateral velocity hold function. Lateral ground speed is derived from the doppler-radar and is summed with a preset lateral velocity demand v_H, which is nominally zero. The resultant lateral speed error is then summed with the additional rate feedback term and the total demand signal sums into the autostabiliser control law at the appropriate point, via the proportional plus integral autopilot loop control law.

2.5 Lateral Axis Control-Yaw Axis

When the transition autopilot mode is engaged the yaw autostabiliser and heading hold autopilot as previously described are also engaged.

2.6 Performance

The transition autopilot was developed and its performance evaluated with the Westland Helicopters Limited simulation facility. However, the development model did not include the Sea State Filter nor did it incorporate the peculiarities associated with the doppler ground speed signal. With those simplifications the transition control laws were developed very considerably to embrace the entry and exit requirements and to include the effects of wind speed and gusts.

It was found that the system was not significantly influenced by gusts but that variations in headwind caused the termination of the manoeuvre to undershoot or overshoot by small amounts. However, the transition control laws were developed to a satisfactory standard within the constraints of the analogue model. During development the gains and time constants were repeatedly adjusted in order to obtain the best empirical compromise.

During initial flight testing, under near ideal conditions, the performance of the transition autopilot was found to be good and closely matched the predicted performance. However, when operated in the intended environment over the sea the problems began to materialize. The main problems were associated with the inadequacy of the Complementary Sea State Filter, and the less than ideal output from the doppler radar. The transition control laws themselves were found to be satisfactory requiring only minor changes of which the most significant was reduction of the time to complete the manoeuvre since in practice, the original designed performance was unnecessarily stately.

A typical result is shown on Figure 8 in which an original simulator result is compared with the in-flight result obtained during early flight trials. The comparison shows that the predicted and actual performance were very close, the difference being partly attributed to a steady headwind in the real flight case as well as an unsteady atmosphere. In the case of the height profile the difference in entry condition also contributed toward the general difference.

Now that the problems associated with the doppler radar output and the Sea State Filter have been overcome, the performance of the transition autopilot is considered to be very good under most conditions.

3. SEA STATE FILTER

For the Transition Autopilot mode to function satisfactory over the sea surface it is essential that

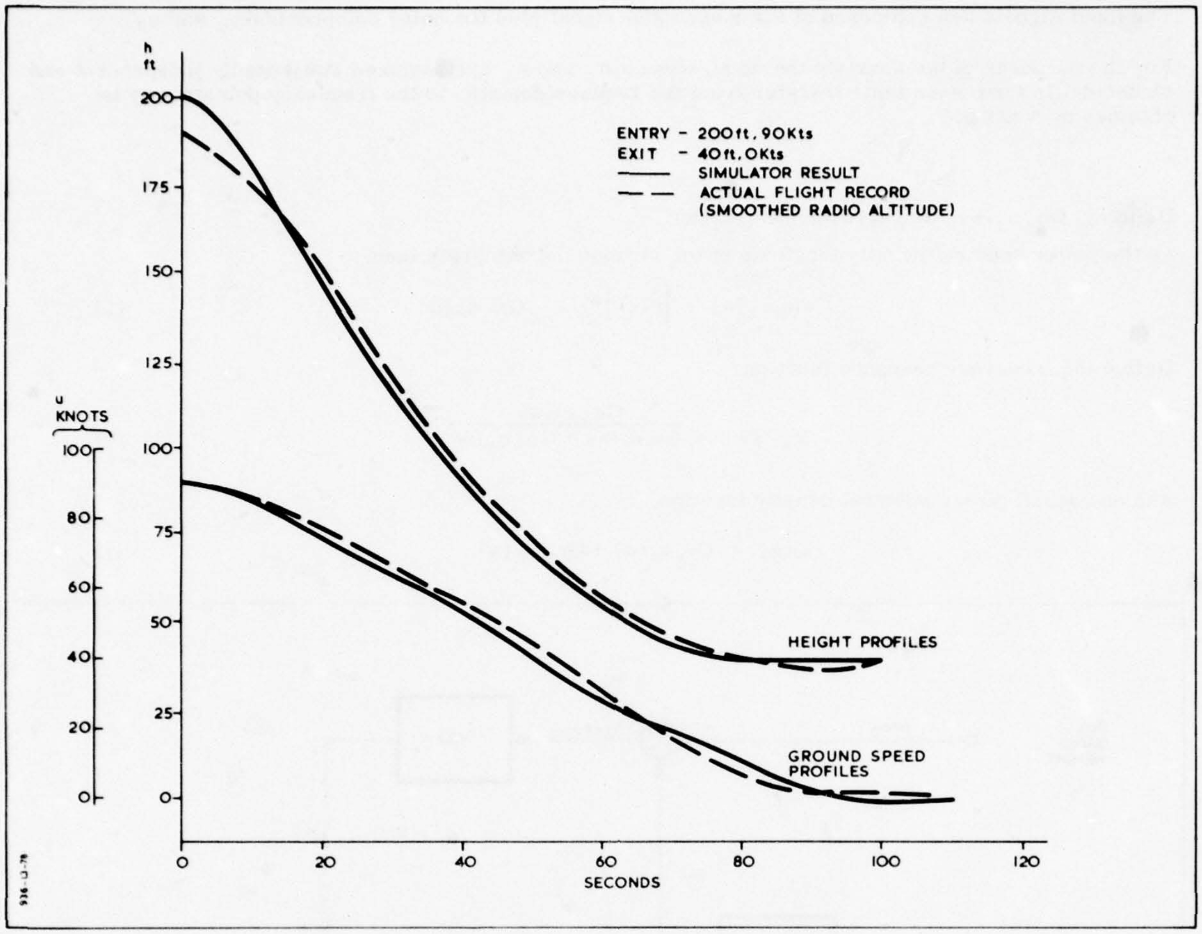


Figure 8 **Transition Height** - Speed Profiles

height information is available which is substantially free of sea motion interference. It is therefore necessary to take raw radio altimeter output information and process it to provide a smoothed mean height signal. This causes very considerable problems with analogue implementation when the system performance requires significant attenuation of sea motion interference. The problem is further compounded when the helicopter is flying against the sea as the bandwidth of the sea motion seen by the radio altimeter is effectively enlarged. In practice the estimated bandwidth of the sea motion is approximately d. c. to 30 Hz with peak to peak amplitude varying from zero to 20 ft. In general the very large amplitudes are associated only with the very low frequencies.

The approach tc the design of the sea state filter was to treat the sea motion interference of the height signal as statistically identifiable noise and to use a complementary optimal filter design technique.

3.1 Filter Design Technique

The design of the complementary filter utilized a technique devised by Bode and Shannon and is described fully in reference 2. In this case the technique is used to filter a combination of radio altitude and normal acceleration signals in such a way as to derive, primarily, a smoothed height signal substantially free from the noise contamination on both input signals. The basic schematic arrangement of the filter is shown in Figure 9.

The transfer functions H (S) and Y (S) are functions of the Laplace variable S and in this case,

$$H(s) = \frac{1}{s^2} \qquad (2)$$

thus, to design the filter it is necessary to define the transfer function Y (s) to minimize the noise error δ on the output height signal. The input signals are radio height h_{RA} and normal acceleration a_z where,

$$h_{RA} = h + n_2 \qquad (3)$$

$$a_z = a + n_1 \qquad (4)$$

The input signals are composed of the meaningful signal plus the noise components n_1 and n_2.

For the purposes of the analysis the noise signals n_1 and n_2 are assumed statistically independent and sinusoidal in form such that transfer from the Laplace domain to the frequency domain may be obtained by writing:-

$$s = jw$$

Defining $Gn_1n_1(w)$, $Gn_2n_2(w)$ and $Ge_2e_2(w)$

as the power spectral density functions of the various noise signals then,

$$Ge_2e_2(w) = |H(s)|^2_{s=jw} \, Gn_1n_1(w) \tag{5}$$

Define the frequency response function

$$Y_o(jw) = \frac{Ge_2e_2(w)}{Ge_2e_2(w) + Gn_2n_2(w)} \tag{6}$$

and an overall power spectral density function.

$$G(w) = Ge_2e_2(w) + Gn_2n_2(w) \tag{7}$$

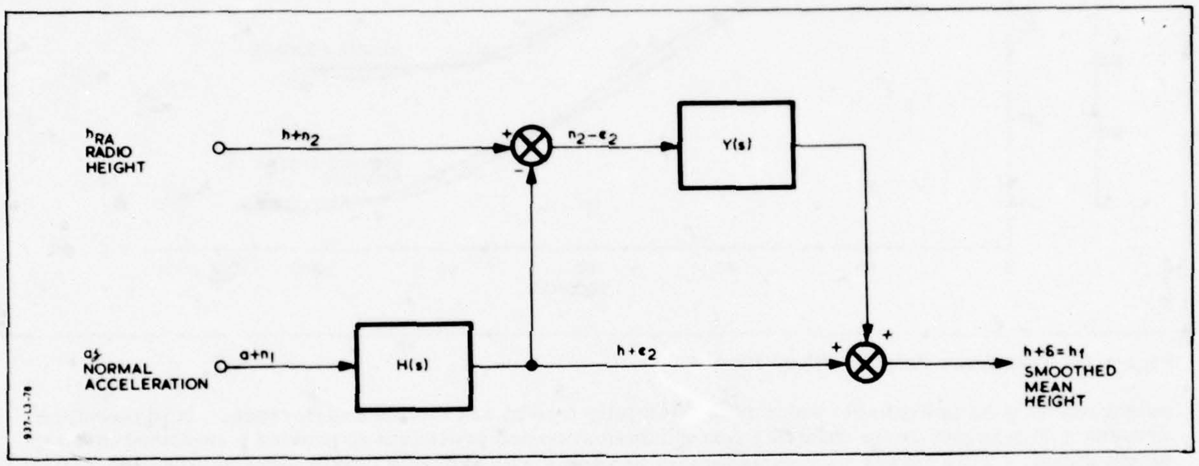


Figure 9 Complementary Filter Schematic

It is necessary to identify a function $Y_1(S)$ such that

$$G(w) = Y_1(jw) \,.\, Y_1^*(jw) \tag{8}$$

where $Y_1^*(jw)$ is the complex conjugate of $Y_1(jw)$. It is then possible to obtain a function,

$$Y_2(s) = Y_o(s) \,.\, Y_1(s) \tag{9}$$

The function $Y_2(s)$ may be expressed in partial fractions some of which will be realisable and others imaginary and hence unrealisable. The realisable component of $Y_2(s)$ is designated $Y_3(s)$.

$$Y(s) = \frac{Y_3(s)}{Y_1(s)} \tag{10}$$

Having thus defined the optimum filter transfer function it is then possible to predict its performance by computing the power spectral density of the output noise component and hence the rms amplitude of the output. The power spectral density function of the output noise is given by,

$$G\xi\xi(w) = \left(Y(s)\right)^2_{s=jw} Gn_2n_2(w) + \left(1-Y(s)\right)^2_{s=jw} \left(H(s)\right)^2_{s=jw} Gn_1n_1(w) \tag{11}$$

3.2 Power Spectral Density Functions

From the foregoing it is evident that the key to the success of this technique is the 'correct' identification of the power spectral density functions of the noise on the radio height signal and on the normal accelerometer signal.

3.3 Accelerometer Noise

In the absence of statistical data defining accelerometer noise it was decided to use a simple model of amplitude limited white noise.

The power spectral density function is given very simply by

$$Gn_1 n_1(w) = \frac{2A^2}{\pi} \qquad (12)$$

Where A is in the half amplitude of the noise.

3.4 Radio Altimeter Noise

A great deal of effort was expended in trying to identify a satisfactory power spectral density function for the sea noise contamination on the radio height signal. A considerable study was made to establish sensible limits for the bandwidth and amplitude of the worst case sea states likely to be encountered. Material for this study was obtained from reference 5 and led to the initial choice of maximum peak to peak amplitudes of 20 ft. with maximum wave period of 15 seconds. Correlation between amplitude and period was not assumed. After some difficulty in designing a realisable filter to cope with these limiting conditions a more realisable worst case wave was defined with lower amplitude and shorter period.

References 3 and 4 give estimated power spectral density functions for sea wave motion. Unfortunately these functions are empirical, based on observation and in general are not analytic functions. They cannot therefore be used directly in the Bode and Shannon filter design technique. This problem was solved by using computer based curve fitting techniques to the empirical functions.

To obtain the final function considerable estimation, approximation and assumption was required, thus it represented a somewhat devalued version of the functions quoted in references 3 and 4.

Evaluation of the transfer function Y (S) was found limiting in as much as a significant element of the function Y_2 (S) was found to be complex and hence unrealisable. It was found that a much simpler bandwidth limited white noise power spectral density function for the sea wave noise gave almost as good results.

3.5 Initial Filter Designs

After considerable investigation and design attempts using variations on the power spectral density functions to obtain a sensibly realisable transfer function for Y (S) the following form was found to be acceptable.

$$Y(s) = \frac{K\,(s + a)}{s^2 + bs + c} \qquad (13)$$

The performance of the filter was evaluated by comparing the error δ on the output with the error n_2 on the radio height input. For sea motion represented by white noise of the appropriate amplitude the attenuation met the requirements. However, for sea motion with the power spectral density function defined by the curve fitting technique, the result was less satisfactory.

The preliminary complementary transfer function for obtaining a satisfactory smoothed height signal was of the form:

$$h_f = \left\{ \frac{K\,(s + a)}{(s^2 + b + c} \right\} h_{RA} + \left\{ \left(\frac{Ts}{1 + Ts} \right) \left(\frac{1}{s^2 + d + c} \right) \right\} a_{\xi} \qquad (14)$$

The long time constant washout was included in the accelerometer term in order to wash-out long term thermal and drift effects which might otherwise prove undesirable. The long time constant was necessary in order not to influence the behaviour of the filter at the frequencies of interest. The analogue realisation of this filter is shown in Figure 10. It will be observed that the height rate output $\dot{h}_f$ is obtained by picking off height rate at the appropriate point and feeding it through a lag.

The optimum design of a complementary filter to produce a height rate signal was carried out along the line described for the height signal. It was found that a very good approximation to the optimum complementary filter for height rate could be obtained simply by adding the lag term into the transfer function, as shown in Figure 10.

3.6 Further Development

Considerable difficulties were encountered when attempts were made to implement the sea state filter design. In general the performance was found to be reasonable, but the washout filter on the accelerometer input caused significant problems in particular at engagement. It was also found

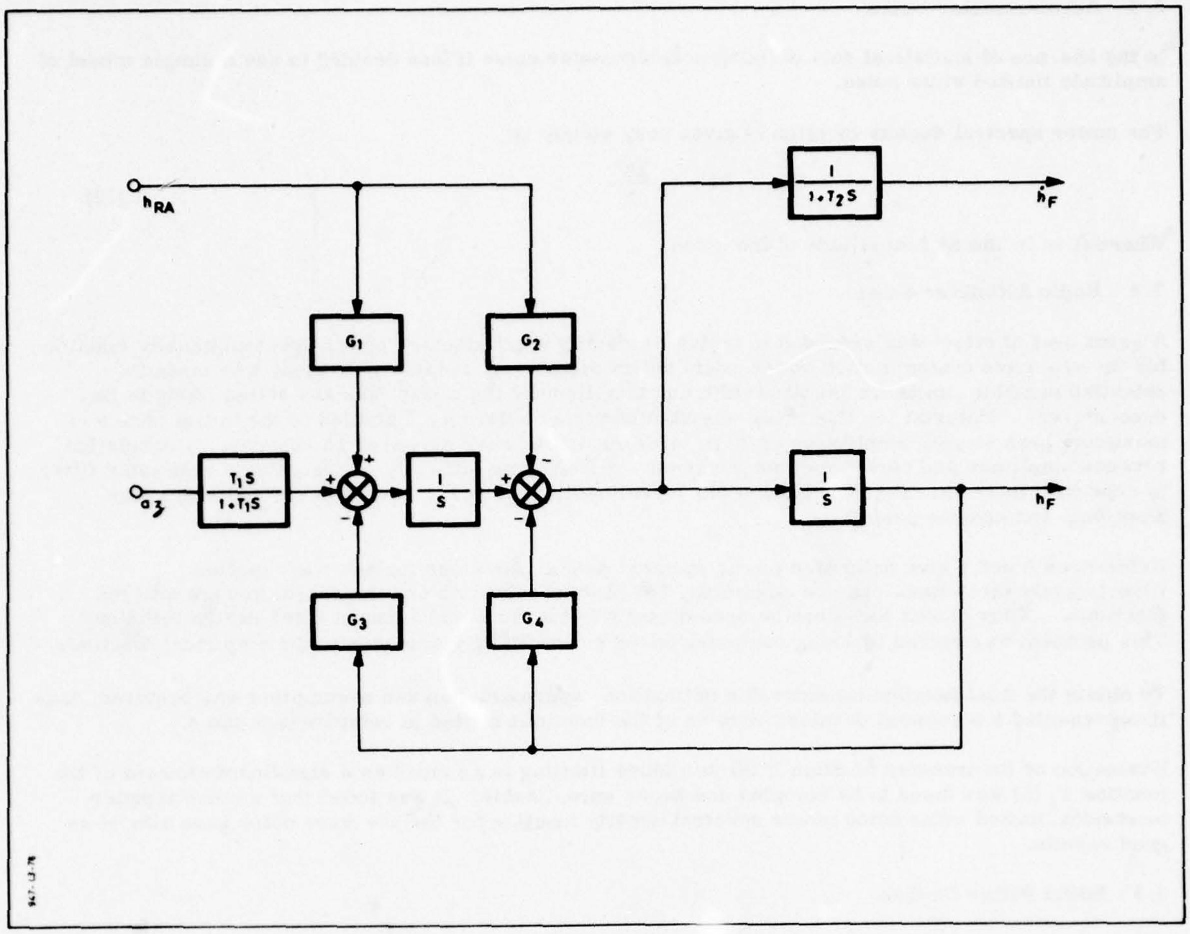


Figure 10 Preliminary Sea State Filter Design

difficult to construct a repeatable analogue filter with the long time constant. Further, totally unacceptable problems were created by having to match the duplex lanes of the filter in the duplex autopilot

Further design effort was of an empirical nature and was based largely on simulation exercises using the Westland Helicopters Limited simulator facility. Initially, most effort was concentrated on modifying the washout filter such that the sea state filter could be engaged immediately after significant manoeuvres which was previously not the case. Extensive modifications of the sea state filter followed in which the gains, time constants and the general order of the filter were changed. In general the performance improvements in the modified filter were negligible, but the engagement problems were largely overcome at the expense of filter complexity.

When flight tested the problems of engagement offsets and transients and lane matching were found to be satisfactory and acceptable transition profiles were obtained. However, excessive short period noise was getting through the filter to the height rate output. As the noise was thought to originate at the radio altimeter output the suggested solution required that the height rate output be complemented independently with the normal accelerometer output signal. The resultant sea state filter control law is shown on Figure 11.

4. AUTOMATIC SONAR DEPLOYMENT AUTOPILOT

The automatic sonar deployment autopilot is operated with the helicopter in a hover hold mode over the sea. The sonar device is suspended, immersed, on a cable beneath the helicopter and its depth and attitude are controlled automatically by controlling helicopter hover height and plan position. The hover height autopilot controls exposed 'dry' cable length to a preset value, normally less than 100 feet. The plan position of the helicopter is determined by controlling the angle of departure of the cable from the helicopter, (cable angle) to a preset trim value. The theory of this method of control is that in the absence of wind and sea drift effects, if the cable angle is zero the sonar device will adopt the required upright attitude in the water.

The design of the 'dry' cable height control law was relatively straightforward since the conventional hover height hold autopilot had already been designed in conjunction with the transition autopilot. The control law was modified accordingly using 'dry cable length' in place of radio altitude.

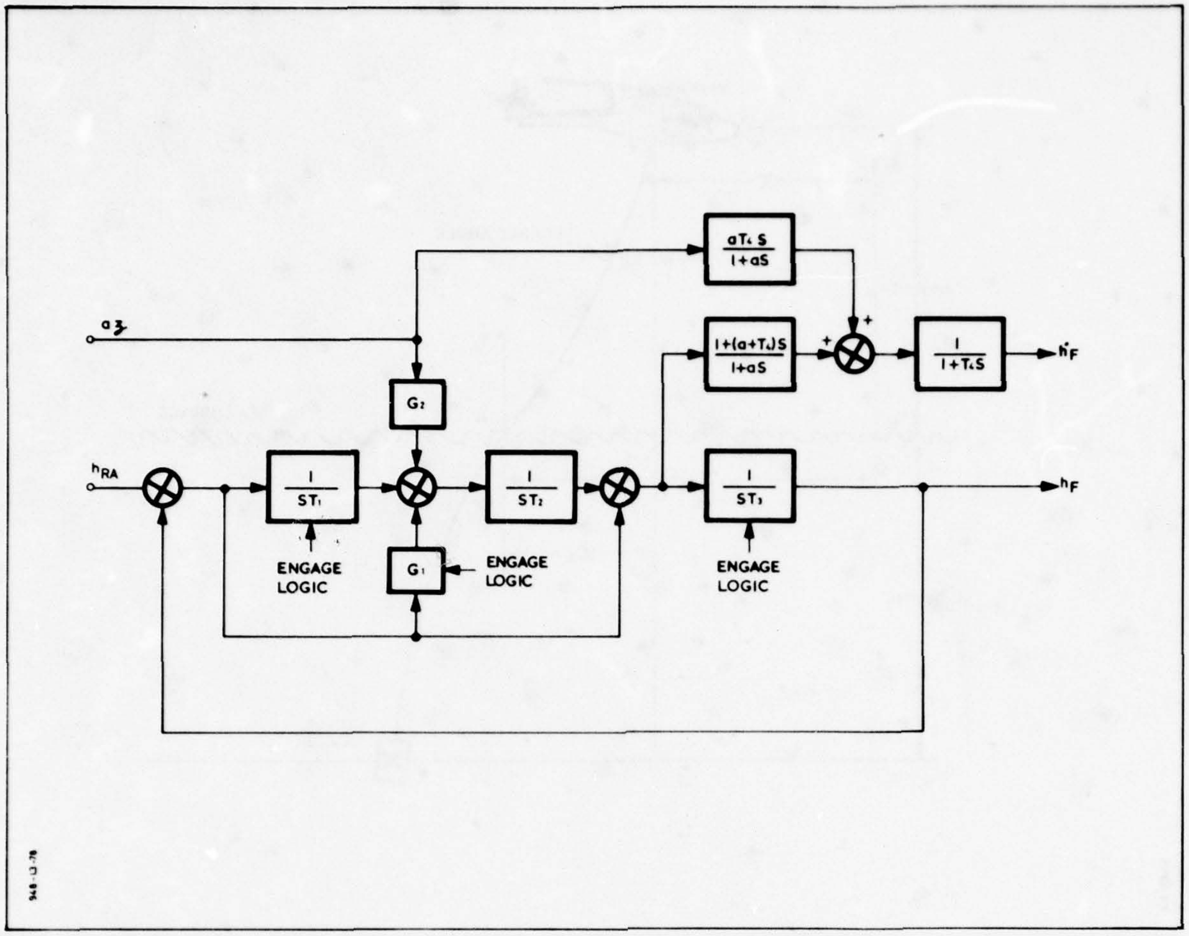


Figure 11 Sea State Filter

The design of the longitudinal cyclic and lateral cyclic control laws to provide the plan position control was not so easy. To evaluate the design the entire system was modelled using the Westland Helicopters Limited analogue simulation facility and depended very heavily on selection of a suitable simulation model for the cable.

4.1 Plan Position Control

Throughout the design phase both the longitudinal cyclic and lateral cyclic control laws were of the same format, as the control requirement is very similar in both axes in what is nominally a steady hover mode. It was found that dynamic differences could be taken care of by suitable choice of gains and time constants for the relevant control laws.

The geometry of the sonar device deployment is shown in Figure 12. Although the geometry shown relates to the longitudinal control axes it applies equally to the lateral control axes. The displacements occur as the result of turbulence effects on the helicopter and cable, steady wind effects on the cable, sea drift and sea motion of the type already discussed in the transition autopilot context. The net result is that the cable defines an arbitrary profile between the helicopter and the sonar device.

In the ideal situation, in the absence of external effects, the cable would hang vertically beneath the helicopter and the sonar device would naturally adopt an upright attitude. This condition defines the datum and is also the condition that the cyclic control system must try to create despite the above mentioned external effects. Therefore, the control system must attempt to reduce to zero the cable angle and the residual horizontal displacement which are related by the cable profile. It is therefore necessary to identify a model defining the cable profile which will obviously be different for every set of external conditions. It was found possible under limited external conditions to define a continuous cable model. However, in the interests of simplicity and the simulation requirements the initial model studied was of a rigid cable followed by a piece-wise linear model, both of which were easier to simulate and proved adequate in the long term.

The initial longitudinal cyclic sonar control system is shown in Figure 13. Cable angle error is obtained by comparing the computed cable angle with a preset value, the error is then passed through the gain stage $G_{\theta c}$ and the proportional plus integral stage, to form the longitudinal cyclic pitch

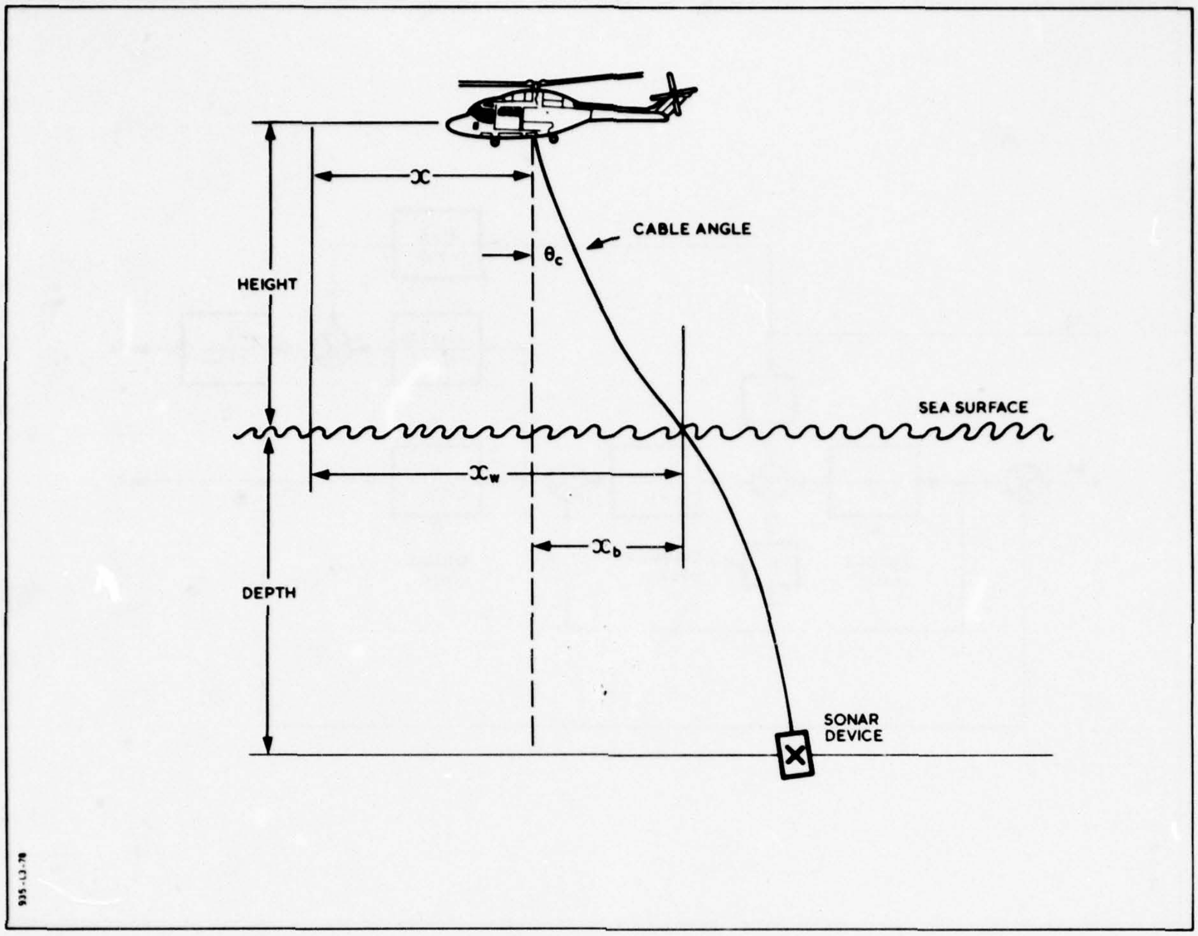


Figure 12 Sonar Deployment Geometry

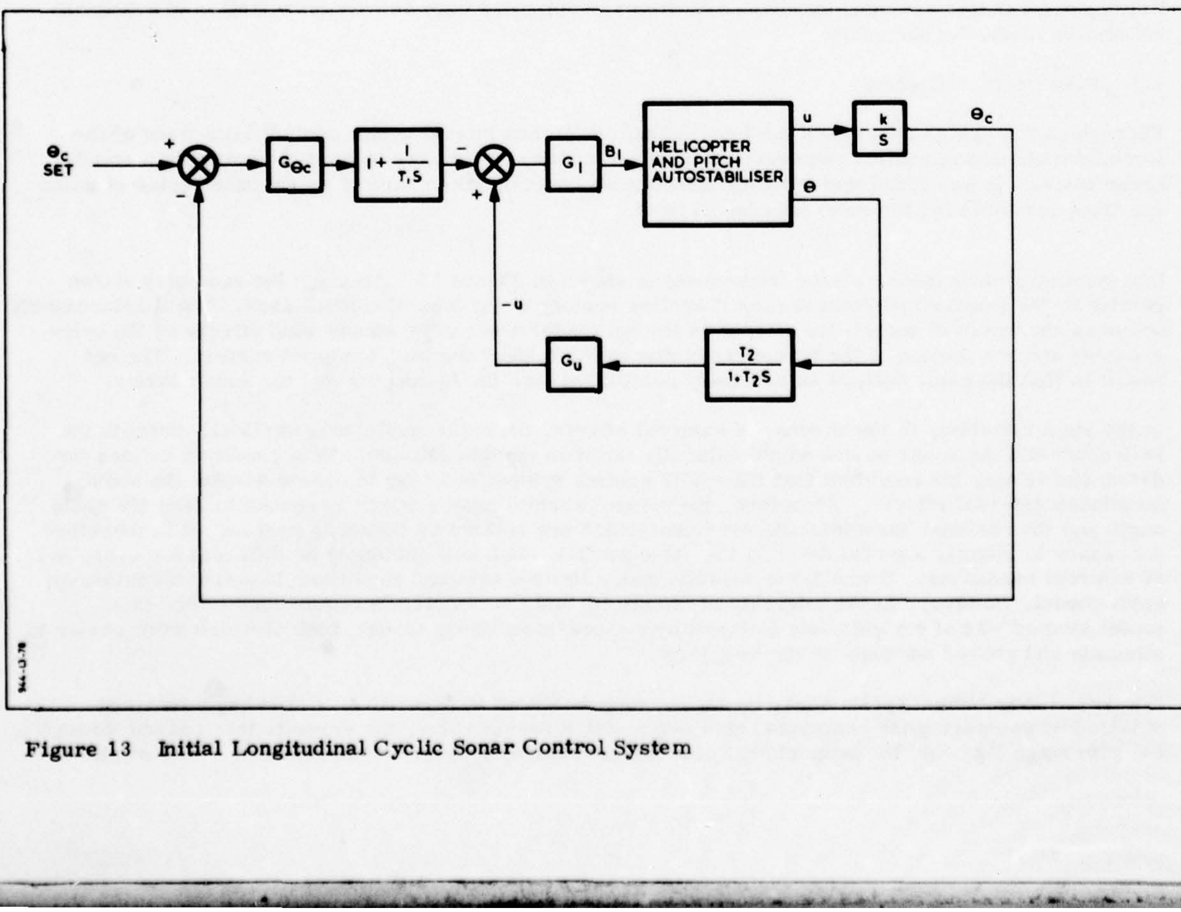


Figure 13 Initial Longitudinal Cyclic Sonar Control System

demand. The inclusion of the necessary integral term was found to be destabilizing, therefore, an additional damping term was included.

Rather than use a doppler derived rate feedback term it was decided to derive a suitable signal from pitch attitude.

By considering the transfer functions,

$$\frac{\theta}{B_1} \quad \text{and} \quad \frac{\dot{u}}{B_1}$$

for the aircraft with autostabilization it was possible to identify

$$G_u = \frac{\dot{u}}{\theta} \tag{15}$$

which was essentially constant for the conditions defined by the sonar hover including steady winds. Therefore a speed damping term was obtained from pitch attitude by feeding θ through a pseudo integrator and the gain G_u.

The stability of the inner loop was evaluated and could only be maintained for a limited range of values for the gain G_1. Similarly for fixed values of G_1 within the stable range and a fixed value of k, the stability of the outer loop could only be maintained for a limited range of values of G_{θ_c}. Choosing suitable values for G_{θ_c} and G_1 would not ensure stability at all conditions as, with a fixed value of k, the loop gain is implicitly a function of hover height. Consequently the stability margins were very small for the range of potential operating conditions and so an improved control law had to be established.

The control system was improved by increasing the gain and hence bandwidth in the outer loop at the expense of stability. Stability was restored by including a phase advance filter in the feedforward path in the inner loop. The resultant control system is shown in Figure 14. The inner loop damping feedback path was modified to include a washout filter to remove steady state trim offsets which would otherwise modify the system performance. The washout filter chosen for this also cancels effectively with the proportional plus integral term now included in the feedforward path of the inner loop, thereby maintaining the correct form of damping feedback.

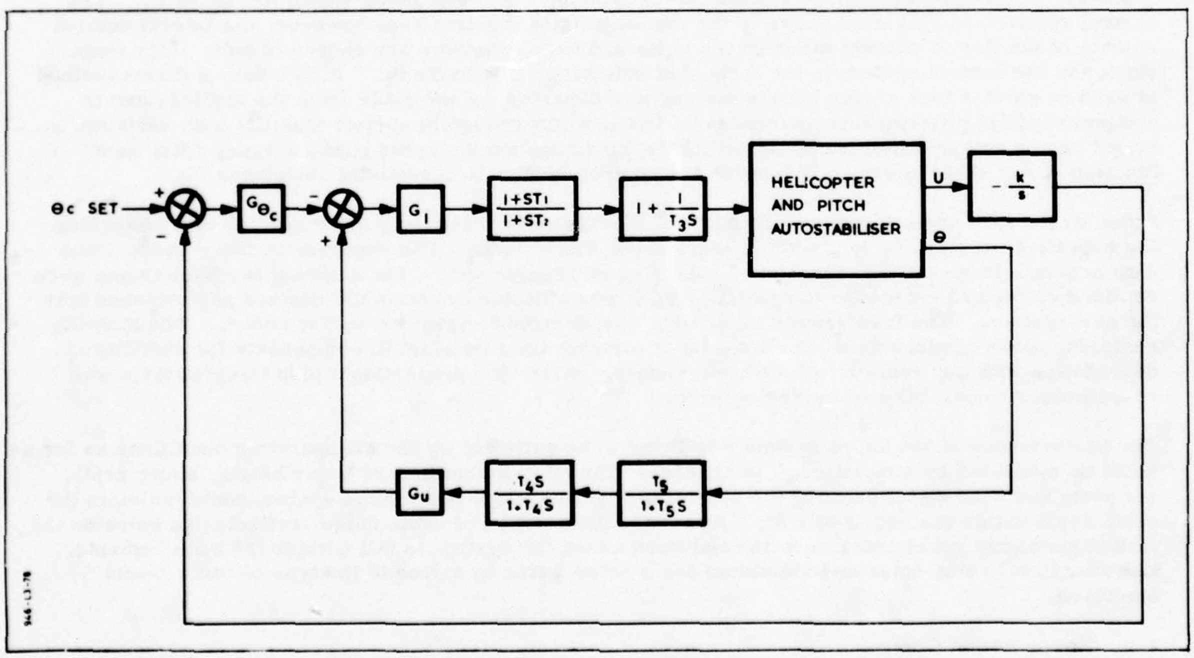


Figure 14 Improved Longitudinal Cyclic Sonar Control System

The performance of this system was evaluated and it was found that the bandwidth was not generally sufficient and also that the attenuation of cable angle disturbances corresponding to typical operational sea states was also insufficient.

Evaluation of the control system defined was carried out on the simulation using the rigid cable model. It was thought that the use of a rigid cable model was a significant factor in the poor performance obtained; it was therefore decided to continue development with a more realistic cable model using a number of discrete elements.

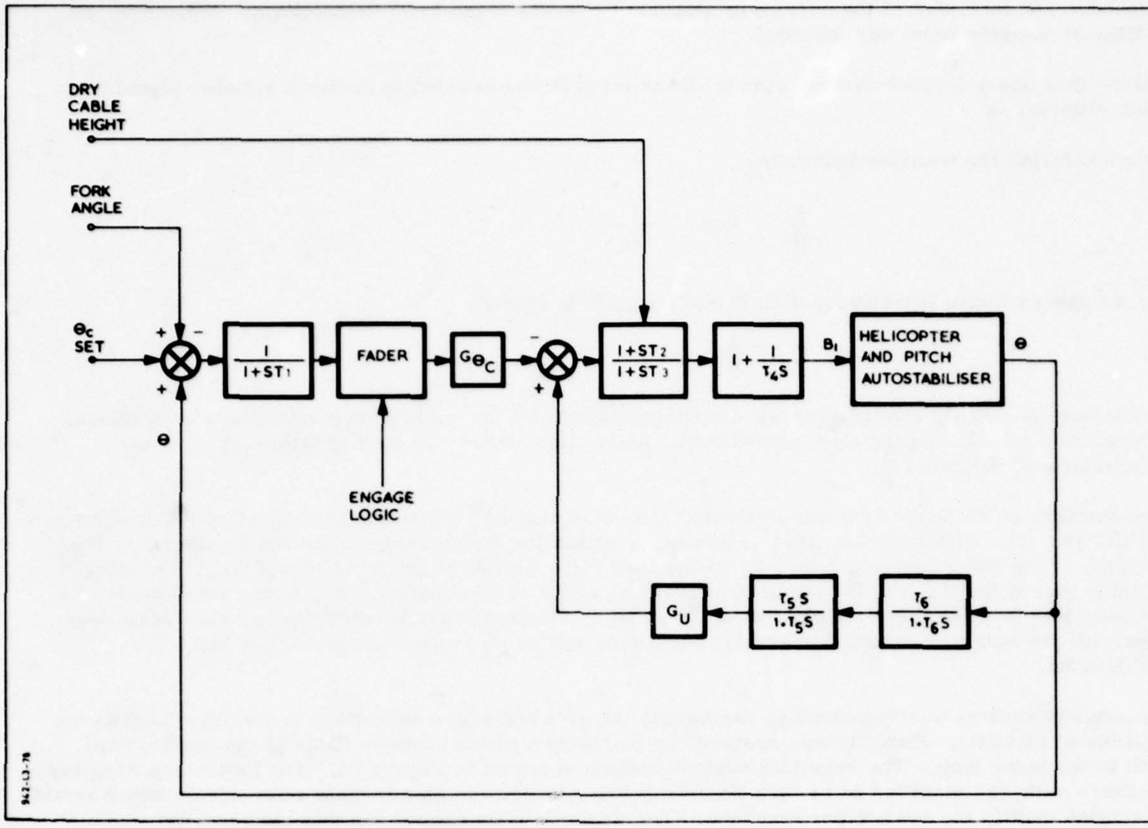


Figure 15 Longitudinal Cyclic Sonar Control System

A number of changes were made to the control system, the result being shown in Figure 15. The control system configuration shown is for the longitudinal control axes however, the lateral control system is similar in concept although the gains and time constants are chosen to suit. The main change in the control system is the method of obtaining cable angle θ_C. In this case a direct method is used in which a fork device senses the angle of departure of the cable from the fuselage and is compensated for pitch attitude variations. The implicit change in system stability with variation in height can be compensated for by adjusting the time constant T_2 in the phase advance filter as a function of dry cable height. Otherwise the control system is essentially unchanged.

Noise on the fork angle signal was found to be a problem necessitating incorporation of a smoothing lag with time constant T_1 in the cable angle error signal path. The engagement fader in the same path prevents large attitude transients occurring on engagement. The damping feedback terms were retained unchanged except for the gain G_u which was adjusted to obtain the desired performance with the new system. The feedforward inner loop was changed to optimize performance. The stability restoring phase advance network allows for a variable time constant to compensate for stability degradation with dry cable length or hover height. Also, the proportional plus integral term was re-optimised to suit the new control system.

The performance of the above system was found to be satisfactory for all operating conditions as far as could be evaluated by simulation. In all cases, for all combinations of hover height, sonar depth, sea state and wind speed defining the operating flight envelope the control system could maintain the cable angle within the required ± 5°. However, the addition of cable noise, reflected as noise on the fork angle signal could under certain conditions cause the system to fall outside the requirements. However, real cable noise and simulated cable noise differ by virtue of the type of cable model employed.

4.4 Hover Height Control

Control of hover height in the sonar autopilot mode was found to be relatively straightforward as much work had already been carried out for the hover hold mode in the transition autopilot. In this case 'height' refers to the exposed or 'dry' cable length and a suitable signal is provided directly for this eliminating the need for the use of the sea state filter. The collective dry cable angle control system is shown in Figure 16.

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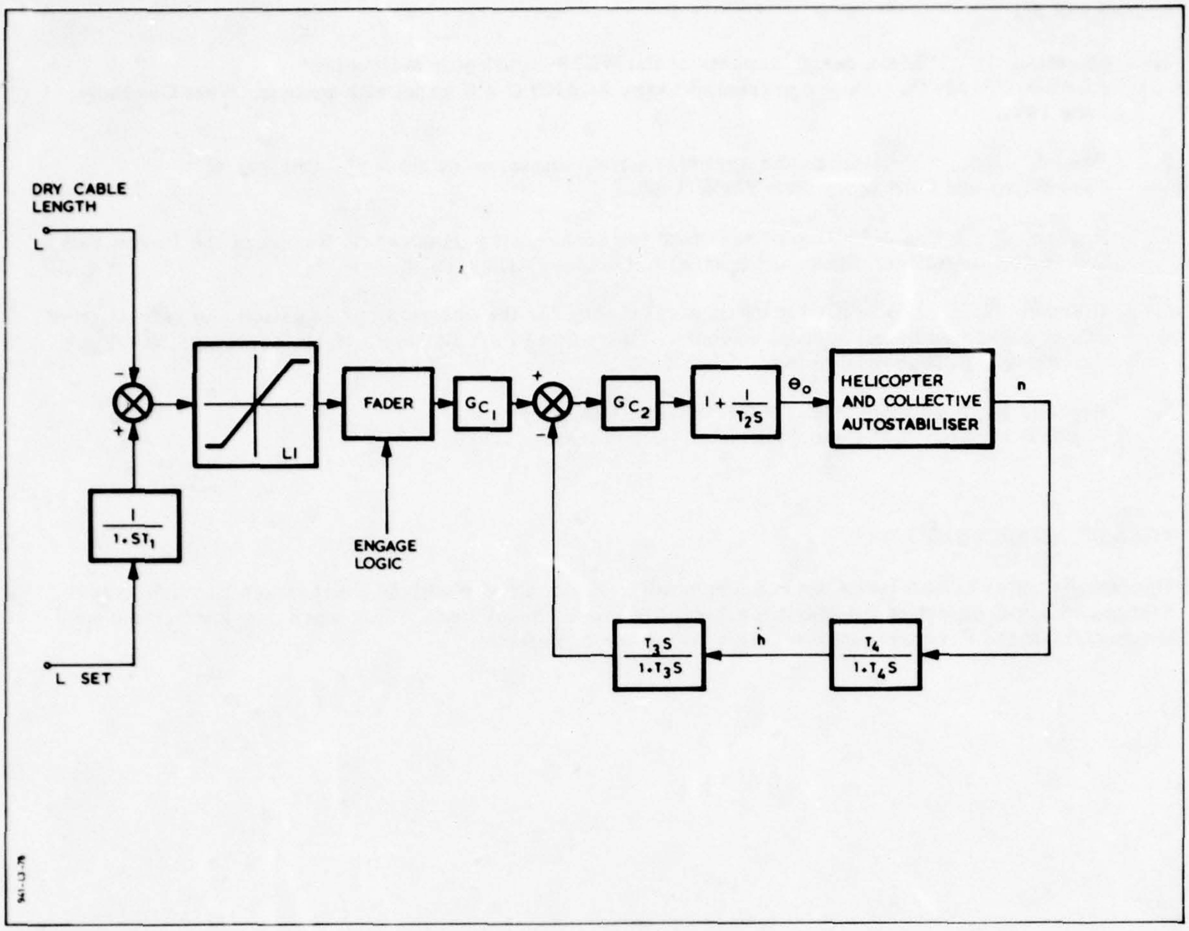

Figure 16 Collective Dry Cable Length Control System

The control system computes a dry cable length error from the actual value and the preset value, the error signal is then passed through the gain stages G_{c1} and G_{c2}, and the proportional plus integral stage to demand collective blade angle. The preset dry cable length signal path includes the lag with time constant T which prevents adjustment transients getting through the control system. The error signal is limited by limiter L_1 to prevent saturation at engagement and is also passed through an engagement fader, which prevents large transients occurring on mode engagement.

The inner loop normal acceleration feedback is modified by a 10 second lag stage to provide a pseudo height rate term for loop damping comparable to the technique used in the cyclic control laws. A long time constant washout filter is also included in this path to prevent steady state accelerometer offsets from modifying the performance of the control system.

The performance of this control system was simulated for all conditions and was found to be very stable with a performance well within the requirements. In this case the requirement is to hold dry cable length to ± 4 ft. including the case where the cable is being deployed at the rate of ± 1 ft./sec. As with the cyclic control system it was thought that cable noise might prove troublesome, but the inclusion of a suitable filter in the dry cable length signal path would overcome this with little effect on performance, as the stability margins were more than adequate in this control axis.

DESIGN AND TESTING OF A REDUNDANT SKEWED INERTIAL SENSOR COMPLEX FOR INTEGRATED NAVIGATION AND FLIGHT CONTROL

by
R. E. Ebner
Litton Systems, Inc.
Woodland Hills, Calif.
and
W. E. H[illegible]well
NASA Langley
Hampton, Va.

ABSTRACT

Requirements for a redundant strapdown inertial sensor complex applied to V/STOL aircraft as developed by NASA are presented. Results of a Litton study for preliminary design of such a system are then shown. Flight test data of a redundant, skewed axis strapdown inertial system are given, demonstrating the feasibility of the primary design aspects of the study program. This data consists of parity equation responses through various flight conditions, showing residual noise levels on redundant gyro and accelerometer comparisons as a measure of minimum failure-level detectability, plus failure isolation and navigation performance through several simulated instrument failures.

INTRODUCTION

The use of helicopters and V/STOL aircraft in applications involving extended complex maneuvering with limited pilot visibility places severe demands on the vehicle flight control system. Sensors used by this equipment extend beyond standard rate gyros and accelerometers into more navigation-oriented devices which provide velocity and position outputs in order to achieve the desired display and control functions. Redundant sensing is then required because of the total reliance of flight safety upon these sensor outputs. Avionics costs could escalate to unacceptable levels unless highly efficient design techniques are employed.

This paper describes the preliminary design of a redundant strapdown inertial navigation unit and subsequent flight testing of these concepts with existing strapdown navigation hardware. The redundancy concepts have thus been shown to be feasible and can lead to the avionics efficiency required in high-performance V/STOL applications.

APPLICATIONS

Redundancy and reliability requirements for a flight-control sensor complex are a function of the flight control modes employed plus the natural stability of the airframe. Table I illustrates the growing importance of redundant inertial data as SAS, CCV, autoland, and improved handling modes are applied to the aircraft FCS. The equipment described in this paper applies to the more demanding vehicles, such as VTOL and CCV, where redundancy is required for most inertial functions. Then, an integrated set of inertial sensors with self-contained redundant preprocessing can prove very cost-effective over separate, redundant rate, attitude, and navigation sensors.

TABLE I – INERTIAL DATA REDUNDANCY REQUIREMENTS VS APPLICATION

Redundant Inertial Data	Conventional Aircraft	FCS, Yaw SAS	3-Axis SAS, FBW, CCV	Autoland, VTOL, Reduced Pilot Workload	All-Weather VTOL
Attitude/heading	X	X	X	X	X
Yaw rate		X	X	X	X
Pitch/roll rates			X	X	X
Body acceleration			X	X	X
Ground speed/track				X	X
Position coordinates					X

SYSTEM REQUIREMENTS

A preliminary system design was performed by Litton[1] under NASA contract number NAS1-13847 for an efficient inertial sensor complex capable of meeting VTOL reliability requirements. Various design concepts were then flight-tested with NASA sponsorship using a redundant strapdown system developed under Litton IRAD.

The motivation for these flight tests and the earlier work that led to them was the foreseeable requirement for improved scheduled operation of commercial helicopters. In a typical situation the helicopter will depart a busy outlying airport, navigate narrow terminal area corridors to avoid air traffic delays, wend its way through a complex flight path to avoid residential areas and obstacles such as buildings which may be higher than its flight path, and finally land on a small pad in the downtown area. A typical flight might last 9 to 12 minutes with a 5-minute layover.

This operating regime is continuously within the most hazardous flight phase – the takeoff, landing, and terminal area operation. Such operations are difficult under the best of conditions; however, under poor visibility this places an undue strain on both pilots and schedule reliability. These operational characteristics and pilot workload considerations require that ultimately these operations be largely automatic, particularly in category II and III conditions. The resultant automatic flight control, guidance, and navigation system characteristics require velocity smoothing in the final landing phase and continuous navigation data during potential data dropouts from external sources. This requires inertial data. The basic requirements for this terminal area navigation are shown in Table II (reference 2).

TABLE II – CIVIL HELICOPTER REQUIREMENTS FOR THERMINAL AREA NAVIGATION

1. Requirements for operation in proximity of obstacles	To within 153 m (500 ft)
2. Accuracy requirements:	
Range	7.8 m (25 ft)
Velocity	1 m/s (2 kt)
Angular	0.05°
3. Multiple aircraft requirements	1 landing/min 1.75 km (1 nm) longitudinal spacing
4. Multiple pad requirements	122 to 244 m (400 to 800 ft) spacing
5. Inertial smoothing requirements	1 m/sec (2 kt) INS for velocity control
6. Reliability/redundancy requirements	Category II dual autopilot or autopilot plus independent monitor Category III triple redundancy
7. Update rate requirements	1 sec
8. Data link requirements	8000 bits/sec
9. Requirements for signal continuity and fidelity, including proximity of obstacles	No multipath Use ICAO ILS standards
10. Inertial/radio-inertial requirements	1 m/s (2 kt) INS for velocity control

A system which depends on automatic operations and precise navigation to the extent described above must have highly reliable data sources; typically this reliability is obtained through redundant sensors. The degree to which future systems will depend on this data far exceeds any other system in use today; therefore certain strategies such as pilot selection of alternate equipment must also be done automatically because insufficient time may exist for the pilot to determine that a failure has occurred, what has failed, and to make an alternate selection.

The degree of redundancy required is driven by two factors: first, automatic redundancy requires at least three units to provide a basis for logical decision making and second, the overall system failure rate must be acceptably low. This latter failure rate is generally taken to be at least as low as 0 (10^{-6}) failures per flight hour. The performance of triple units can be obtained from the same physical hardware as two units if the redundancy management in software is properly implemented.

An additional requirement is that the redundant system be separated physically into at least two units which can each stand alone as an operational unit. This is to preclude simultaneous loss of the entire system from accidental causes such as a minor electrical fire. For civil application this separation should be in the order of 1/2 meter. The impact of this requirement is to make the automatic redundancy management much more difficult because of the likelihood of dynamic misalignment between the two units due to vibration and flexure of the intervening structure. Additional separation could have an impact on control system design since physical location in a given aircraft can influence gain and filter characteristics.

Automatically redundant systems for inertial measurement were first studied at MIT in the late 60s (reference 3). The space program has been the first to use redundant concepts with the Shuttle avionics representing about the ultimate that can be accomplished today. Long-term space missions which rely on redundant systems generally count on the ability of the vehicle to make certain maneuvers or use long integration times to isolate ambiguous failures. The major concern for future aircraft avionics is relatively quick, automatic, unambiguous failure detection and isolation (FDI). For flight control purposes, today's FDI may be adquate since flight control accuracy requirements are relatively lax. However, for precision navigation around obstacles, the FDI techniques require refinement. The principal problems are the reduction of false alarms and missed alarms. The false alarm problem must be solved through sophisticated techniques while maintaining a low alarm threshold for early detection of true failures, i.e., low missed alarm rate.

System probability of failure (POF_s) may be divided into three categories:

POF_t = Probability that a failure occurs and is detected (true alarm)

POF_f = Probability that no failure occurs, but one is detected (false alarm)

POF_m = Probability that a failure occurs and is undetected (missed alarm)

Therefore

$$POF_s = POF_t + POF_f + POF_m = 0\,(10^{-6})$$

or on the order of 10^{-6}.

It is reasonable to expect the equipment of a mature, tripley redundant system to achieve a true failure rate on the order of 10^{-7}, that is

$$POF_t + POF_m = 0\,(10^{-7})$$

This means that false alarms must be somewhat less than this number if the pilot is to have any confidence in the resultant system. It would seem appropriate that the false alarm rate be at least as low as one in 10; that is, if the failure indicator comes on, 9 times out of 10 there has actually been a failure. This implies

$$POF_f < 0\,(10^{-8})$$

In order to achieve this system level performance, the reliability of any one redundancy test must be *significantly* greater than the reliability of the hardware being tested.

The experimental results obtained in these flight tests (and others, reference 4) would indicate that additional effort is required in this area.

SYSTEM DESCRIPTION

The navigator designed to meet the proceeding requirements was configured to consist of four interchangeable plug-in units. Each unit contains one of the redundant channels of hardware consisting of a TDF tuned-gimbal gyroscope, two linear accelerometers, a computer with I/O, and a power supply. The gyro/accelerometer axes are skewed within each chassis as shown in Figure 1 so that when the four channels are installed, as in Figure 2, the four gyro and eight accelerometer axes are distributed in space. This insures that normal operation continues regardless of which two sensors fail. Accurate interchannel alignment is maintained by the aircraft mounting provisions. For applications requiring physical separation of units, channels may be separated by pairs. If significant aircraft flexure occurs between pair locations, however, redundancy management performance of attitude, heading, velocity, and navigation outputs would degrade.

A simplified system block diagram is shown in Figure 3. Each channel consists of: one TDF gyro with pulse rebalance electronics, two accelerometers with pulse rebalance electronics, one GP digital computer, external I/O, internal and intercomputer I/O, and power supply.

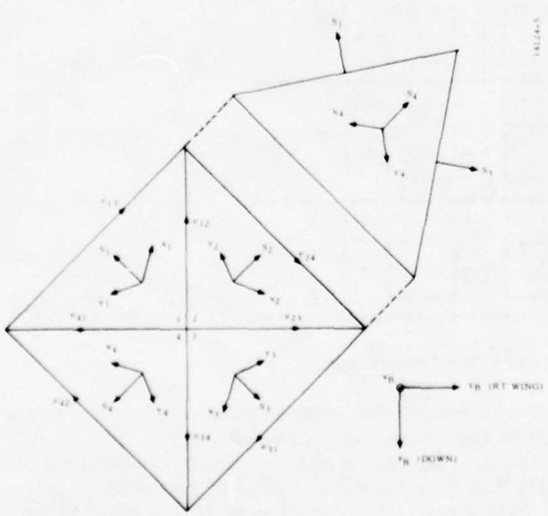

Figure 1. Semioctahedral Gyro-Axis Orientation

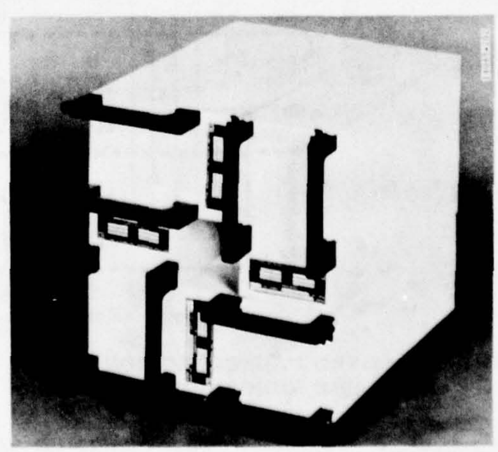

Figure 2. Mock-Up of the Redundant Inertial Navigation System

The amount of cross-strapping between channels has been minimized for reduced cost. Inertial instrument data are transferred to other channels after computer processing; power supplies affect only one channel. Interchannel I/O and software synchronization are provided for sensor redundancy comparisons and to allow selection of sensors for further processing.

The system characteristics are summarized in Table III. Values are approximate, especially in the area of computer requirements, navigation accuracy, and thus system cost, since they are a function of specific user requirements. Some increase in weight and volume may be expected in a more demanding thermal environment requiring highly efficient heat exchangers rather than impingement cooling.

TABLE III – SUMMARY OF SYSTEM CHARACTERISTICS

Size*	0.33 X 0.033 X 36 m (13 X 13 X 14 in.)
Weight*	28 kg, 61 lb
Power	540 w
LRUs	Four identical channels
Redundancy	Fail-op/fail-op
Failure probability (3 h)*	Less than 10^{-8}
Gyros	
Type	G-6/7 tuned rotor
Quantity	4
Accelerometers	
Type	A-1000
Quantity	8
Computer (1 of 4)	150 KOP, 8K-words memory
Outputs	Angular rates, acceleration, attitude, heading, velocity, position
Inertial nav accuracy	1.8 to 18 km/h (1 to 10 nm/h)
Cost, system	$100,000 to 180,000

*Commercial version

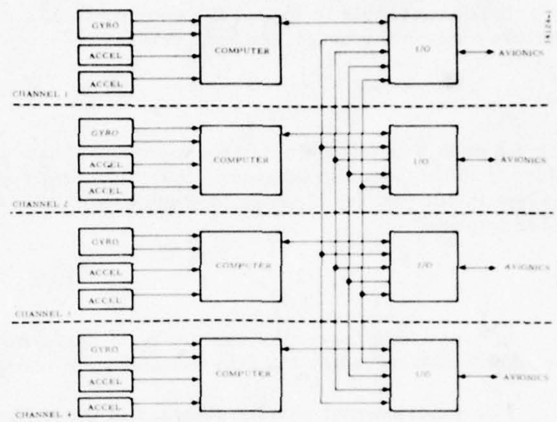

Figure 3. Simplified System Block Diagram

COMPUTER MECHANIZATION

The four-channel computer mechanization which has been selected is shown in Figure 4. Following compensation of the channel's local instruments for constant error coefficients, measurements are cross-fed between computers. Each computer then performs FDI/RM to select instruments to be used for subsequent calculations. Table IV shows instrument selection logic based on FDI failure indications where each channel uses a different pair of gyros. The advantages of this approach are that 1) error buildups which occur during FDI in one or two channels may be eliminated by reseting to a known-unaffected channel, and 2) very small drifts, inside of FDI thresholds, can be detected and isolated by reference to external data such as position updates or terminal error, for subsequent maintenance action.

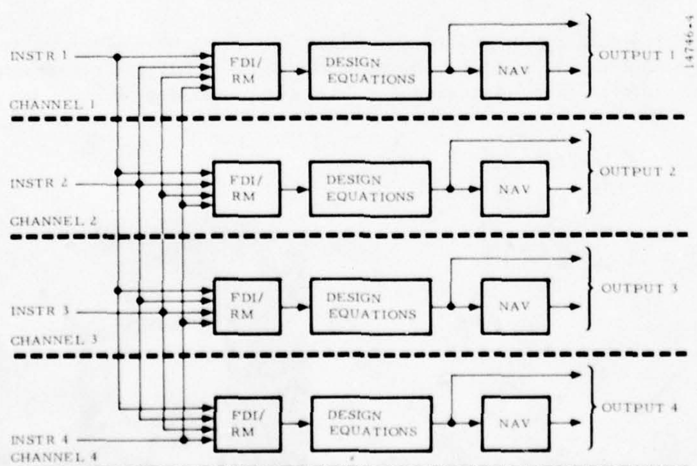

Figure 4. Four-Channel Software Mechanization

TABLE IV – GYRO PAIR SELECTION VS FAILURE INDICATION

Chan	0	1	2	3	4	1/2	1/3	1/4	2/3	2/4	3/4
1	1,2	1,2	1,3	1,2	1,2	1,3	1,2	1,2	1,4	1,3	1,2
2	2,3	2,3	2,3	2,4	2,3	2,3	2,4	2,3	2,4	2,3	2,1
3	3,4	3,4	3,4	3,4	3,1	3,4	3,4	3,2	3,4	3,1	3,1
4	4,1	4,2	4,1	4,1	4,1	4,3	4,2	4,2	4,1	4,1	4,1

Requirements for strapdown navigation are not unusal in the redundant implementation. Computer requirements are for a computational speed of approximately 150 KOPS and a memory of 8K words. Of the total, only about 10% of the speed is due to FDI/RM which indicates the excellent cost-effectiveness of the approach.

SYSTEM FAILURE PROBABILITY

The probability of system failure depends upon the basic component failure rates, redundant element interconnections, and the probability of recovering from each failure condition through FDI (coverage). Two failure rates have been calculated, one for commercial environments with a channel MTBF of 4000 hr and the other for more extreme environments with a channel MTBF of 2000 hr. This latter quantity is equivalent to a full INS MTBF of 1600 hr, by adding one gyro and one accelerometer.

System attitude and navigation failure (NAV) is assumed to occur if gyro error exceeds 1°/hr. Angle rate output failure (FCS) is assumed to occur if gyro error exceeds 1°/sec.

Figure 5 shows the probability of exceeding these two performance levels (NAV and FCS) without recovery, i.e., these system fails, as a function of channel MTBF and flight time.

Further detail on probability calculations is given in reference 5.

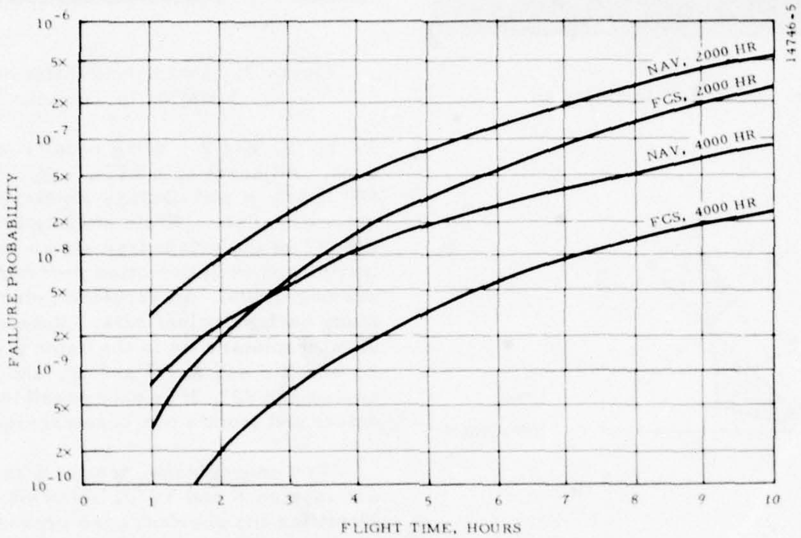

Figure 5. System Failure Probability vs Flight Duration

FLIGHT TEST SYSTEM CONFIGURATION

Several of the techniques employed in the preceding redundant system design were demonstrated in a NASA-sponsored flight test using Litton-developed equipment in November 1977. These techniques were:

1. Four TDF gyros capable of strapdown navigation, were in operation simultaneously. The gyro axes were skewed relative to one another so that any combination of two gyros could be used to provide the full three axes of rate information needed for inertial navigation. Thus complete failure of any two gyros could be tolerated.
2. Six accelerometers were used, also skewed relative to each other. Tolerance of only a single failure was demonstrated due to geometrical limitations of the test sample.
3. Parity equations, filtering, failure detection and isolation, and reselection of instruments for data input to the strapdown inertial navigation based on the results of FDI were solved in real time by a GP digital computer.
4. Two navigation solutions based on a different selection of inertial instrument were maintained simultaneously in the digital computer. Reinitialization of one solution to the variables of the other was performed in flight, demonstrating a means of elimination of transients which can occur in navigation while faulty instruments are being identified and eliminated from the navigation solution.

The redundant test system was configured on a Litton IRAD program and consisted of two orthogonal LN-50 inertial measurement units, one of them skewed relative to aircraft axes. Figure 6 shows the installation of the two IMUs on a flight test pallet, and Figure 7 shows the pallet installed in the Litton test aircraft, a Merlin IV, along with the flight computer.

Figure 8 shows the geometry of the instrument axes. The skewed set of axes, X', Y', and Z', are rotated by 90° from the nominal X, Y, Z set about the R axis. This results in the four gyro spin axes,

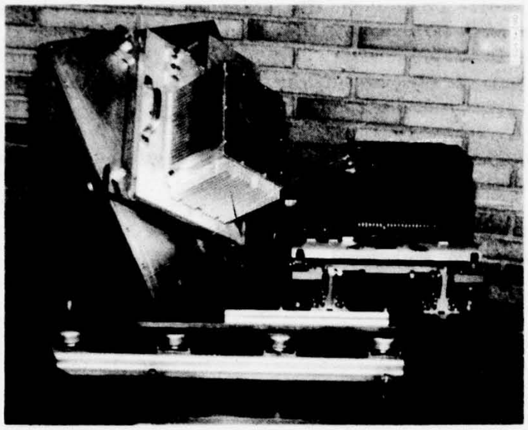

Figure 6. Dual IMU Installation

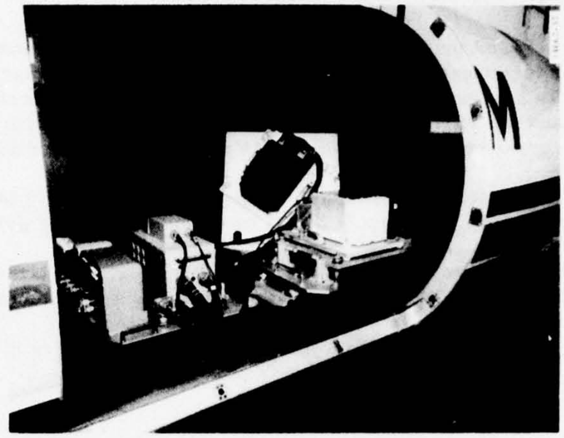

Figure 7. Two LN-50 IMUs on Test Pallet Installed in Test Aircraft

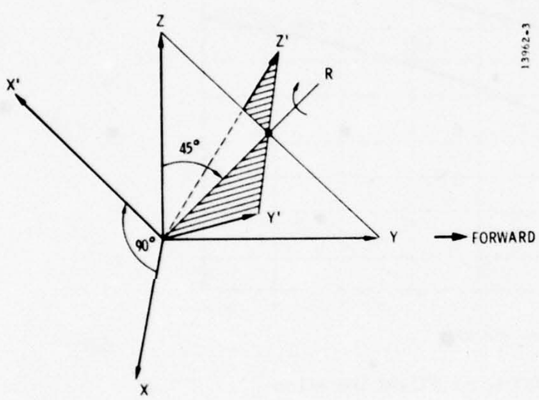

Figure 8. Dual IMU Geometry

Y, Y', Z, and Z', being equally spaced on a 90° cone. Adjacent spin axes, e.g., Y' and Z, are 60° apart, a satisfactory separation for 3-axis navigation data. While slightly less optimum than the 71° of the octahedron shown in Figure 1, navigation and fault detection performance degradation are negligible. An accelerometer is oriented along each principal axis. Since groups of three accelerometers lie in the same plane, e.g., Y, Z, and X', and X, Y' and Z', and many inter-axis angles are 90°, it is only possible to unambigiously detect and isolate one accelerometer failure.

For convenience, the input axes of every gyro are labeled X and Y. A following subscript then identifies the physical gyro involved. Table V shows the relationship between instrument axes and the body axes of Figure 8. The outputs of the two IMUs are input to the same LN-50 computer, software processing is structured as shown in Figure 9.

The predictable errors of each instrument are removed by compensation at an iteration rate of 64 Hz. Simulated gyro or accelerometer errors are manually injected by means of the LN-50 control display unit. The resulting redundant measurements are compared in failure detection and isolation equations to determine which measurement is in error.

TABLE V
AXIS IDENTITIES

Gyro	Accelerometer	Body
X_1 Y_2	X_1	X
Y_1 S_2	Y_1	Y
S_1 X_2	Z_1	Z
X_4 Y_3	X_2	X'
Y_4 S_3	Y_2	Y'
S_4 X_3	Z_2	Z'

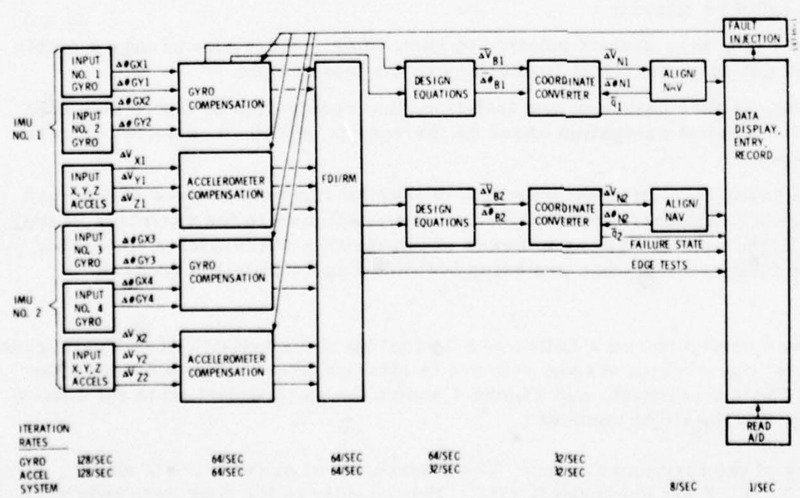

Figure 9. Dual IMU Mechanization, Software

Two completely separate strapdown solutions are then formed. One is a reference solution using the nonskewed IMU without instrument faults injected. The second solution is based on selectable (manual or automatic via FDI results) pairs of gyros and sets of accelerometers. Design equations perform coordinate transformations and account for the redundant measurement data contained in two 2-degree-of-freedom gyros. Two separate sets of quaternion coordinate transformations and inertial navigation equations are then available for comparison. Transients induced into the second solution by manual fault insertion prior to FDI response are thus directly observable.

FLIGHT TEST RESULTS

The flight test consisted of five flights of over 2 hours duration. During these flights, simulated gyro and accelerometer faults were injected into the second navigation solution according to the timing shown in Figure 10. The reference solution, undisturbed by these simulated failures, produced excellent navigation performance, ranging from 0.4 to 1.2 nautical miles (radial) per hour over the five flights. Velocity errors were allowed to propagate on the ground after the flight. Peaks of the Schuler oscillations are shown in Table VI, again indicating satisfactory inertial navigation performance.

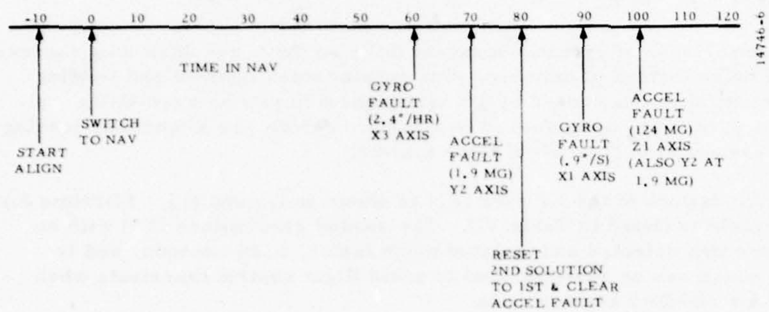

Figure 10. Test Plan for NASA/Langley Flight Demonstration

TABLE VI – RLN-50 FLIGHT TEST VELOCITY PEAKS (REFERENCE SOLUTION)

Flight Pattern	X-Velocity (ft/sec)	Y-Velocity (ft/sec)
East-West	6.0	-4.4
East-West	-3.2	-8.9
North-South	-2.1	-3.7
North-South	-7.3	4.1
Box pattern	-9.0	3.1
Static	-0.1	1.6

Figure 11 shows the navigation performance of the second solution minus the reference solution of one of the flights (14 November 1977). During the first hour of flight, the second solution was operating with gyros and accelerometers of the skewed IMU (60° off level). Navigation error growth is less than 1 nm/hr per axis.

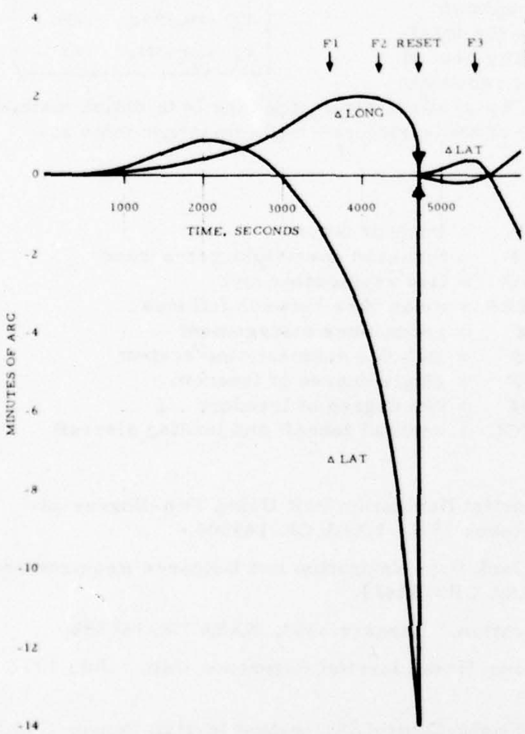

Figure 11. Delta Latitude/Longitude Through Simulated Failures

When a 2.4°/hr failure of gyro number 3 is injected into the computer, approximately 300 seconds elapses before the failure is detected, the correct gyro is identified, and gyro number 1 (from the level IMU) is substituted for the failed gyro. During the detection and isolation time interval, tilt and velocity errors not removed by simple substitution of gyros cause large position errors to accumulate.

The simulated 1.9-mg accelerometer failure was detected and isolated in approximately 430 seconds. Some additional tilt and velocity errors accumulate during this interval, increasing the navigation transient.

At a time of 4846 seconds from entry into the navigate mode, all variables of the second navigation solution were set equal to those of the reference solution during an aircraft turn of 180°, including position, navigation direction cosines, velocities, and quaternions. This technique is possible with redundant systems due to the multiplicity of instruments and computers. Note the minimal buildup of navigation error shown in Figure 11 following the reset.

This is further illustrated in Figure 12, which shows the difference in velocity measurements between the two solutions.* During the 10 minutes after the reset, navigation equations use one gyro in each IMU, the accelerometers of the skewed IMU, and initial conditions from the nonskewed solution — a complex set of conditions which still results in excellent navigation performance. An operational redundant system would thus employ such a computer reset mechanism immediately after failure detection and isolation to avoid most of the transients shown in Figures 11 and 12.

Each of the gyro parity equations represents the combination of the outputs of two TDF gyros along a common vector direction so that vehicle rotations are cancelled, exposing errors in either gyro. The parity equation output, illustrated in Figure 13, is then filtered by a first order filter with a time constant of 512 seconds. For short time durations, the filter acts as an integration of the angular rate error of the two gyros along the vector direction.

Filtered parity equations for the 14 November 1977 flight are shown in Figure 14 through 22. All six gyro equation outputs are shown, and three of the nine accelerometer equations. I_{12} and I_{34}, Figures 14 and 19, are direct comparisons between redundant axes of the same IMU. Disturbances in the plots are the results of normal gyro errors, such as bias, noise, scale factor, g- and g^2-sensitive errors, and residual dynamic errors acted upon by the aircraft environment and flight profile. They present an ideal lower bound on the detection of failures.

I_{13}, I_{14}, I_{24}, and I_{23} involve comparisons of gyros in separate IMUs so there are additional sources of fluctuations including four axes of noise instead of only two, plus angular misalignment and bending between IMUs. The largest transient at takeoff is caused by a misalignment in yaw between IMUs, calculated to be on the order of 0.1°. A gyrocompassing method was used to derive yaw alignment, leading to this limitation. FDI thresholds were set at 0.1° to avoid false alarms.

The effect of the simulated 2.4°/hr failure of the X1 gyro (F1) is shown in I_{13} and I_{14}. FDI time for both functions to reach the 0.1° threshold is listed in Table VII. The second gyro failure (F3) with an amplitude of 0.9°/sec into the X1 gyro was detected and isolated much faster, 0.14 seconds, and is representative of the switching time which can be accomplished to avoid flight control transients when the inertial angle rate data are used for stability augmentation.

Accelerometer FDI thresholds were set at 2.44 m/sec (8 ft/sec), well above the transients in accelerometer parity equations, as shown in Figures 20 through 22. FDI times for the 1.9 mg and 0.124 g failure amplitudes are given in Table VII.

TABLE VII — FDI TIME VS FAILURE AMPLITUDE

Failure	FDI Time (sec)
F_1 – gyro, 2.4°/hr	298
F_3 – gyro, 0.9°/sec	0.14
F_2 – accel, 1.9 mg	429
F_4 – accel, 0.124 g	4.5

CONCLUSIONS

The successful flight testing of skewed, redundant strapdown inertial measurement units of navigation quality has shown the feasibility of use of these methods in achieving the high reliability needed for flight control sensors. Data showing the noise level of redundant comparisons is presented in relation to FDI thresholds. The primary remaining task is to define methods of achieving stable alignments between sensors and means of measuring yaw angle misalignments accurately at a reasonable cost.

NOMENCLATURE

- CCV = control configured vehicle
- CEP = circular error probable
- FBW = fly-by-wire
- FCS = flight control system
- FDI = failure detection and isolation
- fo-fo = fail-operative/fail-operative
- GP = general purpose
- IMU = inertial measurement unit
- INS = inertial navigation system
- I/O = input or output
- KOP = thousand operations per second
- LRU = line replaceable unit
- MTBF = mean time between failures
- RM = redundancy management
- SAS = stability augmentation system
- SDF = single degree of freedom
- TDF = two degree of freedom
- VTOL = vertical takeoff and landing aircraft

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*The scatter of data points in Figure 12 is due to a time skew of 3 seconds in data recording, and aircraft heading changes.

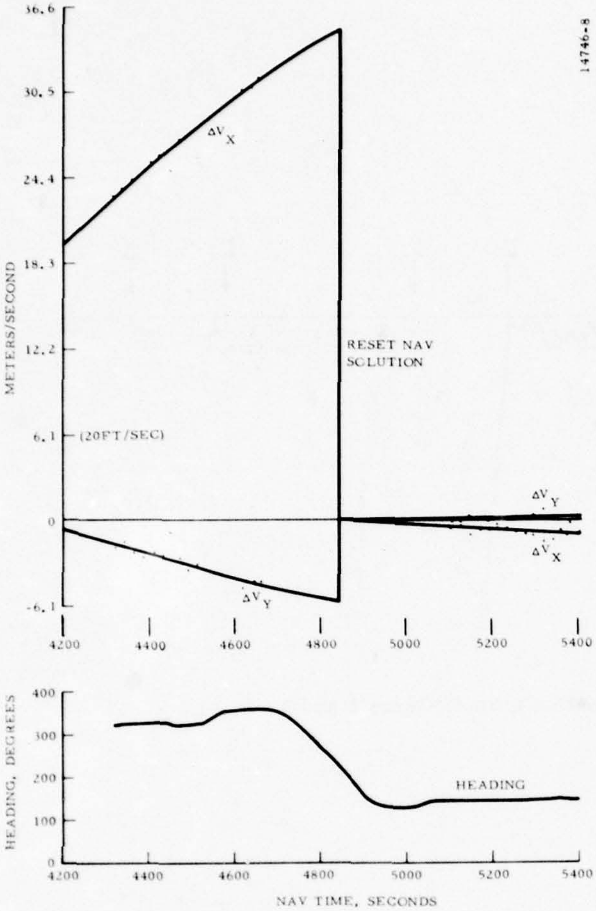


Figure 12. Delta Velocities Through In-flight Reset

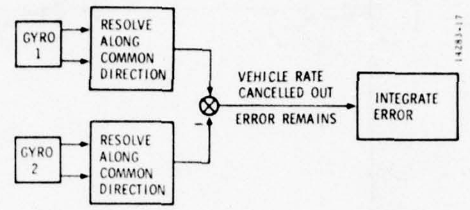


Figure 13. Formation of Gyro Parity Equation

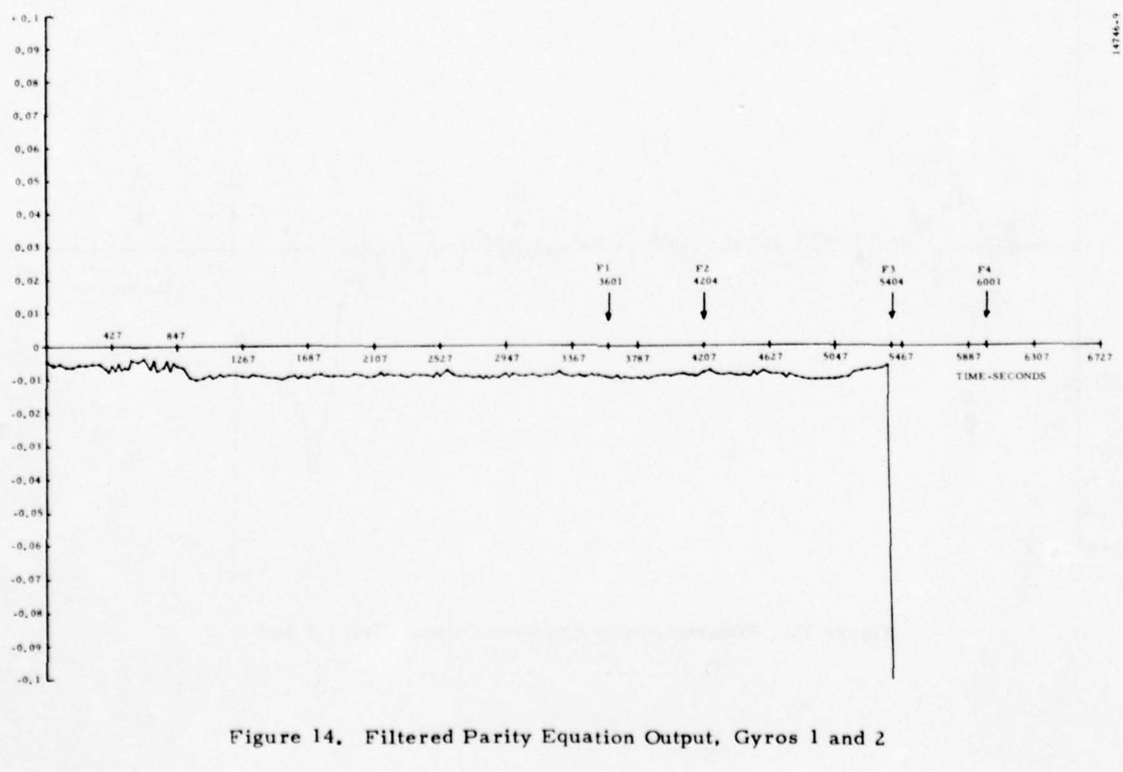


Figure 14. Filtered Parity Equation Output, Gyros 1 and 2

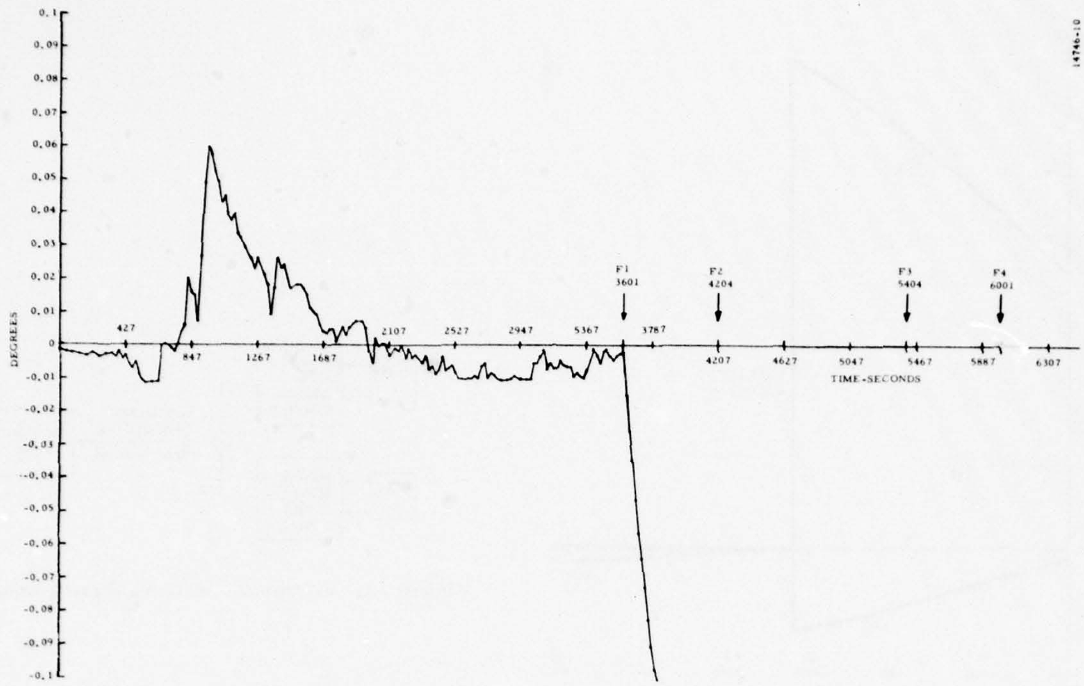


Figure 15. Filtered Parity Equation Output, Gyros 1 and 3

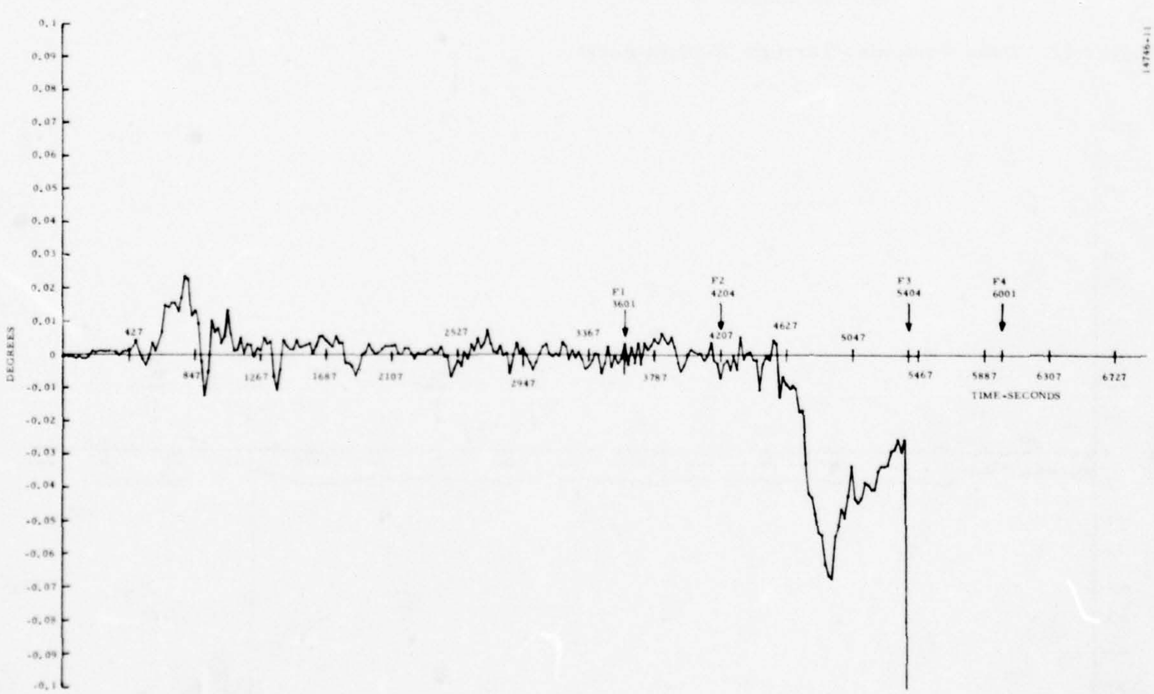


Figure 16. Filtered Parity Equation Output, Gyros 1 and 4

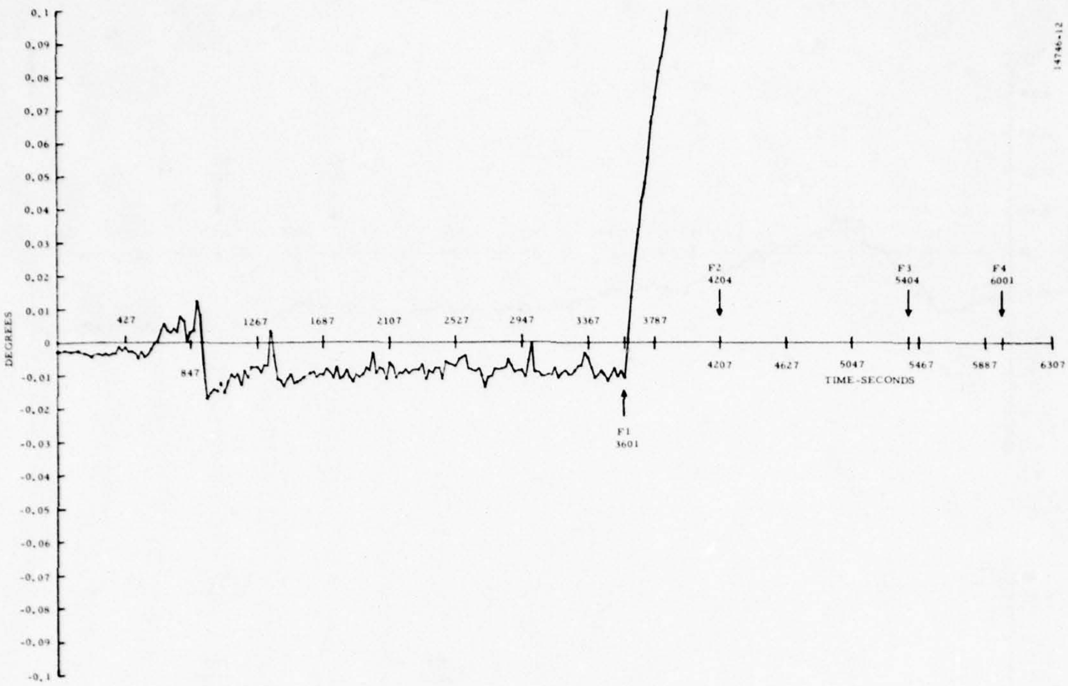


Figure 17. Filtered Parity Equation Output, Gyros 2 and 3

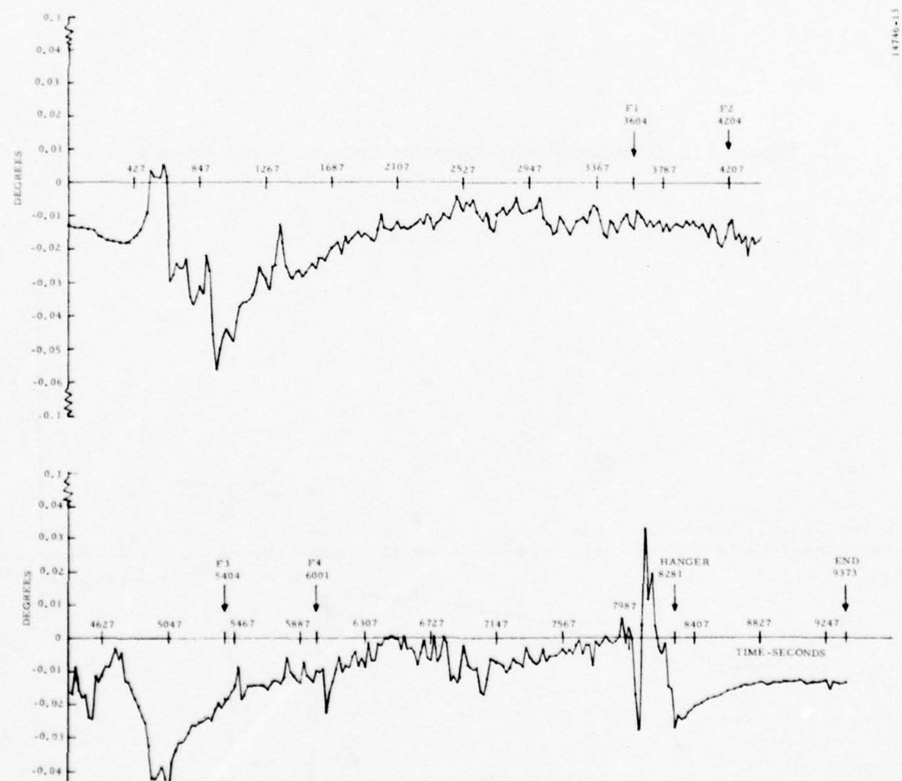


Figure 18. Filtered Parity Equation Output, Gyros 2 and 4

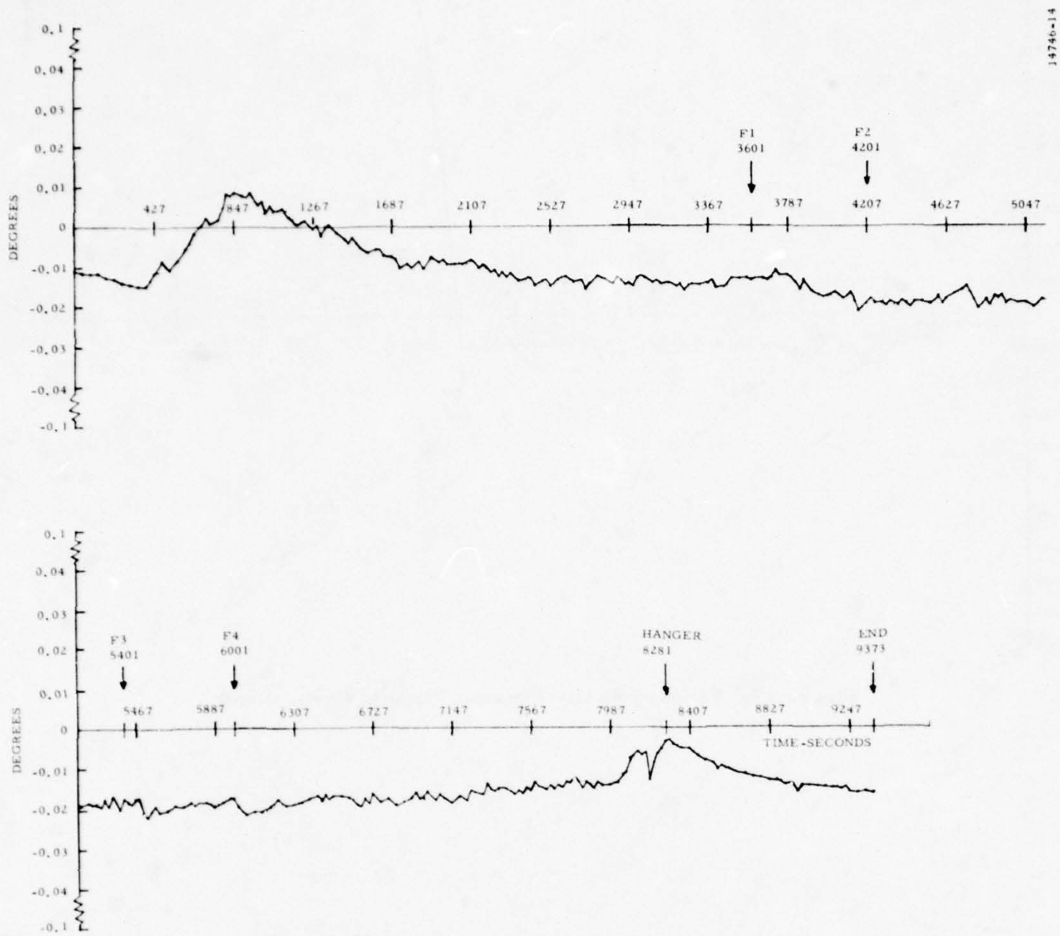


Figure 19. Filtered Parity Equation Output, Gyros 3 and 4

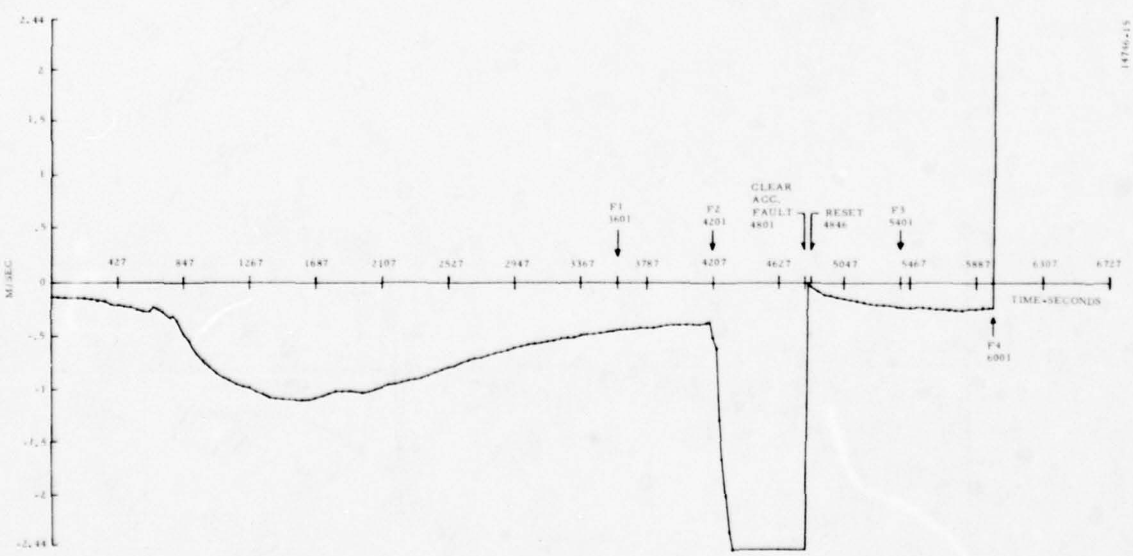


Figure 20. Filtered Parity Equation Output, Accelerometers X1, Y1, Z1, Y2

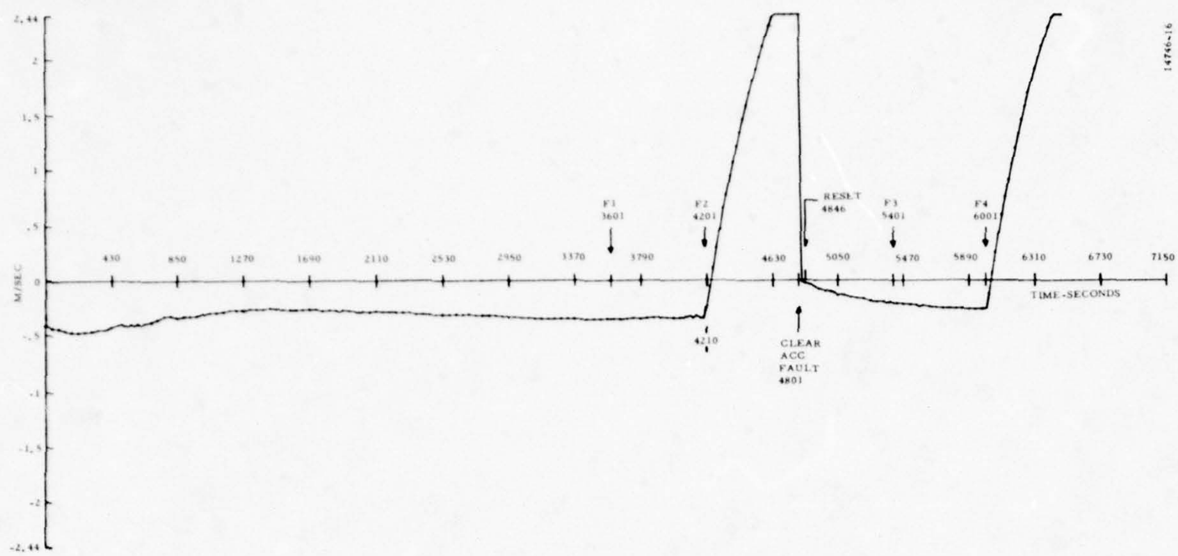


Figure 21. Filtered Parity Equation Output Accelerometers Y1, X2, Y2, Z2

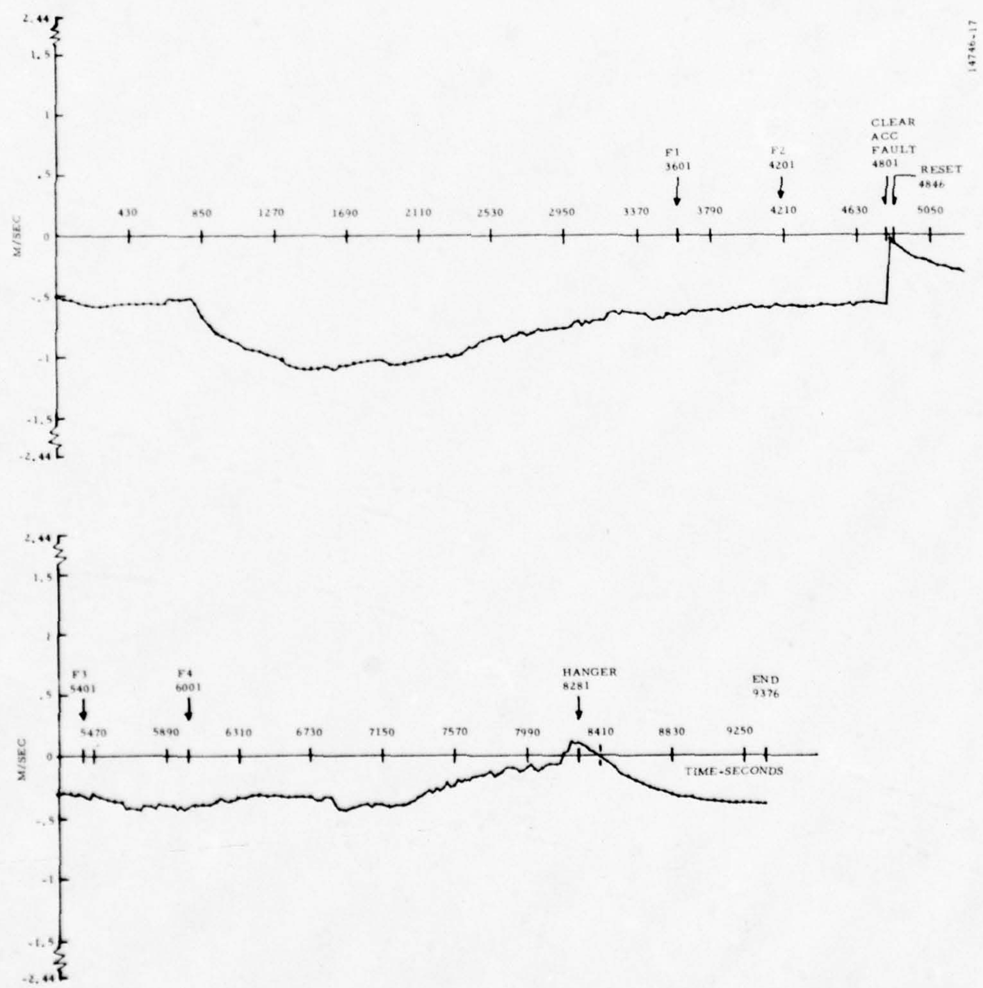


Figure 22. Filtered Parity Equation Output, Accelerometers X1, Y1, X2, Z2

SCAN CONVERTER AND RASTER DISPLAY CONTROLLER FOR NIGHT VISION DISPLAY SYSTEMS

by

H.W. Killian and W. Voswinckel
Electronics Division
DORNIER SYSTEM GMBH
FRG 7990 Friedrichshafen

SUMMARY

The paper describes a modular Raster Display System and its functional moduls. The system allows for digital scan conversion of images of electro-optical sensors and for digital storage of images with several gray tones. With the help of the digital symbol generator digital information can be converted into adequate symbology which again can be superimposed on the sensor image. Superposition of two sensor images can be performed with the same equipment.

Some technical features are discussed which exceed the common modes of current display systems and may help to support missions of military helicopters flying low level at poor visibility conditions.

1. INTRODUCTION

Extensive studies, simulation experiments and flight trials have been performed in recent years in order to find adequate solutions to the mission requirements which call for low level flying under poor visibility conditions.

This paper will not discuss the availability of onboard flight information and other relevant problems for the above missions, but concentrate on technical means in the display system area, which has considerably advanced with the progress of semiconductor technology.

The most suitable display system for the application in mind will depend on helicopter type, number of crew, mission requirements etc. but also on the availability of adequate equipment.

In general it can be said that the requirement is for the presentation of either a terrain image in raster technique or a computer generated symbology on its own or for a combined image consisting of a terrain image with superimposed symbology.

Displays which can be used for dynamic terrain images are available as cathode ray tubes working in raster mode. While monochromatic valves are state of the art for cockpit application, colour systems will be available soon.

Most raster display systems in use apply the standards of the commercial video technique, i.e. 525 lines at 30 Hz or 625 lines at 25 Hz with interlaced fields (CCIR-standard).

The superimposition of symbology can be done in two different ways: The symbols are generated in video raster and mixed with the terrain image signal or the symbology is generated by a stroke write generator and drawn during the vertical retrace period of the electron beam.

Both methods have their advantages and shortcomings. The raster-symbology looks not so smooth as the stroke written one, but the number of symboles which can be drawn in the retrace period is limited. Moreover, for future systems other video rasters such as non-interlaced fields may be used with more lines and shorter retrace periods. Therefore raster symbol generators are more advantageous for modern display systems.

The terrain video is derived from electro-optical sensors such as a LLLTV or FLIR camera or from a scanning radar sensor. The format conversion which may be necessary can be done with the help of a scan converter. In case a digital scan converter with an image storage is available, the latter can perform some extra tasks.

The image storage feature may become most interesting for reconnaissance tasks and target acquisition. It may also play a role as digital map storage of a gray tone map, which can be used to support the navigation.

A further requirement is to superimpose the terrain image or gray tone map image with the computer generated symbology representing flight data etc. The problem of hiding the background image by the symbols can be solved by proper symbol arrangement and symbol shape and size.

Furthermore one has to make sure that the symbols are readable under all illumination conditions i.e. means will have to be provided to ensure sufficient contrast between background image and symbology.

An even more sophisticated image superposition can be thought of: LLLTV and FLIR cameras have their shortcomings and benefits in some area, while imaging radar sensors have others. A combination of two terrain images taken by different types of sensors will possibly enlarge the information contents of a combined image.

It must be understood that this superposition cannot be the simple sum of image elements but must be performed with the help at an intelligent combiner.

Various types of symbol generators allow to meet all requirements regarding symbol types, positioning accuracy, economic coding etc..

Reliability in terms of fail-safe-behaviour of the display system must not be underestimated. Any failure in the system must allow the pilot to return to a flight mode where he can operate the aircraft with reduced information and return to his base. This requirement leads to a display system consisting of at least two monitors and display controller functional blocks which normally have their dedicated tasks, but can replace the second part to a certain degree.

In Fig. 1 a typical scenario is shown which gives an impression of the environment the system will have to work in. The reader will have to add just one component, poor visibility.

2. FUNCTIONAL ELEMENTS OF AN INTEGRATED DISPLAY SYSTEM

2.1 Scan Converter / Image Storage

Because of the physical principle of electro-optical sensors, the output signal does not necessarily meet the format requirements of a given video-raster system. The FLIR camera e.g. may have a scan mechanism carrying numerous detectors. Only thus sufficient dynamics of the terrain image can be achieved with state of the art detectors requiring comparatively long integration time. A format or scan conversion becomes necessary so that special monitors for each of the sensors can be avoided. By now mainly analog type scan converters have been applied. But late semiconductor buffers together with fast logics allow already now reasonable solutions to the problem. Full advantage of digital converters can only be taken when next generation memories will be available. There are two major ways in digital scan conversion:

The first one stores the digitized sensor image in the format in which the sensor works. The conversion into the format of the display system is done from the stored data.

The second way is to process the incoming sensor data directly and feed them to a storage which is organised in the display format, i.e. each pixel of the screen is represented by one or more bits in the storage.

For each application a trade off has to be performed as regards refresh rate of imaging sensor and display (the latter is the video refresh rate), resolution of sensor and display, memory organisation, work load on conversion hardware or processor resp.. Other aspects such as multiple usage of the same image storage for other tasks than scan conversion may also influence the solution.

Fig. 2 shows the block diagram of a scan converter with its typical elements.

Analog to digital conversion of the incoming video signal normally requires a fast converter, especially if more than one analog channel must be processed. The scan/format conversion will be sensibly done on the digitized data and the logic will be special to the video-sensor. It will have to take into account all relevant features of the sensor, such as scan pattern, resolution, non-linearities in scanner motion etc.. One can even think of a compensation for the position changes or vibrations of the helicopter with time, which otherwise may result in distortions of the image.

The size of the image storage is again a function of a number of parameters. One of these is the display format, i.e. the number of lines and pixels to be displayed. The number of "layers" of the storage for each pixel depends on the intensity resolution of the sensor and of the monitor. In case of colour displays additional layers of the storage are needed.

The storage contents will be fed linewise to the monitor via a digital to an analog converter. The read out of the image storage will not erase the storage contents so that an image in the storage can be replayed over an arbitrary long period (single shot mode).

The micro-processor shown in Fig. 2 performs the scan conversion. It will also allow for additional manipulation of the stored image. The manipulation can comprise all kind of information enhancement or even pattern recognition and data reduction.

The latter feature can become most important in the context of image transmission on low bandwidth radio links.

2.2 Symbol Generators

In the area of raster symbol generation there is the choice between different approaches, each of them with specific advantages and shortcomings. The three major types will be presented. They can be combined depending on the task to be performed.

In Fig. 3 the general block diagram of a symbol generator / display controller is shown. The main functional elements are the control unit with the symbol generator memory and the refresh storage. The input data enter the system via an interface block which in some applications also comprises an input buffer. The pixels which build the symbols are stored in an output buffer, then they are d/a converted and possibly superimposed on a video-signal before the analog data is fed to the monitor.

More details on the different types of symbol generators are given below.

2.2.1 Full Image Buffer Type Generator

This type of symbol generator involves a background storage with at least the same number of cells as the number of pixels shown on the screen. With reference to the image storage of the scan converter this background storage can be considered as one layer of the displayed image. The contents of this buffer will be transfered to the monitor in a line by line manner. Loading of the buffer is to be done e.g. with the help of a microprocessor controlled symbol generator, which converts the incoming coded data into pixels (dots) of the symbols. This method allows for the generation of extensive images with rather small dynamics. A symbol can be erased by erasing all pixels, from which it is constituted.

In case symbols are superimposed, erasing of one symbol results in a missing pixel of the other unless adequate means will be provided to overcome this.

This type of generator does not necessarily need an input buffer, since the image is stored pixelwise.

Fig. 4 shows the block diagram of this type of controller. It's main hardware characteristic is the bulky image storage.

2.2.2 Line by Line Type Generator

This type of generator is best suited for highly dynamic symbol images with a limited quantity of symbols, such as a roll/pitch symbol on a fighter aircraft display. The block diagram is given in Fig. 5.

In the symbol generator only those pixels of symbols are generated and transfered into a switched buffer which will be presented on the screen in the raster line succeeding the one in progress. Thus, one half of the buffer is transfered to the monitor in accordance with the TV standard applied, while the second part of the buffer is filled with the information to be shown in the next line. While the second part of the buffer will be fed to the monitor, the first one will again be filled with the information of the line succeeding then.

The control unit will have to check all coded information in the refresh storage for elements of the next following raster line every line period, which is in the order of 40 to 60 µsec depending on the format applied.

Thus the number of symbols which can be generated during this period is limited by the generator capacity. This restriction can be reduced by i.a. proper organisation of the refresh storage, extension of the symbol generator capacity by dedicated generators and/or by a larger line buffer storage with e.g. ten raster lines instead of two. While the increase of the storage size will reduce the dynamic capability to the same degree, the higher number of special symbol generators increases the complexity with almost no loss in dynamics. In the extreme case, each symbol type may have its own generator.

The image quality of this generator type is the same as for the full image buffer type. Fig. 6 shows a typical image of the above two generators.

2.2.3 Chess Board Type Generator

This type of generator (Fig. 7) assumes the screen to be partitioned in a number of cells built by an imaginary net of lines and columns. Each cell may consist of $n_x \times m_y$ pixels and can be identified by its reference coodinates or reference number resp..

The symbol generator output signal is directly connected to the intensity control input of the monitor. Therefore no line buffer is required. The generator is best suited for alpha numeric displays. In case a sufficiently large set of symbols is available in the symbol memory this type of generator can also produce simple graphics as shown in Fig. 8 and 9.

This type of symbol generator is the simpliest one discussed here. Since the refresh storage is also organized in the chess board manner, for the generation of the pixels of one raster line only the codes of one row of the image have to be searched for relevant information. Because of the extremely economic coding this generator type is also of interest in the area of data transmission.

2.2.4 Symbol Generation Techniques

There are two ways of symbol generation used in raster display technique. While the full image buffer type generator can live with both of them the other two can only work with the method described first.

Each symbol or character which can be selected is stored in the symbol memory in matrix form e.g. 5 x 7 or 7 x 9 pixels (see also Fig. 10). From this given set of symbols an image can be composed. The number of symbols will depend on the size of the memory.

While the chess board type controller can place a symbol with reference to the fields only, the others can position it at any arbitrary point of the screen.

More flexibility in regard to symbol and character size and complexity is provided by the second method, where symbols are composed by adding minivectors to each other. This way any size of a symbol can be achieved by multiplication of the standard lengths by a constant factor.

2.2.5 Superposition of Images

The requirement for superposition of symbols and E/O-images arise for cockpit application in the vertical situation display and the horizontal situation display area. Mainly the two controller types discussed first will be used for this purpose. While the symbols are in general presented at one brightness level only, E/O sensor images will contain various tones of gray.

Good readability of the symbols has to be provided, independent whether the symbol is shown in front of a bright or a dim background.

The technical means to achieve this are i.a. a relative brightness control of the two image contributions, inversion of the symbol brightness in case of bright video image or punching a dark background field behind a bright symbol. The dark background field can be reduced to a small space at the left and the right of the symbol.

Fig. 11 presents a video-image with superimposed symbology.

Mixing of images does not only mean superposition of symbology on a background image, but may also be the superposition of two or more symbol images or the combination of several sensor images. The latter is in a very early stage of development.

In Fig. 12 the images of two different E/O-sensors are combined in such a way that the outer part may be provided by a sensor with a poor resolution, while the centre is taken from a high resolution narrow angle sensor. In the case of scan converters with image buffers being involved in the display system, also more sophisticated combinations of images can be thought of.

2.2.6 Special Features of Raster Display Systems

The main characteristics of the raster technique are that the images are replayed line by line. For digital images the lines are again divided in a number of pixels.

All symbols including characters and the more complex ones such as circles, vectors, etc. are composed of a number of dots. The number required to get a legible symbol depends on the symbol size and the resolution of the screen.

The screen can be refreshed either in the interlaced mode (commercial TV) where the image is built up from two half pictures which are displaced to each other by one video line or the same half image is shown twice. The observer will experience either a flicker or a highly digitized image unless a sufficiently high repetition rate as well as a high number of lines are given. This will result in display systems for shophisticated applications which exceed the present standards i.e. 525 lines/30 Hz and 625 lines/25 Hz.

Working with the above standards improvements can be achieved by e.g. showing horizontal symbol lines by at least two adjacent raster lines or by interpolation algorithms and gray tones.

3. APPLICATIONS

After this excursion into technical details let us return to the application, i.e. the helicopter display system.

First of all the pilot needs a screen which provides him with the visual sight. Because of the type of mission in mind the outer world will be sensed by a suitable electro-optical sensor. It will be converted in a scan converter and displayed on the monitor. The screen must be mounted at an exposed place in the cockpit. Independently whether it is a Head Up Display or a Head Down Display it occupies a significant amount of space in the panel area which has earlier been used for the mounting of conventional instrumentation.

Furthermore the pilot will not be able to watch a large number of aircraft parameters while flying i.a. nap of the earth at night while all his attention is concentrated on the picture on the screen.

The limitation in panel space as well as the load on the pilot can be reduced by superposition of most relevant flight data on the Vertical Situation Display.

On a second monitor, a Multifunction Display, which will also be mounted on the panel, all kind of alphanumeric data can be shown. This display may also be used to present the text of a command or a report which arrived via a data communication link. Transmission of text is most economic as compared with voice as regards time duration. Shorter radio transmission times may help to make the detection of the aircraft more difficult. The Multifunction Display can also be used to optionally present detailed information on flight status, aircraft status, weapons, targets etc.. In case of a failure of the display described first the Multifunction Display can serve as a replacement. The presentation will be limited to the most important information.

A third monitor as a Horizontal Situation Display may support navigational tasks.

Map information can be displayed as gray tone image, while reference points or targets can be represented by suitable symbology. The map can be generated from the coded data stored in a magnetic cassette system.

Fig. 13 shows the block diagramme of an integrated display system. It consists not only of the displays but also of a scan converter/image storage, a symbol generator subsystem consisting of all three types of generators discussed earlier and a number of interfaces and input devices. Controls will allow the crew to operate the system.

With the equipment discussed, it will be possible to not only meet all requirements in the display area but do a great bit more.

The following will give some additional features:

As seen earlier the image buffer allows to replay an image on the display screen over an arbitrary long time. This feature may support the crew in so far that during periods of fast changes of the outer world image, a static image can be achieved by an electronic switch between electro optical sensor and buffer. Thus, the human eye can evaluate the image before the next one is presented. The same feature can be utilized when single camera shots are taken in a bob-up-mode. Since the evaluation can be performed afterwards the exposure time can be minimized.

The image buffer is a digital one. Thus it becomes possible to transmit the image via a radio link in a bit by bit manner. The receiving station which can be either a ground post or another helicopter will recompose the image. For a 625 line video system the number of bits to be transmitted will be in the order of 3 Mbit with 6 gray tone bits for each pixel. A reduction can be achieved by either a smaller number of gray tone bits, a worse resolution or a combination of both. 3 bits of gray and a resolution of one quarter of the original will provide a reasonable image with about 400 Kbit of information which means a reduction of the transmission time by eight. Unfortunately the poorer resolution complicates the evaluation. As a compromise the system allows for transmission of the image which poor resolution and gray tones while a certain area can be transmitted with full resolution and all gray tones. Thus, the receiver can get a general impression of the environment and all details on the subjects of interest. Fig. 14 illustrates the procedure. The picture shows a high way crossing with a truck on the exit road.

In case targets need not to be presented in detail, but only be marked, a graphic symbol agreed upon will suffice. It can be transmitted in code together with the target coordinates. This method will require not more than 30 bits instead of about 200 Kbit in the above case for the high resolution sector.

By now work with the map represents a major problem area for the missions in mind. An improvement may come up with maps where altitude contours are represented by gray tones. This kind of map can be stored on a tape in coded form. Presuming all units of a platoon working with the same map it becomes possible to reduce the data transmission to just a small number of graphic symbols which represent relative position of the aircrafts involved, target positions, obstacles etc..

The codes are a few bits long only, complete messages can be transmitted in an extremely short time. Thus lengthy verbal information exchanges can be avoided.

The transmission link can also be used in the opposite direction. All kind of data can be generated onboard and be transmitted to one or more receiving stations, where the information will be displayed on the Multifunction Display. Alphanumerics and/or graphics or a combination of both can be transmitted in coded form. This method allows for a minimization of the operation time of the radio equipment and this may help to make the detection more difficult.

Alphanumerics is also a very economic and powerful tool in information exchange. Each character consists of not more than 10 bits if a chess board type display controller is used (e.g. ASC II). The information to be transmitted can be composed with the help of the Multi Function Display. Formating of reports and other regular messages may help to generate them.

The key board shown in Fig. 13 stands for all the input devices such as key board joy stick, roll ball etc..

Superposition of two video images can be done in several ways. Fig. 15 shows the image of a high resolution sensor in the lower part and a poor resolution image in the upper part. Superposition is done in the image storage with two different input stages, one for each of the sensors. The sensor for the lower part may show good performance in the short range while the other may be a far range sensor. In Fig. 12 another method of superposition is illustrated where the high resolution segment can be defined by the operator.

4. HARDWARE

The applicability of the above techniques will strongly depend on the availability of suitable hardware. Fig. 16 shows an advanced model of a raster display controller/symbol generator which contains also the major elements for image manipulation. The set up consists of - front to back - the sophisticated video mixer (1), the image buffer with 512 x 648 bits (4), the micro processor with symbol storage (2), the input stage (1) and the timing unit (2) which also generates the synchronisation signals for the TV-monitor.

The numbers in parentheses represent the printed boards. The functional moduls build part of a family which has successfully been applied in a number of military programs. The power dissipation of the above controller is about 20 Watts.

The scan converter which was used for the image manipulation consists mainly of the same functional moduls, i.e. image buffer micro processor etc.. The frontend stages and the output interfaces will be special to type. The system will consist of about 28 printed boards depending on the interface requirements. The power requirement is about 30 Watts.

Because of the low power consumption forced cooling is not required. The board size allows for mounting in 1/2 ATR boxes.

First flight tests of an experimental set up are planned for the time after the simulation tests in 1979.

FIGURES:

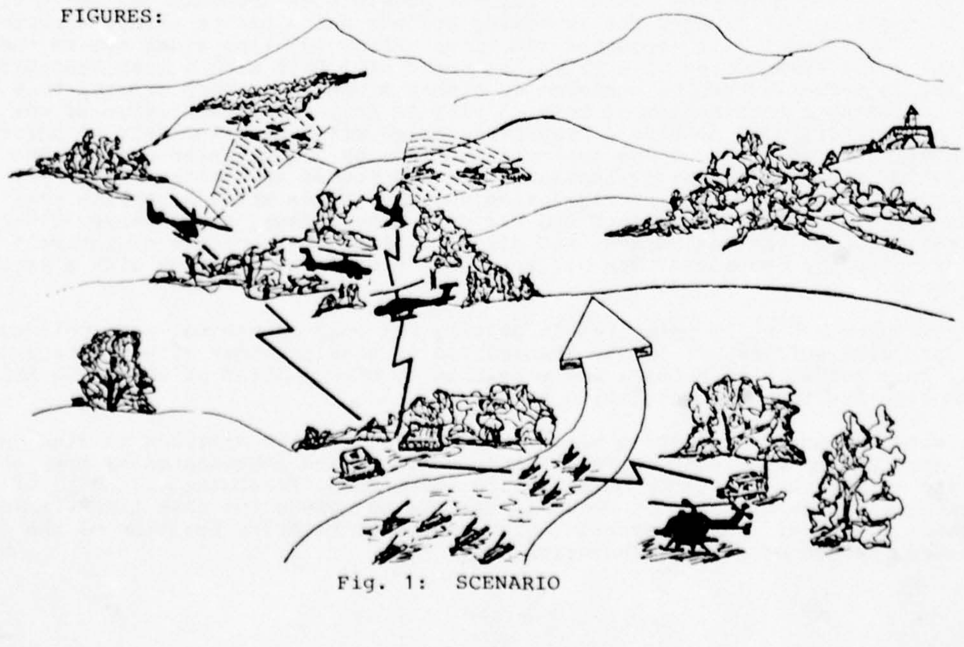


Fig. 1: SCENARIO

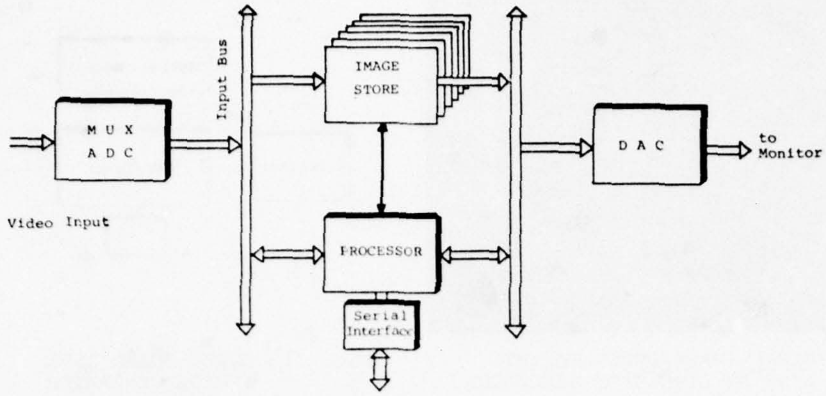


Fig. 2: SCAN CONVERTER BLOCK DIAGRAMME

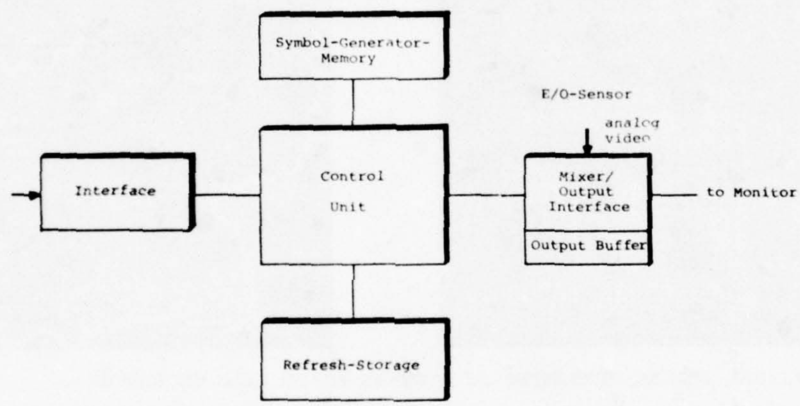


Fig. 3: SYMBOL GENERATOR, TYPICAL BLOCK DIAGRAMME

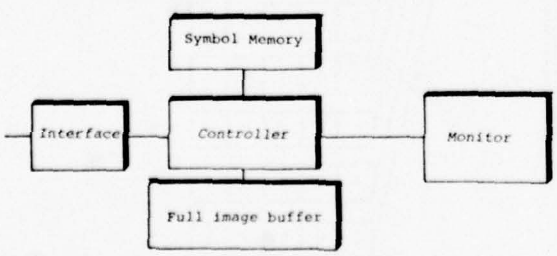


Fig. 4: FULL IMAGE TYPE SYMBOL GENERATOR

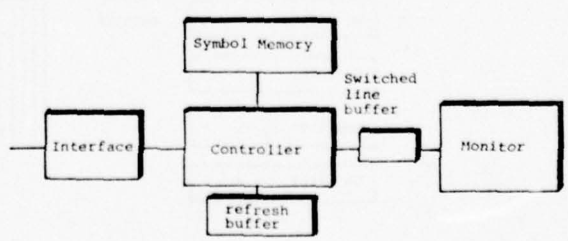


Fig. 5: LINE BY LINE TYPE SYMBOL GENERATOR

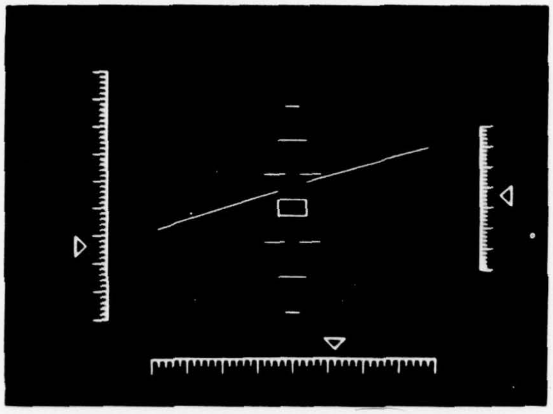


Fig. 6: TYPICAL IMAGE GENERATED BY A LINE BY LINE TYPE GENERATOR

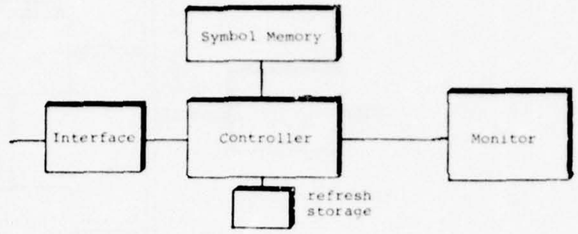
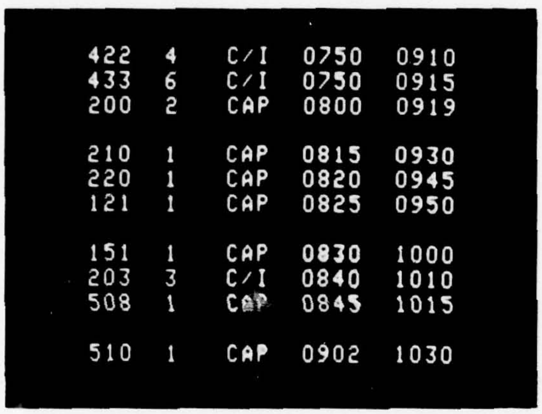


Fig. 7: CHESS BOARD TYPE SYMBOL GENERATOR



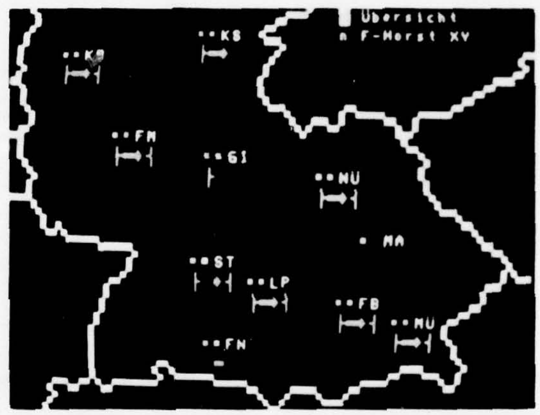


Fig. 8/9: TYPICAL IMAGES GENERATED BY A CHESS BOARD TYPE GENERATOR

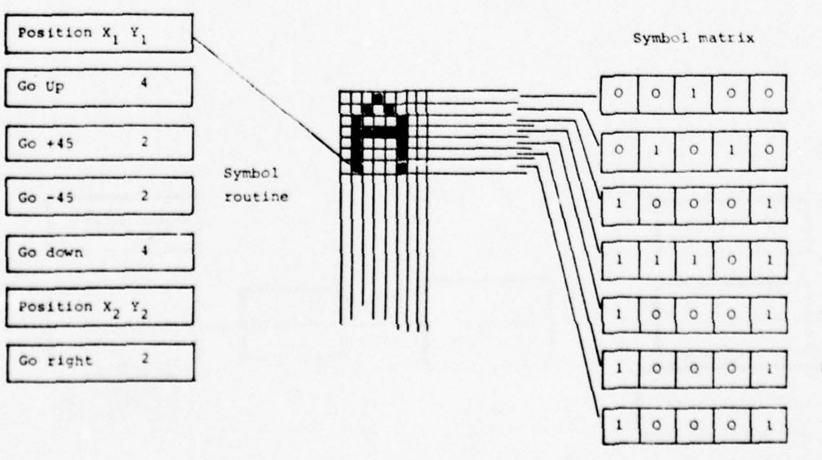


Fig. 10: SYMBOL GENERATION TECHNIQUES

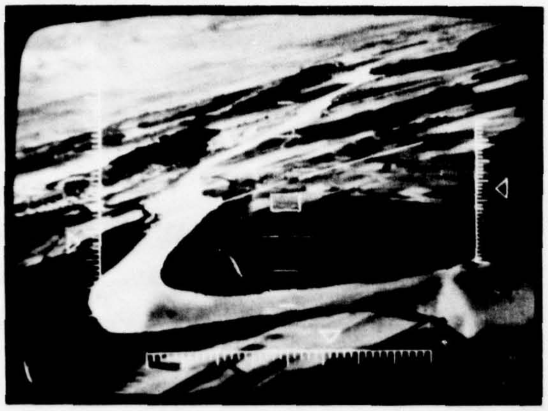


Fig. 11: VIDEO IMAGE WITH SUPER-IMPOSED SYMBOLOGY

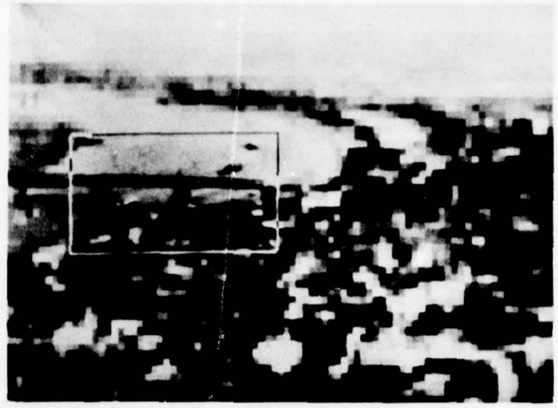


Fig. 12: SUPERPOSITION OF TWO VIDEO IMAGES

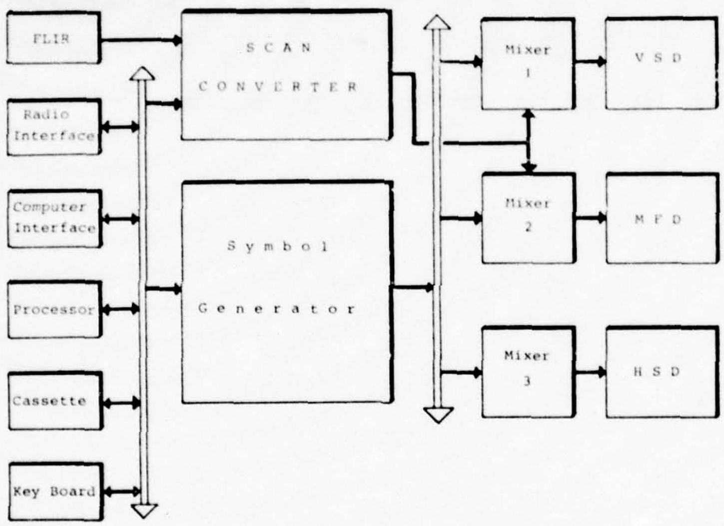


Fig. 13: INTEGRATED DISPLAY SYSTEM

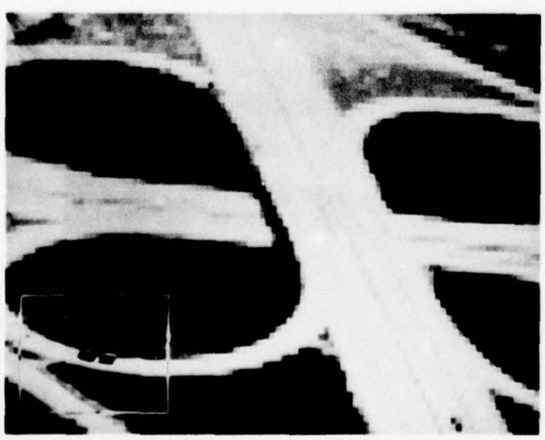


Fig. 14: 8x8 RASTER IMAGE WITH HIGH RESOLUTION SECTION

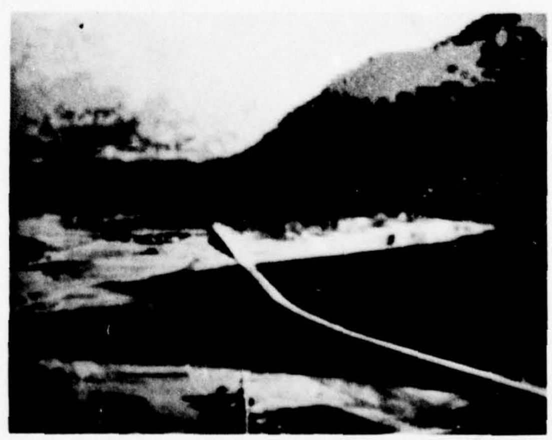


Fig. 15: TERRAIN IMAGE TAKEN BY TWO DIFFERENT E/O SENSORS

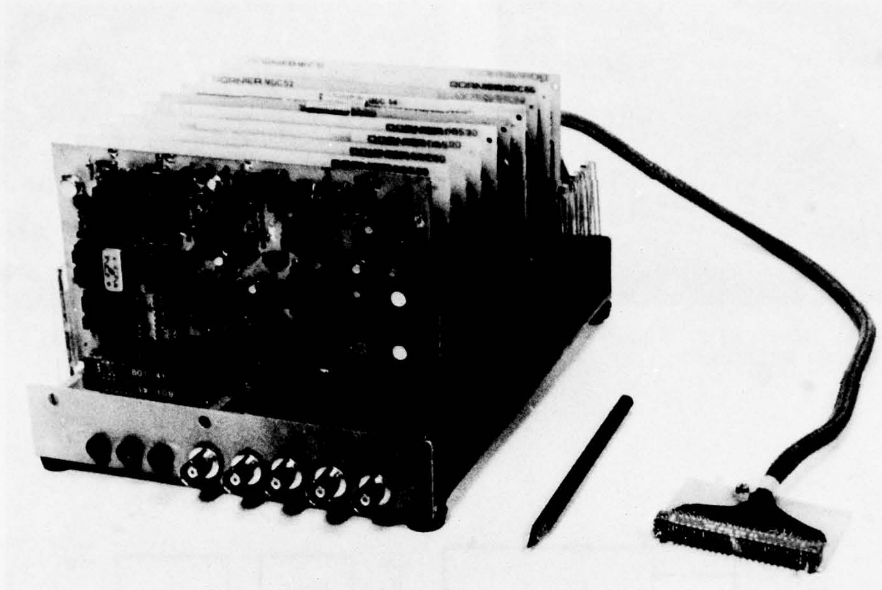


Fig. 16: DISPLAY CONTROLLER / SYMBOL GENERATOR

APPLICATIONS OF PATTERN RECOGNITION SYSTEMS FOR DAY/NIGHT PRECISION AIRCRAFT CONTROL

by

Alfred Kleider
Physicist
US Army Avionics R&D Activity
ATTN: DAVAA-E
Fort Monmouth, NJ USA

SUMMARY

A wide variety of military and civilian tasks requiring the precision control of helicopter position and movement can be addressed by the judicious employment of pattern recognition systems. This paper explores the philosophical and practical foundations for the utilization of advanced technological developments in lasers and Charge Coupled Devices (CCD) to provide light weight, low cost solutions to these problems in day/night operations. The special case of weather limited visibility is presented and projected operational parameters are discussed relative to near term availability of infra-red devices (IRCCD).

The concept of feature extraction is explored using the complimentary one dimensional characteristics of wire/wire-like obstacles and a 1728 linear CCD Array as the foundation of the Wire Obstacle Warning System (WOWS) presently in development. In this instance the appropriate geometrical, optical, and electronic compatibilities were combined to provide a real time automatic pattern recognition system for Nap-of-the-Earth (NOE) helicopter operations.

In a similar manner cooperative "targets" are considered with appropriate sensor geometries such that recognition can be effected digitally. Thus, in conjunction with a modest on-board computer composed of LSI semiconductor structures, position and pilotage data can be provided at modest cost. The employment of a low power, modulated illuminator provides a compatible day/night system with high background rejection.

1.0 BACKGROUND:

The rapid advances in technology, specifically in the areas of Large Scale Integrated (LSI) silicon devices and lasers, have provided the aviation community with a wide variety of potentially useful components and concepts. Specifically the developments in Charge Coupled Devices (CCD) and microprocessor-based microcomputers are of particular interest in this area. The uses of these components are only limited by our imagination, and as shown in an earlier paper [1] in Ottawa, Canada, the data processing capability of CCD's can be utilized to accomplish pattern recognition and pattern discrimination.

Time and the intent of this conference does not lend itself to a discussion of a physics of CCD's, however, one simple comment is germane to the concepts to be addressed in the remainder of this paper. Whereas the components normally encountered in an electronics device usually provide a controlled variation in current, voltage or frequency and thereby result in the transfer of information in terms of modulation-demodulation signals, the CCD is uniquely different. In the CCD it is a data stream which is outputted by the device. In essence the CCD is truely a digital device which can be variably quantized to yield either a dual or multilevel logic depending upon the thresholding techniques employed. Thus, as in the Wire Obstacle Warning System (WOWS) [2][3] a binary logic is used to develop an algorithm to recognize the presence of a wire in the field of view of the system. Similarly by selecting a suitable CCD filter to follow the CCD sensor a shades of grey device can be developed to impart three dimensional information to a scene in a two dimensional display.

To date the CCD has been employed in an imaging system [4] in either an NxM matrix of elements or in a linear format which senses the scene in swept strips which are reformatted in a scan converter. A second broad use of CCD's is in the area of a tapped filter which can be readily shaped and tailored to fit specific system requirements. In the broadest sense the concepts of pattern recognition must focus on dissimilarities; i.e. those characteristics of geometry and placement which provide a means for identification. In a similar manner pattern recognition techniques can be utilized in CCD sensor based systems to provide a wide variety of aircraft guidance and control functions: The successful and perhaps simplest system matches the sensor geometry to compliment the target asymmetries. To illustrate, if one considered a one dimensional object such as the image of a wire in the focal plane of an optical sensor it can be characterized relative to resolution as an orthogonal set with exclusive high and low spatial frequency requirements. Thus in selecting a sensor a single linear CCD with a large number of small size elements (8μ x 17μ) was chosen to provide high frequency resolution perpendicular to the array and it had a low frequency response parallel to the array. The low frequency response was determined by the mechanical scan mechanism and the scan to scan separation within the total field of view.

When we address the area of aircraft Guidance and Control with an eye to utilizing a pattern recognition system, the normal problems associated with pattern recognition are at one and the same time nonexistent and totally under the control of the system designer. To illustrate, in a recognition system the goal may be to distinguish a tank from a truck so the presence of a large projection in the middle of the vehicle can be used as an identifier since most tanks carry a large turret mounted gun. However, in a guidance and control situation it is not an object which must be distinguished but rather a position or a set of conditions. For those applications it is an apriori assumption that the "target" will be cooperative. In essence both the target itself and the target placement can be counted as system elements which exhibit a synogistic relationship.

2.0 APPLICATIONS AND SYSTEMS:

In arriving at an application for a pattern recognition system it is required that (1) a need must exist, (2) the need is unfulfilled or the solution is costly and/or significantly raises the pilot workload.

The applications herein considered are related to helicopter aircraft because in the first case the problem is unique to nap-of-the-earth (NOE) operations; in the second, the operation is only meaningful for a helicopter, and lastly the update rates for a helicopter are sufficiantly slow to permit an optical evaluation and interaction in a man/machine interface. These applications in the order given above are (1) Wire Obstacle Detection, (2) Precision Hover and (3) Formation Flying.

2.1 WOWS:

The first item, the Wire Obstacle Warning System (WOWS) has proceeded beyond the concept stage and the system fabrication is nearing completion. The contractor, the Fairchild Camera and Instrument Co. of Syosset, NY is scheduled to deliver WOWS on or about 1 Dec 78 and flight testing is projected to begin in Apr '79. In this application the object to the recognized was not within the control of the system designer, however, the unique signature of the wire was the forerunner of the concept and provides the key to other applications.

The WOWS is shown conceptually in figures 1 and 2. It is a night vision device incorporating a GaAs laser illuminator which is synchronously pulsed with a gated image intensifier. The system is designed to operate over a wide spread of light levels, however, its maximum effectivity is below 10^{-2} ft candles in a clear weather situation. Operation at 1.06 microns is also possible with a slight improvement over the .85 micron GaAs system for light haze operations. The most promising low light level, all weather system would use a pulsed 10.6 micron CO_2 illumination system, however, the present status of infrared CCD's (IRCCD) is limited to arrays composed of larger size individual elements. For the purposes of wire recognition, the elemental size limits the spatial frequency capability of the device and thus negates the pattern recognition feature paramount to the WOWS operation. Low reflectivity of wire materials at 10.6μ similarly reduce the usefulness of this wavelength for this type of system. As will be shown in the following paper by Dr. DelBoca of US Army Avionics Research & Development Activity, Fort Monmouth, NJ, a successful 10.6μ system requires refined detection techniques such as heterodyning to provide adequate signal to noise. However, it must be remembered that in this application the target is small (approximately 10^{-5} radians) and is non cooperative.

The WOWS is classified as a non-imaging system. As is shown in figure 3 WOWS is a feature extraction device in that the objects imaged at the focal plane of the scanned linear array are processed by a digital algorithm corresponding to the pattern of a wire. Thus the system is blind to the trees and even the towers that support the wire, and the on board computer dedicated to WOWS can only recognize and display the presence and location of the wire.

This then is the key to other signature sensitive systems. For a given purpose the target must be specificable in a unique digital format which can be filtered by the associated computer to provide the necessary sensorial outputs to derive the necessary signals for aircraft guidance and/or control. The systems which are treated in the following sections are still in the realm of concepts. It remains to be seen if they will have sufficient need or cost effectiveness to be built and tested.

2.2 Precision Hover:

In a variety of situations, both military and commercial, it is highly desirable to position a helicopter over a specific ground point. Even under daylight and calm weather conditions the pilot is severely taxed to maintain a precise hover without visual corrective inputs from the ground. To illustrate, in the installation of roof mounted cooling towers and other equipments in hard to reach places, the helicopter has been used as a flying crane. In fact, in the final placement of equipment, riggers such as those employed in conventional crane operations are used to provide the guidance function for the pilot in the form of hand signals. A system capable of providing this "guidance" function independent of a ground crew member would be advantageous.

The utilization of a pattern which is readily described in a digital format, and one whose asymmetry is deliberately included in the system design, can be matched to a CCD sensor to provide a highly sensitive positioning system capable of both day and night operation.

The selection of a CCD format should reflect an interplay with the target design. For the precision hover sensor a target design, as shown in figure 4, has been selected with a mixture of asymmetries and cueable symmetries which will allow the sensing of orientation as well as positionally corrective diognostics. The target design conforms to the conventional 3x4 aspect ratio common to television frame format. This selection of signature is thus consistent with an area CCD such as the Fairchild 211 which is a 244x190 element matrix whose photosensors are 18μ by 14μ with a center to center spacing of 30μ. Using a one (1) meter target whose design is implemented with cat eye retro-reflectors about 3mm diameter and a 244x190 CCD in the focal plane could be effective for heights up to 30 meters. The target image in the focal plan would encompass 165x220 elements at the maximum hover altitude.

In operation the Precision Hover Device (PHD) would be mounted on a low grade stabilized platform. The requirements for the platform are simple since the expected variations on the three axis are small and do not involve rapid responses with incumbent high G-loads. A dual lens system which would accommodate an acquisition field of view as well the higher magnification optic for accurate position maintainance would be used to put the aircraft on station. For both daytime and night operations a low power CW laser illuminator is used to provide both sufficient signal for the CCD recognition system and to allow narrow band optical filtering to accommodate the background conditions in daylight operations. The PHD is not intended to function as a long range acquisition device and therefore no provisions are included to enhance navigation to or selection of the target. Thus, the maximum operational range of the system is perhaps twice the maximum hover height, ie. 60 meters. Obviously at these short ranges the laser power requirements are low and well within cost and safety limitations. A 10 milliwatt CW system should conservatively yield a S/N of approximately 6db for daylight operations with a simple optical system employing a 50Å to 100Å optical bandpass filter. Initially it was felt that a sophisticated filter such as the Birefringent Optical Discriminator (BOD) would be required for daylight operations. However, since I have elected to limit the acquisition requirements, the BOD, which is derived from the Lyot filter, is no longer necessary.

Operationally, the pilot locates the target visually or via other external clues and with the PHD in its wide field of view mode the pilot views the target on the PHD display. When the target is within the proscribed bounds the pilot moves to the high magnification mode positioning the aircraft until the target intercept coincides with the display reticule. These are shown in figure 5.

With the aircraft in position and the hover height selected to keep the target image size smaller than the CCD sensor array, the pilot enables the PHD. In the enable mode the PHD digitally locates the keyed intersections and stores them in memory. Once so fixed, displacements and/or changes in image size and position are seen as aircraft deviations. Based upon these perceive deviations corrective signals can be either displayed to the pilot or fed directly into a flight control system. The sensitivity of the system is such that it can detect digital address changes of one element which translates as positional changes of 6 millimeters on the ground and a change in altitude of 10 centimeters. For practical purposes the vectorial rates of change are sensed and sampled over an appropriate time interval to provide the pilot corrective signals which can be meaningfully implemented. However, the basic system sensitivity permits anticipatory data inputs to permit precise hover where it is critical to a given mission.

The PHD is similar to a closed circuit TV except that the target image is translated to a set of asymmetric digital coordinates. In the pattern shown in figure 4 there are ten (10) keyed intersections with a precise spatial and geometric relationship. Thus, the system can be operated with a high probability of detection and a high false alarm rate (corresponding to low S/N) for each of the points and still result in an over all system whose detection probability is high with a very low false alarm rate. In essence the TV image is dually sensed. In one instance the raw video is combined to yield a conventional scene and in the other instance the raw video is filtered non destructively to perform feature extraction according to a prescribed set of instructions entered into the PHD memory. The incorporated program is flexible to the extent that it is capable of scaling in size as well as for initial x, y offsets, however once these factors are determined the act of enabling the system sets the specific data addresses and changes are sensed as deviations to be reported to the display or automatic flight controls.

2.3 FORMATION FLIGHT AID (FFA):

Several years ago helicopter rotor-tip lights were developed to provide visible cues for night time formation flight. [5] It was shown in a test program conducted at Fort Hood, TX that in conjunction with light panels placed strategically about the aircraft, such an operation was feasible. The apparent size and shape of the elliptical pattern described by the whirling rotor tip lights provided the pilot with both range cues and position cues. Since the rotor blades have a fixed and known diameter the size is directly relatable to range in a given optical system and the size of the minor diameter of the ellipse gives an indication of the pilots position relative to the plane of the rotor blades.

While the rotor tip light system seemed useful it was limited in application because of the variability in the human factors area. Contrary to the PHD, the target of interest possesses a high degree of symmetry and regularity. Thus the Formation Flight Aid System (FFAS) capitalizes on the pattern regularity to incorporate a simpler sensor format. Here again the sensor is a CCD, however, the moderate resolution of a series of 1x100 linear arrays are used in a orthogonal high and coarse resolution mode similar to the WOWS. In the FFAS a series of such linear arrays are used in the focal plane and no mechanical scanning is required. The format is shown in figure 6. The system optics are designed to insure that the image of the rotor tip circle or ellipse covers less than the full extent of the sensor. If the major axis of the ellipse gets larger than the image size appropriate to a 13 meter diameter rotor at a distance of 30 meters the FFAS provides warning to the pilot. Actually it is possible to set both maximum and minimum alarm limits to aid the pilot in maintaining his position in the formation.

In operation the pilot closes to the preselect range. The target image then falls on the sensor array and the system is switched from an acquired mode to a lock-on status. For the lock-on mode several things are caused to happen. First, the image of the target ellipse is centered in azmuth such that an equal number of arrays on the left and right of the sensor "see" the target. Thus as the lead aircraft varies in direction the FFAS sensor tracks the change and a rotary shaft encoder provides heading information to the pilot. Second, changes in range, as sensed by changes in diameter of the major axis, are compensated by a change in system optical magnification, In this manner the image size remains fixed and the range variations which are inversely proportional to the magnification are provided as pilotage data.

The FFAS is a night time device. It is not necessary for the entire ellipse to fall on the sensor. The location of several adjacent rows in the region about the extremes on the major axis are used to provide data to the system microcomputor which in turn calculates the minor axis dimension and displays the full ellipse.

3.0 CONCLUSIONS:

The concepts and systems which have been described perform a wide variety of tasks, however, throughout there exist a physical commonality which is the kernel of this presentation. In each instance the task to be performed has been matched to conform to the characteristics of an array or matrix of quantized information elements. The tasks have been reduced to a pattern which is digitally described and processed in a multi microprocessor based system. In this manner the microprocessor, LSI hybrid semiconductors, and the CCD, all of which are products of a single technology, have been exploited to perform necessary tasks for helicopter control and/or guidance. While the tasks described are representative, the future application of this methodology and technology can go far to reduce the pilot work load while simultaneously improving the safety and utility of the helicopter.

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(3) Kleider, A. U. S. Patent #4068124, Jan 1978.

(4) Hirschberg, I.; Balopole, H. "CCD Modular Camera Program" EASCON-76 Sep 1976.

(5) Wilkin, M. "Test Report on Modified Electronic Binocular and Formation Marking Light Kit for Night Formation Flight of Army Helicopters" Tech Report ECOM 4067 Jan 1973.

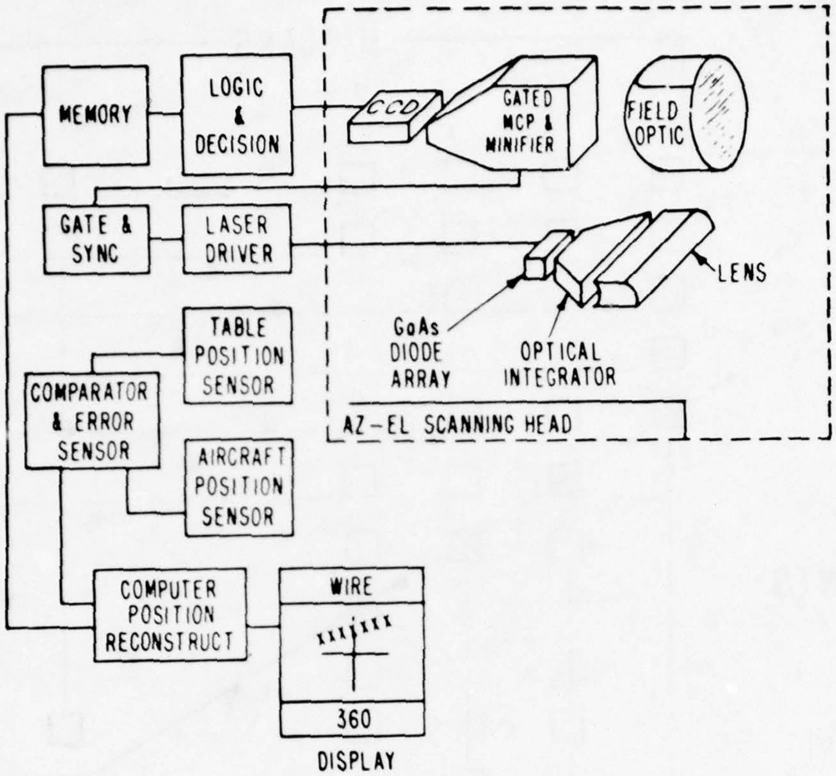

Fig.1 Wire obstacle warning system schematic

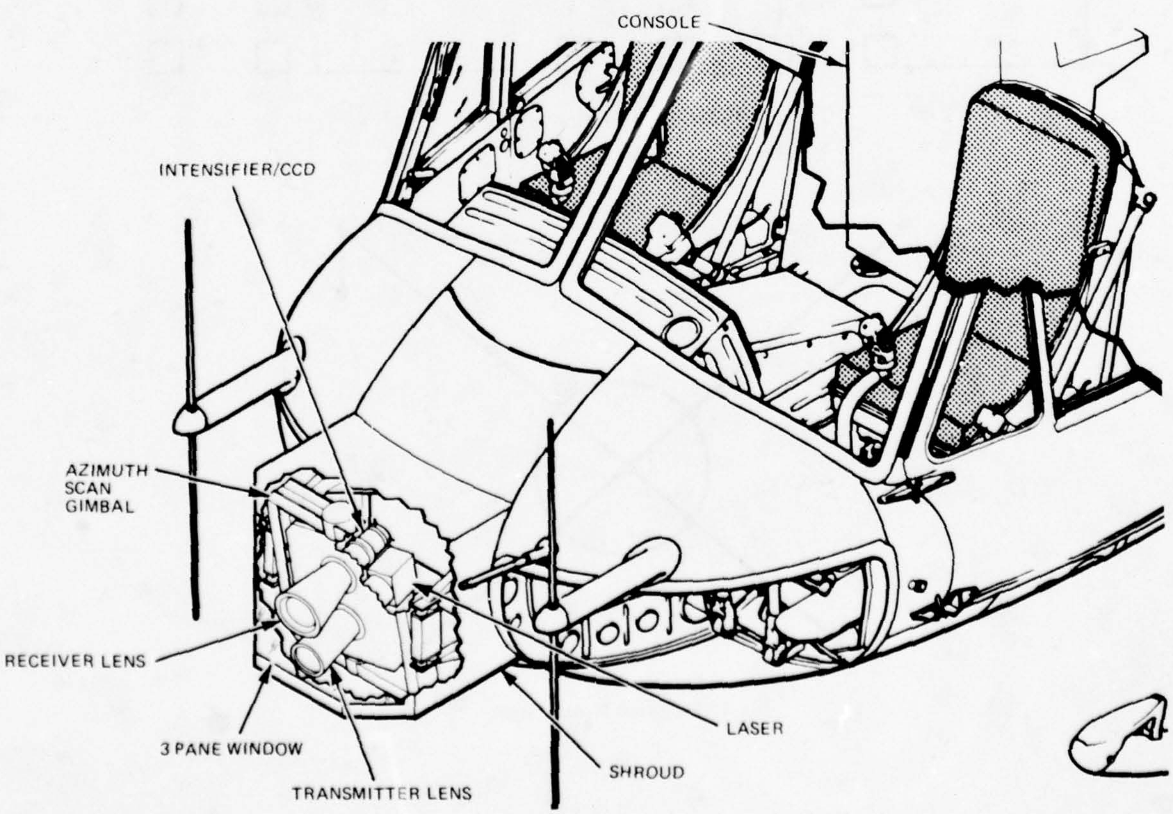

Fig.2 Wire obstacle warning system (WOWS)

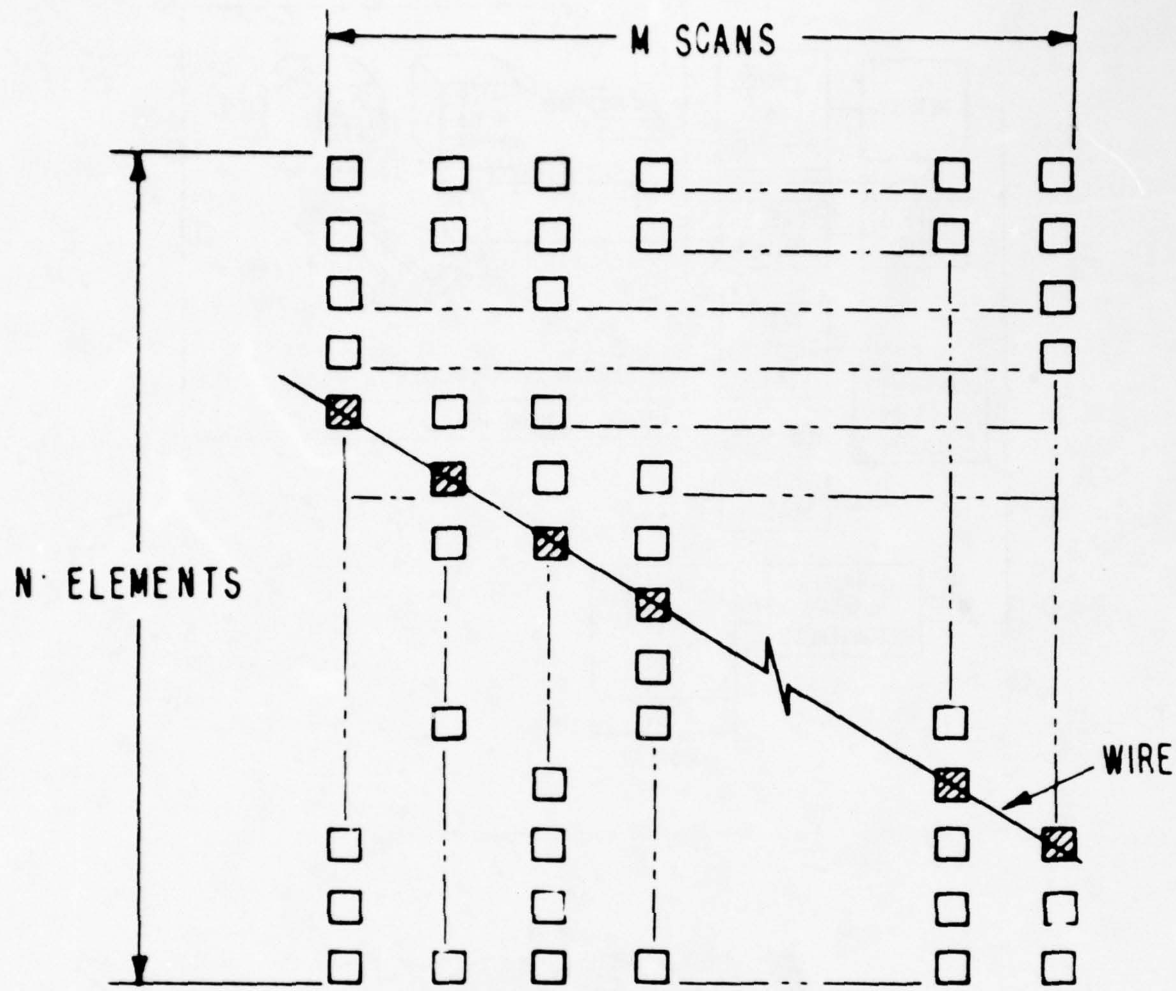

Fig.3 WOWS detection matrix

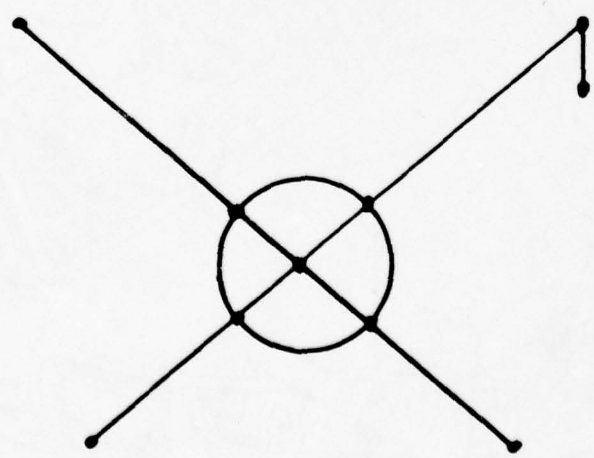

Fig.4 Precision Hover target

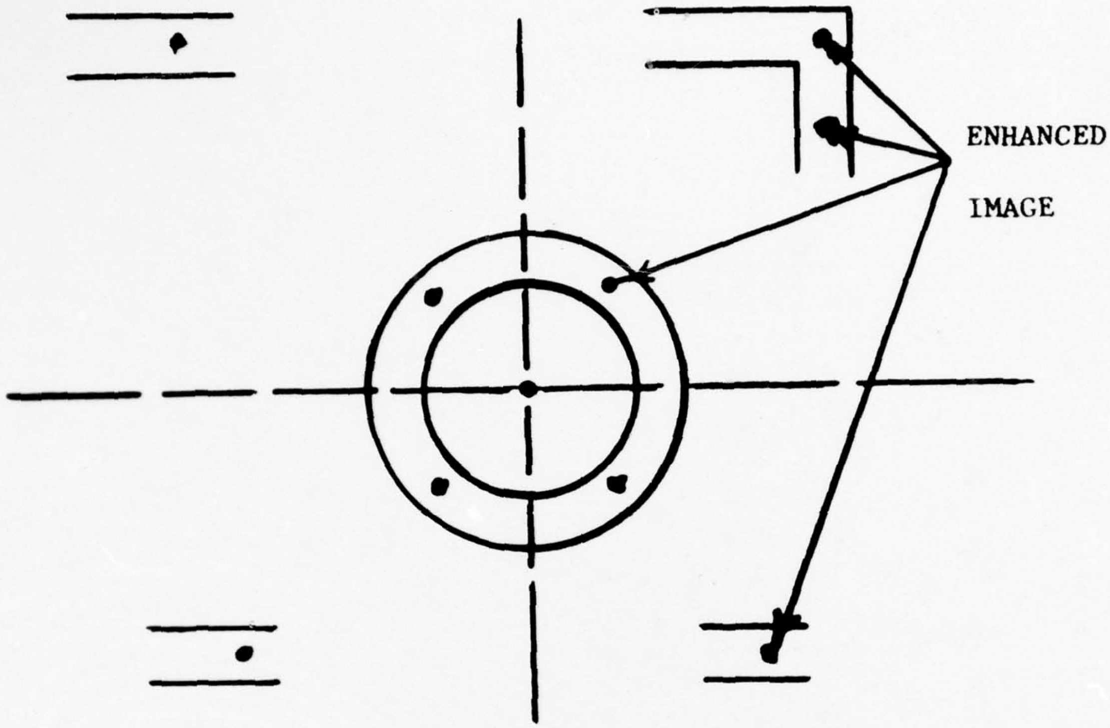

Fig.5 Aligned return

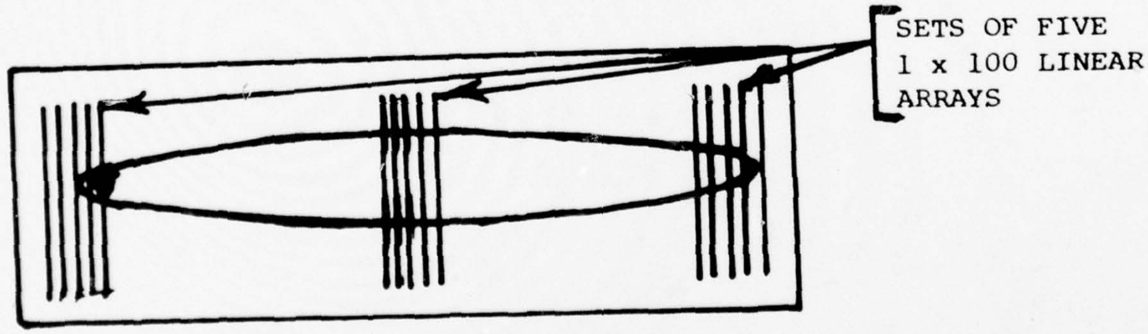

Fig.6 Focal plane array for formation flight aid

HETERODYNING CO_2 LASER RADAR FOR AIRBORNE APPLICATIONS

by
R. L. Del Boca
US Army Avionics Research and Development Activity
Fort Monmouth, New Jersey 07703

R. J. Mongeon
United Technologies Research Center
East Hartford, Connecticut 06108

SUMMARY

Analysis of tactical scenarios indicates a number of radar requirements that are optimally solvable with low power 10.6 micron laser systems. Specifically the high detectivity realizable with optical heterodyning receivers coupled with the efficiency and versatility of 10.6 micron sources makes CO_2 heterodyne laser radars unique candidates for airborne applications. This paper addresses the characteristics of optical heterodyning radars and considers the applications pertinent to pulsed and CW sources.

Exploratory development efforts by the US Army Avionics Research and Development Activity in conjunction with United Technologies Research Center have been directed at advancement of the state-of-the-art in airborne CO_2 heterodyne laser radar systems. This paper will discuss the design considerations, hardware configuration, and test results of flyable breadboard models which have demonstrated the feasibility of employing CO_2 scanning laser systems for wire detection, precision hover, Doppler navigation, and terrain following.

LIST OF SYMBOLS AND ABBREVIATIONS

- S_{min} - Minimum receiver detectable signal for a given bandwidth
- h - Planck's constant = 6.63×10^{-34} Joule sec.
- f - Frequency
- B - Bandwidth
- n - Quantum efficiency
- λ - Wavelength
- Po - Peak transmit signal
- S/N - Signal to noise (power) ratio
- τ - Pulsewidth
- P_{av} - Average power
- PRF - Pulse repetition frequency
- P_{min} - Minimum receiver detectable signal for a given PRF
- T_{sys} - System transmission
- T_{atm} - Atmospheric transmission
- R - Range
- $D_{T,r}$ - Diameter of transmit, receive aperture
- σ - Cross section
- D_w - Diameter of wire
- ρ - Target reflectivity
- g - Heterodyne efficiency
- LN_2 - Liquid Nitrogen

1. INTRODUCTION

1.1 Operational Scenario

In order to achieve mission effectiveness in the present threat environment, Army helicopter operations have focused on terrain flying. Specifically, terrain flying is the tactic of employing aircraft in such a manner as to utilize the terrain, vegetation, and man-made objects to enhance survivability by degrading the enemy's ability to visually, optically, or electronically detect or locate the aircraft. This flight regime is divided in three categories as sketched in Fig. 1. Nap-of-the-Earth (NOE) flight (A) is conducted as close to the earth's surface as vegetation or obstacles will permit with variations in aircraft airspeed and altitude. Contour flight (B) conforms generally to the contours of the earth, at low altitude, and is also accomplished by varying airspeed and altitude. Low level flight (C) occurs at a given altitude, generally along a straight line, and with constant airspeed.

1.2 Problem Definition

In order to enhance operational effectiveness throughout the flight regime described above, high accuracy sensors are necessary since warning times can be small, and room for maneuver limited. Additionally, the number of cues are far less than those available at higher altitudes. Rivers, road crossings, and hill tops can often not be seen simultaneously in a large aerial view, but rather are crossed (or missed) serially. When obstacles are included as hazards to these low altitude flights, the overall sensor requirement becomes that of achieving high resolution over modest line of sight ranges. As will be demonstrated shortly this is the expected performance capability for optical radars.

2. OPTICAL RADAR

Some of the overall characteristics of optical radars are noted in Fig. 2. In this paper the term optical radar will generally imply radar wavelengths from the visible through the infrared portion of the spectrum. As contrasted with microwaves, these shorter optical wavelengths permit higher spatial resolution for a given antenna aperture, better Doppler sensitivity, and increased temporal resolution. In order to select an appropriate optical frequency, however, one must consider the factors pertinent to total system performance (Fig. 2).

2.1 Advantages of CO_2 Optical Radar

While a detailed analysis of each criterion over the entire spectrum of optical wavelengths is beyond the scope of this paper, a brief summary is presented to indicate the advantages inherent in CO_2 heterodyne laser radars. The characteristics of low power 10.6 micron heterodyning radar are shown in Fig. 3 with the conclusion to be drawn that CO_2 provides the optimum wavelength optical radar with respect to source, receiver, and atmospheric considerations.

2.1.1 Source

Most importantly, 10.6 micron lasers are versatile transmitters permitting modulation techniques familiar from microwave systems to be employed. The ability to operate CW, FM-CW, pulsewidth and/or pulse rate variable enable the system designer to choose an efficient source with an optimum transmitter format matched to a given airborne scenario.

2.1.2 Receiver

Present 10 micron detector technology has made available durable, high quantum efficiency, wide bandwidth photovoltaic detectors for incorporation into optical heterodyne receivers. As a consequence of heterodyning, receiver sensitivity is not limited by electronic or detector noise, or by background photons, with the minimum detectable power being described by

$$S_{min} = \frac{hfB}{n} = \frac{hcB}{n\lambda} \sim 4 \times 10^{-20} W \qquad \begin{array}{l} B = 1 \text{ Hz} \\ n = .5 \end{array}$$

for a 1 Hz bandwidth and a detector quantum efficiency of 50%. It is this near theoretical receiver sensitivity, realizable in practice, that mades many low power 10 micron missions possible.

In contrasting 10 micron and shorter wavelength radars it is appropriate, therefore, to consider the theoretical and practical aspects associated with achieving efficient heterodyne laser radars. First, receiver sensitivity improves with wavelenght because the photons per unit power are proportional to wavelength. Secondly, efficient heterodyning infers a spatial coherence at least equal to the receiving aperture of the radar. Air turbulence and resulting index fluctuations partially destroy the transverse coherence of the reflected signal. The longer wavelengths are less sensitive to this atmospheric aberration, as shown, by the following representation. Consider a one kilometer optically heterodyning radar operating in mild turbulence; it would be restricted to a .1 meter aperture operating at .63 microns whereas a 10 micron radar could utilize up to a 3 meter antenna for the same range and turbulence. Increased range and/or turbulence decreases the coherent aperture for both wavelengths but the ratio improvement for the long wavelengths remains approximately $(\lambda_L/\lambda_S)^{6/5}$. And, finally, from a practical viewpoint the mechanical precision, alignment, and stability of the optically heterodyning receiver are easier to realize at the longer wavelength, a particularly important consideration for airborne optically heterodyning radars operating in intense vibrational environments.

2.1.3 Atmospherics

The above paragraphs outlined the favorable 10 micron source and receiver characteristics and it now remains to consider the final major factor in determining optical radar feasibility, namely atmospheric scatter and/or absorption. For 10.6 micron radars, molecular absorption caused by either CO_2 or H_2O is a consideration, but does not seriously effect short range considerations. For example, a 30°C temperature and a 90 percent relative humidity results in 3 dB/km (one-way) attenuation. The limiting atmospheric consideration of all optical radars is fog, with longer wavelengths once again proving to be superior. A dense fog can prove totally limiting, with 100 dB/km attenuations, but moderate fogs having visibilities of approximately 450 meters (13 mg/m^3 liquid water content) result in only 10dB/km attenuation at 10.6 microns. It is important to recognize that while sensor performance can be degraded by atmospheric conditions, to the point of totally defeating long range all weather optical radars, there are many short range (< 10 km) tactical scenarios that can be satisfactorily addressed. As will be shown shortly this is primarily a result of realizing a sensor with a large clear weather margin.

2.2 CO_2 Sensor Configurations

For the terrain flying scenario described earlier, a number of specific sensor tasks have been identified as optimally solvable with coherent CO_2 laser techniques. Figure 4 provides a partial list which will serve as the basis for discussion in this paper. As shown, the transmitter format is selected depending on the task. For range resolution of wires, particularly those close to a background, a short pulse system is preferred. A high frequency clock, synchronous with the transmit pulse, provides accurate time of flight range measurement. In addition, a short pulse system (1) simplifies the design of the Doppler tracking receiver which tunes the narrowband intermediate frequency (IF) amplifier section, and (2) provides temporal isolation of transmitter feedthrough in common (transmit/receiver) aperture systems.

Since optical wavelengths permit order of magnitude improvements over microwave Doppler sensitivities, a CW CO_2 system can yield the high resolution three axis Doppler velocity information necessary for accurate hover, and NOE navigation. A system design employing an acousto-optic cell as a frequency translator provides a frequency offset local oscillator beam while reducing the complexity of the transceiver. In order to achieve isolation from transmitter feedthrough a dual (transmit/receive) aperture or aperture shared design would be incorporated.

2.3 Heterodyne System Design Considerations

Understanding the fundamental heterodyne sensitivities discussed previously, the system designer can now perform analyses pertinent to performance margin for the set of sensor tasks outlined above. For a pulsed heterodyne system, the power ratio of peak transmit signal (Po) to minimum receiver detectable signal S_{min} (=NOISE) is given by $(S/N)_{power} = Po/S_{min} = Po/hfB/n$ where B is the bandwidth of the matched filter $1/\tau$ (τ = pulsewidth). Relating peak power to average power by $P_{av} = Po \times \tau \times PRF = Po \times 1/B \times PRF$ where PRF is the pulse repetition frequency yields

$$(S/N) = \frac{P_{av}}{\frac{hf}{n}(PRF)} = \frac{P_{av}}{P_{min}}; \qquad P_{min} = \frac{hf(PRF)}{n}$$

For a 10 watt average power system, a PRF of 1 Hz, and an ideal (100 percent) quantum efficiency detector (n = 1) detector; the value of this ratio represents 207 dB of gain (Fig. 5). If we assume the target intercepts the entire beam, then the received detector signal (P_r) can be derived from P_{av}, and the received signal to noise (power) ratio can be expressed in terms of P_{av} by

$$(S/N)_R = \frac{Pr}{\frac{hf(PRF)}{n}} = \frac{P_{av}\,T_{sys}}{\frac{hf}{n}(PRF)}\,T_{atm}\,g\rho\,\left(\frac{D^2}{4R^2}\right);$$

D = diameter of receiver aperture
R = range to target
ρ = target reflectivity
T_{sys} = system transmission
T_{atm} = atmospheric transmission
g = heterodyne efficiency

Expressed in decibels this can be written

$$(S/N)_{dB} = 10\log\left(\frac{P_{av}}{hf}\right) + 10\log(T_{sys}\,g\rho n) + 10\log\left(\frac{D^2}{4}\right) + 10\left(\log T_{atm} + \log\frac{1}{R^2} + \log\frac{1}{PRF}\right)$$

For an assumed system having the following values

P_{av} = 10 watts
n = .5
T_{sys} = .1
ρ = .1
g = .1
D = 10 cm

substitution yields,

$$(S/N)_{dB} = 207\text{ dB} - 33\text{ dB} - 26\text{ dB} + 10(\log T_{atm} - 2\log R - \log PRF)$$

where the log formalism has been used on the right hand bracket terms to demonstrate that losses will be associated with $T_{atm} < 1$, range, and PRF. One additional factor must be entered to address the probability of false alarm. This term can be considered a signal threshold to noise value which provides a level of noise immunity, thus decreasing the system false alarm rate. The remaining S/N after all of the above factors are taken into account is referred to as the $(S/N)_{dB}$ margin; written,

$$(S/N)_{dB_{MARGIN}} = (S/N)_{dB} - (S/N)_{dB_{THRESH}}$$

Assigning a value of $(S/N)_{dB_{THRESH}} = 15$ dB as representative for this analysis, yields,

$$(S/N)_{dB_{MARGIN}} = 133 - 20 \log R + 10 (\log T_{atm} - \log PRF)$$

The semi-log plot of fig. 5 is intended to graphically portray the physical and analytical considerations examined above. The crosshatched area represents the $(S/N)_{dB}$ margin available for weather and/or PRF considerations as a function of range. For a PRF of 40 kHz, the remaining weather margin as a function of range is shown as doubly crosshatched.

3. CO_2 WIRE DETECTION SENSOR

As mentioned earlier a prime application for CO_2 pulsed heterodyne airborne laser radars is the area of wire detection. Safety statistics to date indicate that the US Army and the Civilian Aviation Community are each recording over two aircraft wire strikes per month.

In an analysis formulated according to the previous recipe, a low average power (2.0 watt) high PRF (40 kHz) system is analyzed as a wire detection sensor (Fig. 6). The various system and target characteristics have been folded into the calculations with the resultant atmosphere/weather margin equal to 16 dB for normal incidence and 5 dB for a 45° angle of incidence at a range of .5 km.

Under an exploratory development program a Laser Obstacle/Terrain Avoidance Warning System (LOTAWS) was developed and tested by the US Army Avionics R&D Activity and United Technologies Research Center. The results obtained in ground and flight testing are consistent with the previous analysis. The heterodyne CO_2 laser radar utilized a 2.0 watt average power, single frequency, pulsed laser transmitter and a CW local oscillator which was slaved to the transmitter. A high PRF (40 kHz) system was dictated by the necessity to scan a wide field of view for the wire detection task. The system was installed in a CH-53A helicopter and demonstrated the detection of .32 cm field wire at typical ranges of 500 meters and power lines at ranges to 1.6 km.

3.1. Major System Description

In its most basic form the LOTAWS mission consists of the identification of wire threats and the incorporation of this information into a display format for presentation to a pilot. Figure 7 presents the major LOTAWS subsystems and their functional interrelation. The pulsed transmitter, delivering a 340 nsec pulse at a PRF of 40 kHz and an average power of 2 W, is coupled via the germanium duplexer to the common transmit/receive telescope and scanner. The electronically programmable wedge scanner is capable of directing the 250 μrad beam anywhere in a conical field of 30 deg, with typical scan patterns consisting of spiral, circular, or linear scans. The return radiation is mixed for heterodyning with the CW local oscillator at the duplexer and detected by a LN_2-cooled HgCdTe photovoltaic diode. The pulsed rf current is amplified by a wide bandwidth preamp prior to initial processing by the narrow bandwidth Doppler tracking receiver. This design enables amplification to occur at a bandwidth of 3 MHz while aircraft velocity induced Doppler shifts as great as 20 MHz are tracked. After video detection of the pulse, the signal processor determines the range to the target return and discriminates wire returns from other background signals. For the initial LOTAWS tests and as an aid for flight testing and evaluation, the LOTAWS returns were displayed on a storage monitor with the scanner position controlling the azimuth and elevation coordinates. To further aid the visualization, the storage display was scan converted into TV format and electrically mixed with the view detected from a nose mounted TV camera. In this manner, a visual scene with low natural resolution is augmented with high resolution wire information.

As shown in Fig. 8, the LOTAWS system was divided into three functional groups for installation into a CH-53A helicopter. The optical transceiver consisting of lasers, telescope, optics, and detectors was mounted behind the copilot's compartment. The scanner was mounted in the nose. The optical path between the transceiver and scanner covered a total distance of 3.3 m and utilized three folding mirrors. The signal processor and display were stationed aft at the observer/data recording station.

3.2. Transmitter

The LOTAWS transmitter is depicted in Fig. 9. The CO_2 gain tube is a conventional low pressure, sealed-off, CW pumped, Brewster-window configuration. Passive Q switching is realized with the SF_6 cell, and the pulse width is determined primarily by the dynamics of the CO_2 medium. The PRF is determined by the available gain in excess of loss with the latter largely determined by the unbleached loss of the SF_6. Since operating frequency determines the available gain for a selected SF_6 pressure, both PRF and optical frequency can be controlled by laser cavity length. A grating is included in the cavity to prevent CW oscillation at lines where the SF_6 has low or no loss and to select the P_{20} operating line. To realize a large bandwidth length control servo, the grating was mounted onto a specially designed, rugged, tilt free, PZT drive with a natural resonant frequency greater than 2 kHz.

A resonator with minimum angular sensitivity and high transverse mode discrimination was used in minimizing the vibrational susceptability of the transmitter. The stable operation, both in frequency and PRF, required that the laser operate single mode. The resonator finally used was the "unstable" resonator depicted in Fig 9b. This low magnification, transmission coupled configuration was realized by a combination of a "flat" LITTROW grating, a flat output coupler, and the weak negative lens resulting from the temperature gradient in the CO_2 gain medium. Without the wave guiding effects of the 5.5 mm gain tube, the losses associated with the magnification would otherwise be too great for this relatively low gain application. With the waveguide enhancement, this configuration proved competitive in efficiency with stable designs but provided considerable improvement in transverse mode discrimination in addition to increased tolerance to angular alignment. A secondary characteristic of the unstable configuration of some importance was the large intracavity mode size, which reduced the power density incident on the low thermal capacity replica grating.

To provide an appropriate housing for the laser resonator and intracavity components, a cylindrical invar structure with excellent stiffness-to-mass characteristics both in deflection and torsion, was configured to force any acoustic resonances to be higher than 300 Hz (Fig. 9c).

3.3. Scanner

The optimum scan pattern and field of view required for a specific obstacle warning scenario is determined in part by the aircraft's operational characteristics including craft velocity, maneuverability, pilot warning time, and the radar constraints including spot size and scan rate induced phase lag. For the system deployed in the helicopter, the maximum scan rate (for a 50 percent pulse-to-pulse overlap, a 40 kHz PRF and a 250 μrad divergence) is 5 rad/sec. Since the program emphasis was exploratory, a versatile scanning capability which would enable a flexible flight evaluation was selected. To enable the electronic selection of a variety of scan patterns and fields, a pair of independently driven rotating optical wedges were used.

To aid in the visualization of the scanning pattern, a pair of visible transmitting wedges were assembled on independent motor mounts and a HeNe laser and Polaroid camera were used to capture some of the simple patterns. A mechanical chopper was utilized to provide a beam modulation of 2 kHz so as to provide an evaluation of the instantaneous scan velocity. Figure 10 portrays a few of the scan patterns that are realizable. In general, counter prism rotation produces a line scan for equal prism speeds and rosette-like patterns for slow relative speed differences, while common prism rotation generates spiral scans for low relative speed differences and slewing circular scans for large relative speed differences. For LOTAWS, 10 cm diameter germanium wedges were independently driven by shaftless DC torque motors, thereby realizing a compact gearless drive assembly with high torque-to-inertia ratios. For the wedges used, the full angular field was 30 deg and, depending on the scan complexity, a full field could be scanned in times ranging from 0.1 to 10 sec. As initially installed, scan generation was realized through velocity servos controlling the individual wedge rotation speeds. Present plans include incorporating position control of the wedges so as to enable control of the instantaneous scan rate, dynamic control of the scan field, and the addition of a track-while-scan mode.

3.4 Vibration Considerations

An overriding consideration in designing the LOTAWS for helicopter operation was the vibrationally intense environment, stability of heterodyning alignment, and the laser frequency which had to be maintained during flight. To minimize the effects due to flexure and vibration, the optical transceiver components were designed for stiffness, and high stiffness-to-mass ratio so that any resonant frequencies would be greater than 300 Hz, beyond the range of significant aircraft vibrational energies. The vibrational susceptability of the transmit and local oscillator lasers were further reduced by incorporating wide bandwidth laser length control servos with closed loop bandwidths greater than 300 Hz. Finally, to minimize impulse noise due to shock excitation of the high frequency transceiver resonances, visco-elastic damping material was utilized at the interface of the honeycomb optical table and the rigid welded support rack.

3.5 Test Data and Results

During the LOTAWS flight tests, data were obtained which showed the effect of vibration isolation and electronic compensation in maintaining transmitter and local oscillator-optical frequency. Figure 11 shows the frequency broadening due to vibrations during hover within ground effect. As previously discussed, a transmitter optical frequency shift results in a change in PRF, which, for the flight data, is approximately 8 Hz in the 40 kHz PRF. The maximum relative optical frequency shift between the local oscillator and transmitter is approximately 100 kHz, which is well within the 3 MHz IF bandwidth.

The main target of interest for the exploratory tests of LOTAWS has been .32 cm Army field wire. In considering a wire target in the radar model, the signal or carrier-to-noise ratio is expected to follow a $1/R^3$ dependence (Fig. 12), since the beam is intercepted completely in one dimension and only partially in the other dimension. The static data have been plotted for normal incidence, but this range dependence should be independent of angle. The maximum target range indicated represents a signal-to-noise ratio of approximately 14 dB for the .32 cm wire at 1200 m.

The ability to successfully detect these small diameter target wires in flight was a major result of the initial flight test program. The results of static tests performed in the aircraft, but on the ground, and the data from the flight program indicate that the LOTAWS system performance did not deteriorate in the airborne environment. This outcome is consistent with the very small optical frequency shifts measured in flight. A plot of wire returns as a function of angle of incidence for .32 cm field wire at 400 m (Fig. 13) indicates the data analyzed. In addition to the agreement between the data sets, it is worth noting that the rapid fall off in carrier-to-noise ratio occurs within the first 15 deg from normal incidence and that the ratio is reasonably constant from 15 deg to 40 deg from normal. This angular dependence is the result of both the specular nature of the wire scattering, resulting in the rapid decrease at small angles, and the diffuse scattering, resulting in reasonably constant values over a broad range of increasing scattering angles.

In order to incorporate the detected wire information into a display format which might be meaningful to a pilot, the location of wire detections was combined with the video display from a camera mounted on the nose of the aircraft. When detections which satisfied the wire identification logic occurred in the scanned field of view, a bright dot would be superimposed at those points on the TV video scene. This was accomplished by viewing a storage display with a vidicon and video mixing this output with that of the forward looking TV camera.

The following sequence of figures shows the scene (Fig. 14), the employed scanning format, and the sequential wire detections which occurred for transmission lines at a range of 1.6 km (Fig. 15). If a spiral scan is employed, a series of wire detections would appear as a horizontal linear array of dots. The display storage time can be controlled such that detections over a certain number of TV frames would indicate a horizontal wire. Photos 1 through 6 of Fig. 15 indicate sequential detection of the wires for the arcs of the spiral scan using the assumed lines. Although the storage aspect of the display was not employed for this video scene, when viewed at the TV frame rate of 30 frames/sec, the linear aspect of the detections is quite pronounced.

4. CO_2 HOVER SENSOR

In order to experimentally assess the potential of employing a CO_2 system as a multifunction sensor applicable to the tasks previously depicted in Fig. 4, we undertook the Multifunction LOTAWS effort aimed at design fabrication and testing of the necessary exploratory CO_2 airborne sensor configurations. For investigation of Doppler navigation and precision hover a CW frequency offset homodyne design was adopted. In the initial exploratory development Multifunction LOTAWS effort, the hover aspect was selected as representative of sensor performance, since the total navigation problem required considerations beyond the scope of the tests.

The main components of the CO_2 transceiver consisted of a one watt average power CW transmitter, an acousto-optic cell for frequency translating a portion of the transmit beam to provide a frequency offset local oscillator reference, and a photo voltaic HgCdTe detector with a quantum efficiency of 50 percent. The transmit/receive apertures ($\sim$ 1 cm) "aperture shared" a 7.5 cm clear aperture germanium wedge scanner. The beam was scanned conically (12° half angle) at a 100 Hz frequency, about a boresight at maximum depression, and the three axis velocity was resolved utilizing F.M. discrimination and phase sensitive detection. The CO_2 sensor derived velocity information was integrated to obtain position, and both the velocity and position information were displayed symbolically to a subject pilot via a CRT. The symbology was developed by the US Army Avionics R&D Activity and included additional information consisting of filtered attitude derived from the vertical gyro and altitude derived from the radar altimeter (Fig. 16). With this sensor and symbology, pilot-in-the-loop hover at a 50 meter altitude with approximately 4 meters radial excursion over a period of several minutes was demonstrated. Doppler velocities of approximately 1 cm/sec have been resolved with this configuration. The above test established the feasibility of Doppler laser radar to serve as a sensitive Doppler navigator, and high resolution day/night hover sensor.

5. CO_2 TERRAIN FOLLOWING SENSOR

Further efforts in pulsed heterodyne CO_2 airborne laser technology are presently underway in the Multifunction LOTAWS effort to demonstrate the terrain following sensor capabilities of CO_2 scanning laser systems via pilot in-the-loop terrain following tests. Earlier work with the LOTAWS system previously described, yielded data on the signal to noise (power) ratio for signal returns from grass as a function of range, for various angles of incidence. Figure 17 portrays the result that even for relatively grazing incidence (75.5°) at approximately 1200 meter a S/N of 25 db is achieved.

For the new multifunction configuration a short pulse 30 nsec variable PRF (20-60 kHz) transmitter should provide range resolution of 5 meters for terrain and also permit the detection of wires close to a background. The exploratory terrain following format will scan a vertical field of $\pm$ 25° about a boresight which can be varied from approximately +15° (elevation) to -30° (depression) from the longitudinal aircraft axis. The frame time can be varied from one-half to one second per scan and the sensor outputs of elevation angle versus range will be interfaced with an SKC-2000 airborne computer for processing in a terrain following algorithm. The sensor information is thus converted to climb/descend "command" information for given aircraft velocities and load factors.

The symbology displayed to a pilot on the CRT is shown in Fig. 18. This cruise mode display presents the climb/descend command via the triangular symbol at left edge of the screen, with additional flight cues and a forward looking image completing the CRT information.

6. CONCLUSION

The concepts and techniques associated with 10 micron optical radar have matured to the level where feasibility demonstrations of flight hardened breadboards are occurring. Specifically low power ($\leq$ 10 watt) 10.6 micron sources can be configured to perform a number of short range ($\leq$ 10 km) tactical and operational scenarios of interest to low flying aircraft (Fig. 19). Flight tests have confirmed near theoretical performance for a number of applications including obstacle (wire) avoidance, terrain following, doppler navigation and hover. The versatility of the sources argue strongly for an expanding range of applications, particularly in the areas of target recognition and identification.

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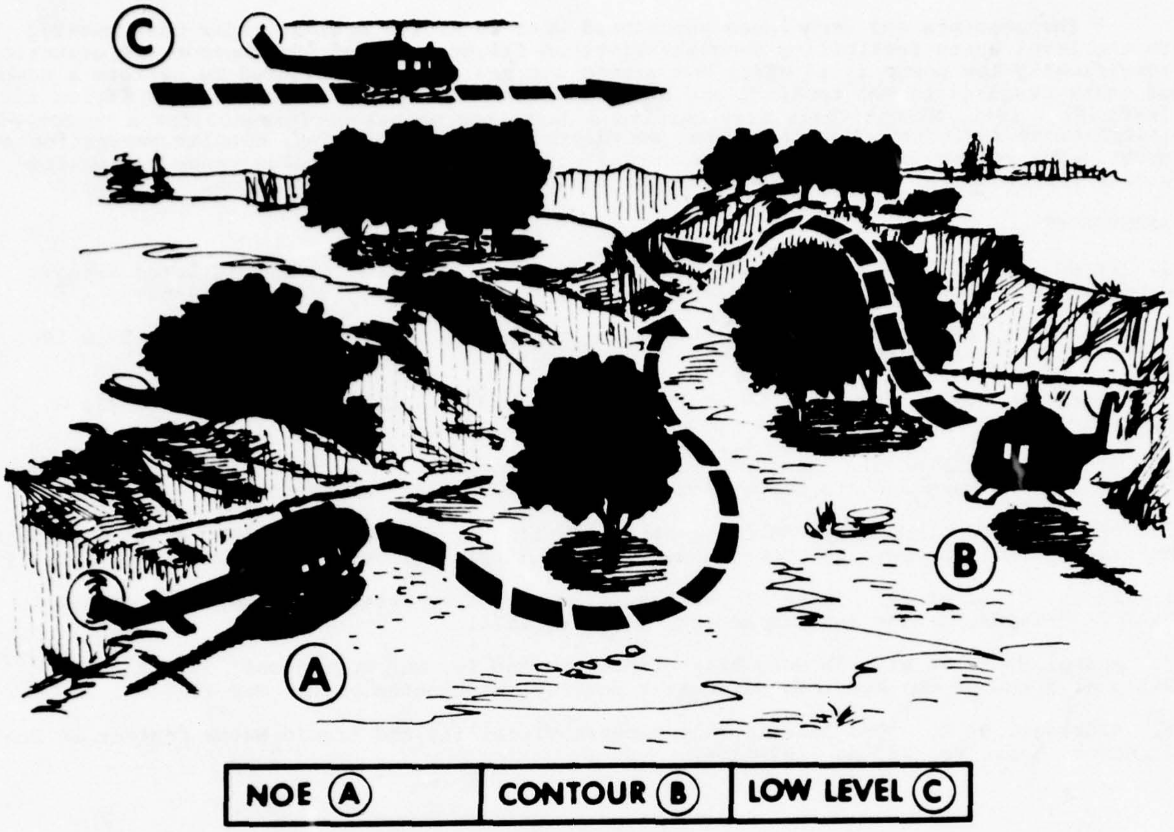

Fig.1 "Terrain flying profiles"

- PROMISE OF SHORT WAVELENGTHS
  - SPATIAL RESOLUTION $\theta_A \approx 2\,(\lambda/D_A)$
  - DOPPLER SENSITIVITY $\Delta f \approx 2V_T/\lambda$
  - TEMPORAL RESOLUTION - PICOSECOND PULSEWIDTHS
- NECESSARY CRITERIA
  - SCENARIO IDENTIFICATION
  - SOURCE
  - RECEIVER
  - SCANNER
  - ATMOSPHERE

Fig.2 Characteristics of optical radar

- SOURCE ($\leq$ 10 W AVERAGE POWER)
  - WIDE SELECTION OF TRANSMITTER FORMATS
  - EFFICIENT ($\approx$10%)
  - APPROACHES THEORETICAL PERFORMANCE (DIFFRACTION LIMITED, SINGLE FREQUENCY)
- RECEIVER
  - HIGH QUANTUM EFFICIENCY, HIGH BANDWIDTH DETECTOR
  - HETERODYNING ALIGNMENT LESS RESTRICTIVE THAN SHORTER WAVELENGTH SYSTEMS
  - HETERODYNING SENSITIVITY: $S_{min} \approx (hfB/n) \approx 4 \times 10^{-20}\ W \Big|_{n=0.5}^{B=1\ Hz}$
- ATMOSPHERE
  - CO_2 AND H_2O (HAZE) ABSORPTION: (30°C, 90% R.H.) $\Longrightarrow$ 3dB/km
  - FOG ABSORPTION AND SCATTERING: (400 m, vis., 13 mg/m^3 WATER CONTENT) $\Longrightarrow$ 10 dB/km
  - WAVEFRONT DISTORTON - COHERENT APERTURES ON THE ORDER OF 1 m

Fig.3 Characteristics of CO_2 optical radar

APPLICATION	CHARACTERISTICS/ADVANTAGES
PULSED HETERODYNE	
WIRE/OBSTACLE DETECTION	PULSED TRANSMITTER/CW LOCAL OSCILLATOR
TERRAIN FOLLOWING	RESOLUTION OF WIRES CLOSE TO BACKGROUND
RANGE FINDING	TIME-OF-FLIGHT RANGE MEASUREMENT
	SIMPLIFIED DOPPLER TRACKING RECEIVER
	TEMPORAL ISOLATION OF TRANSMITTER FEEDTHROUGH
FREQUENCY OFFSET HOMODYNE	
NAVIGATION	CW TRANSMITTER/A-O OFFSET LOCAL OSCILLATOR
PRECISION HOVER	MAXIMUM DOPPLER SENSITIVITY
	SIMPLIFIED TRANSCEIVER
	SPATIAL ISOLATION OF TRANSMITTER FEEDTHROUGH

Fig.4 Sensor tasks

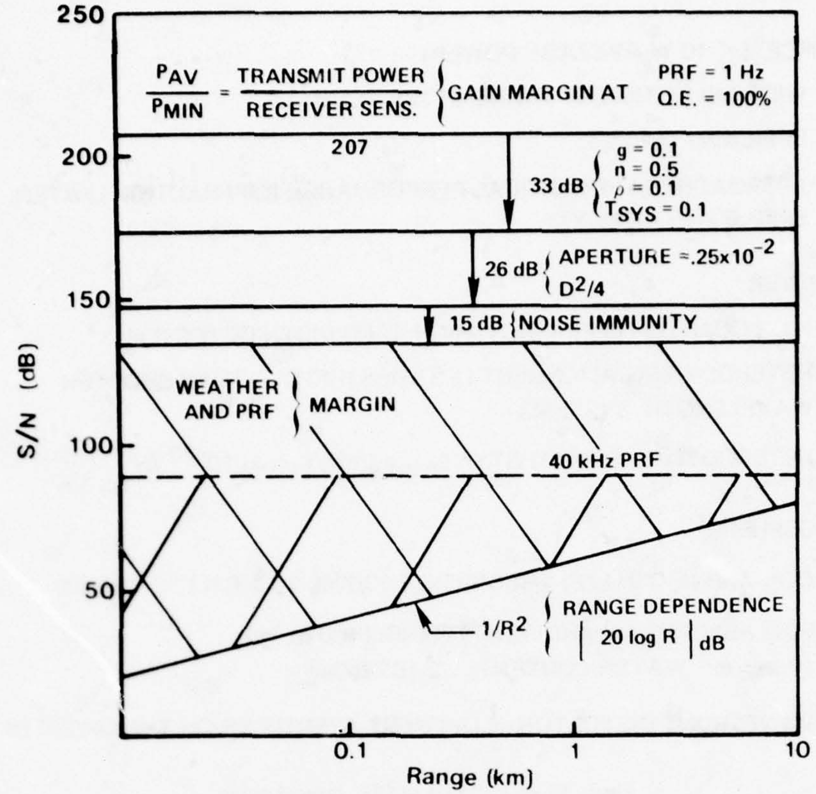

Fig.5 System design considerations

- **TRANSMIT TO THRESHOLD MARGIN**

  P_{av}=2W
  PRF=40 kHz
  Q.E.=50%
  $(S/N)_{thres}$=15 dB

  $$\left[\frac{P_{av}}{P_{min}}\right]_{dB} - \left[\frac{S}{N}\right]_{thres}$$ ➧ GAIN MARGIN 136 dB

- **TARGET EFFECTS** — LOSSES dB

  RELATIVE CROSS-SECTION
  D_W=0.32 cm
  D_T=10 cm
  $$\sigma_w = \left[\frac{2D_w}{\pi}\right]\left[\frac{D_T}{R\lambda}\right]$$ 14

  REFLECTIVITY $\rho_{normal} = 0.243$; $(\rho_{45^o} = 0.022)$ 6 (17)

- **RECEIVER**

  APERTURE $D_r^2/4R^2$ 80
  D_r = 10 cm
  HETERODYNE EFF=0.1 10
  OPTICAL TRANS=0.1 10

- **TOTAL LOSSES** 120 (131)

- **ATMOSPHERE/WEATHER MARGIN** 16 (5)

Fig.6 CO_2 sensor wire detection analysis

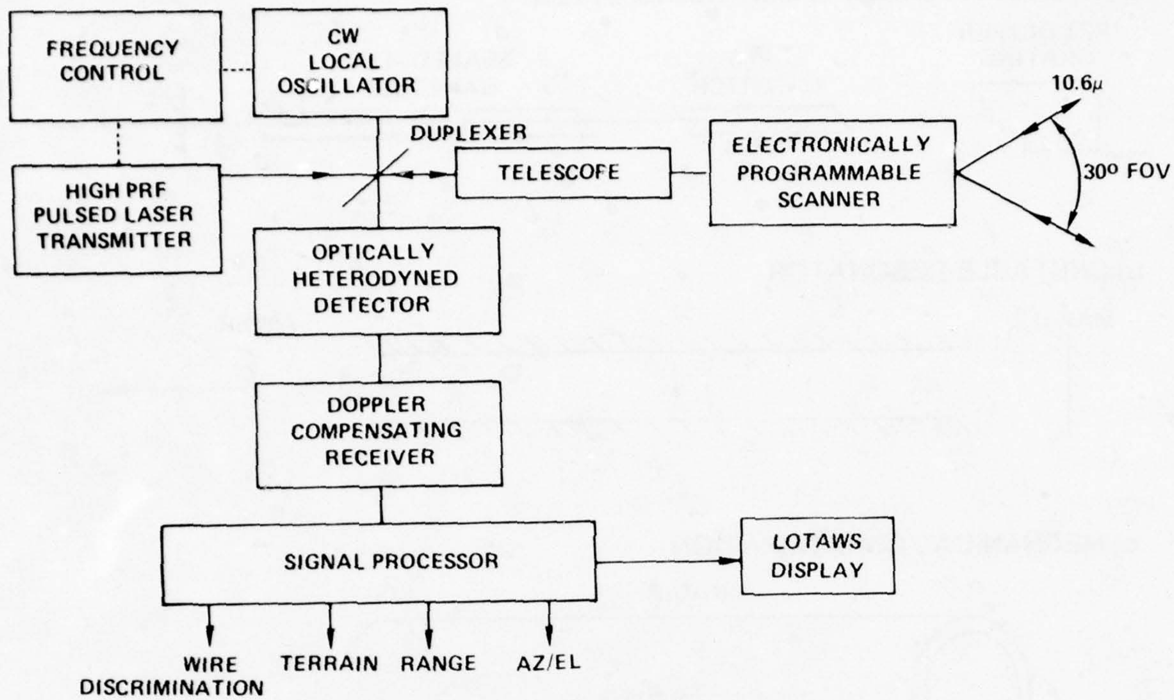

Fig.7 Major LOTAWS subsystems

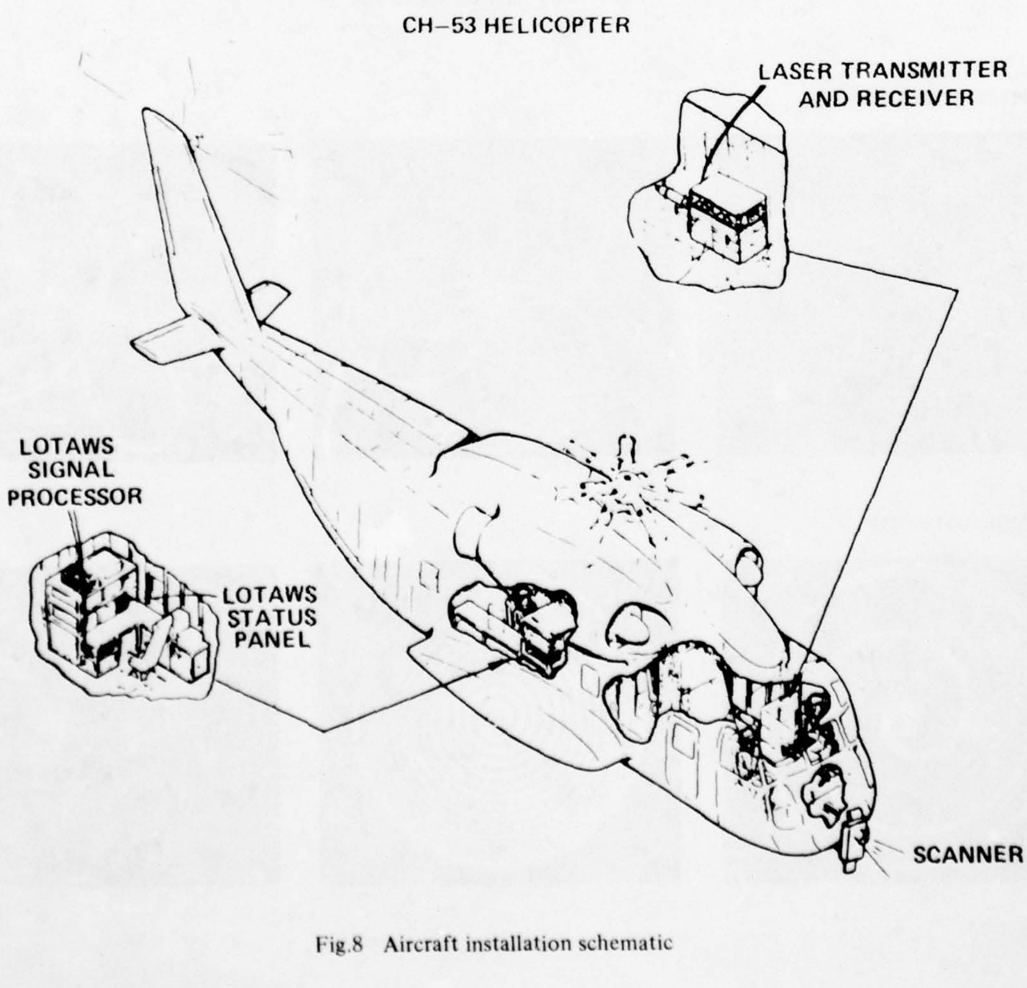

Fig.8 Aircraft installation schematic

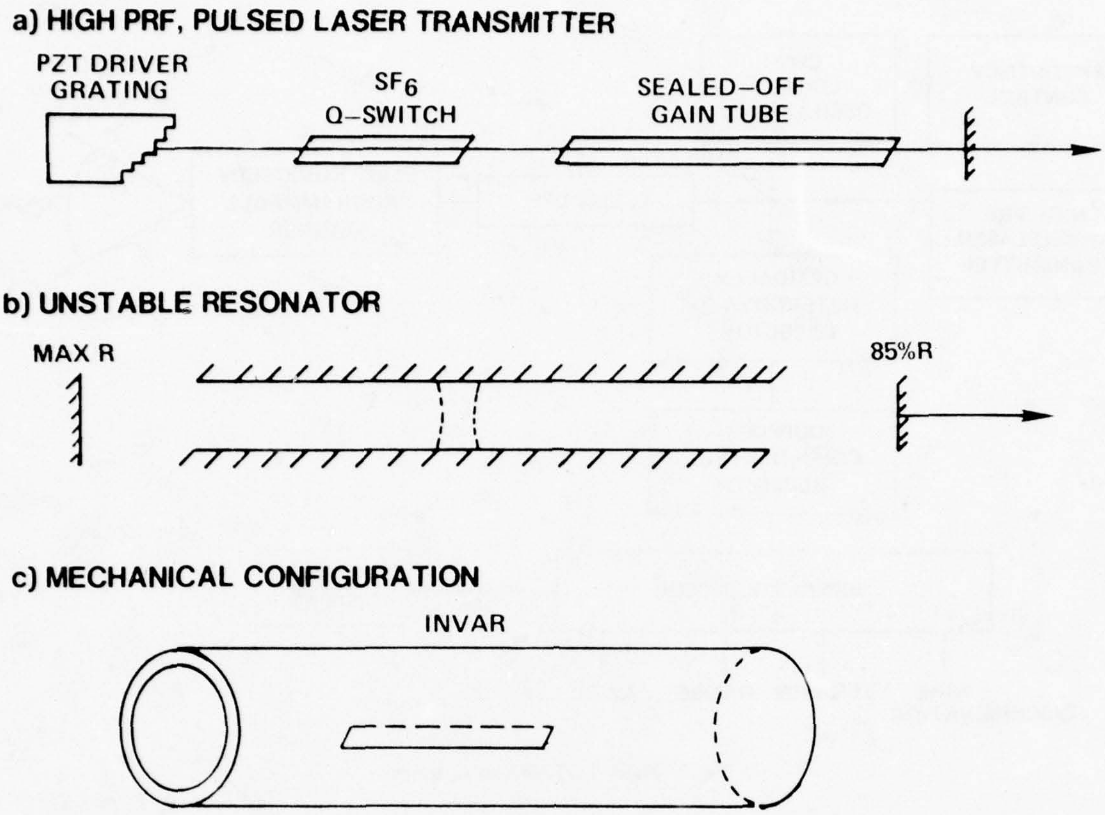

Fig.9 CO_2 laser configuration

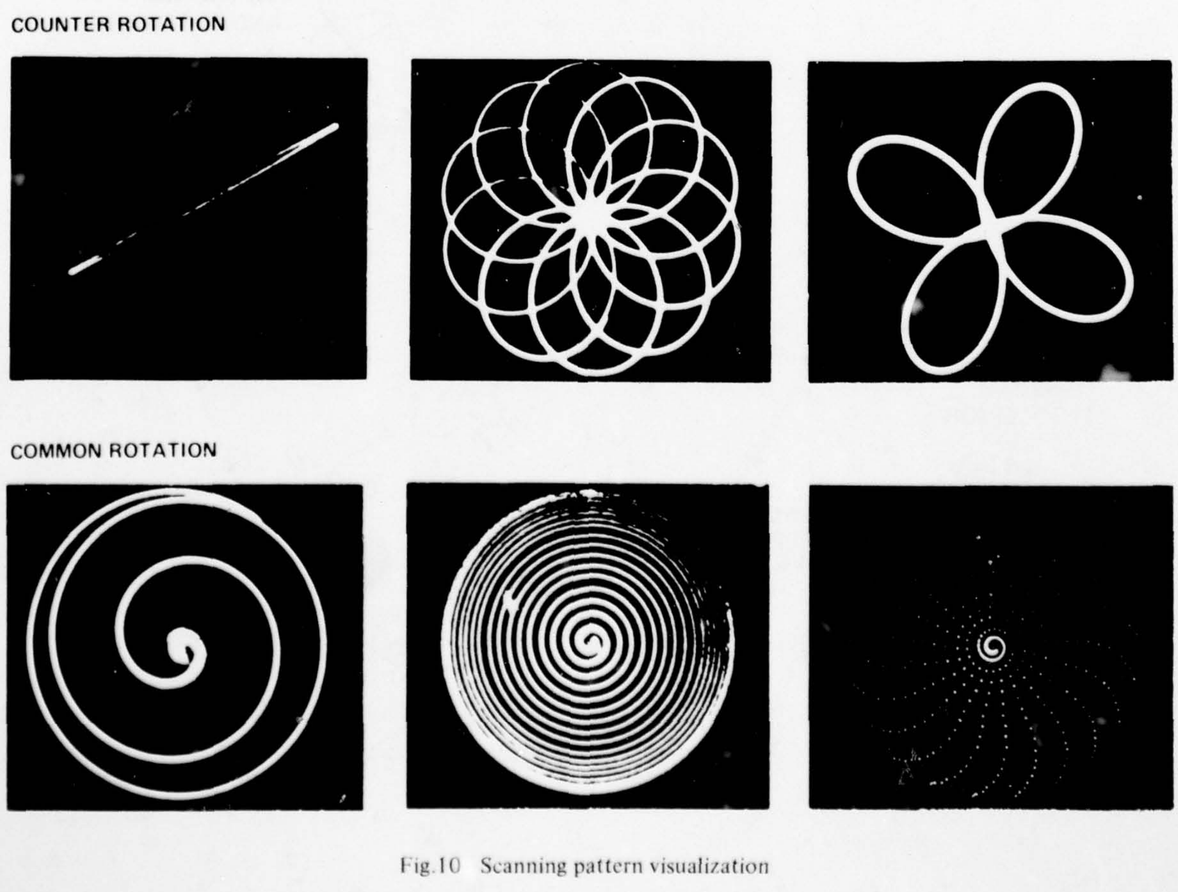

Fig.10 Scanning pattern visualization

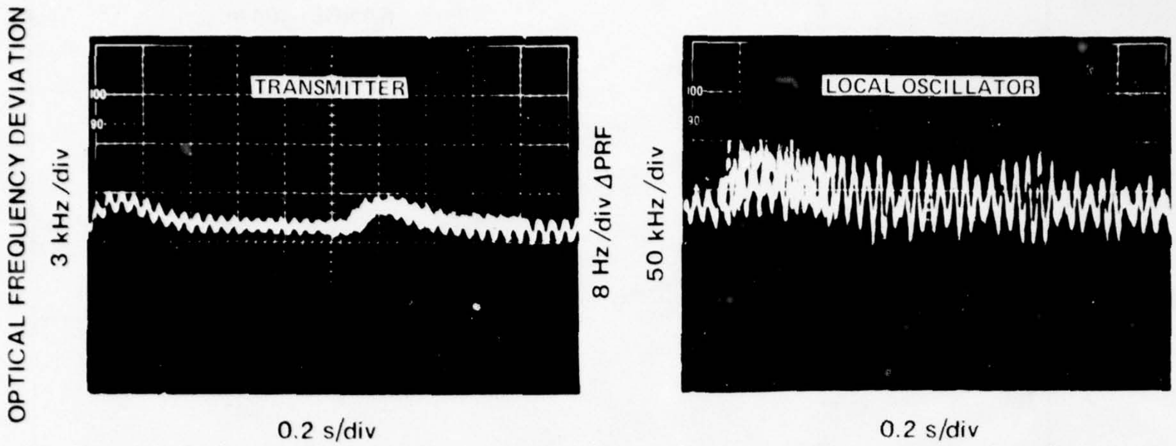

Fig.11 In-flight vibrationally induced optical frequency excursions

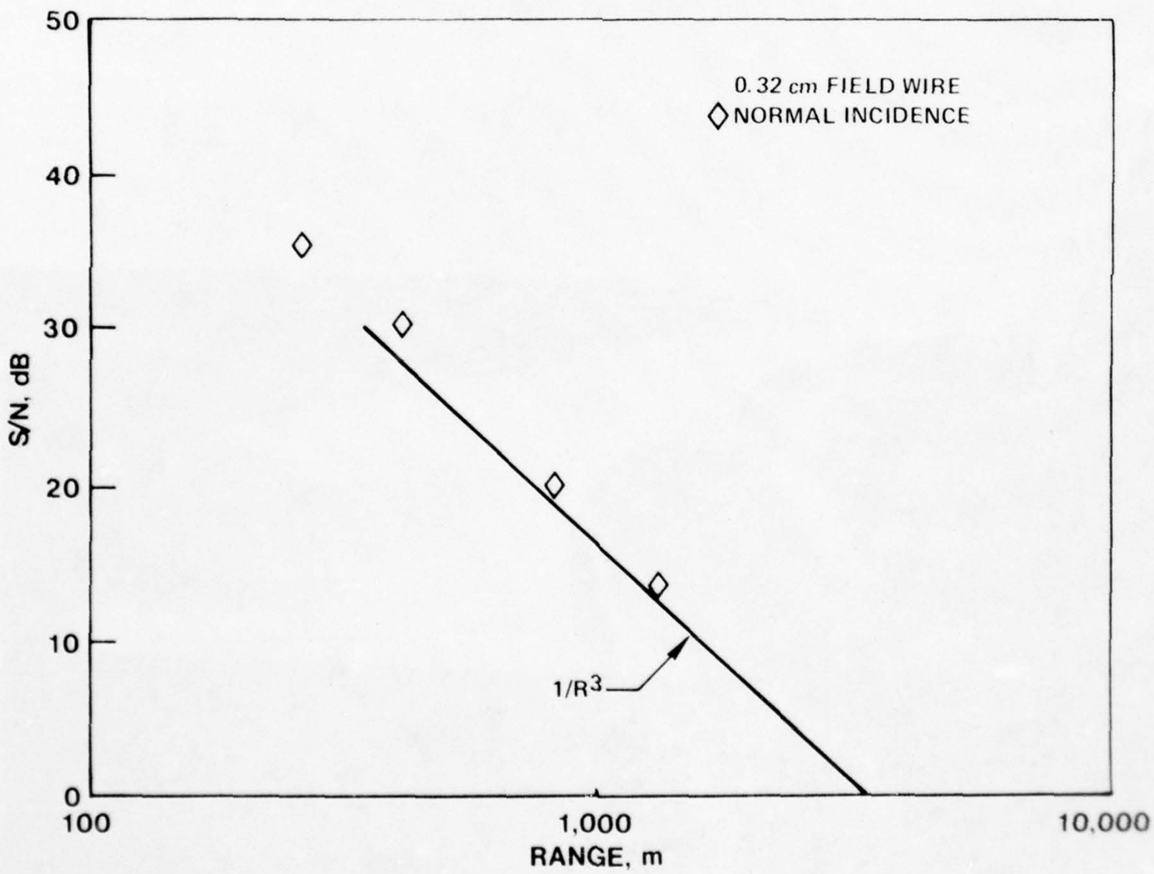

Fig.12 Signal to noise ratio vs range

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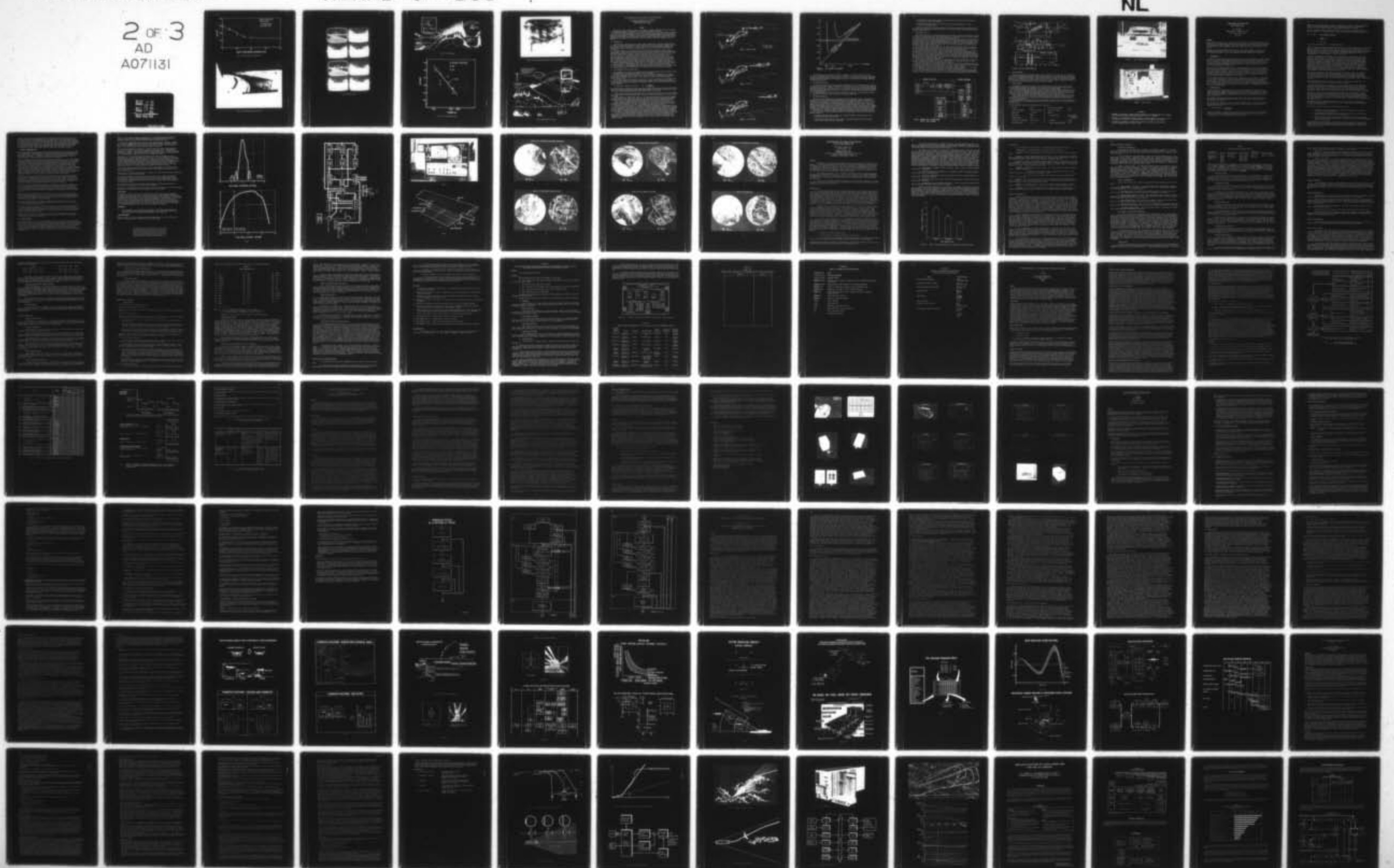

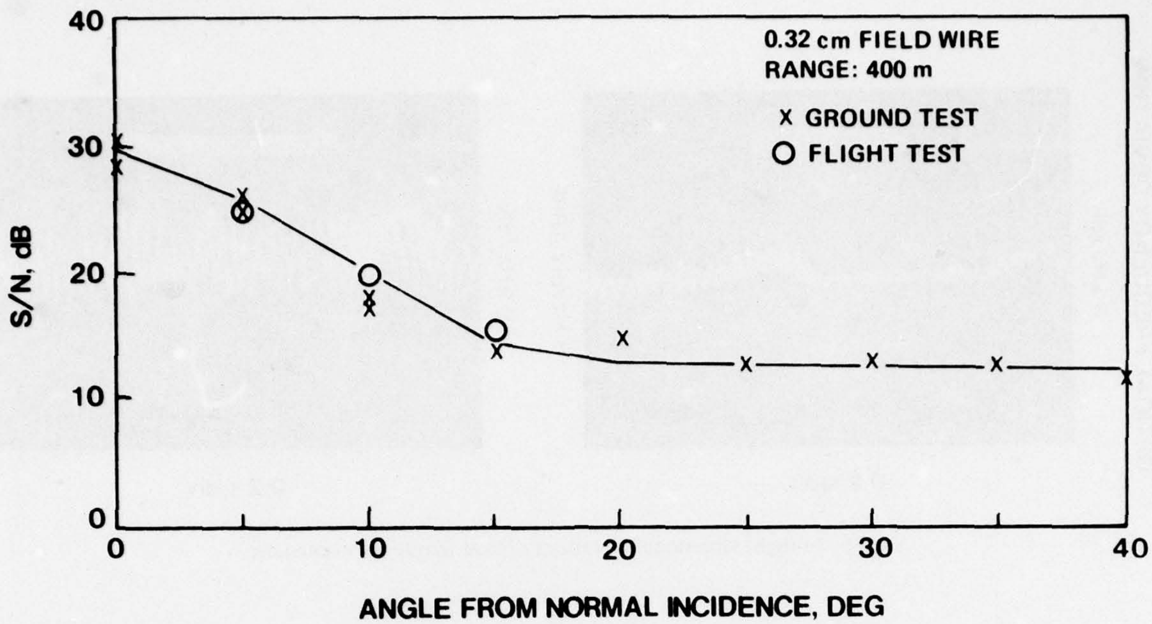

Fig.13 Angular dependence of signal return

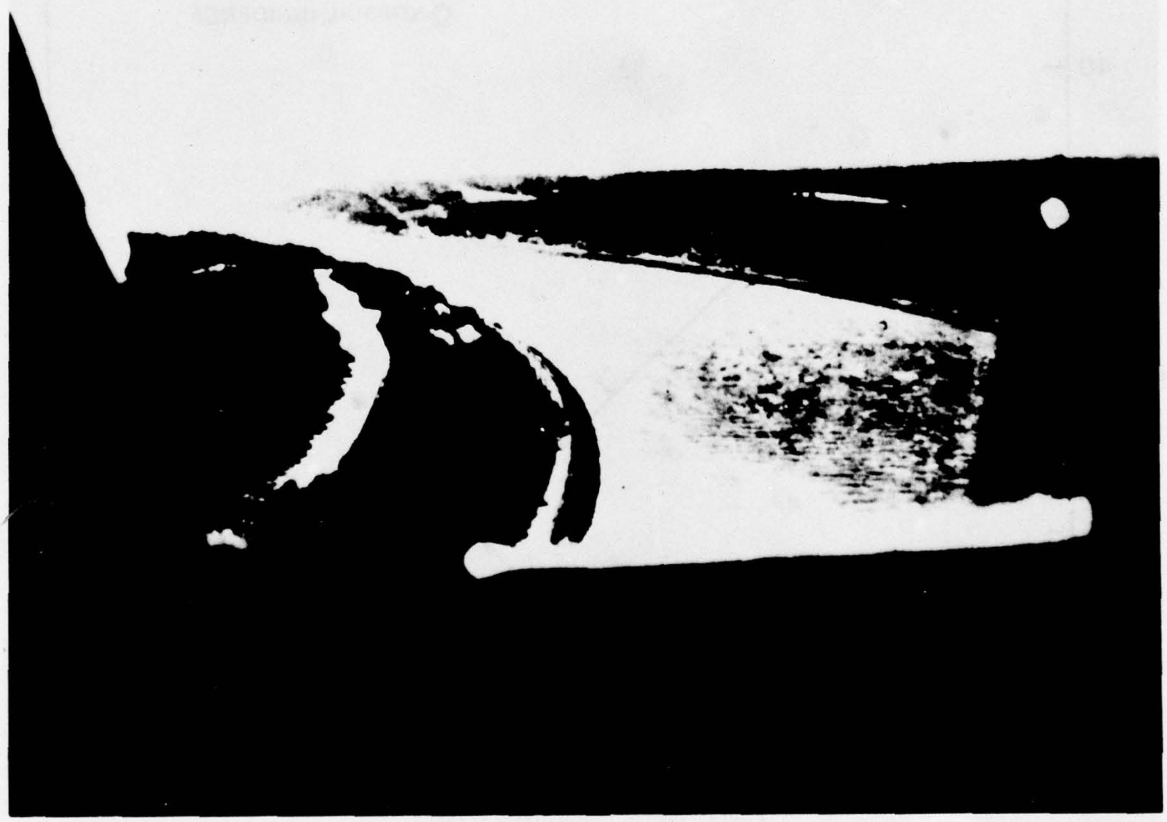

Fig.14 35 mm photo from aircraft cockpit

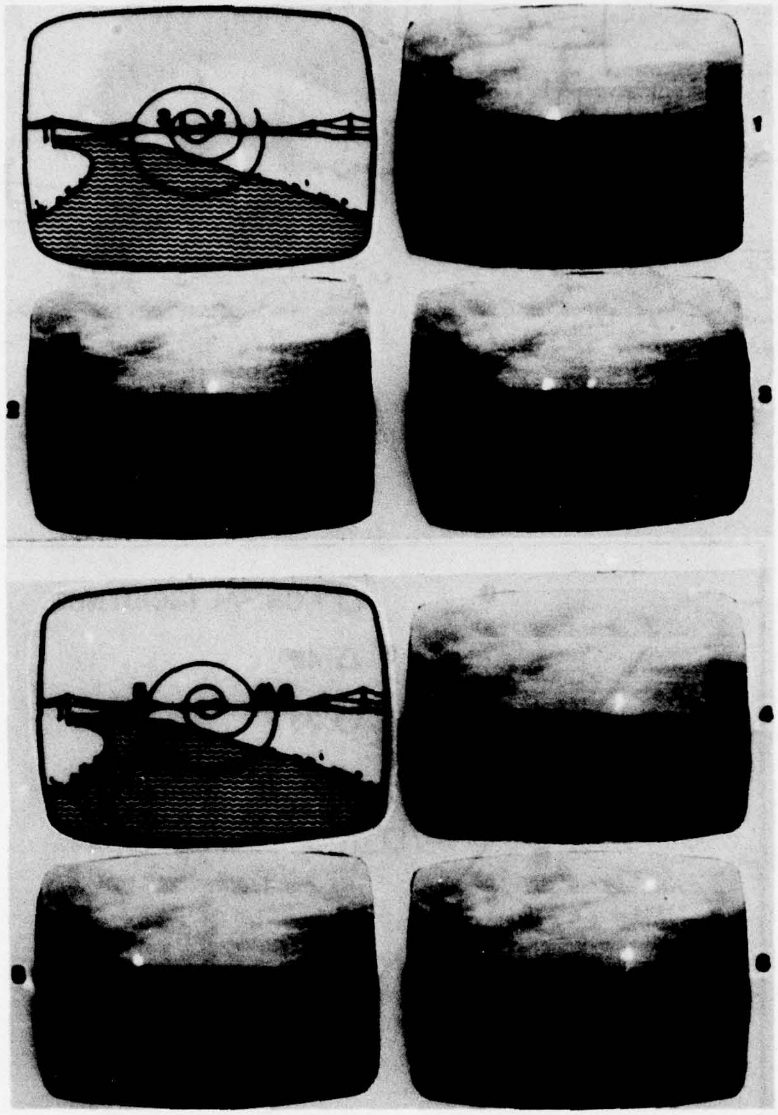

Fig.15 Wires at 1.6 km

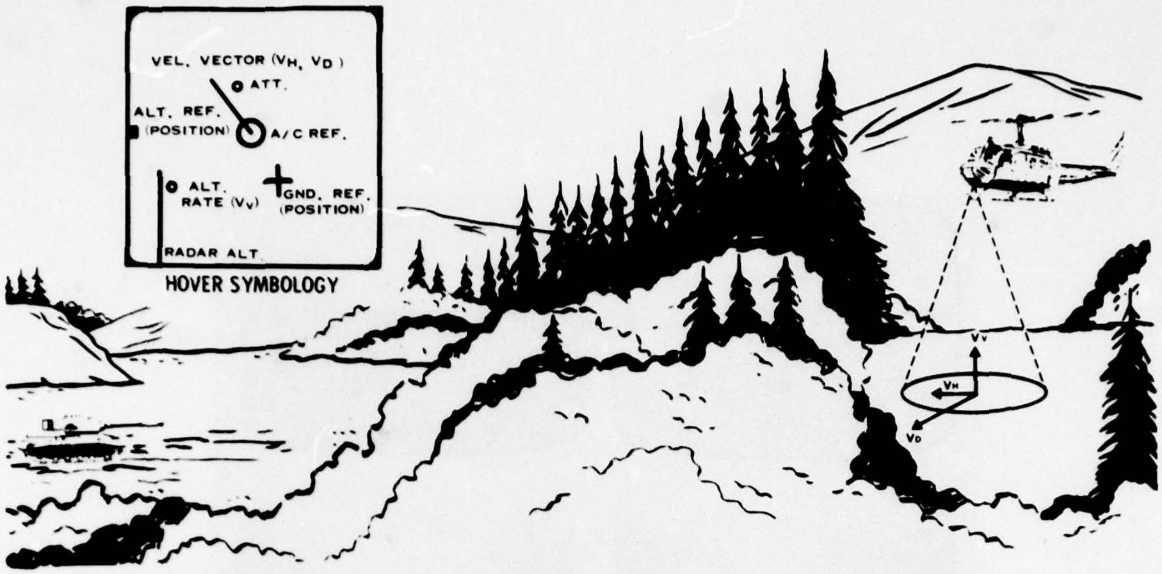

Fig.16 CO_2 Hover sensor

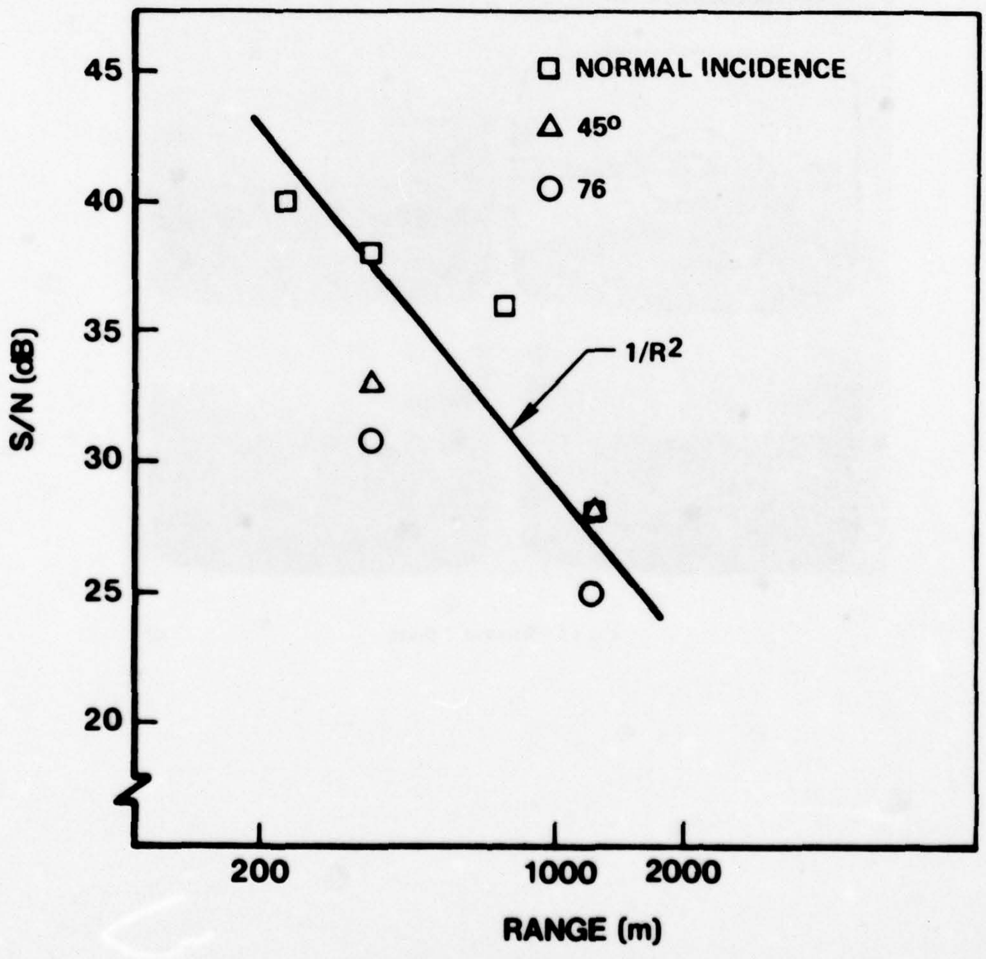

Fig.17 CO_2 sensor terrain returns

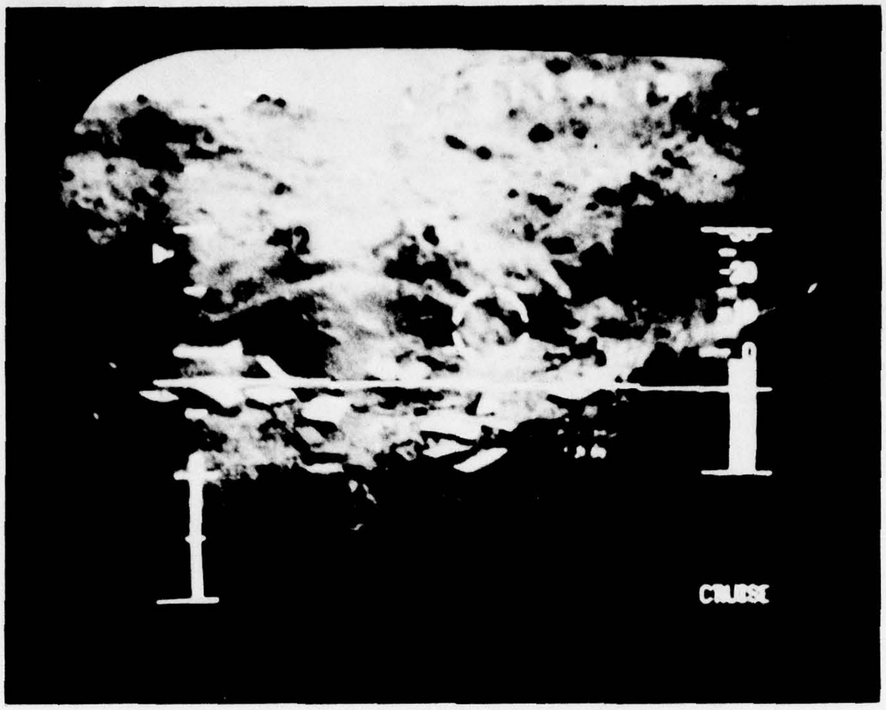

Fig.18 Terrain following symbology and forward looking imagery

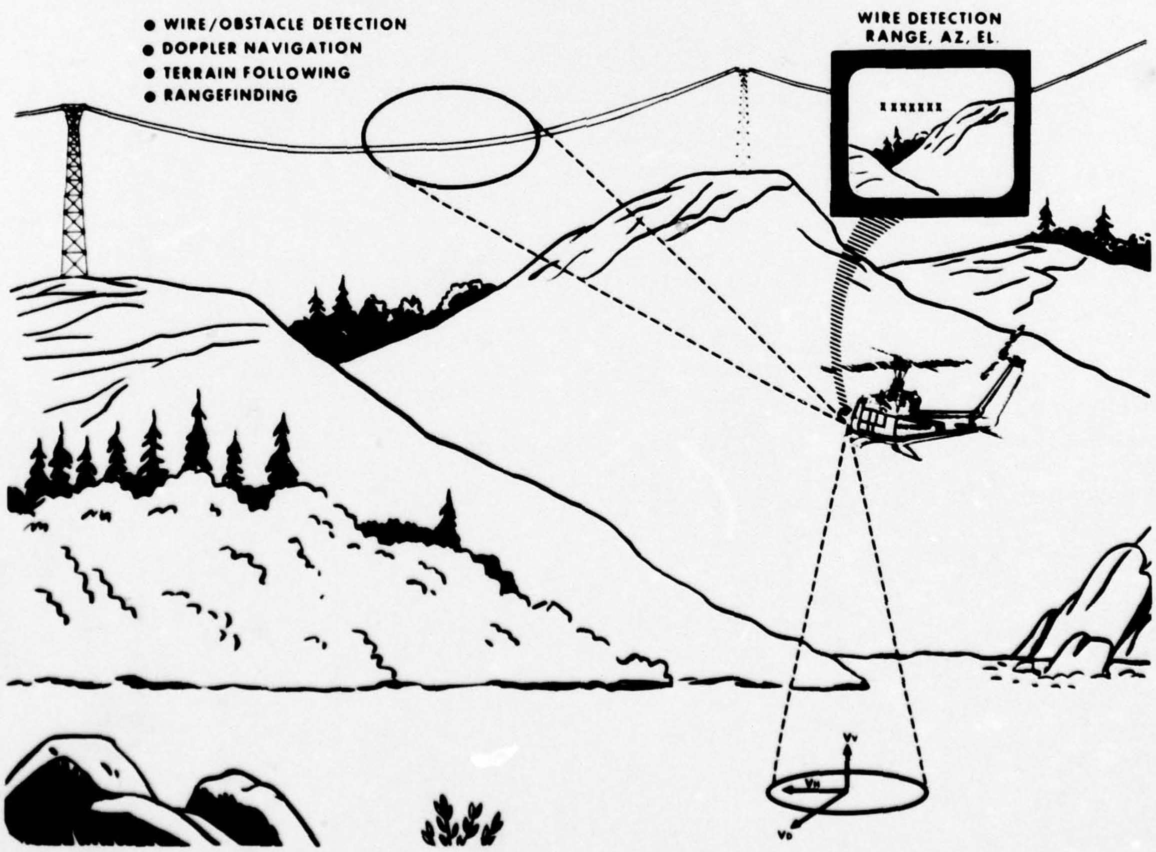

Fig.19 Multifunction CO_2 sensor

A SELF CONTAINED COLLISION AVOIDANCE SYSTEM FOR HELICOPTERS

by S. Bloch, G. Hoefgen and D. zur Heiden
Standard Elektrik Lorenz AG (ITT)
7000 Stuttgart 40, Germany

SUMMARY

The paper describes a proposed solution to the problem of obstacle detection and collision warning pertaining to the operation of helicopters at low altitude and within a formation.

A simple yet effective algorithm is used in order to assess the collision hazard posed by stationary obstacles as well as other helicopters in the proximity of the protected aircraft. The raw data needed (range and closing speed) is provided by a 35 GHz versatile low-cost FM/CW-radar that has been originally developed for the protection of road vehicles. With the exception of the antenna assembly only minor modifications are needed in order to make this radar fit the requirements of an airborne anti-collision system.

1. INTRODUCTION

Apart from their traditional roles in reconnaissance, communication, and troop transportation etc. helicopters are increasingly deployed for ECM and weapon delivery. Attack helicopters nowadays may carry guns, rockets, and anti tank missiles, as well as anti surface vessel and submarine torpedoes. This increased significance of helicopters in modern warefare calls for improved night and poor-weather capability.

The high vulnerability of helicopters often requires the pilot to maintain a flight course at minimum height in order to avoid early detection. This task becomes extremely difficult when performed under conditions of deteriorated visibility. Without appropriate assistance the pilot will have to climb to a "safe" height, clear of potential obstacles, thus exposing himself to the hazards posed by unfriendly air defense. The two dimensional FLIR information as well as Low Light Level TV are certainly useful tools in helping to maintain a flight course at lower height, but they cannot, however, provide complete information pertinent to collision avoidance. Another significant draw-back of optronic systems is the reduced performance, particularly when penetrating (smoke) fog and precipitation. In such cases an anti-collision radar would be more appropriate, as microwave propagation is considerably less effected by adverse weather conditions (and temperature gradients) than light and IR.

A very significant advantage of a radar system is its versatility, providing slant range, azimuth, elevation and doppler (closing speed) data thus permitting highly effective hazard assessment.

The price paid for this augmented capability is the increased vulnerability to ECM. For this reason appropriate measures need to be taken in order to minimize the risk of ECM detection and jamming.

2. THE ASSESSMENT OF THE COLLISION THREAT

The requirement for a small-size, low-weight and cost-effective airborne collision avoidance system dictates the need for a relatively simple method of threat assessment.

A sophisticated signal processing system, in which a multitude of different data (e.g. range, relative speed, elevation, azimuth, height, flight course, wind correction angle etc.) is analysed may well provide very reliable collision warning. Here, however, the cost of the airborne equipment is going to be prohibitive.

In contrast, an over-simplified system, indicating a collision hazard whenever the helicopter nears an obstacle, would be easier to implement, the performance, however, would be quite unsatisfactory. A viable compromise between the two options mentioned above seems to be a radar using the so called TAU-hazard assessment.

The factor TAU is defined as the ratio: $$\text{TAU} \cong \frac{\text{Range}}{\text{closing speed}}$$

Using TAU, no angular information or flight course data is needed. The system will operate on a real time basis, with the threat evaluation being performed instantaneously, with no need for tracked data storage. The following simple examples demonstrate the performance of the TAU-criterium:

Figure 1 shows two helicopters separated by 100 m. The relative velocity of 20 m/s, which is identical in this specific case to the closing speed, indicates that the two helicopters are on a direct collision course.

Figure 2 depicts an almost similar case, but now the relative velocity and the closing speed are no longer identical. This situation will lead to a near miss with the two aircraft reaching a minimum distance of 20 m. At the given closing speed this configuration could still be regarded as hazardous.

Figure 3 shows a situation where the two helicopters will at no point be closer than 45 m.

Figure 4 shows the behaviour of TAU vs. the range R. As long as the helicopters are sufficiently separated, TAU decreases almost linearly. For configuration #1 the linearity will be maintained until the impact point is reached. In this case, TAU represents the time-to-go to collision.

For the two other configurations, TAU reaches a minimum value followed by a renewed rapid increase (with TAU reaching infinity at R_{min}). An alarm will be initiated whenever the calculated TAU falls below a pre-set threshold. In the example shown in figure 4, collision warning will be given for configurations #1 and #2 only. The geometrical shape and size of the protected space determined by TAU will automatically be adapted to the magnitude and direction of the relative velocity and the closing speed involved.

The choice of the alarm threshold determines the sensitivity of the collision warning. It should be set so as to provide sufficient time for evasive action. The sensitivity has to be matched to the maneouverability of the helicopter and the reaction time of its pilot. Setting an excessively high sensitivity, on the other hand, should be avoided as it will cause an increase in the number of inappropriate alarms.

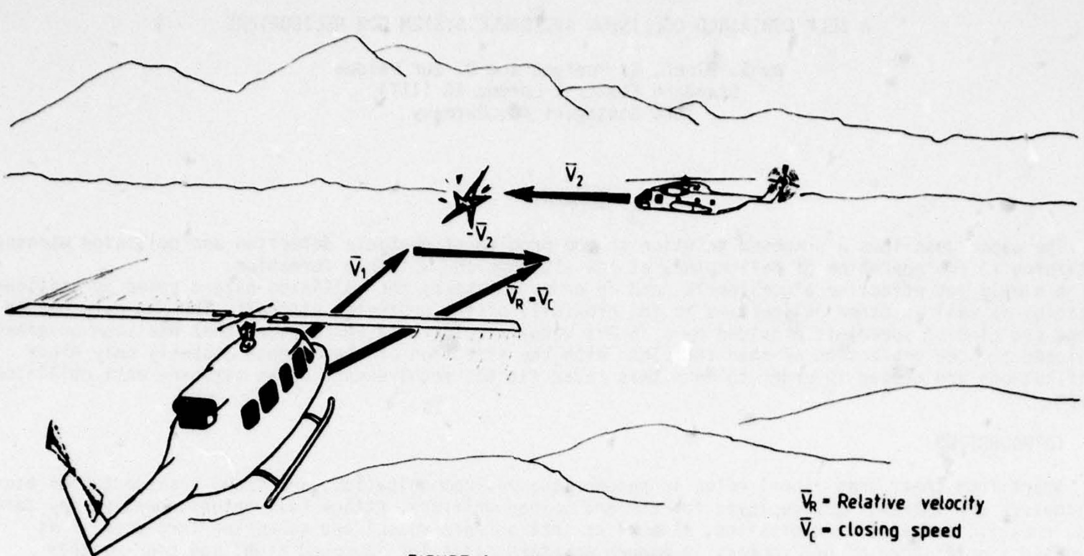

FIGURE 1 : COLLISION COURSE

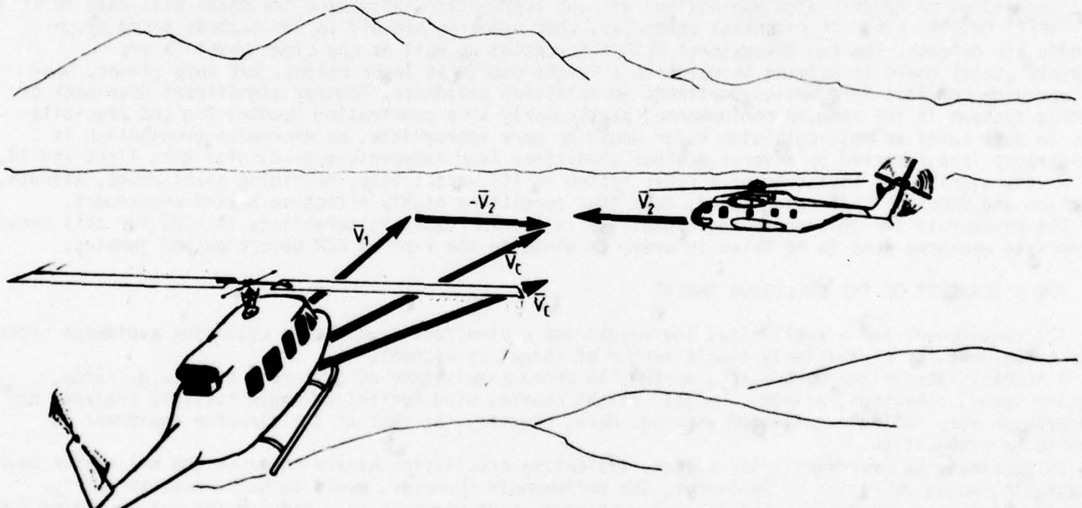

FIGURE 2 : NEAR MISS

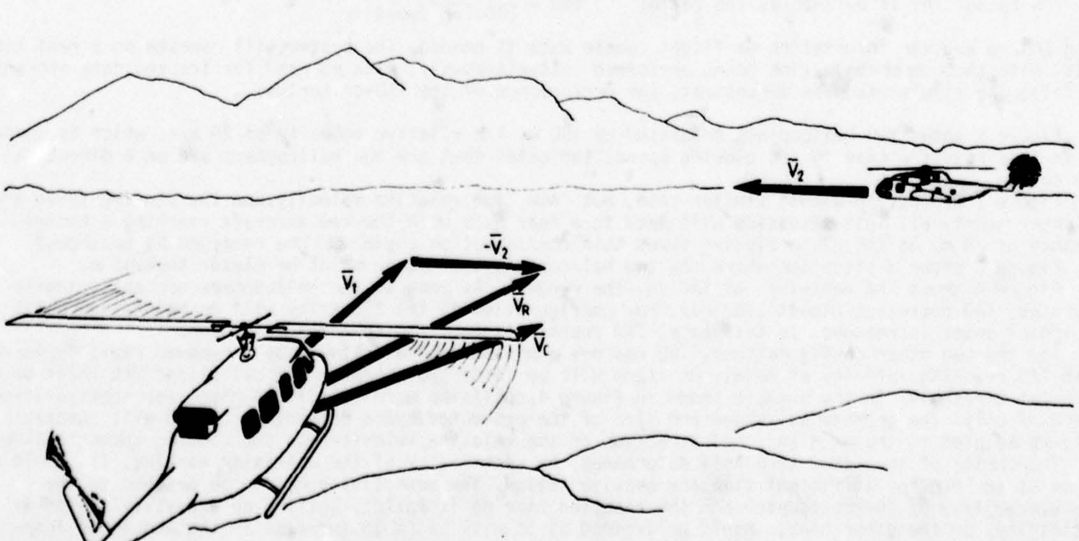

FIGURE 3 : SAFE SEPARATION

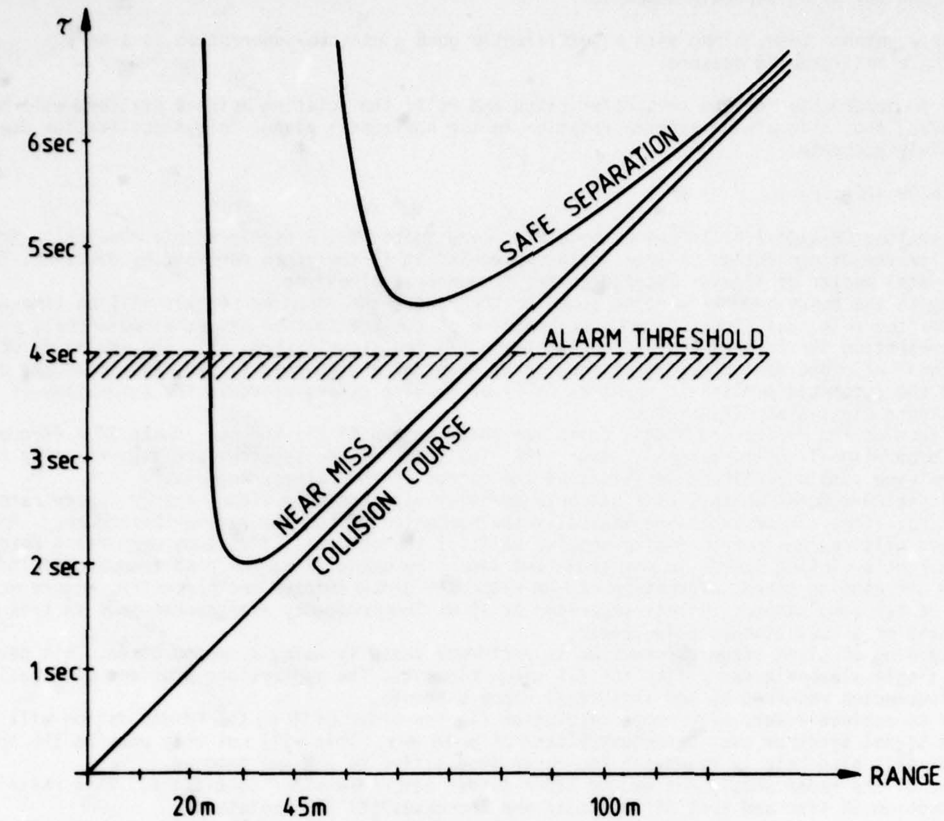

FIGURE 4: TAU HAZARD ASSESSMENT

The performance of the TAU-hazard assessment is degraded in cases where very small relative velocities are involved (while hovering or within a formation). In such cases small errors in measuring the closing speed might cause a significant inaccuracy of TAU. This deficiency necessitates independent range monitoring that generates an alarm whenever an obstacle is detected within a pre-determined space surrounding the helicopter.

3. THE RADAR CONFIGURATION

The choice of a simple algorithm for the threat assessment is the first step towards the realization of a low cost anti-collision system. The second step is the design of a simple and rigid radar capable of providing high resolution, in range and doppler, whilst being insensitive to ECM and jamming. An FM-CW radar apears to be the best possible choice, with its relatively low peak power permitting the implementation of a completely solid state system that will need neither a stabilized high-voltage-power-supply nor fast RF-switching devices.

The transmitted power is almost undetectable by alien ELINT receivers, thus making the radar much less susceptible to ECM, than a pulse system of corresponding capabilities.

4. ANTENNA CONSIDERATIONS

One of the crucial problems facing the designer of a collision warning system, is the question of coverage that is to be provided. The unique maneouverability of helicopters basically requires the monitoring of the entire space surrounding the aircraft. Unfortunately this aim is difficult to realize. In particular the "illumination" of the upper hemisphere is expected to be technically difficult. An appropriate confinement of the collision protection to the lower hemisphere will help to simplify the hardware implementation without causing excessive degradation of the overall performance. This statement is based on the assumption that most critical collision hazards are expected to be at moderate or negative elevations. (In case of formation flights some helicopters might be "blind" to hazards posed by higher flying aircraft. Here, however, the higher helicopters will adequately monitor the collision threat.)

Despite the fact that the TAU-hazard assessment does not require any angular resolution, the radar will still have to use a scanning directional antenna. The following main reasons explain the need for antenna directivity:

a) The angular resolution reduces clutter and cuts down the number of targets that have to be processed simultaneously by the radar.

b) The additional angle information is very useful in deciding the evasive action to be taken once a collision hazard has been established.

c) The directivity helps to achieve good isolation between the CW-transmitter and the receiver front-end (by using separate antennas).

d) A narrow antenna beam, along with a sufficiently good side-lobe-suppression is a very effective anti-jamming measure.

In order to compensate for the helicopter pitch and roll, the rotating antenna platform will have to be stabilized, thus maintaining antenna rotation in the horizontal plane. This stabilization does not need to be unduly accurate.

5. RADAR SENSOR (Fig. 5)

The transmitted CW-signal is frequency modulated (saw tooth-FM). A highly stable modulation frequency is important for the determination of the closing speed and it is therefore derived, by division, from a stabilized crystal master oscillator located in the frequency synthesizer.

According to the range and the closing speed of the target the incoming signals will be time-delayed and doppler-shifted (Fig. 6a). Instantaneous hetrodyning of the transmitted and received signals yields an output representing the frequency difference between the two signals (Fig. 6b). The amplitude of the difference signal of close-in-targets is attenuated by means of a high pass filter. This measure, corresponding to the automatic sensitivity control (STC) of a pulse radar, prevents the saturation of the receiver by strong signals at close ranges.

The spectrum of the difference signal (with the sweep return f* blanked out) peaks at a frequency Δf which is proportional to the target's range. The side lobes of the spectrum are significantly suppressed by applying simple amplitude weighting at the output of the hetrodyning mixer.

The periodic FM-modulation implies a discrete spectrum with the main signal energy concentrated in very few spectral lines. These lines are basically the harmonics of the modulating FM-frequency. Any relative motion will cause a corresponding doppler shift of the spectral lines thus permitting determination of the target's closing speed (in magnitude and sign). By means of narrow band frequency filtering the range and the closing speed information can be extracted as two independent parameter. Figure 6c shows the spectrum of two simultaneous targets separated by 10 m. The frequency axis corresponds in this case to the time axis of a conventional pulse radar.

The processing of slant range information is performed serially using a second mixer. This permits the use of a single steep-flanked filter for all range elements. The synthesizer provides all local oscillator frequencies required by the individual range elements.

In order to achieve a very high range resolution (in the order of 10 m) the FM-modulation will have to spread the signal spectrum over a frequency band of $>$ 50 MHz. This will not only provide the necessary resolution but will also help in rendering the radar insensitive to ECM and jamming.

The range of the radar should not exceed 250 m (under adverse weather conditions). This restriction permits a reduction in size and cost of the radar and increases its ECM resistance.

The close range operation and the requirement for a small-size low-weight system make the Ka-Band (35 GHz) or W-Band (94 GHz) most appropriate for the collision avoidance task. Broad-band narrow-beam systems are readily achieved at these frequencies, whilst ECM detection and jamming become very difficult.

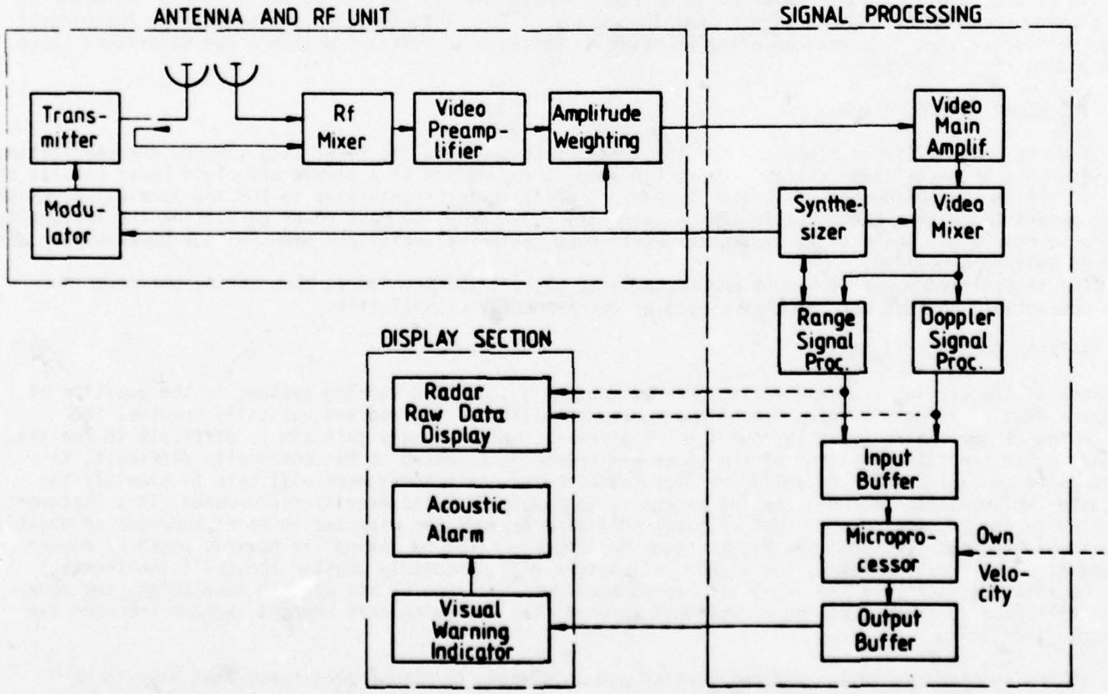

FIGURE 5 AIRBORNE ANTI COLLISION RADAR
OVERALL BLOCK DIAGRAM

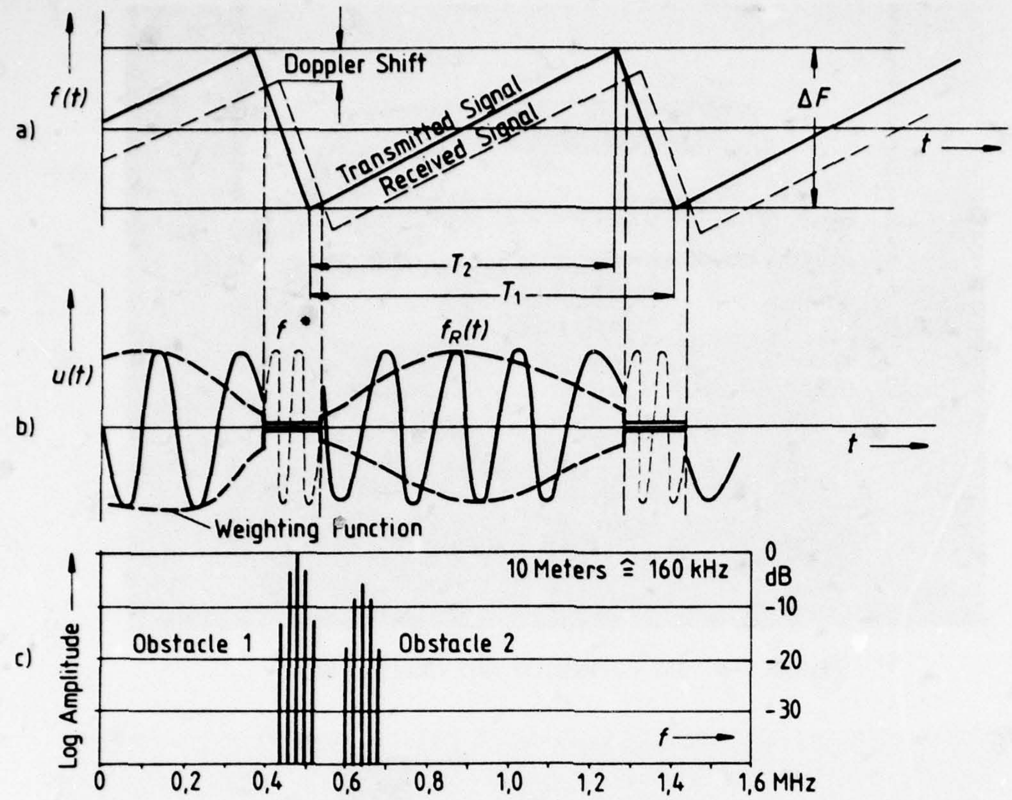

FIGURE 6 : FM/CW RADAR PRINCIPLE

6. HAZARD INDICATION

The threat evaluated by the system's computer (micro-processor) will raise an acoustic intermittent tone alarm which becomes intensified the further TAU falls below the pre-set threshold. The azimuth of the threat will be simultaneously indicated on a display consisting of a circular LED array with its elements spaced in 10° increments. For formation flights it might be useful to display all the adjacent helicopters on a PPI scope or an equivalent ρ - ϑ-indicator. In case of a collision hazard, an acoustical alarm will be sounded and the threat will be shown as a flashing blip on the scope.

7. RADAR PARAMETER

The helicopter radar will be a derivative of a Ka-Band automotive anti-collision radar that has been developed and successfully tested over the last three years at Standard Elektrik Lorenz AG. This project has been sponsored by the German Ministry of Science and Technology (BMFT).

This FM-CW-radar, dubbed RAS, has shown its merits in numerous tests and under a variety of different traffic situations. Over the next two years extensive testing of 10 vehicles fitted with RAS radars will take place, in order to demonstrate system operability under normal traffic conditions. The RAS radar is light and rigid and its project price (large scale production) is in the order of $ 500.

Figure 7 shows the two small 35 GHz RAS antennas which are barely larger than the car's head lights.

Figure 8 shows the RF unit, using modular integrated techniques. The module incorporates the transmitting antenna (rear view), a gun oscillator with its modulator (left hand side), as well as the RF-mixer and video pre-amplifier.

The photographs give a good impression of the compactness of the RAS radar. Due to its high resolution and accuracy, as well as its mechanical features, RAS is ideally suited for use in an airborne collision avoidance system.

Following parameters summarise the proposed anti-collision radar:

CW power	50 mW	Closing speed resolution	2,5 kt
Operating frequency	35 GHz (or 94 GHz)	Dwell time on target	0.01 s
Band width	> 50 MHz	Antenna:	
Modulation	FM/CW	3 dB beam width	3.5° (azimuth) 15° (elevation +5°/-10°)
Maximum range	250 m		
(range of) closing speed	-15 to 160 kt	Rotation	60 RPM
Range resolution	< 10 m	Weight (without antenna)	10 lb

FIGURE 7 : RAS - AUTOMOTIVE ANTI COLLISION RADAR

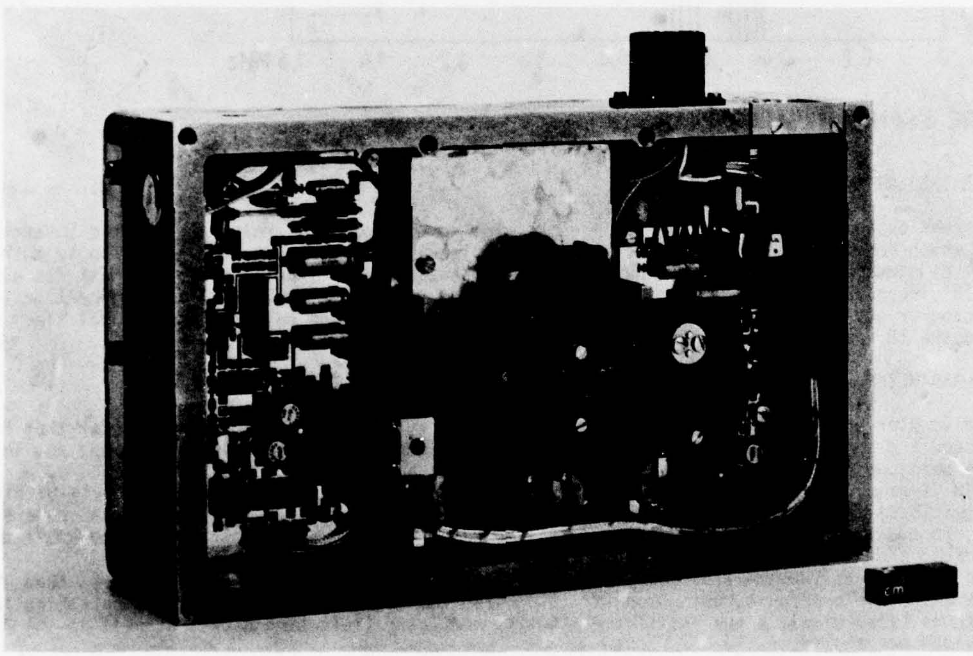

FIGURE 8 : RAS - RF UNIT

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A HELICOPTER HIGH DEFINITION ROTOR BLADE RADAR

by

C.M. Stewart
RADAR GROUP, FERRANTI LIMITED
Crewe Toll, Edinburgh
Scotland

SUMMARY

By installing a radar aerial within the rotor blade, a sensor having unique capabilities for the operation of helicopters in darkness or bad weather can be provided. A description of such a system is given outlining the factors affecting the choice of the principal radar parameters and their interaction with the helicopter rotor design.

Examples of the high definition pictures from the radar display are shown with the appropriate section of map for comparison and some suggestions are made on operational roles where such a radar system would have particular advantages.

1. INTRODUCTION

In an effort to improve the operating capabilities of the helicopter in poor weather and bad visibility considerable efforts in recent years have been devoted to sensors which, in general, have attempted to augment the visual scene as it appears to the pilot and crew. Consequently these sensors are usually operated in the optical and infra-red wavelengths where the limitations on weather penetration, field of view and range can be significant.

Radar with its longer wavelengths offers unique capabilities in all these respects particularly when advantage is taken of the scanning property of the helicopter rotor to mount the radar aerial within one of the blades. The map-like presentation available from such a radar, even at the lowest altitudes, can be an invaluable aid and the high resolution, 360° uninterrupted field of view and high data rate obtainable enhance the performance to a level difficult to achieve with any other form of airborne radar or indeed any other sensor.

To investigate the potential of the rotor blade radar and to assess its capability in the field, Ferranti Ltd. under contract to MOD (PE) constructed an experimental version which was coupled to an aerial designed by the Royal Signal and Radar Establishment and installed within a special rotor blade by Westland Helicopters Ltd. Flight trials are still being carried out by RSRE utilising a Wessex helicopter but it is the intention in the later part of this paper to show samples from the results achieved to date.

2. SYSTEM DESIGN

In the selection of system parameters certain constraints were imposed by the static and dynamic properties of the rotor head, in particular the rotor rotation speed which in the Wessex is approximately 230 rpm. This represented a refresh data rate from a single aerial mounted within the blade of about 4Hz and a scan rate of 1380°/sec.

The refresh rate therefore lay well within the flicker rate detectable by the eye and this had to be taken into account when selecting the type of display to be used with the radar.

A high scan rate interacts with the azimuth beamwidth (Bw) of the aerial and the pulse repetition frequency (p.r.f.) so that for any patch of ground within the radar's feld of view,

$$\text{hits per scan} = \frac{\text{Bw} \times \text{p.r.f.}}{\text{scan rate}}$$

Clearly the number of hits per scan should never be less than one and preferably be higher since the returning echoes from each particular patch of ground can be integrated to give improved detection sensitivity.

However, for a high definition radar it is essential to use a narrow azimuth beam-width to achieve good angular resolution, despite the penalty of reducing the amount of time the beam can spend illuminating each patch of ground.

For this radar an azimuth beamwidth of 0.5° was chosen and a minimum number of hits per scan of 2. Inserting these numbers in the above relationship shows that:-

$$\text{p.r.f. (min)} = \frac{\text{scan rate x hits/scan}}{\text{Bw}}$$

$$= \frac{1380 \times 2}{0.5}$$

$$= 5.52 \text{ KHz}$$

In practice a minimum p.r.f. of 6.2KHz was used.

Since the interval between successive pulses fixes the maximum time for a transmitted pulse to travel to a target and the echo to return, the p.r.f. determines the maximum range or more correctly the largest range scale which can be used. Allowance must be made for settling time in the radar and display between successive pulses and taking this into account, the maximum range scale became 10Km. This was supplemented by further scales of 5,2,1, and 0.5Km.

A further consequence of mounting the aerial within the rotor blade is that the effect of the application of both cyclic and collective pitch on the elevation pointing direction must be allowed for in addition to the usual aircraft pitch changes. The simplest solution was to ensure that the elevation beamwidth was sufficiently great to encompass these variations since in any case the available elevation aperture was too small to allow much control of the aerial pattern in this plane. This was not necessarily the most efficient technique since some aerial gain was sacrificed but it had the merit of simplicity. The elevation beamwidth achieved in practice was about 40° but as can be seen in figure 1 it was possible to produce a sharper cut off on the upper side of the beam than the lower thereby concentrating most of the available RF energy on the ground instead of dissipating it uselessly above the horizon line.

The choice of frequency to be used lay between I band and J band i.e. covering the range 8-16GHz since lower frequencies make the aerial design increasingly difficult (in particular the length becomes excessive) and higher frequencies incur significant losses both in the waveguide system taking transmitter pulses and received signals to and from the aerial and in the atmosphere itself, particularly in rain.

These considerations led to a choice of I band at approximately 9GHz which for 0.5° beamwidth resulted in an aerial 4 metres in length.

Since a primary aim was to achieve a high resolution radar map, the good angular resolution of 0.5° had to be matched by similar range resolution and two pulse widths of 50 and 100ns were provided giving equal range and angular resolution at 0.8 and 1.7Km. These were paired with p.r.f.'s to maintain the mean power as constant as possible in the interests of easing the transmitter design problems.

The use of a short pulse also permits good minimum range provided the recovery time of the receiver protection system can be kept short. This problem becomes increasingly difficult as the peak power level is raised and some design difficulties were encountered as a result since a peak power of 80KW had been chosen to give good signal/noise ratios on a single pulse basis without relying on integration from scan-to-scan.

Nevertheless careful design permitted a recovery time of 50ns to be achieved, so that the obscured area of ground close to zero range did not exceed 80 metres in radius from this cause.

The choice of display took into account several factors:

(1) The need to choose a system which removed the 4Hz flicker resulting from the rotor rotation speed.

(2) A high definition capability was required to match the good range and angular resolution of the rest of the system.

(3) The range scales, particularly the shortest implied very high writing speeds on the cathode ray tube face. These had to be achieved simultaneously with high brightness.

The requirement to remove flicker indicated a need for storage and the two most promising possibilities appeared to be a scan converter feeding a television type display or a direct view storage tube (DVST) which incorporates a storage mesh within the tube itself.

The 4Hz refresh rate and high definition requirement made the specification for the scan converter very demanding although just within the state of the art at the time. On the grounds principally of economy the D.V.S.T. was chosen although it is doubtful if, in view of the rapid decrease in the cost of digital components, this would now be the best choice. A small sacrifice in resolution was accepted on the longest range scales as a consequence but full brightness was retained by the use of a short term digital buffer store which enabled the read out to the display to be at a lower rate than the input from the radar receiver.

3. SYSTEM IMPLEMENTATION

Because the radar was intended for experimental purposes only, provision was made for varying many of the parameters in flight from the control unit although this also resulted in a larger and more complex system than would be necessary in an operational equipment. In addition, a special airborne photographic recorder was developed to retain permanent records from the display.

A block diagram of the major units of the system is shown in fig. 2. The displays offered a wide selection of operating modes including, ground stabilisation, north or heading stabilisation, PPI and offset PPI, marker facilities and four different range scales. This necessitated a considerable amount of processing of angle data mainly in digital form and this was performed in the Angle Generator Unit. A view of the principal units in the helicopter is shown in fig. 3. The only units external to the radar which were required were a doppler navigator to provide along and across track rate signals together with drift angle and a compass bearing input for north stabilisation. To establish the pointing angle of the aerial beam in azimuth a shaft encoder was fitted to the main rotor shaft but this angular output was modified by the lag angle of the aerial rotor blade as measured by a separate sensor to arrive at the true azimuth bearing of the beam.

The problem of providing an RF connection between the aerial and the Transmitter/Receiver was solved by running the waveguide up the inside of the hollow rotor drive shaft. To complete the path across the rotor hinge assembly to the aerial was however more difficult but by the use of a rotary joint and a carefully sited flexible waveguide a relatively simple system was evolved which has proved to be entirely satisfactory providing good life and low RF losses.

The rotor blades of the Wessex helicopter are normally of all-metal construction but it proved possible by taking advantage of the newer fibreglass techniques used in later helicopters to incorporate the aerial within the trailing edge of the blade without in any way changing the weight or mass balance characteristics to the extent that the radar blade was completely interchangeable with the normal blades.

A cross section of the blade aerial is shown in fig 4. The use of a trailing edge aerial was permitted by the use of I band in contrast to the higher frequency leading edge aerial used by earlier workers, giving the significant advantage that this almost completely avoids the deterioration in radar performance caused by erosion of the surface of the fibreglass.

The aerial itself which is a length of waveguide with angled slots cut in its narrow side can be withdrawn to permit inspection and to allow it to be re-used if a blade change becomes necessary.

To assist in fault finding on the radar an extensive built-in-test facility (B.I.T.) was included from the beginning. This provides not only test signal injection but extensive monotoring of parameters to localise the fault to a particular unit. Further diagnostic indications within each unit allow fault tracing to particular modules or printed-circuit cards.

A very important facility is provision for simulating the rotation of the rotor to allow the radar to be tested without the necessity of starting the aircraft engine and driving the rotor blades.

4. TRIALS RESULTS

In the first phase of the trials the purpose has been to study the radar profiles of as large a variety of terrain combinations as possible within easy flying distance of base, which in this case was the Royal Aircraft Establishment airfield at Bedford. Later phases of the trials programme will examine other aspects of the operational use of the radar equipped helicopter including the assistance to be afforded in landing, control of other (non radar-equipped) aircraft and the possibility of obtaining station-keeping situation displays when a number of aircraft are operating in concert.

Since all the flying to date has taken place in the south of England where the land is fairly flat but manmade landscapes are both widespread and varied an extensive range of terrain situations has been available for study.

A selection of photographs taken from the radar's photographic recorder is shown but it must be emphasised that the lack of the dynamic effect resulting from the use of still snapshots robs the display of much of its effect. Moving objects such as cars and aircraft are much more obvious in reality.

Two identical displays are available in the aircraft each using a 10cm diameter D.V.S.T. one visible to the pilot, the other in the rear cabin. The special photographic recorder uses a repeat display which appears on a 2.5cm CRT optically coupled to a camera. The radar operator can select single shot, continuous bursts at 2 frames/second or dwell times of 2 or 1 seconds per frame.

Some aspects of the accompanying photographs from the test flights are worth high-lighting. At lower altitudes such as 350 ft. it is interesting to see that shadowing can enhance a terrain feature, in some cases creating almost a 3-D image, so that identification is particularly easy. This can be seen, for example in Flight 21 where the annular shape of the earthworks making up the ancient defensive works called Old Sarum in the bottom right of the picture is clearly visible. Similarly the road on the left side of the photograph shows up as a bright line edged with black, it is thought because the road is lined with trees.

The contrast between grass and concrete is very advantageous in the location of airfields as seen in Flight 24 and this makes it even possible to detect aircraft on the runway itself.

Roads, and to a slightly lesser extent, railways are almost always visible particularly on the range scales below 2Km/radius.

The high definition of the radar is clearly evident in the views of coastlines and estuaries in Flights 21,23 and 24. Even with a scale of 5Km/radius, buoys and small craft can be distinguished and this is even more evident when the scale is increased to 2Km/radius as in Flight 24 where many yachts and similar vessels can be seen at anchor.

This standard of presentation suggests that coastguard applications might be a particularly suitable role for the Rotor Blade Radar.

This view is strengthened by the fact that a high scanning speed is a well established technique for the decorrelation and suppression of sea clutter.

CONCLUSIONS

Although the flight trials are not complete, sufficient data has been gathered to suggest that the wide field of view, high definition radar displays obtainable from the Rotor Blade Radar both over land and coastal regions would offer significant advantages for the operation of helicopters in darkness or bad weather. The equipment installation has been found to make very modest demands in terms of space and location within the helicopter when competing with other avionics and the aerial and rotor blade assemblies have proved to be relatively simple to build and maintain and show good operating lives. It is believed therefore that a Rotor Blade Radar is an entirely practical and useful sensor for modern helicopters.

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Thanks are due to the Royal Signal and Radar Establishment for the illustrations used in the paper and for information on the blade aerial.

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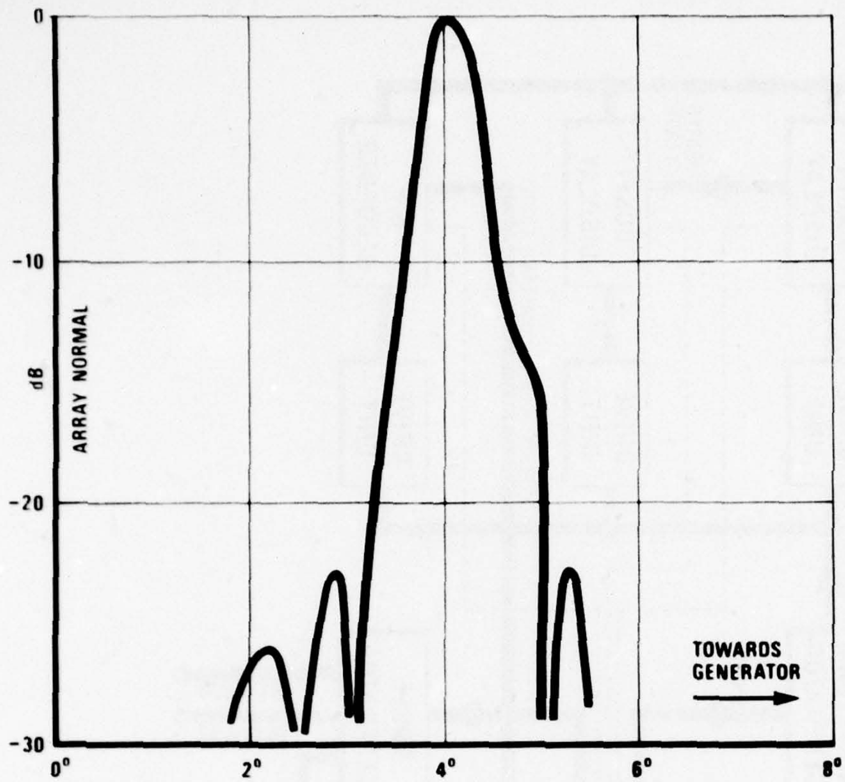
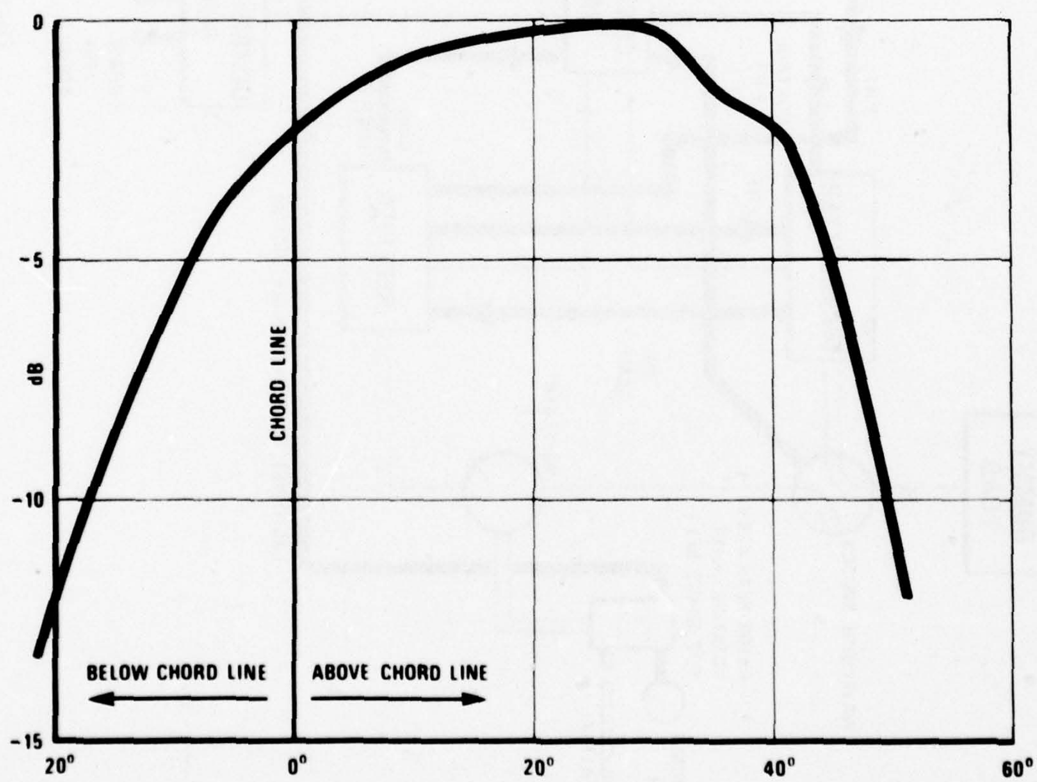
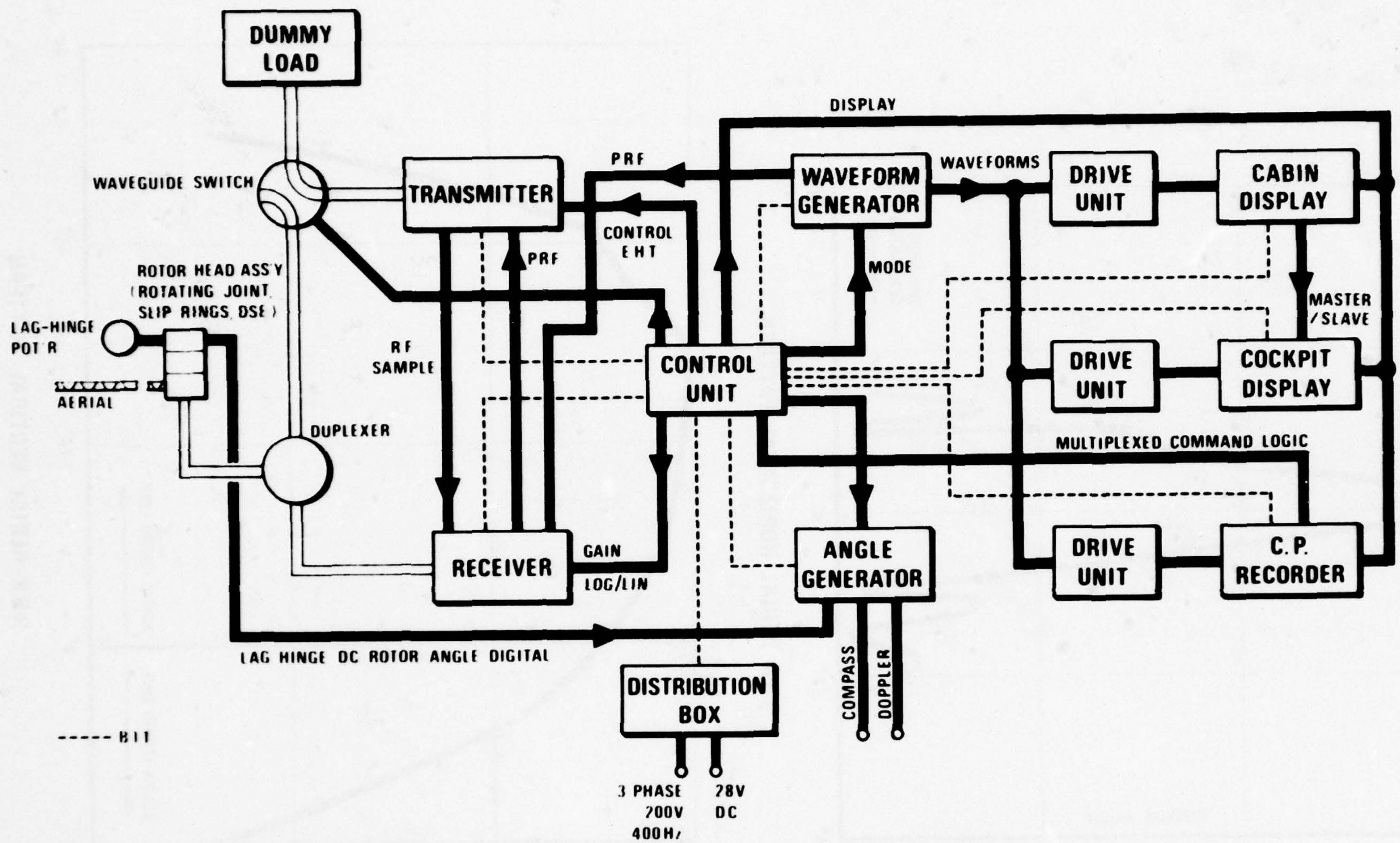


RBR AERIAL HORIZONTAL PATTERN



RBR AERIAL VERTICAL PATTERN

Figure 1



DUMMY LOAD
DISPLAY
WAVEGUIDE SWITCH
TRANSMITTER
PRF
WAVEFORM GENERATOR
WAVEFORMS
DRIVE UNIT
CABIN DISPLAY
CONTROL
EHT
PRF
MODE
ROTOR HEAD ASS'Y (ROTATING JOINT, SLIP RINGS, DSE)
MASTER /SLAVE
LAG-HINGE POT'R
RF SAMPLE
CONTROL UNIT
DRIVE UNIT
COCKPIT DISPLAY
AERIAL
DUPLEXER
MULTIPLEXED COMMAND LOGIC
RECEIVER
GAIN
LOG/LIN
ANGLE GENERATOR
DRIVE UNIT
C.P. RECORDER
LAG HINGE DC ROTOR ANGLE DIGITAL
DISTRIBUTION BOX
COMPASS
DOPPLER
BIT
3 PHASE 200V 400Hz
28V DC

Figure 2

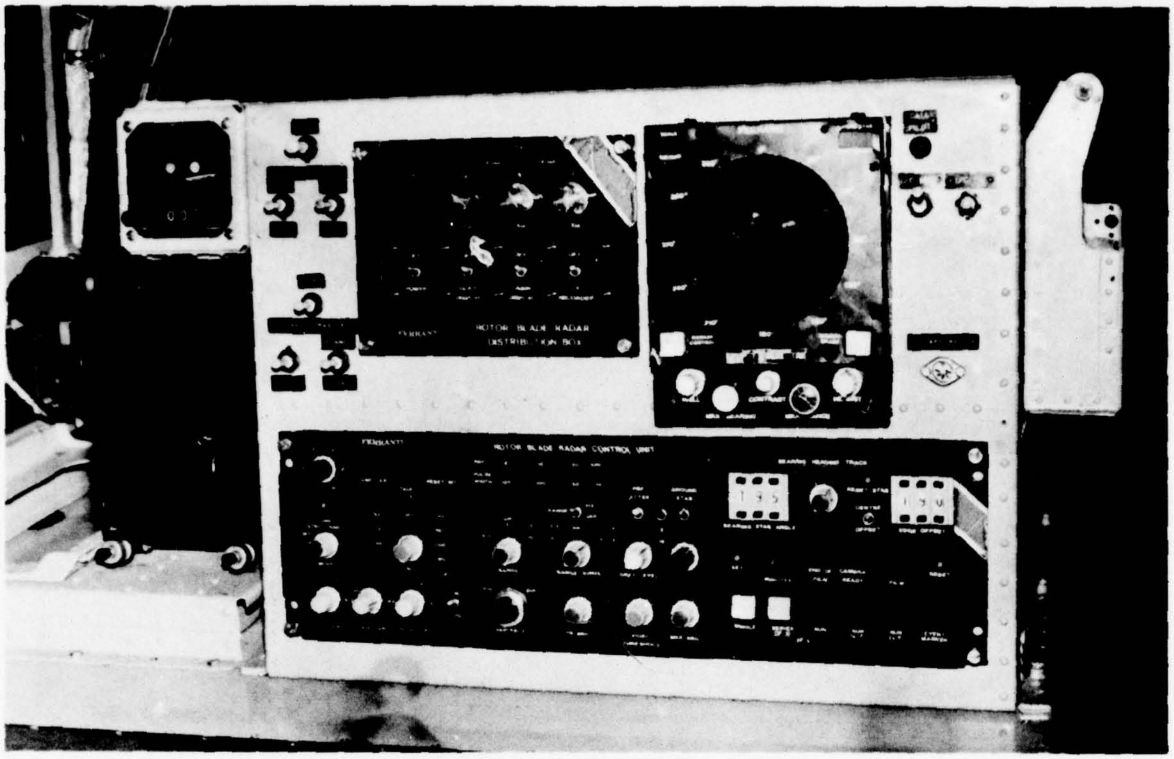


Figure 3

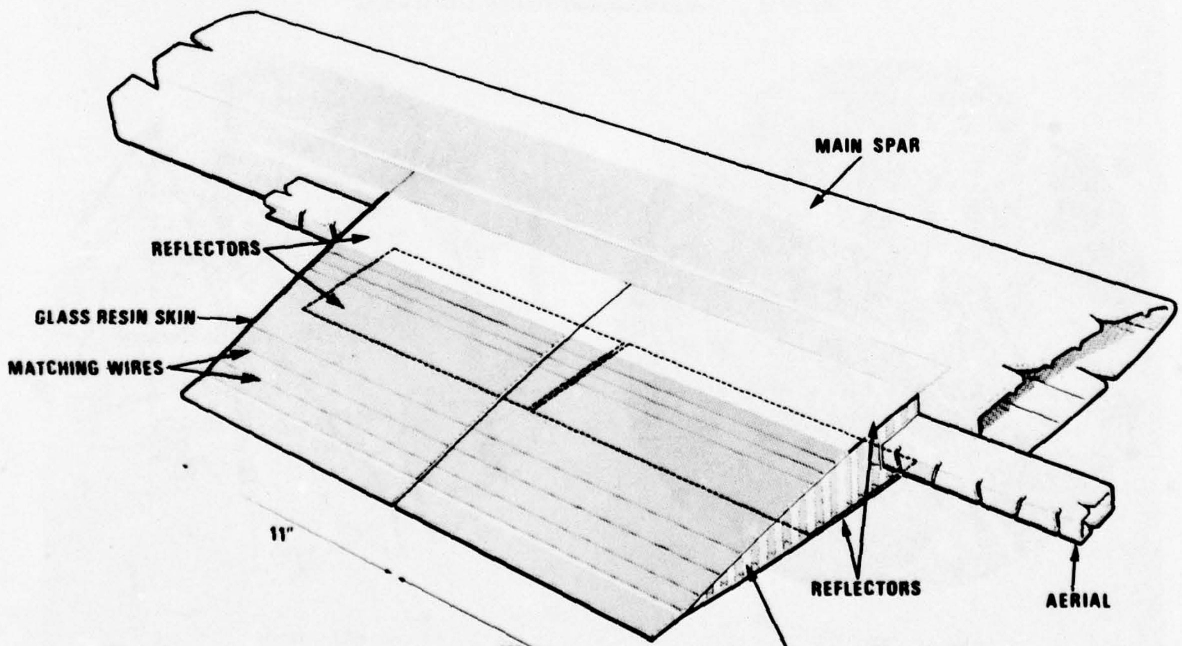
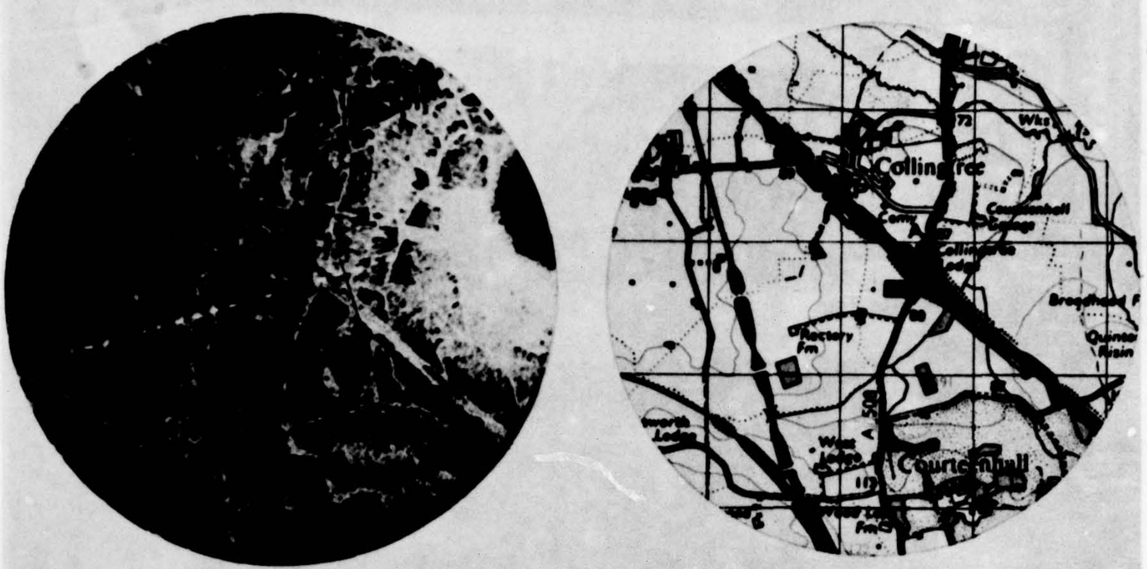


Figure 4

FLIGHT 18 10/10/77 INTERSECTION 15 OF M1 MOTORWAY (NORTHAMPTON)

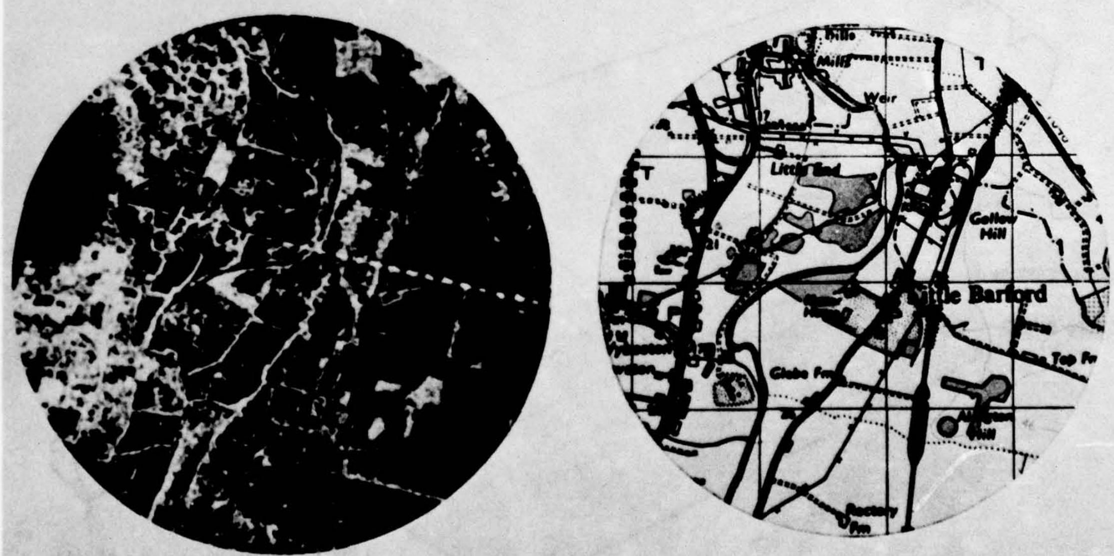


Height 350'
Scale 2 km/radius

P.R.F. 20 kHz
Pulse 50 nsec

A

FLIGHT 18 20/3/78 BARFORD POWER STATION (NEAR ST NEOTS)

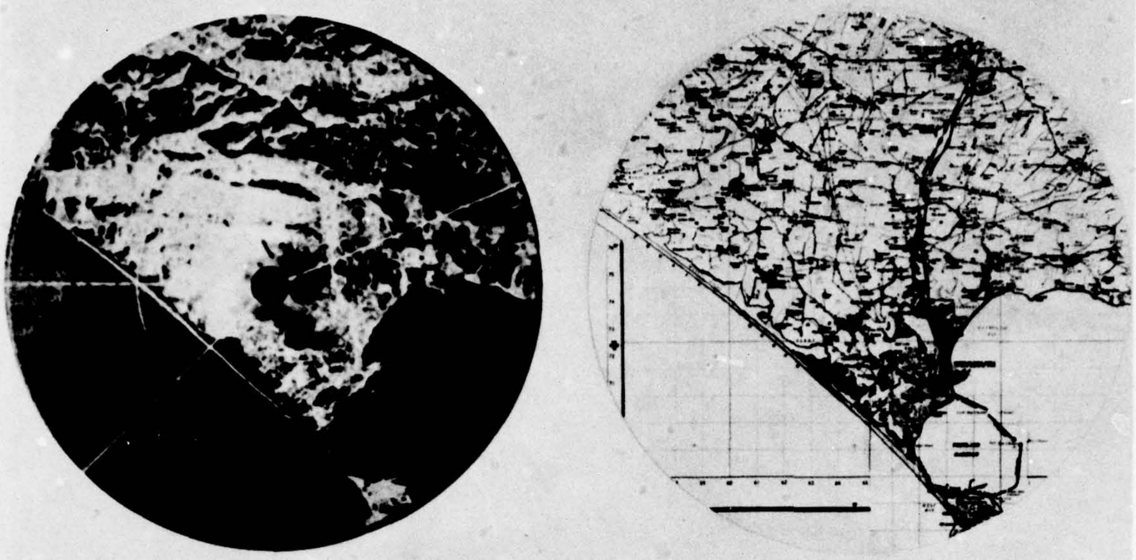


Height 350'
Scale 2 km/radius

P.R.F. 20 kHz
Pulse 50 nsec

B

FLIGHT 21 14/4/78 CHESIL BEACH, WEYMOUTH, PORTLAND HARBOUR

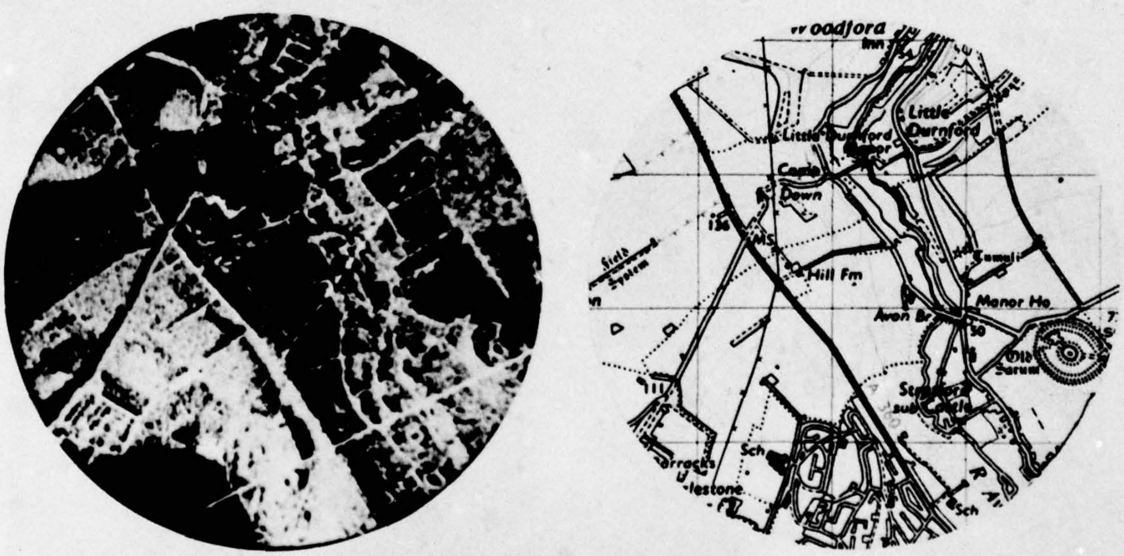


Height 1500′
Scale 10 km/radius

P.R.F. 10 kHz
Pulse 100 nsec

C

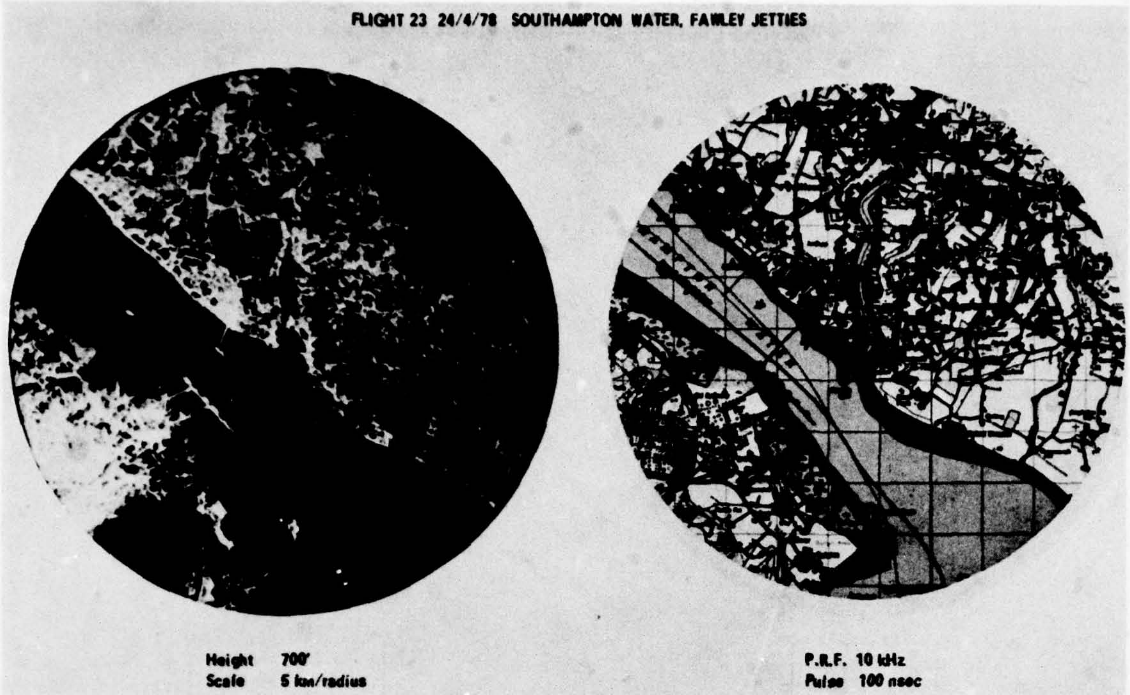
FLIGHT 21 14/4/78 OLD SARUM HILL FORT (SALISBURY)



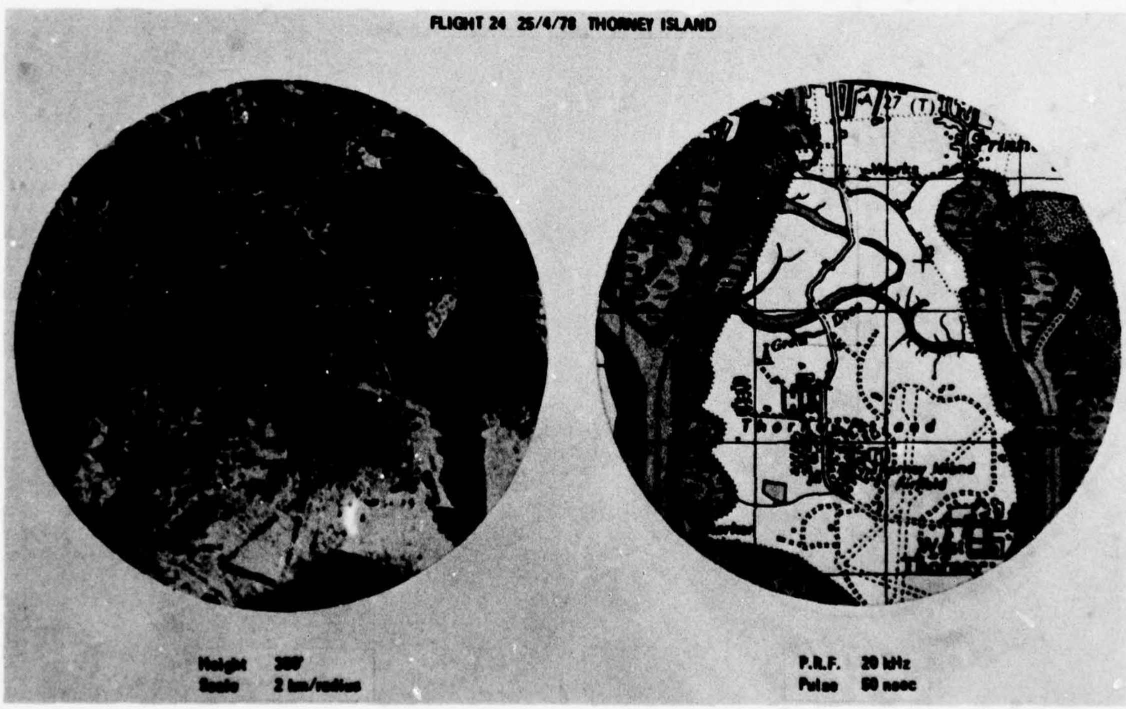
Height 350′
Scale 2 km/radius

P.R.F. 20 kHz
Pulse 50 nsec

D


FLIGHT 23 24/4/78 SOUTHAMPTON WATER, FAWLEY JETTIES
Height 700'
Scale 5 km/radius
P.R.F. 10 kHz
Pulse 100 nsec

E


FLIGHT 24 25/4/78 THORNEY ISLAND
Height
Scale 2 km/radius
P.R.F. 20 kHz
Pulse 50 nsec

F

DESIGN PROCEDURE FOR AIRCREW STATION LABELING SELECTION AND ABBREVIATION*

Patrick M. Curran, Ph.D.
Commander, MSC, USN
and
Norman E. Lane, Ph.D.
Commander, MSC, USN
Aircraft and Crew Systems Technology Directorate
Naval Air Development Center
Warminster, Pennsylvania 18974 USA

SUMMARY

Guidelines for crewstation labeling contained in several military documents are subject to diverse interpretation, thus leading to inconsistent labeling. The purpose of this paper is to describe a design procedure for the selection and abbreviation of labels for crewstation controls and displays. It is asserted that this improved procedure for increasing information flow between an operator and the systems/subsystems with which he is engaged will contribute to improved mission effectiveness, safety, and training by reducing operator errors, response times, and learning times.

The design procedures set forth require the identification of specific operator involvement with displays and controls. From this identification, functional statements are prepared for each control and display with respect to what, where, when, and how the operator should act upon gathering information on a characteristic of a system/subsystem or component.

This report describes methods of preparing these functional statements, the usage of common labeling of associated controls and displays, as well as specific procedures for abbreviating display/control labels.

INTRODUCTION

In modern aircrew stations the increased employment of new sensor/display systems and sophisticated onboard computers have presented operators with increasing amounts of data in new and unusual formats. These evolutions have considerably altered the traditional man-machine information transfer process that existed in older aircraft. New equipment technologies greatly extend the human sensory, motor, and mobility range and provide aircraft/environment information and resolution of flight/tactical dynamics which would otherwise exceed aircrew capabilities. At the same time, they impose time-critical challenges to operators to cope with information generation capabilities far beyond the wildest expectations of information needs. To minimize this discrepancy between data needs and data availability, it is essential that methods be developed to provide aircrew information acquisition, process, and transmittal in the most rapid, accurate, and reliable manner.

Limited space in airborne crewstations necessarily dictates some form of compression or abridgment of system information. Experienced pilots, in general, represent a relatively homogeneous group with respect to familiarity with aviation language. This group becomes less homogeneous when different levels of training, experience, or other types of crew members are included. This heterogeneity expands when support personnel (maintenance and servicing) are considered. The diverse needs of different personnel with varying functions brings about a breakdown in standardization goals. The development of a systematic procedure which will result in suitable labeling selection that can be efficiently utilized by this diverse group should be of major benefit in crewstation design.

Currently, various documents (e.g., MIL-STD-783B) covering the labeling of aircrew stations or airborne equipment serve as guidelines for making and identifying the various displays and controls in aircraft. These documents do not contain objective rules for the generation of suitable labeling, and provide inconsistent direction for the establishment of abbreviated crewstation labeling. A review of labeling in aircraft cited in an operational survey in 1968 (Reference 6) revealed a total lack of standardized procedures in the labeling information, composition, and abbreviation that have been derived in accordance with these document requirements. A recent unpublished survey of the crewstation labeling in four aircraft verified these earlier findings and concluded that applicable standards are:

a. subject to varying interpretation;

b. do not result in consistent labeling structure; and

c. in the four aircraft surveyed, more than 50 percent of the abbreviated labels did not conform to the intent of the design criteria specified in these standards.

* Opinions or conclusions contained in this paper are those of the authors' and do not necessarily reflect the views or endorsement of the Navy Department.

In current aircrew stations, display, control, and panel labeling provides too much or too little detail which is irrelevant and frequently misleading. Speed and accuracy of information acquisition and required control response, even by experienced aircrew, is severely affected by this improper device labeling.

Most often, the labeling is structured in engineering terms, which describe what a device is, rather than in terms of the function it is intended to serve. Thus, the operator is required to determine or recollect functions and, then, translate these into direct information inputs and outputs. These interface deficiencies have not stemmed completely from an absence of adequate control and display hardware technology. Rather, they have emerged from the lack of a systematic methodology for determining and using the functional operational requirements assigned to the operator and the detailed operator information and response requirements involved in his task. Display and control terminology from the hardware engineering development for each system (often carried over from previous systems) is used in place of device functions, reflecting a lack of human factors engineering analysis and evaluation during the design and integration phases, which could provide the appropriate information requirements for aircrew decision making and control actions. Previously cited studies of crewstation labeling have identified a number of common problems:

a. Unnecessary or redundant words in display, control, and panel labels leading to clutter or illegibility.

b. Labeling content that identifies the equipment and <u>not</u> the <u>direct function</u> that the operator is concerned with.

c. The use of inappropriate adverbs and verbs that do not provide explicit information or the desired function.

d. Individual labels for display and control groups when a single label for the group would be adequate.

e. An excessive use of acronyms and symbols in lieu of proper word descriptions that facilitate information acquisition and response.

Ideally, the labeling for all aircrew station equipments should present the complete word(s). This is not ordinarily practical given the critically limited crewstation space available for label presentation. Labeling abbreviation will frequently be required; the shortening of the label should be minimal within the limits of the available space. Recent U.S. Navy studies (unpublished) have shown that the relative time to read labels is a function of the extent of their abbreviation. More extensive abbreviations require more time for the operator to absorb the intended original-word information. Figure 1 depicts this relationship.

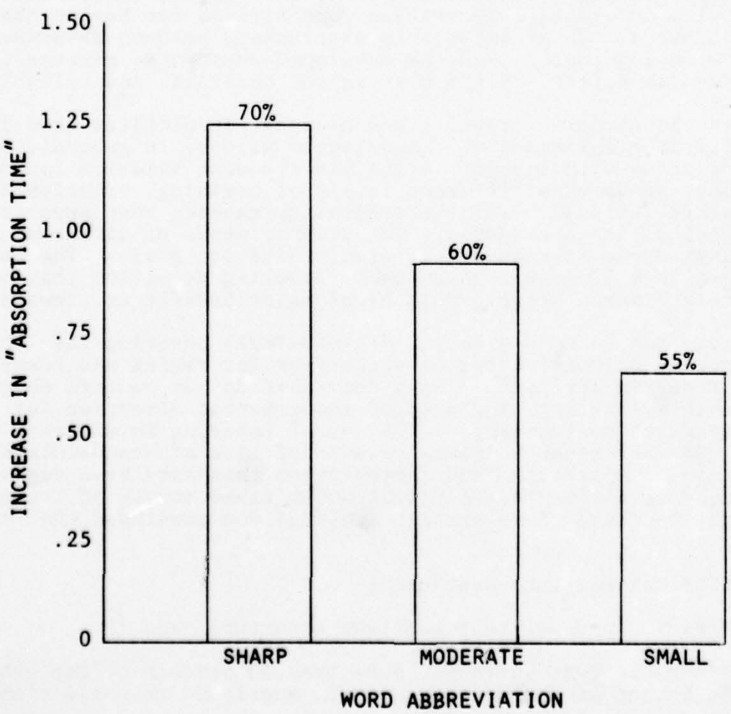


Figure 1. Effect of Word Abbreviation on Information-Processing Time

DEFINITIONS

For the purpose of this report, the following definition of terms apply:

Abbreviation - a shortened form of a word(s) used in the label.

Acronym - a word usually formed from the first (or first few) letters of several words.

Control - a device that permits operator inputs to the system/subsystem hardware. These devices include levers, knobs, switches, etc. that are either continuous or discrete. The continuous control has values and the discrete control has settings.

Display - a device that provides system/subsystem activity information or outputs to an operator. These devices will include dials, lights, readouts, radar scopes, etc. and may be continuous such as a fuel gauge, or discrete such as a warning light. A continuous device has values, a discrete device has settings.

Labeling - refers to the choices of size, type, and location of words, abbreviations, acronyms, and symbols to:

1. identify the information being conveyed from the system to the operator and back to the system from the operator; and

2. describe to the operator when and how the various displays and controls are to be used.

Symbol - a written or printed mark or letter standing for or representing an object, quality, process, or quantity that is usually a widely accepted and utilized information designator in one or more specialized fields.

System - the collective entity that is the major independent element (machine) which the operator is controlling (i.e., aircraft, helicopter, ship, submarine, etc.)

Subsystem - the collective components that provide operator capability(ies) (i.e., communications, navigation, flight control, etc.) to accomplish the system purpose(s).

LABELING REQUIREMENTS

In the crewstation design displays and controls (devices) are usually arranged on panels as functional device groups and/or sub-groups serving a common system or subsystem purpose. These groups are intended to provide the operator with all of the information and control required for the employment of various subsystem capabilities (i.e., flight control, propulsion management, navigation, communications, etc.). In most cases, the display panels that provide the status or directive information concerning a subsystem are located apart from the control panels that contain the operator action devices associated with these systems. This design practice is primarily due to the highly limited space in the crewstation and to operator considerations which dictate placement of displays in forward areas for rapid viewing, and control placement in areas of effective operator reach. As a result, it is essential to provide clear, concise, and consistent labeling to enhance rapid and accurate association between information acquisitions and appropriate control actions to overcome problems created by physical separation.

A major difficulty encountered in the design and integration of crewstation displays, controls, and panels is the selection and presentation of the information the aircrew will need to effectively accomplish their assigned tasks. The specific requirements that are identified when operator roles are established form the basis for defining displays and control requirements for throughput of information between the operator and the hardware he employs. During this phase of identifying operator interface requirements, operator functions (tasks) must be broken down into finer sub-tasks until all information the operator must absorb (from displays), process or transmit (through controls) is identified. It is from this information that labeling for both normal and emergency flight operations must be derived.

A major requirement of labels and legends is to identify for each device the information being transferred to and from the operator so that he does not have to rely on memory for this purpose. Proper labeling will greatly reduce the amount of time needed to train an operator as to where each information transfer point is located in his crewstation and will also increase his understanding of how his system may be controlled. A second and equally important function of labeling is to inform the operator when and how various information can be transferred.

The incorporation of computers in aircraft has resulted in an ever-increasing degree of system/subsystems integration with a concomitant increase in quantity and complexity of information transfer between operators and the system. In earlier less-integrated systems, the relationships between information and control responses were fairly well-defined and comprehended. The current emphasis on integration has resulted in a major change in the man/equipment interface. Advanced systems with tabular message formats, tactical and strategic situation plots, and multiple sensors make it essential that an effective and consistent labeling methodology be developed.

LABELING DERIVATION PROCEDURE

Purpose and Application

The purpose of this section is to present a systematic procedure for the determination of appropriate labels for aircrew station displays and controls. It is anticipated that this procedure can also apply effectively for operational and emergency instructional placards.

It is intended that these procedures be applied to major subsystems (e.g., communications, flight control, navigation, radar, sonar, etc.) during their integration into crewstations in the aircraft design and development program as well as individual subsystem developments for which no specific aircraft application is immediately contemplated. In this latter development case, application of the procedures should result in display/control/panel labeling which will minimize change due to different labeling philosophies when integration with a specific aircraft platform is made.

Further, where possible, this labeling procedure should be employed in conjunction with analytical methods that serve to determine the required panel space for displays and controls as a function of information content, criticality, and frequency of usage (CUBITS) (Reference 7). During this design process, adequate panel space for the most appropriate labeling should be allotted. Also, since crewstation panel space is normally at a premium, it will be desirable and, in many cases, necessary to consider the substitution of shorter words (e.g., "SEND" vs. "TRANSMIT) without sacrificing meaningfulness for the labels selected in accordance with the procedure specified herein. The Aircraft Cockpit Environmental System Control Panel shown in Appendix A was designed with the consideration of the above factors.

Types of Information

Display and control labels should be derived from the following types of information:

a. Entity Terms - The word or word combinations which unambiguously identify the physical element (system, subsystem, component, device, etc.) about which information is to be transferred to or from the operator.

b. Attribute Terms - The general and specific features or characteristics of the entity which is being presented by the display or changed by the control.

c. Setting/Scale Terms - The word, word combinations, and/or numerical designators which identify the particular settings, range of values and/or the units associated with the displayed information or control action.

d. Time of Usage Terms - The words which identify the particular mission segments or situations when the specific display/control should be used.

e. Method of Usage Terms - The word or word combinations which explain how the operator uses the display or control device to gather or change the information of concern.

f. Information Source Terms - The word(s) which identify the source from which the information being transferred is derived.

Development of Functional Statements

As a first step, operator involvement with each display, control, and display/control grouping during normal and emergency phases of flight should be determined and the appropriate types of information, as defined in the previous section, should be compiled. To obtain the pertinent key information listing, functional statements of the specific information or transmission/change the operator is to obtain from the displays or perform on the controls shall be prepared. In these statements, answers to the questions of what, when, where, and how the operator is involved with the display/control should be clearly and readily apparent. With displays, the operator is concerned with gathering information on the state/setting of some feature or characteristic of something. With controls, the operator is concerned with taking action on, or changing, the state/setting of some feature and/or characteristic of something. The "entity," "attribute," and "setting/scale" terms provide the what information; and the "method-of-usage" is the how information.

In these statements and in the subsequent key information listing (Table 1), the appropriate term words should be arranged so that each succeeding word is part of the term (terms) identified by the preceding word to facilitate the analysis. Where applicable, these statements shall include all the types of information terms as defined above; examples of the above procedure may be noted in each of the defined terms in the following paragraphs and in Appendix A. Following the preparation of the functional statements, the appropriate key information terms shall be extracted and presented in tabular form as shown in Table I.

Entity Terms

The words which identify the information transfer involved in the display/control utilization shall be selected and arranged so that each succeeding word is part of the e-quipment identified by the preceding word (e.g., "Helicopter Radio Receiver"). These terms

TABLE I

DISPLAY/CONTROL INFORMATION LIST

Display/ Control Device	Entity	Setting/Scale	Time-of-Usage	Information Source	Method-of-Usage
(Indicator, Toggle Switch, Pushbutton, etc.)	(System, Subsystem, Component, Device)	(Discrete, Continuous)	(Mission Segment, Emergency, Primary, Alternate, etc.)	(None, if obvious)	(None, if obvious)

identify the control/display requirement for the operator's acquisition or selection of the "Frequency" which is the attribute of the "Receiver" entity-component which is part of the "Radio" subsystem which in turn is part of the "Helicopter" system. In the example shown in Appendix A, the entity term for the "Aircraft Cockpit Environment Control Panel" describes the panel as a grouping of controls for the management of the "Cockpit Environment" subsystem which is part of the Aircraft.

Attribute Terms

Words shall be selected that describe what is being changed or the state of the characteristics (attributes) of the physical element (entity) of concern. Examples of attributes are the "sweep rate" of a radar antenna, the "attribute" of an aircraft, the distance "range" selection for a sonar display control. In the example in Appendix A, the "pressure" is the attribute associated with the cockpit air.

Setting/Scale Terms

The words selected should describe the discrete and/or continuous aspect of the particular display or control. In the case of the cockpit temperature selector control in Appendix A the setting terms are "cool" and "heat" with a continuous operator selection capability between these two limits.

If numerical units/symbolic designator(s) are to be used, particularly for display scales, the proper units shall be determined from the required reading and/or setting accuracy requirements associated with each display and control. Numerical/symbolic designators that are widely accepted or easily understood should be selected (e.g., 1000 meters, % RPM, Hz, etc.). In Appendix A the units selected for the cockpit pressure presents cockpit altitude in 500-foot increments from 0 to 50,000 feet. The unit labeling is "Feet-Equivalent Altitude."

Time-of-Usage Terms

Information concerning times during which a particular device or device-setting should be used may be important to the operator. Usually these times correspond to various mission segments (e.g., start-up, taxi, take-off, climb, attack, etc.). Appropriate words for this purpose should be selected.

Words shall also be included to describe the specific situation(s) of designated display and control functions, so that upon setting up the control device properly, the system performs all the required system functions to enable the various subsystems involved to be adjusted for that mission segment.

Where control devices and control settings are used only in emergency situations, the selection of the word "emergency" is desired. It is also desirable to use the word "normal" as part of the control setting labeling to distinguish it from either the "emergency" situation or from other special situations which are not emergency ones but rarely-occurring ones. However, "normal" should not be used for more accurate words such as "primary" or "in-flight."

Information Source Terms

Where operator confusion may exist, words which describe the source of the information of concern should also be selected (e.g., altitude obtained from the radar, distances from sonar, etc.). In Appendix A the various sources of cockpit air are listed and provided for operator control.

Method-of-Usage Terms

In most cases, the method of display/control is readily apparent to the operator by the display/control configuration. Where specific action is not readily apparent (e.g., "pull," "push," or "turn" to accomplish a required action) word(s) describing method of use should be selected and employed in the labeling. Also, where operator action procedures in the use of displays/controls is complicated or not readily apparent to the operator, describing word structures shall be formed and appropriate labeling presented to the operator.

In those cases where word structures will be required to inform the operator how he is to employ the display or control, modifying or action words such as "a," "the," "it," "to," "turn," "set," "adjust," "control," "enable," etc., shall not be selected when they do not provide required information or are obvious to the operator.

Where the title of the device, and the appropriate operator action is required in the labeling, the verb (describing operator action) selected should convey the most direct and maximum information possible to: 1) obtain the shortest possible verb-entity/attribute term(s); and 2) convey the most direct information concerning the required operator action(s) in the verb.

Displays and controls are usually grouped into major assemblies called panels. These panels are considered to be discrete components which provide the operator with information (through displays) and action (through controls) required to fulfill a specific subsystem capability. There may be one or more than one display or control panel grouping for the various characteristics (attributes) associated with the specific system/subsystem (entities). An example of these multiple display/control groupings is a communications subsystem which has several major component divisions (UHF Radio Transmitter, UHF Radio Receiver, Automatic Direction Finding, Voice Security, Intercommunications, etc.). In addition, dual independent equipment (versions) for one or more of these entity/attributes may be provided. In these cases, they should be grouped and labeled No. 1, 2, 3, etc. The type of information to be listed for these panels for labeling consideration should include descriptive titles of the component (entity) that forms the panel and the major attribute associated with the component. In Appendix A the entities are the (aircraft cockpit air) and the attributes are the (temperature and pressure).

LABELING SELECTION

For each display, control, and display/control group (this may be either a major component in a subsystem or sub-groups of displays or controls in a large group such as a control panel) the compiled list of types of information (Table 1) are used as the basis for appropriate labeling selected in accordance with the following procedures and requirements.

GENERAL REQUIREMENTS

Labeling should not include words that identify the system, subsystem, component, or device when the shape or other obvious feature conveys this information.

Where possible, the labeling shall contain words that identify the appropriate display or control entity/attribute (e.g., Radio Frequency, Fuel Flow, Compass Heading, Antenna Sweep Rate, Cockpit Temperature, etc.). Major equipment titles should not be used (e.g., Air Conditioner).

The same words shall be employed in the labeling of displays that are used to label their associated controls.

Where possible, only singular forms of the labeling word(s) shall be used.

Punctuation marks such as periods, hyphens, apostrophes, and slants should be minimized; if used, they should be consistent and should correspond to the punctuation meanings under the abbreviation section.

DETAILED REQUIREMENTS

From the types of information listing prepared for each display, control, and display/control grouping, appropriate labels shall be selected as follows:

DEVICE GROUP LABELING

The label for the display/control group should include the entity and attribute word(s), where applicable, that identify the major component and characteristic the operator is involved with. Where the component(s) (panel) title is very obvious with respect to the system or subsystem involved, the entity term should not be included in the label. For example, in the display for aircraft airspeed, "Aircraft" should not be included unless confusion with the speed of another entity such as target, ship, etc. can occur. In the example shown in Appendix A, the entity term of "Aircraft" was not selected because "Cockpit" sufficiently identifies the aircraft interior location and the entity term "Air" when used with cockpit is the major feature of the aircraft interior environment.

In many cases, there are display and control sub-groups within a panel that can be identified with an adequate label that will apply to all of the devices in this group. An analysis of the listing for each display/control will reveal this label-device relationship. In these cases, one appropriate label should be selected for the title of all of these related devices. For example, if a number of displays and/or controls which display or allow operator action for the testing of devices A, B, C, D, etc. a "TEST" label with appropriate bracketing shall be used in lieu of separate "TEST" labels for each display or control as shown below:

Generally, only entity and attribute terms for the display and control groups will be in the types of information listing. However, where applicable, "setting/scale," "time-of-usage," "information source," and "method-of-usage" terms should be included. In these cases, as shown above and in Appendix A, the labeling shall include clear, concise, and consistent word(s) which describe these terms.

Individual Display and Control Labeling

Labeling requirements for individual displays and controls consist of title labels to identify the display or control and discrete or continuous setting labels or scales for operator action. Appropriate words that describe the attribute and/or the entity shall be selected for the title label. Where the device grouping (panel) title adequately identifies the equipment, the entity term should not be included in the title. This label should also include the "setting/scale," "time-of-usage," and "method-of-usage" if this information is required and is not to be included in the displayed scale or control setting labeling.

For the display scale or control setting labeling, the appropriate entity, attribute, settings/scale, time-of-usage, and method-of-usage word(s) from the types of information listing should be selected in accordance with the requirements outlined in the previous paragraphs.

Labeling Design

The style, size, and arrangement/location of the letters, numbers, and symbols used in the labels should be in accordance with the requirements of NATO STANAG 3329AI, "Numerals and Letters," and U.S. Military Specification MIL-M-18012, Markings for Aircrew Station Displays.

LABELING ABBREVIATION

Purpose and Application

The purpose of this section is to establish requirements and guidelines for abbreviation of labeling on aircrew station displays and controls. The guidelines presented here attempt to reduce the time required for interpretation of abbreviated labels by providing consistent rules for the formation of abbreviations.

General Requirements

Where possible, sufficient display and control panel area shall be provided to eliminate the need for labeling abbreviation.

Since labeling abbreviation shall only be employed where there is insufficient space to present the complete labeling word structure, this may result in the same label being presented in both complete and abbreviated form on the same display/control group.

When abbreviation is required for two or more identical labels with a display/control group, the same abbreviation shall be used for all such abbreviated labels. The same label shall be used for the singular and plural forms of the word(s).

Punctuation marks in labels should be avoided except for selected marks used to indicate abbreviation structure in the circumstances described below. These selected marks are: 1) the dash (-), used to identify compound-word abbreviations; 2) the slash (/), used to indicate rates and ratios; and 3) the period (.), used to separate letters or acronyms.

Detailed Requirements

Short Words - Single-word labels of four letters or less should in general not be abbreviated. Such short words appearing in multiple word labels may be abbreviated when an approved abbreviation exists and when context is sufficient to ensure that this is entirely clear.

Approved Label Abbreviations

Single word and word combination abbreviations, particularly those associated with critical and/or emergency display and control uses, that have received long standing use and wide aircrew acceptance and generally conform to the abbreviation requirements specified herein should be compiled and considered for adoption in future aircrew stations. A format for this compilation is shown in Table A-II. The use of these abbreviations should be based on the lack of sufficient space to present the complete labeling structure.

Formation of Labeling Abbreviations

Formulation of abbreviated labels shall be based on available panel space and on the guidelines described below. Where abbreviation is required, the longest abbreviation consistent with available space should be used, subject to constraints imposed by other uses of that label on the same display/control panel; e.g., requirements for standardized abbreviations.

Single-Word Labels

Although "approved" abbreviations are contained in several U.S. Military standards and specifications, many of these are inconsistent and are not efficient in the sense of transmitting information to the operator. The basic goal of abbreviation is, for a given amount of space, to maximize the information input to the operator and thus to minimize the time required to translate the abbreviation into the parent word(s). Among the principles which help achieve this goal are: 1) consonants carry more information than vowels; 2) letters toward the beginning of a word carry more information than those toward the end; 3) letters toward the beginning of a syllable carry more information than those toward the end; and 4) specific suffixes on a label are of little or no utility in conveying the function of the control or display being labeled. The rules which follow incorporate these principles. Abbreviations derived from these rules should be given precedence over label abbreviations derived in arbitrary or inconsistent ways.

ABBREVIATION PROCEDURES

General Rules

1. Break word into its component syllables.

2. Delete any suffix listed in Table II

3. Retain first syllable intact until all succeeding vowels and consonants have been deleted.

4. Vowel Deletion Operations

   Stage I: The following operations do not apply, at this stage, to those vowels which are themselves syllables, or are the first letter of syllables.

   a. Go to the right-most (last) syllable of the word, delete the right-most vowel of this syllable.

   b. Proceed to the next syllable (to the left) and, again, delete the right-most vowel. Continue this leftward-moving syllable, right-most vowel deletion process until the first syllable is arrived at. Retain first syllable intact.

   c. Again, go to the right-most syllable of the word and delete the right-most remaining vowel. Do this until only those vowels which are themselves syllables, or are first letters of syllables, remain. To repeat, leave the first syllable intact.

   Stage II: Should further contraction be necessary, delete in a left-moving process remaining vowels, except for the vowels in the first syllable.

5. Consonant Deletion Operations

   At this point, the remaining word structure will consist of the intact first syllable and remaining syllables containing one or more consonants.

   Those syllables that contain only one letter shall be retained until multiple-consonant syllable structures have been reduced to single-consonant syllables. This will be accomplished by the following procedure.

   a. Look for syllables containing three or more consonants starting from the last of these syllables, delete the left-most consonant (provided it is not the first letter of the syllable), continue this process until the first syllable is reached. Repeat this process until remaining word structure is reduced to the first syllable intact, and remaining syllables consisting of one or two consonants. Delete the right-most consonant of any two-consonant syllables, moving in a right-most to left-most fashion.

   b. Now only the first syllable and single-consonant syllables remain. Delete the remaining single-consonant syllables in a right-most to left-most fashion until only the first syllable remains.

6. First Syllable Operations

   If further abbreviation is necessary, operate on first-syllable vowels and consonants as outlined previously until first letter is reached and retained.

TABLE II

SUFFIX DELETION LIST

1) -s	14) -ure	27) -able
2) -izer	15) -ance	28) -ible
3) -ating	16) -ence	29) -er
4) -izing	17) -ince	30) -ing
5) -ization	18) -ment	31) -ic
6) -ation	19) -ent	32) -al
7) -ator	20) -ical	33) -ate
8) -icate	21) -ity	34) -or
9) -ize	22) -ater	35) -nes(s)
10) -matic	23) -ency	36) -eou(s)
11) -atic	24) -ed	37) -u(s)
12) -mum	25) -ary	38) -y
13) -tude	26) -ion	

Note: 1. (-s) is dropped from endings of other suffixes
2. (-er) is not deleted for words of six or fewer letters
3. (-y) is not deleted if preceded by vowel, b, d, or p

Appendix B gives an example of the use of the above procedure.

Compound-Word Labels

Frequently labels will consist of words which, although written and treated in normal usage as one word, are actually compounded from two or more complete word or words and prefix modifiers. Examples are such words as infrared, megahertz, etc. Since the information contained in these compounds requires both words, the deletion of letters on a right-to-left basis is sometimes inappropriate. Such compounds require special handling. The general rules for compounds are: first, avoid abbreviation if possible; second, attempt abbreviation as if the compound were a single word. If this results in an ambiguous or unrecognizable abbreviation; third, abbreviate each part of the compound separately and combine with a dash (-) (one-stroke width). Thus, under approach two, "infrared" becomes "INFRR," "INFR," or "INF;" under approach three, "INF-RD" or finally "I-R." Similarly "afterburner" becomes "AFTBRN," "AFTBN," or "AFTB," then "AFT-BRN," "AF-BN," or "A-B." Note the substitution of the (-) for the symbol (/) previously used. The latter mark is reserved for rate and ratio abbreviations.

The distinction between compound-word and multiple-word is not always clear. The two differ primarily according to whether the compound word is treated in common usage as a single word. For handling multiple-word labels, see section entitled "Multiple-Word Labels."

Labels Indicating Quantity and Units of Measure

Words which are compounds composed of a prefix indicating quantity and a word indicating units of measure require special notation for abbreviation. Typical prefixes of quantity are "milli-," "micro-," "deci-," "centi-," and "kilo-." Typical units words are "meter," "gram," "Hertz," "volt," and "ampere." Rules described in "Single Word Labels" will not provide satisfactory abbreviation for these quantity unit compounds since the units term will be deleted before the quantity terms. Optimum abbreviations will require a special notation system which is only partially completed.

Whenever space permits, labels involving quantity-units compounds should be written out in full. Where abbreviation is essential, use only well-known and widely-recognized notation. Avoid inconsistent usage (as in mm for millimeter and mfd for microfarad). Retain NATO STANAG 3647AI Nomenclature in Aircrew Stations.

A special notational system for quantity prefixes which utilizes exponential notation is currently being developed. When complete, this notation could replace other guidelines for quantity-units compounds. The essence of this proposed new notation is the use of a subscript attached to the units abbreviation to indicate the quantity involved. The subscript has the meaning of "10 to the nth power," times the units of the abbreviation. Thus, if "meter" is abbreviated "M," millimeter becomes M_{-3}, (meter x 10^{-3}); centimeter, M_{-2}; kilometer, M_3. For "second," millisecond would be SEC_{-3}; microsecond, SEC_{-6}; and nanosecond, SEC_{-9}. One goal of this notational system is to avoid special Greek symbols for terms such as "micro" and to reduce confusion resulting from the use of one letter for multiple terms, such as "M" or "m" for "micro," "milli-," "Mega," "meter," etc.

Like all labels, those devoting square and cubic measures should be spelled out whenever possible. When abbreviation is required, two forms may be used. The preferred method, space allowing, is to precede the units abbreviation with "SQ" for square measure and "CB" for cubic measures. For more compact abbreviation, a superscript (exponent) may be attached to the units abbreviation. Thus, if inches is abbreviated "IN" square inches is SQ IN or IN^2, or, under the special notation proposed above, SQ M_3 or M_3^2.

Labels Indicating Rates and Ratios

Labels which denote units as a function of time (feet per second, revolutions per minute) or ratios of units (pounds per square inch, grams per square centimeter) will generally require some form of abbreviation due to space limitations. The symbol slash (/) should be used to separate individual units abbreviations, and should be read as replacing the word "per." Thus, "FT/SEC" and "LB/IN^2" (pounds per square inch). This is the only appropriate use of the (/) symbol.

Multiple-Word Labels

When space does not allow presentation of the full label, abbreviation of multiple word labels can be formed in one of two ways--individual word abbreviations and acronyms. For individual word abbreviations, each word of the label is abbreviated according to the rules for single words and combined, with spaces separating individual words. Thus "Power Amplifier" becomes "PWR AMP," "Cabin Pressure" might be "CBN PRES," "Alternating Current" might become "ALT CRNT."

An acronym is an abbreviation formed from the first (or first few) letters of significant words of the complete label. Acronym abbreviations can be indicated by individual letters either with a space (two-stroke width) or by a period (.) (one-stroke width). Separating by spaces is consistent with the rule for individual word abbreviations. "Alternating Current" might be "A.C.," or "A C," but not "AC;" "Outside Air Temperature" might be either "O.A.T.," or "O A T," but not "OAT."

Abbreviations formed from the individual word rule may combine any of the types of abbreviation in previous sections. "Shaft Horsepower" could be "SHFT HP," or "SHFT H-P," or "S HP," or "S H-P." Appendix C gives examples of abbreviations formed by the rules of this section and preceding sections.

Inappropriate Labels and Abbreviations

For systems already in existence, a lack of appropriate guidelines has resulted in labels and abbreviations which are less than optimal and which are frequently not in compliance with the spirit of existing documents and standards. Separate and independent development of systems and component subsystems creates an inconsistency of labeling across panels within a system and encourages an abundance of separate pseudo-acronym labels to describe individual system components. Frequently these labels are developed primarily for the pronounceability or buzz-word characteristics of their acronyms, rather than for the communication of functions in a fashion consistent with labels for other displays and controls of the total weapons systems. Typically, such labels include words such as "display," "indicator," or "control." Use of these terms is specifically counter to the guidance in STANAG 3705AI and U.S. Military Standard 1472B. To further complicate the labeling and abbreviation structure, components with the same function developed separately may have different labels. For example, both "Vertical Situation Display" and "Vertical Situation Indicator" are more appropriately labeled "Vertical Situation" without the display or indicator tags, and abbreviated labels should not reflect the VSD or VSI nomenclature.

A key purpose of this document is to provide guidelines sufficient to reduce or eliminate inappropriate and inconsistent labeling and abbreviation in future systems developments. A single poor label or abbreviation assesses a small, but definite, time charge on the operator in terms of increased familiarization and training time and an increased time to read and translate, and increases the likelihood of errors of operation which can cause accidents and system damage. These small inefficiencies, multiplied by thousands of system elements and thousands of operators, represent a substantial and wholly unnecessary degradation of total system effectiveness.

CONCLUSIONS AND RECOMMENDATIONS

The following conclusions and recommendations as a result of this study are made:

a. Modern aircraft weapon systems have considerably altered the traditional man/equipment information transfer process that existed in previous aircraft.

b. The operator/equipment interface in the determination of the information required by the operator and categorization of his control response must be optimized to meet the ever-increasing information transfer requirements in modern aircraft.

c. Current U.S. Navy human factor engineering guidelines and requirements in regulation documents do not result in clear, concise, and consistent aircrew station display and control labeling.

d. Ongoing research should be completed and the data from such compiled.

e. The design procedure and requirements for aircrew station labeling selection and abbreviation contained in this report, and from findings from ongoing research, is recommended as a departure point in developing a military standard for use in the design of effective display and control labeling in aircrew stations.

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APPENDIX A

AN EXAMPLE OF THE DESIGN PROCEDURE FOR LABELING SELECTION FOR AN AIRCRAFT DISPLAY AND CONTROL PANEL FOR AIRCRAFT COCKPIT ENVIRONMENTAL SYSTEM

Display

Five digit numerical readout.

Controls

The following controls are to be grouped on this panel:

1. Four, pushbutton ("ON"-"OFF") switches (interlinked so that only one switch is activated at any time)

2. One, two-position ("ON"-"OFF") switch

3. One, two-position ("ON"-"OFF") switch (one position is lever-lock guarded)

4. One, three-position (momentary "ON," "OFF," momentary "ON")

5. One, Thumbwheel Control (continuous)

To determine the appropriate labeling for these displays and controls, the following steps shall be taken:

A. Prepare operator functional statements on each display and control as follows:

1. Indicator

Read indicator to obtain aircraft cockpit air pressure altitude in 500-foot increments from 0 to 50,000 feet. Since the reading accuracy must be 500', the tens and unit scale markings are fixed.

2. Four Pushbuttons

Push one of four pushbuttons to select aircraft cockpit air conditioning/ pressurization air sources from aircraft external (ram) door, or both engines, left engine, or right engine bleed air valves.

3. Three-Position Switch

Move toggle switch from center off position to one of two positions momentarily to open or close the aircraft external air door to increase or decrease the air flow when the automatic engine air flow source is not being employed.

4. Two-Position Switch

Move toggle switch in one of two positions to either obtain automatic aircraft cockpit air pressurization altitude or dump the cockpit air pressure to exhaust air. Since this latter action can be critical under some situations, this position is guarded.

5. Two-Position Switch

Move toggle switch in one of two positions to either automatically maintain aircraft cockpit air temperature at the comfort level selected by another temperature control or manually select the cockpit temperature as airspeed and/or altitude changes occur.

6. Rotary Control

Turn control to increase (heat) or decrease (cool) the aircraft cockpit air temperature.

B. Analyze the operator functional statements and extract (or revise to more concise word(s)) the major entity, attribute, setting, information source, time-of-usage, and method-of-usage terms as shown on the attached Display/Control Information List (Table A-I).

C. From the functional statements and the information terms in the attached tables, identify those displays/controls that are related and have the same entity-attribute descriptive terms. These displays/controls should be located in close proximity to one another and one appropriate label title with appropriate bracketing over the related displays and controls shall be selected.

D. From the display/control information list for each display and control, select the most appropriate labeling for the device title and scales/settings. The display and control panel labeling extracted for this sample aircraft environmental subsystem is shown in Figure A-1. It is to be noted that the control for automatic or manual air temperature selection is directly related to the air temperature level selector control. Accordingly, a common label title for these controls is used.

The four pushbuttons used to turn on the air conditioning/pressurization and selection of the source of cockpit air are interconnected so that the activation of one disengages the previous selection. Accordingly, only the source term information is required for the setting labels of these controls as may be noted in Table A-I and Figure A-1.

It is also to be noted that the entity-attribute description of "Aircraft Cockpit Air" is contained in all of the individual controls in Table A-I. Since these controls are grouped in a panel, the panel title label selected is "COCKPIT AIR" and is not used in the label title for the individual display/control labeling. If these controls applied to other crewstations, the term "Interior" would be used in lieu of "Cockpit" in the group title.

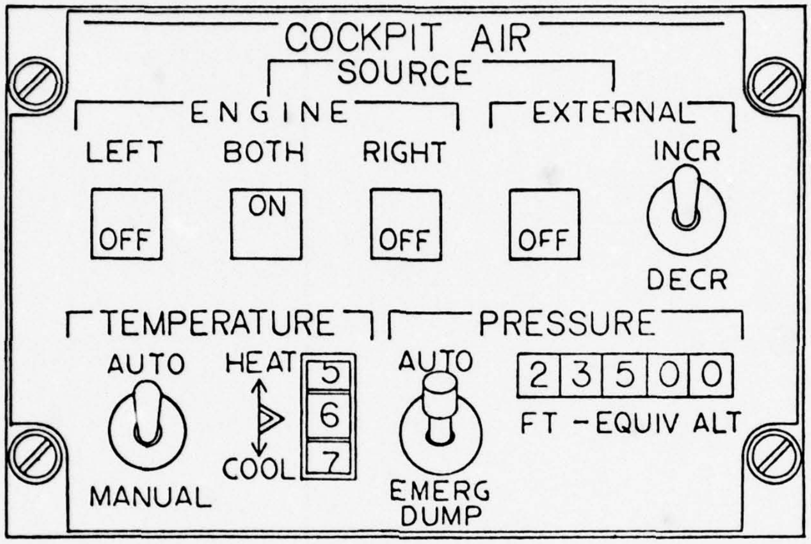

Figure A-1. *Aircraft Cockpit* Environmental System Control Panel Labeling

TABLE A-I

DISPLAY/CONTROL INFORMATION LIST FOR AIRCRAFT COCKPIT ENVIRONMENTAL SYSTEM

Display/ Control Device	Entity	Attribute	Setting/Scale	Time-of-Usage	Information Source	Method-of-Usage
Pushbutton	Aircraft Cockpit Air	Source	External (Ram) Door	Alternate	N/A	Obvious "Push"
Pushbutton	Aircraft Cockpit Air	Source	Both Engines	Normal	N/A	Obvious "Push"
Pushbutton	Aircraft Cockpit Air	Source	Left Engine	Alternate	N/A	Obvious "Push"
Pushbutton	Aircraft Cockpit Air	Source	Right Engine	Alternate	N/A	Obvious "Push"
Toggle Switch	Aircraft Cockpit Air	Flow	Increase Decrease	Alternate	N/A	Obvious
Toggle Switch	Aircraft Cockpit Air	Pressure	Automatic Dump	Normal Emergency	N/A	Obvious
Indicator	Aircraft Cockpit Air	Pressure	0-50,000 feet (equivalent) altitude)	Normal	N/A	Obvious "Read"
Toggle Switch	Aircraft Cockpit Air	Temperature	Automatic Manual	Normal	N/A	Obvious
Thumbwheel Control	Aircraft Cockpit Air	Temperature	Increase (heat) Increase (cool)	Normal	N/A	Obvious

TABLE A-II

FORMAT FOR

APPROVED LABEL ABBREVIATIONS FOR AIRCREW STATION DISPLAYS AND CONTROLS

WORD(S)	LABEL

APPENDIX B

EXAMPLE OF ABBREVIATION PROCEDURE USAGE

TRANSMISSION	- WORD
Trans·mis·sion	- SYLLABLE BREAKDOWN
Trans·mis·sion	- PRIORITY STEPS
Trans·mis·si~~o~~n	- Protect first syllable, right-most vowel of right-most syllable is deleted
Trans·m~~i~~s·sin	- Next syllable vowel deleted in a left-moving fashion
Trans·ms·s~~i~~n	- Return to right-most syllable, delete remaining vowel
Trans·ms·s~~n~~	- Right-most consonant of last syllable deleted
Trans·m~~s~~·s	- Right-most consonant of next syllable deleted in leftward-moving fashion
Trans·m·~~s~~	- Final consonant deleted
Trans·~~m~~	- Remaining consonant deleted
Trans	- First syllable
Tr~~a~~ns	- Vowel deleted
Trn~~s~~	- Right-most consonant deleted
Tr~~n~~	- Next right-most consonant deleted
T~~r~~	- Last consonant deleted
T	- Ultimate abbreviation

APPENDIX C

EXAMPLES OF ABBREVIATION FORMULATION FOR MULTIPLE AND COMPOUND WORDS

Label	Abbreviations
Airborne Early Warning	A.E.W. A-BRN ERLY WARN
Automatic Direction Finding	A.D.F. AUTO DRCT FIND
Auxiliary Alternating Current	AUX ALT CRNT AUX A.C
Undercarriage	UNDCAR UN-CAR U-C
Supercharge	SUPCHRG SUP-CHRG S-CHRG
Miles per Hour	MI/HR
Automatic Flight Control	A.F.C. A F C AUTO FLT CNTRL
Kilograms per Square Kilometer	KG/SQ KM KG/KM^2 or $G_3/SQ\ M_3$ G_3/M_3^2

SUBJECTIVE ASSESSMENT OF A HELICOPTER APPROACH SYSTEM FOR IFR CONDITIONS

by

H Howells

Flight Systems Department
Royal Aircraft Establishment
Farnborough
Hants
UK

SUMMARY

This paper outlines the contribution of subjective assessment techniques used in the flight evaluation of 3 azimuth approach guidance laws presented on crosspointers of electromechanical instruments in a fully stabilised helicopter. The flight evaluation conducted at RAE, Farnborough, UK, was reported in 1974 (Ref 1). Attempts were not made to simulate IFR conditions but the pilots were instructed to confine their attention to monitoring the instruments within the cockpit as a safety pilot was carried. The author's contention is that the use of subjective assessment techniques, in conjunction with the interpretation of radar plots and comments of an airborne trials observer, provide a type of information not obtainable from other sources. The technique, although used here on electromechanical instruments, would be equally applicable to an assessment of electro-optical displays.

INTRODUCTION

Equipment designers tend to consider the use of such subjective methods as interviewing or questionnaire techniques only as a last resort when it has not been found possible to measure or quantify the performance of the equipment. In terms of the normal academic and professional training of designers this is understandable so that if the designer has to resort to using some form of subjective assessment, the results are often disappointing to all concerned. Perhaps unknown to many designers of aircraft systems, persons engaged in certain fields of research and areas of study, such as clinical psychologists and even market researchers, often possess no other tool other than some form of systematic subjective assessment. In the hands of such practitioners, it is often a far more systematic and precise tool than when used by an equipment designer who lacks an appreciation of the various rating scales, checklist techniques etc available. In the field of aerospace R&D, it is usually possible to find a specialist with experience in the use of such methods and it is advisable to seek such professional help if available (Ref 2). In the case of assessment of equipment by test pilots, the insights into equipment performance which the experienced test pilot can offer are often not readily assimilated by the designer as he usually does not possess the systematic means of categorising and codifying such assessments. The flight evaluation by RAE test pilots of approach guidance information on crosspointers appeared to be a suitable context in which to apply a more systematic means of collecting such pilot opinion.

TRIALS APPROACH TASK

The aim of the trial (Ref 1) was to evaluate 3 types of azimuth presentation by assessing tracking accuracy from radar plots. The trial was based in the earlier discussion of the implications of operating helicopters in poor visibility (Ref 3). The 3 types of azimuth presentation were:

1 Raw guidance signals.

2 Rate aided display.

3 Composite law formulated to improve joining of approach path by constraining the intercept angle to a nominal 40° with an associated demanded bank angle.

These azimuth display laws were used in conjunction with raw elevation error signals and the 3 laws were examined for a fixed glideslope angle of 6° to establish what approach accuracies could be achieved and whether qualitative differences in pilot workload between the guidance laws were experienced. The approaches were initiated at 1000 ft, at a constant descent speed of 60 kn and with a break-off height of 100 ft.

DISPLAY CONFIGURATION

A Newmark type 4055K Attitude Director Indicator (ADI) and a Smiths type S6 Horizontal Situation Indicator (HSI) were fitted to the starboard instrument panel, to replace the standard instrument fit, and modified to display approach guidance information on crosspointers, data linked from a Bell SPN 10 lock-follow radar system. The ADI presented the usual attitude information together with elevation errors on the horizontal crosspointer and the azimuth guidance signal on the vertical crosspointer. Similarly, in addition to basic heading information on the HSI, the elevation displacement signal was assigned to the horizontal pointer moving vertically against graduated dots on the instrument face whilst the azimuth displacement pointer was located on a central carriage which moved left or right of a course marker.

The technique required for following an instrument approach with these displays was to fly the ADI central aircraft symbol and the circle on the HSI, towards the intersection of the crosspointers.

REASONS FOR USE OF SUBJECTIVE ASSESSMENT

The main means of assessment used by the team running the trial were tracking accuracies from radar plots together with comments of an airborne trials observer. A questionnaire previously used by the trials team during the early stages of the study was limited to such details as date, time of day, trials number, meteorological information and the pilot was encouraged to make any comments he wished to make on the conduct of the day's flight programme. Understandably, this method did not provide adequately detailed information to supplement the interpretations made of the tracking errors indicated by the radar plots. In a similar type of helicopter crosspointer evaluation conducted by NASA (Ref 4) where radar plots formed the basis of an approach aid assessment, findings from pilot opinion were reported but the method of opinion collection was not reported. The author was approached to obtain a more structural means of obtaining detailed information on pilot opinion for the RAE trials team.

BACKGROUND TO CHOICE OF SCALING METHOD

At the 18th Conference of the Human Factors Society (1974), Helm (Ref 5), pointed out that in an increasing number of system evaluations, the operators were required to make complex judgements using a variety of scaling methods on such factors as: operator workload, task difficulty and man-machine effectiveness. Helm indicated that perhaps the most familiar rating scale in the field of aviation was the Cooper-Harper Scale (1969 - Ref 6) with the McDonnel Scale (1968 - Ref 7) being probably the next most widely known, although many Human Factors specialists constructed their own variations of non-ordinal, non-adjectival rating scales (ie 10 cm lines). To obtain the final numerical rating in the Cooper-Harper Scale the qualities being assessed are considered ordinal in the sense that they are ranked in order of decreasing acceptability and adjectival in that adjectival phases are used to structure the decision process (Fig 1). The disadvantage of the Cooper-Harper Scale is that being an aggregate scale, it is not considered to be linear and so was not designed to facilitate averaging of ratings. The claims made for the McDonnel 7-point scale appear to overcome this limitation and so permit the performance of mathematical operations on the ratings due to the greater linearity of the scale. The object in this trial was to identify the range and type of problems associated with using the guidance displays and so produce recommendations for acceptance, modification or re-design of the equipment. Precise mathematical quantification of pilot opinion was not therefore a realistic goal and so the Cooper-Harper Scale was considered for its potential to form a basis or starting point for the study of a suitable method of subjective assessment which would suit the needs of the trial.

TAILORING THE BASIC SCALING PROCEDURE TO MEET TRIAL AIMS AND TO MAXIMISE TEST PILOT INVOLVEMENT

From anecdotal evidence it was known that test pilots often believed that their expertise in assessing airborne systems was not being fully utilised by the designers. To exploit this aspect, the syllabus of the Empire Test Pilots School, Boscombe Down, UK, was examined to discover just how the pilots were instructed to evaluate airborne systems. The course syllabus stressed that in spite of attempts to quantify aircraft handling qualities, pilot opinion still remained the most reliable source of available information on the subject. An explanation of the development and use of the Cooper-Harper Rating Scale for the evaluation of aircraft handling qualities was also contained in the syllabus together with a reference to its possible extension to the assessment of avionics systems.

It was this familiarity of the test pilots with the Cooper-Harper type scales which influenced the author in the choice of questionnaire format to be used. The Cooper-Harper Rating Scale was to be used for the overall assessment of each azimuth approach guidance law and the scale's sequential decision format for the questions on the more detailed aspects.

The attractiveness of using the decision format of the Cooper-Harper Scale was that as the pilots would be conversant with its operation, it would form an acceptable method for systematically collecting the detailed opinion over the questions covering the main areas of interest had been constructed. The list of questions was drafted from joint meetings between the pilots and the trials team where anticipated problem and interest areas were discussed. The draft was circulated for comment on accuracy and acceptability of terms used and question wording.

The pilot version of the questionnaire was used in the very early stages of the trial to identify any areas of bad questionnaire design. It essentially consisted of rows and columns on an A3 sheet. The first wide column contained the question being asked, followed by the forced choice responses, with sufficient branching as required to cover the broad dimensions of the particular question, (ie using the sequential decision aspect of the Cooper-Harper Scale with which the test pilots were acquainted). The next columns were reserved for the pilots to register their preferred choice category to each question by reading through the decision chain and ticking the final response which corresponded to their opinion to any given question. A column was provided for each azimuth guidance law under test that day so that the test pilot was making a comparative assessment in reading from the initial question, through the decision chain, before making his final response by ticking the appropriate row in the respective columns for the guidance law under consideration. Under the column heading for each guidance law, the pilot was asked to allocate a Cooper-Harper numerical rating as an overall assessment of the guidance law being considered. The final wide column, which extended to the edge of the paper, provided an opportunity for the pilots to qualify the force choice responses just made.

MODIFICATION OF QUESTIONNAIRE FOLLOWING PILOT RUNS

During the pilot runs of the revised questionnaire it was found that the pilots were a little reluctant to qualify their response categories they had ticked by supplying written details in the qualifying comments column provided but they were quite prepared to discuss such explanations at length. This reluctance to write down details had been experienced by the trials team on the earlier 'report sheets'. It had been thought that by structuring questions by which the pilots were to review the day's flight programme, this difficulty experienced in previous trials would be overcome. This had been achieved in broad terms in that by responding to the forced choice categories of the branched questions, the pilots were evaluating the system in far more detail than before. It had been noticed that whilst the pilots were completing their questionnaires up to the comments column, whilst not filling in the comments column, they were more ready to explain to the person administering the questionnaire, the reasons for their choices rather than writing them down. They were therefore asked if they had any objections to their comments being tape recorded. Having established that they possessed no such objections, it was decided that the comments column be deleted from the questionnaire and substituted by a tape recording of these comments. The procedure then became one of the pilots reading aloud through the questions and associated branched responses, ticking the final response categories corresponding to their opinion and the 'open microphone' on the tape recorder ensured that all their detailed comments and explanations were recorded at minimum inconvenience to the pilot. In this sense the re-designed questionnaire resembled more a self-administered structured interview schedule, with taped responses. (Fig 2).

POST DE-BRIEF ANALYSIS

Detailed content analysis was performed on the taped explanations for the boxes ticked at the end of each branched question. Findings were presented to the designers in the form of bar charts (Fig 3) of the Cooper-Harper Ratings for each pilot and each guidance law and for the responses to the forced choice questions (Fig 4) together with the summary sheets outlining the problems, limitations, advantages and recommendations from the content analysis of the taped explanations given by each pilot (Fig 5a, 5b). The pilots were far more enthusiastic in participating in the structured, taped de-brief which evolved from the shortcomings identified during the pilot run of the original self-administered questionnaire. Depending upon the experience of the test pilot, the taped de-brief lasted from ½ to 1½ hours. Incorporation of new pilots not originally assigned to the trial did not present any difficulties as an introductory leaflet explaining the assessment procedure to be followed was read by each pilot before participating in the taped de-brief.

This combination of varying levels of detail from:

1 Overall Cooper-Harper Ratings for each guidance law, Fig 3,

2 Bar charts of responses to the branched, forced choice questions to illustrate any difference in answers to the questions between different guidance laws, Fig 4,

3 Summaries of content analysis of the taped explanations to such questions, Fig 5b,

provided a hierarchy of explanations and detail so that the designers were able to decide which level most suited their requirements.

CONCLUSIONS

The encouraging aspect of the use of the subjective assessment method eventually evolved from the initial pilot runs with the more conventional self-administered questionnaires and finishing with the taped de-brief was that it allowed the pilot to describe in greater detail aspects of display design and operation without feeling unduly restrained by a rigid question-response format. After completing the day's flight programme, the pilot would be asked to consider all the approaches performed that day by addressing himself to the same questions, presented in the same sequence, with the same range of responses but with the possibility of describing in detail his comparative assessment by means of the taped de-brief. The immense detail obtained by this method was therefore easy to obtain and when reduced by content analysis, categorised into sufficient detail to provide the designers with easy to assimilate information on the problems, limitations, advantages, disadvantages and recommendations of the display system and its operation. When taken together with the radar plots and the comments of the airborne trials observer, a far more complete assessment of the system was possible than with the previously loosely structured sheet used on early occasions by the trials team. It is suggested that whilst this method of using a structured, taped de-brief was used to assess an electromechanical guidance display in a helicopter, the principle would be equally applicable for assessing an electro-optical aid in similar circumstances.

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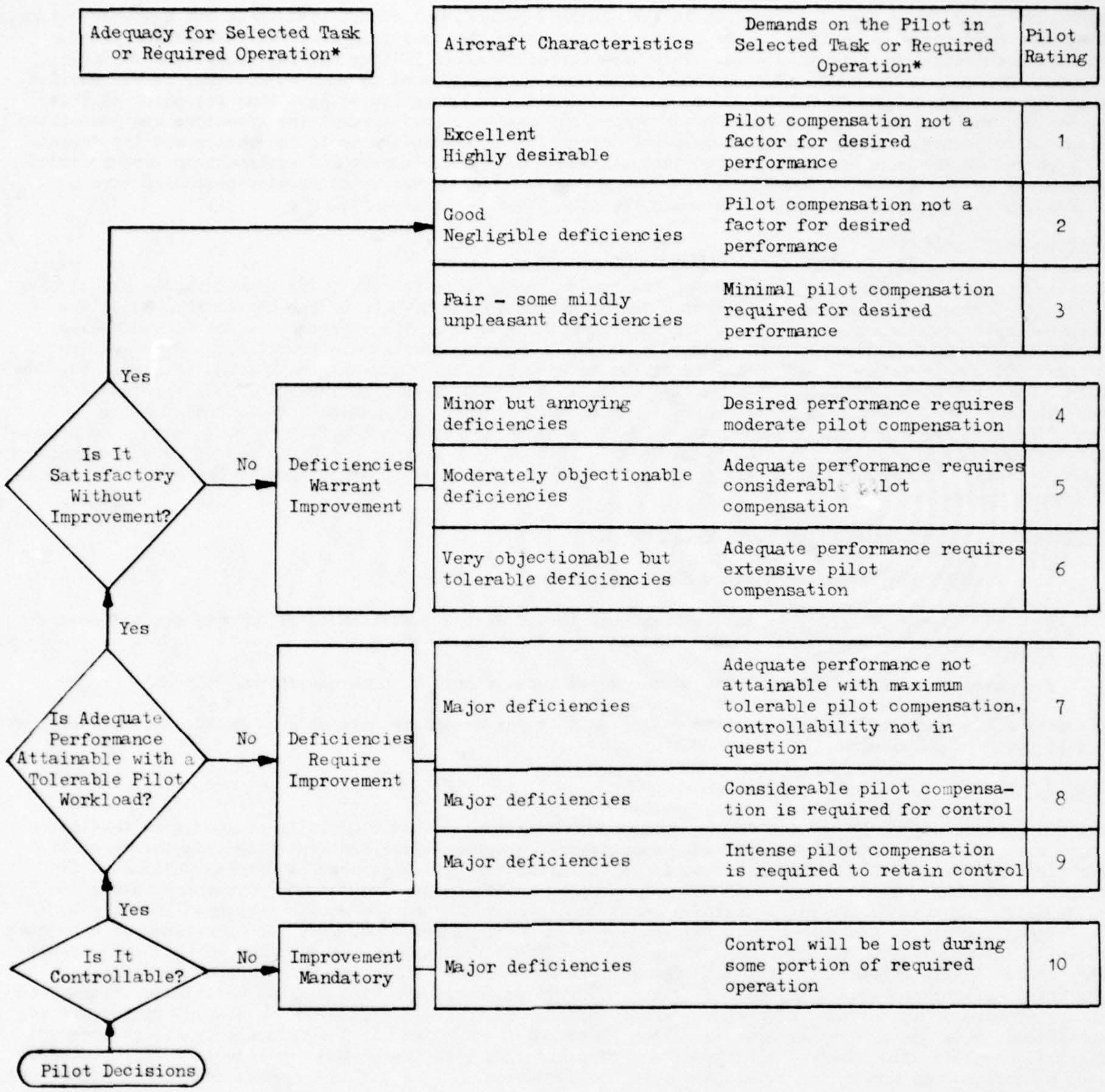

*Definition of required operation involves designation of flight phase and/or subphases with accompanying conditions.

Fig 1 THE COOPER-HARPER RATING SCALE
(For assessing aircraft handling qualities)

	QUESTION	ANSWER CATEGORY	COOPER -HARPER RATING / TRIAL CONDITION			
			AZIMUTH LAW eg RAW			
1	Was difficulty experienced in maintaining required: Glide Path?	NEVER				
		SOMETIMES				
		OFTEN				
2	Was difficulty experienced in maintaining required: Centreline?	NEVER				
		SOMETIMES				
		OFTEN				
3	Would you classify the movement of the ADI crosspointers as:	HIGH GAIN				
		MEDIUM GAIN				
		LOW GAIN				
4	Did crosspointer direction of movement ever appear to be in the wrong sense?	YES				
		NO				
5	At any time was difficulty experienced when integrating the information: Provided by ADI crosspointers?	YES				
		NO				
6	At any time was difficulty experienced when integrating the information: Between crosspointers and AI ball?	YES				
		NO				
7	At any time was difficulty experienced when integrating the information: Between ADI and HSI?	YES				
		NO				
8	Any problems in using horiz. and vert. dot graduations on HSI?	YES				
		NO				
10	Was the attention required, by the ADI/HSI display, to maintain the desired flight path:	MINIMAL				
		DEMANDING				
		V. DEMANDING				
11	Was the time available to perform the usual routine monitoring tasks:	NEGLIGIBLE				
		INSUFFICIENT				
		SUFFICIENT				
13	Considering the accuracy required from the radio altimeter, were scaling intervals:	TOO COARSE				
		ADEQUATE				
		TOO FINE				
14	Was the effort required to maintain constant airspeed along the flight path:	MINIMAL				
		DEMANDING				
		V. DEMANDING				
16	Were problems of conflict ever experienced between visual and motion cues?	YES				
		NO				
17	How would you rate the task difficulty of these simulated low visibility approaches with: Helicopter GCA	MORE DIFF				
		ABOUT SAME				
		EASIER				
		NO EXPERIENCE				
18	How would you rate the task difficulty of these simulated low visibility approaches with: Fixed Wing ILS	MORE DIFF				
		ABOUT SAME				
		EASIER				
		NO EXPERIENCE				
20	During localiser capture, was demanded azimuth sufficient to ensure smooth interception?	NEVER				
		SOMETIMES				
		ALWAYS				
21	Compared with beam holding, was attention required by displays during localiser capture:	INCREASED				
		ABOUT SAME				
		DECREASED				

Fig 2 EXTRACTS FROM SELF-ADMINISTERED, TAPED, DEBRIEF SCHEDULE

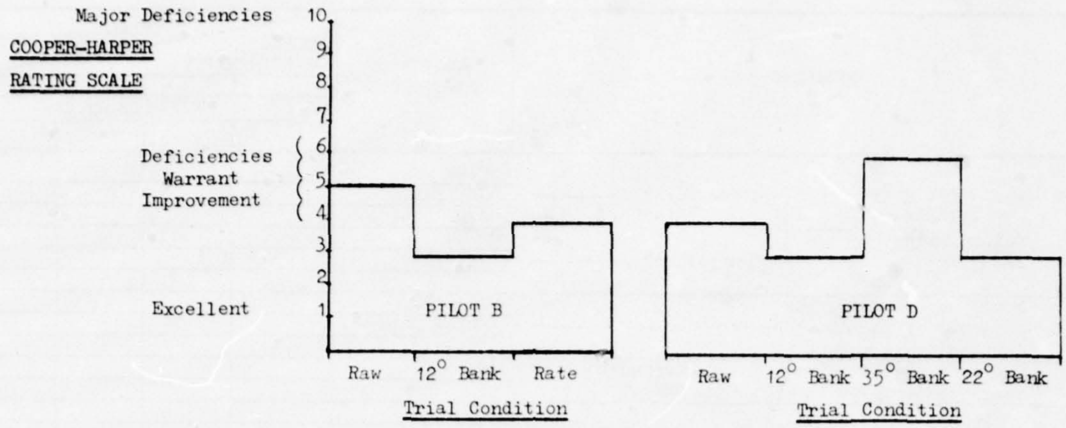

Fig 3 SPECIMEN OF PRESENTATION OF OVERALL ASSESSMENT OF DIFFERENT TRIAL CONDITIONS (AZIMUTH LAWS) BY TWO PILOTS USING THE COOPER-HARPER RATING SCALE.

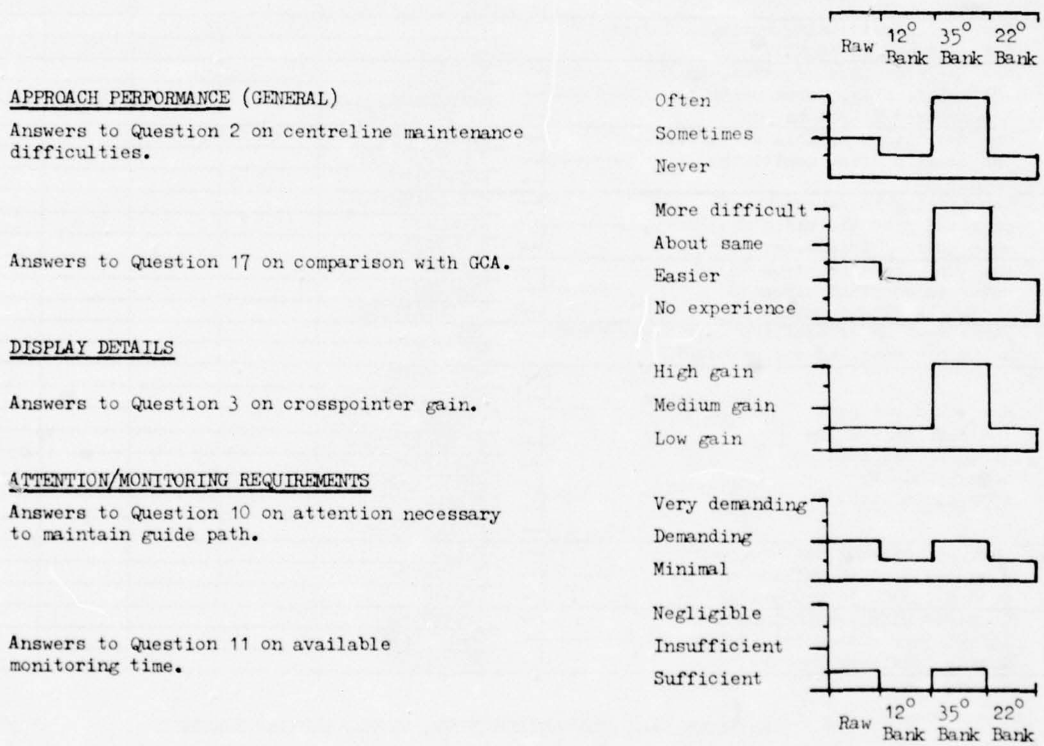

Fig 4 SPECIMEN OF PRESENTATION OF FINDINGS FROM RESPONSES TO FORCE CHOICE QUESTIONS BY PILOT D TO ILLUSTRATE ANY QUALITATIVE DIFFERENCES BETWEEN 4 TRIAL CONDITIONS (AZIMUTH LAWS).

APPROACH PERFORMANCE (GENERAL)

eg Glide path maintenance (Q1); Centreline maintenance (Q2); Comparison with GCA (Q17); Comparison with ILS (Q18).

LOCALISER CAPTURE

eg Adequacy of azimuth demand (Q20); Attention comparison between beam holding and localiser capture (Q21); Reasons for overshoot (Q22).

ATTENTION/MONITORING REQUIREMENTS

eg Attention necessary to maintain flight path (Q10); Available monitoring time (Q11); Effort to maintain airspeed along flight path (Q14).

DISPLAY DETAIL

eg Crosspointer gain (Q3); ADI/HSI integration (Q7).

SUPPLEMENTARY INFORMATION SOURCES

eg Vision and motion cue conflict (Q16); Radio altimeter scaling intervals (Q13).

Fig 5a EXAMPLES OF HEADINGS (ASPECTS) UNDER WHICH THE TAPED DEBRIEF EXPLANATIONS WERE SUMMARISED

Aspect	Trial Condition	Question Number	Problems/Limitations	Advantages	Recommendations
DISPLAY DETAIL Crosspointer Gain	Raw and Bank	3	ADI higher gain than HSI. ADI high gain when unsatisfied, low gain when satisfied. With 35^o Bank angle, problem was exaggerated by far too high gain both when unsatisfied and satisfied together with needle bounce.	Apart from bounce, 12-22^o Bank not far from optimum in terms of control responses.	Fractional damping required so that pointer would only start to move in response to a definite Bank demand.
SUPPLEMENTARY INFORMATION SOURCES Radio Altimeter Scaling Intervals	Raw and Bank	13	Adequate as no difficulty in realising 100 ft break-off height was approaching.	Expanding scale increased the impression that descent rate was becoming critical, even with constant power setting.	To really assess display deficiences once the laws have been sorted out, trials ought to be run without stabilisation and the helicopter levelled off at 100 ft and 100 ft maintained without visual reference.

Fig 5b EXTRACTS FROM EXPLANATION SUMMARY SHEET (FOR PILOT D) SHOWING FORMAT PROVIDED TO DESIGNERS

THE IMPACT OF A MULTI-FUNCTION PROGRAMMABLE CONTROL DISPLAY UNIT IN AFFECTING A REDUCTION OF PILOT WORKLOAD

by
Bernard S. Gurman
U.S. Army Avionics Research and Development Activity
Fort Monmouth, New Jersey 07703, U.S.A.

SUMMARY

This paper describes three recently developed digitally-addressed Multi-Legend Display Switches (MLD/S) which employ different mechanizations, and the incorporation of these MLD/S's in a multi-purpose, programmable Control Display Unit (CDU). The paper cites several programs which logically led to the development of the Multi-Legend Display Switches and the Control Display Unit, describes these programs and discusses possible applications. In addition, the paper points out that these developments can lead to the reduction of pilot workload and improve overall pilot/vehicle performance.

INTRODUCTION

Technologists proudly state that as operational needs have become more demanding, science has been able to produce novel solutions, albeit, sometimes very sophisticated. Often it appears as though the need arises as a result of a technologist-developed solution. In any event, as aircraft operational requirements increase, solutions are gradually found. Now, the need to fly at nap-of-the-earth, in marginal weather and reduced visibility has introduced extraordinary demands on the crew.

DISPLAYS

Since 1945, when human engineering is said to have begun as a formal discipline (10), aircrew station designers have tried many ways to improve the pilot's lot. The Army-Navy Instrumentation Program (ANIP) begun in 1952, combined two separate programs and provided the first major effort to simplify in-flight management through display integration (Figure 1). While some ANIP studies (1) (both human factors and engineering) investigated control functions, the primary thrust was toward concepts of improved-integrated displays. This work was reported to AGARD in 1958 and 1959 (2). Many of those concepts have been translated into equipment employed in the modern aircraft of today. Examples of those concepts are the Helmet Mounted Display, Vertical Situation Display, Horizontal Situation Display and the integrated Control Computer employing improved microcircuitry techniques. Other ideas, such as Flat Panel Displays, are still in development; in particular, those employing thin film electroluminescent material.

It was obvious then, as it is now, that with increased mission requirements, the introduction of more sophisticated equipment, and the need to fly in adverse weather, the increased workload could adversely affect the crew's performance.

CONTROLS

The Joint Army-Navy Aircraft Instrumentation Research (JANAIR) program, successor to ANIP, directed a number of studies and investigations (dealing with avionic, sensor, and weapon systems) in order to achieve reduction of pilot workload through integration of displays and controls and the automation of selected tasks. The Integrated Cockpit Research Program (6, 11) addressed the problem of workload and in a conceptual design described an integrated multi-function control display unit which employed changeable switch indicators and which would handle most sub-system control functions (Figure 2).

Later, an adaption of the Integrated Cockpit Research Procedure was used to define cockpit control and display requirements for the then next generation utility transport helicopter under consideration by the Army (7). This study determined control and display implications of different avionics equipment and illustrated alternate recommendations in a full scale mockup of the proposed vehicle. A control and display requirements analysis was conducted based on the derivation of specific functions necessary for accomplishing four specified missions. In turn, a control/display mechanization was derived from specific mission functions. For this program, "inventions" were prohibited, i.e., to the extent possible, existing, (i.e., in-development or in-production) hardware was to be employed. The use of dedicated control panels not only required increased panel space over the "workhorse" UH-1, but resulted in high workload situations. This was further complicated by scan and reaction times, anthropometric considerations, viewing angles, and a myriad of panel design problems.

While extensive success had been accomplished through integrated displays, the burden on the pilot was not diminishing; instead data and housekeeping control was even more demanding. It was evident that a need existed for a totally new approach to cockpit instrumentation, control layout, and integration.

One proposed improvement called for a multi-function control unit which would handle most sub-system control functions. Several mechanizations were proposed. A key element was the use of a changeable legend display switch as the principal Input/Output (I/O) device.

Subsequently, the Cockpit Switching Study was funded by JANAIR. This study developed a procedure for design and application of multi-function switching concepts to cockpit controls, thereby reducing the number of cockpit switches required and the "switching" workload of the pilot or crew. This study led to a published procedure (4) which was validated through simulation (5). The key device required was a solid-state, multi-legend display switch. (While 25 mm round CRT's, and incandescent - projection lamp-display switches were available, the CRT volume was too large and expensive and the projection devices did not emit sufficient light to be seen in even moderately bright ambients).

A state-of-the-art review conducted as part of the Cockpit Switching Study concluded that "flat panel" technologies such as liquid crystal, plasma, thin film electroluminescence, and fiber optics, although appropriate, were not yet ready for this application.

MULTI-LEGEND DISPLAY SWITCH (MLD/S)

In 1976, the author initiated three programs to demonstrate the feasibility of a solid-state multi-legend display switch. Each program was to combine a switch and a solid-state alpha-numeric display to provide two rows of four characters which could be changed from a programmable source. The switch would provide command or "choice selection" feedback to the programmable source. Feasibility was demonstrated by all three approaches.

The first MLD/S is an all solid-state display switch made up of 2 four character light-emitting diode chips (Figure 3). Each character is a 5 x 7 dot matrix about 3.8 mm (0.15 inch) high. The MLD/S incorporates a Hall-effect switch and the force feel can be controlled to provide as much as 500 grams pressure, if desired. The device is approximately 25 mm wide, 32 mm high and 10 mm deep. The LED's used are standard commercial displays (8) and, as such, are not readable in bright daylight. All pins are brought out to the rear of the device. Except for the display, the unit can meet the necessary military switch specifications.

The second MLD/S also employs LEDs (Figure 4). The display element is an LED mosaic module having a resolution of 33 diodes per inch. The actual display field used consists of a array of six rows of four characters each. Each character is formed by a 5 x 7 diode matrix with a one-diode space between characters. The display field is divided into three functional legend areas of two rows of four characters. The switch element is composed of a three contact point membrance switch and a pressure plate. Each contact point is centered over one of the three legend areas of the display. While the display module is actually a mosaic of 48 x 96 element, only the central area of 23 x 47 elements were used for this demonstration. The display area size is 38 mm wide by 76 mm high.

The third MLD/S is made up of an electroluminescent display module and an overlay switch (Figure 5). While the switch performs similarly to the previous one, it is not transparent. The switch visibility has not interfered with the acceptability. (Until you ask the user his opinion, acceptability is often not a factor. The display element is a conventional powdered EL phosphor with a thin-film-transistor circuit addressing layer on the back. Each display area (20 mm wide by 18 mm high) consists of a crossed grid matrix of 30 by 30 lines. As with the other two devices, the characters are a 5 x 7 matrix and each display switch area provides two rows of characters.

As stated earlier, feasibility was demonstrated. As presently constructed, none satisfies the military envirionment, but efforts are underway to develop a fully compatible device which will meet the total ambient and envirionmental requirements.

In addition, there are other mechanizations which have been proposed - at present, they are all display device limited. The future is not constrained to any particular display technology, although, at present, the most likely candidate is a black field - high-contrast thin-film electroluminescent display.

MLD/S APPLICATIONS

There are several potential applications of the Multi-Legend Display Switch. The description of one follows:

The advancement of digital technology has been rapid; microprocessors, memories, displays and even sensors. The introduction of the digital data bus now allows for a quantum improvement in aircraft-system integration through novel digital system architectures. The employment of the U.S. MIL-STD-1553 data bus is currently being pursued by all three U.S. services.

The solid-state multi-legend display switch as the key device in the multi-function switch procedure was discussed earlier. We now have all the ingredients for an integrated Control Display Unit - the means to interface with the system (MIL-STD-1553 data bus), the means to handle the data (microprocessors), and the means to interface with the operator - the multi-legend display switch.

In September 1977, the U.S. Army Avionics R&D Activity began a program to develop a Programmable Control Display Unit (CDU). The intent was to demonstrate that by employing Multi-Legend Display Switching, one could provide a programmable keyboard/control panel which could be changed to suit both the varying functions which had to be controlled and the particular aircraft installation. If successful, the Programmable CDU would obviate the need for many of the dedicated control panels which now occupy most of the console area. Of equal importance, the unit could be reiteratively programmed to optimize the task flow operation, even after hardware development.

One can design a panel or work station where each switch/display element is dedicated to a single function; obviously an impractical procedure. At the other limit, a single display switch (or at least a very few) could be employed. Since Gurman's Law (a varient of Murphy's Law) says "what you are looking for is always at the other end", that is also an impractical approach. It is obvious that a middle ground must exist. The Cockpit Switching Procedure mentioned earlier does provide a suitable vehicle and was employed, in part, in this program.

The Programmable Control Display Unit, which has been developed, integrates the functions of six dedicated military avionics control units. To accomplish this integration, each dedicated unit was examined in detail to define <u>all</u> operations to insure that all functions could be performed and to insure consistancy <u>of</u> operation, that is, all similar or like functions would operate in a similar or like manner. In addition, certain basic rules were applied: all functions of all units would be retained, the procedures should be orderly and designed for ease of operation, prompting techniques should be applied wherever practicable, workload should be minimized, the operator should <u>always</u> know the "state," and automated steps should be applied to the extent possible and reasonable.

The CDU employs the second of the Multi-Legend Display Switches described earlier. It is essentially the same display module used in the Interactive Display Terminal described by Gärtner and Heizhausen (3, 9).

In this application, three mosaics are combined to yield a display approximately 76 mm (3.0 in) high by 114 (4.5 in) wide (Figure 6). The membrane switch matrix provides switches at 12.7 mm (1/2 in.) spacing, thus allowing for 56 switch locations. The display provides up to 9 rows of twenty-four characters. In addition, areas can be outlined in such a manner as to yield any number of varying formats (Figure 7). Figure 8 shows the primary executive control status page, while Figures 9 through 14 show the submode or principal control page for each unit. In addition, each unit has a status page such as the VHF status page (Figure 15) and the Doppler status page (Figure 16). The complete panel including 10 fixed-legend switches is shown in Figure 17.

The CDU can store up to 10 preset frequencies for each communications set as well as 10 checkpoints and 10 targets for the doppler navigation set. Through the use of prompting and appropriate display feedback, the operator can select the unit, select modes, store or call up prestored frequencies for the communications equipment or checkpoints and targets for the navigation equipment, and can conduct a manual fault or status review.

All aircraft in a group can have all preset communications and navigation data entered quickly by using cassette or other bulk loading techniques which would eliminate extensive preflight preparation. Thus in-flight data entry can be drastically reduced.

The principal function of the CDU can now be that of basic operations - turning units on or off, selecting or changing modes, selecting or changing preset frequencies or checkpoints. The functional capability of all units is retained. The number of units which can be controlled is limited only by the amount of memory employed and the specific dedicated operations required.

The U.S. Army, Navy and Air Force are conducting technology and concept demonstration programs for integrated cockpit displays for future aircraft programs under the DIMAP (Digital, Modular Avionics Program), AIDS (Advanced Integrated Display Systems), and the DAIS (Digital Avionics Information System) programs. These programs seek the design and development of a new approach to cockpit instrumentation to improve information transfer to a pilot (crew) by use of a limited number of programmable electronic displays. An inherent part of all three programs is the use of integrated multi-function control display units. A series of studies conducted by the Air Force in support of the DAIS program has examined the use of multi-legend display switches. Reising reported that "the preferred --- hardware configuration for cockpit use is one in which the legends appear <u>on</u>, rather than adjacent <u>to</u> the switches." (12)

In addition, the Army currently has in development the Integrated Avionics Control System (IACS), which integrates the control of up to 10 units using currently available technology.

The fielding of fully programmable control display units only awaits acceptable display technology.

OTHER CDU MECHANIZATIONS

There are two other CDU mechanizations using display switch technology which should also be considered.

Because the mosaic display/switch provides little or no physical feedback, a programmable CDU employing a matrix of single-multi-legend display switches may be more applicable to the helicopter environment. Figure 18 shows one possible configuration. On the other hand, a combination of both the larger inter-active panel and single MLD/Ss may offer flexibility in the way that information and control functions may be handled.

ANOTHER APPLICATION OF MULTI-LEGEND DISPLAY SWITCHING

The other principal application for the Multi-Legend Display Switch is for Fault Warning Advisory Panels. Some aircraft have 60 or more legend lamps, taking an inordinate amount of space. Further, when more than a very few are lit, the appearance of a new fault may go undetected. By replacing the large panel with a small number of the display switches - say 12 to 16 - the messages can be prioritized, acknowledged by the crew, and then tallied for post-flight diagnostics and maintenance.

CONCLUSION

It is our intent to expose the programmable CDU to extensive evaluation. First, to assess acceptability of the user community to integrated controls and second, to determine the operational impact.

It has been demonstrated that the engineer can operate the CDU with speed and accuracy. But can the crew conduct all operations without additional burden? Will the burden be decreased?

Through careful application of the Multi-Function Switch Procedure, the designer can avoid imbedding cetain functions too deeply in the operation. This concern was addressed by Gartner and Heizhausen ... "Although a large number of switching functions can be combined into a relatively few multi-legend switches, satisfactorily ergonomic designs of integrated switching arrays are very difficult to achieve because not all switch functions are available at any time. This means that some switch functions may not be available when needed." (3). The solution they chose, i.e., using a large access surface does lend itself to a satisfactory solution. Essentially, Gärtner and Heizhausen used more than a "relatively few" multi-legend switches. Indeed, as noted earlier, there is a middle ground. It is acknowledged that the use of too few switches will not work.

The focus of design must be on the in-flight procedures -

- Operations must be simple and direct.

- Opportunity for error must be held to a minimum.

- Where errors do occur, the procedure for correction must be obvious and straight forward.

- In-flight functions which need to be dedicated or accessed quickly must have the highest priority.

- Prompting should be used as much as possible in order for the operator to always know the status of the operation and what to do next.

It is expected that the limit -- to integrate or consolidate functions into a common control unit -- will not be determined by the electronics, but rather, by an inverse Gestalt. There may be a limit on the number of differing subsystems which can be incorporated and a limit on the number of functions or operations. For example, it might be possible to integrate as many as 20 communications sets because of the similarity of operations. It might only be possible to integrate two communications sets, two navigation sets, and one weapons (stores and delivery) set.

There may also be a limit to the number of dissimilar operations a crew member may handle without getting "lost" in the CDU system procedure. The measure of this will have to be determined through stress analysis of both simulation and actual in-flight testing.

FINAL SUMMARY

The rapid advancement of digital technology has reached the state where new avionic system architectures are being established around digital data bus concepts. While allowing an almost quantum improvement in system integration, the traditional role of the aircraft crew is being severely affected. Many human-factors programs have laid the foundations for pilot (crew) unburdening. Recent developments have now demonstrated means by which in-flight system and subsystem display and control functions can be greatly reduced.

Although in the past the major effort was directed toward integrated displays, the emphasis is now on the control of data and functions.

The programmable CDU which has been described is only one of many possible configurations. Although the display presentations appear somewhat similar to CRT "pageing" on a menu basis, virtually all control functions are performed via operation of Multi-Legend Display/Switches.

This paper has cited several programs which logically led to the development of Multi-Legend Display/Switches which, in turn, expand the I/O capability of control display units. Through versatile programming techniques, "universal" CDU's can be tailored for varying functions, varying in-flight procedures, or varying aircraft applications.

Through reprogramming of the function flow procedures, new opportunies are provided for pilot unburdening. Since many of the operating procedures can be automated, the pilot (crew) can assume higher orders of command and control of the vehicle.

This novel approach to control/display unit design and operation will provide workload reduction relief and enhance pilot performance.

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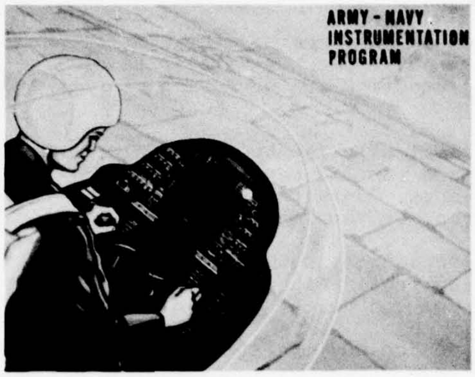

FIG. 1

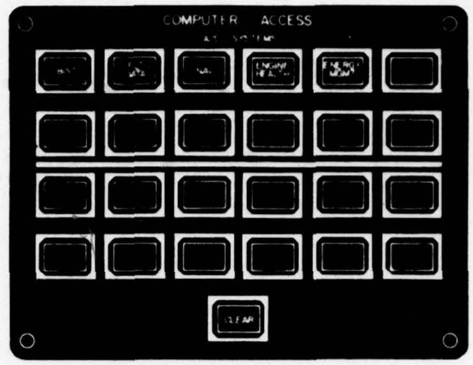

FIG. 2

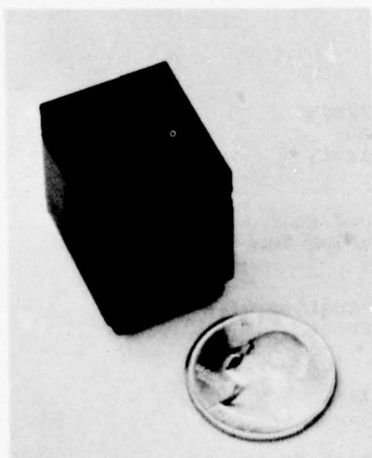

FIG. 3

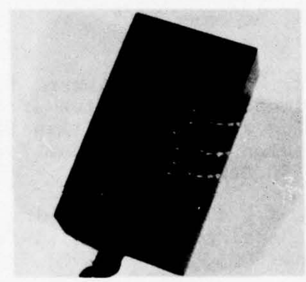

FIG. 4

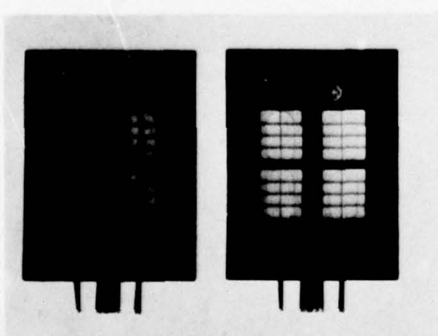

FIG. 5

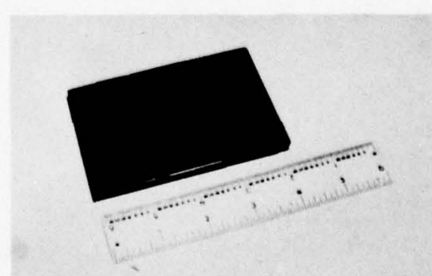

FIG. 6

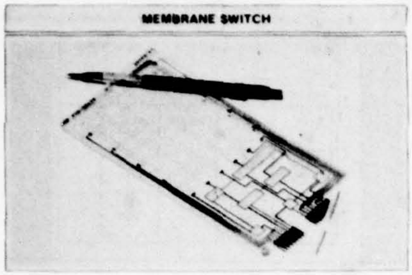

FIG. 7

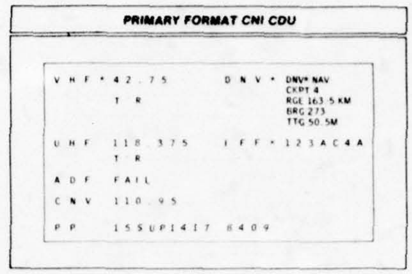

FIG. 8

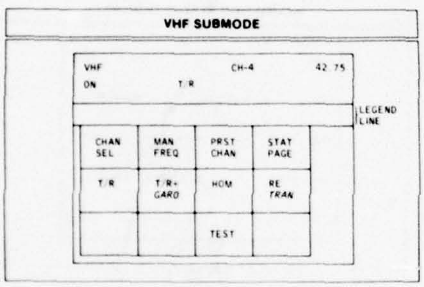

FIG. 9

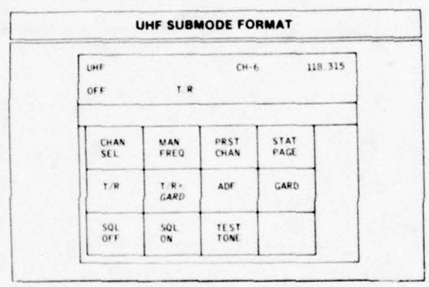

FIG. 10

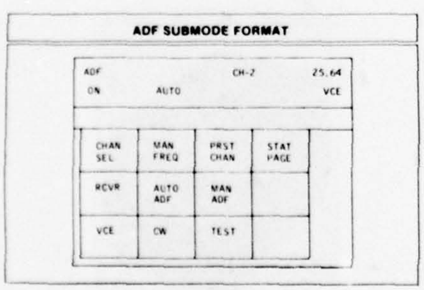

FIG. 11

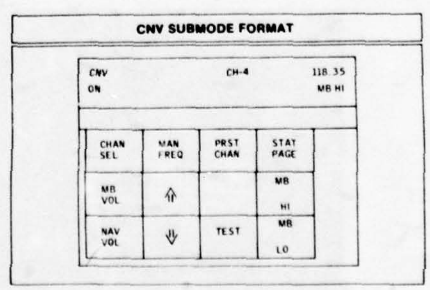

FIG. 12

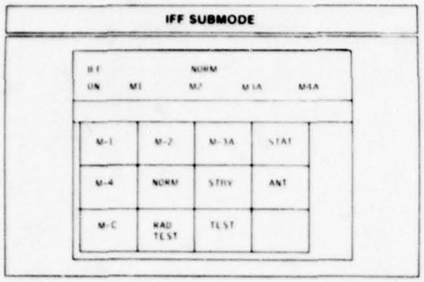

FIG. 13

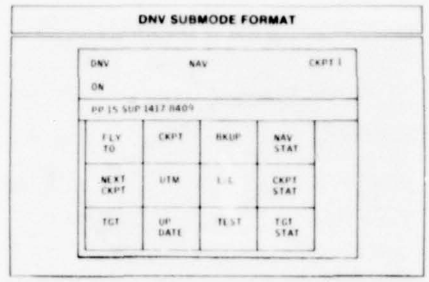

FIG. 14

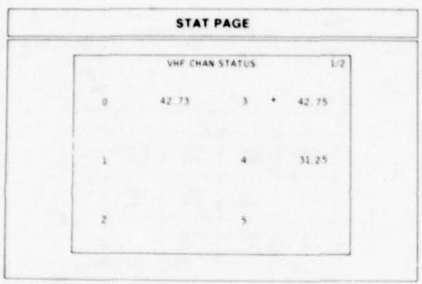

FIG. 15

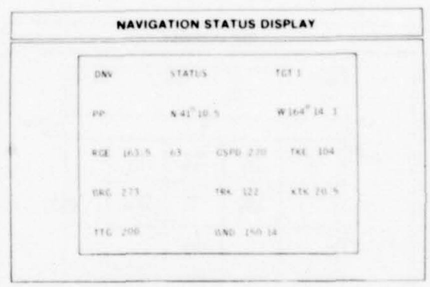

FIG. 16

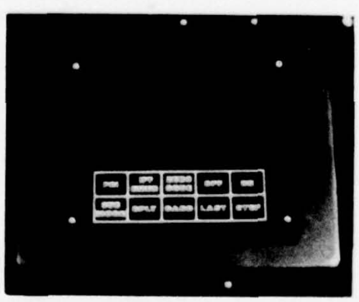

FIG. 17

FIG. 18

ETUDE D'INTERFACE EQUIPAGE-SYSTEME DANS UN HELICOPTERE ANTI-CHAR DE NUIT

Par

C. FERLET
J.L. MASCLE

GIE ISPENA
11 Villa Croix Nivert

75015 PARIS - FRANCE

RESUME

La complexité croissante des missions confiées aux aéronefs modernes a pour conséquence directe un accroissement de la charge de travail des équipages qui peut atteindre dans bien des cas un seuil critique.

Il est important de considérer ce problème dès le début de la phase de définition d'un aéronef afin de proposer, pour l'interface équipage-système, des solutions véritablement intégrées qui garantissent une efficacité opérationnelle maximale.

Une telle démarche nécessite l'adoption d'une méthode de travail rigoureuse et systématique, qui doit être capable de prendre en compte des données aussi diverses que les contraintes techniques ou les exigences opérationnelles pour aboutir à la définition de solutions optimales.

La méthode de travail retenue pour mener l'étude d'interface équipage-système d'un hélicoptère anti-char capable de réaliser sa mission de jour comme de nuit est présentée ici à titre d'exemple.

La méthode distingue cinq étapes dans le cadre d'un processus itératif qui permet d'améliorer progressivement la définition de l'interface équipage-système en fonction des difficultés considérées aux différentes étapes.

1 - INTRODUCTION

Le rôle de l'avionique à bord des aéronefs n'a cessé de croître ces dernières années au point de devenir essentiel à la satisfaction d'une mission et de poser des problèmes importants d'encombrement, de poids et de coût.

Une intégration des différents systèmes est apparue nécessaire. L'apparition des calculateurs digitaux a permis un progrès important dans ce sens en autorisant un dialogue beaucoup plus fructueux entre les différents équipements.

Mais la nécessité d'une intégration est également apparue au niveau de l'interface équipage-système, l'expression système étant prise ici dans un sens très général, devant l'encombrement des postes d'équipage et devant la charge de travail croissante imposée à l'équipage.

Une telle intégration doit tenir compte d'un certain nombre de contraintes d'ordres divers (fiabilité, coût, sécurité) et garantir une meilleure efficacité opérationnelle.

On peut définir quatre directions principales dans lesquelles peut porter cet effort :

- Mise au point de nouveaux modes en utilisant largement les possibilités de traitement de l'information - Exemple : Mode de pilotage ou de navigation,
- Automatisation des tâches répétitives,
- Amélioration ergonomique des postes d'équipage,
- Regroupement des visualisations et commandes.

Les dernières réalisations technologiques dans le domaine des visualisations électroniques ou des commandes multiplexées permettent, en particulier, d'envisager la communication homme-système sous un autre jour. Les travaux en cours dans le domaine des viseurs et visuels de casque et de la commande vocale viendront prochainement compléter ce potentiel.

Mais deux difficultés majeures, dues à la nature même de l'être humain, doivent être surmontées :

A/ La première est la résistance naturelle de l'homme au changement. Cette résistance est d'autant plus importante que les échanges entre l'équipage et les systèmes revêtent une importance considérable dans les aéronefs et qu'il est bien plus difficile de modifier des réflexes ou des automatismes acquis par une longue pratique que de remplacer, par exemple, un calculateur analogique par un microprocesseur.

B/ La deuxième difficulté apparait lorsque l'on cherche à définir une interface équipage-système dans laquelle le dialogue entre l'équipage et les systèmes soit simple et facile tout en évitant à l'équipage de se sentir exclu du travail opérationnel réalisé par les matériels embarqués et de voir son rôle réduit à celui de simple exécutant. Il s'en suit, à l'égard des systèmes un malaise et une méfiance qui nuisent à l'efficacité de la mission. Il est donc essentiel que ce dialogue soit organisé de manière à préserver le pouvoir de contrôle de l'équipage sur les systèmes embarqués ; ceux-ci doivent rester les assistants de l'équipage dans la réalisation de la mission.

Devant la nature des difficultés rencontrées et la diversité des solutions à envisager, on conçoit qu'il soit nécessaire d'adopter une méthode de travail rigoureuse et systématique tentant une approche système du problème posé.

1.1. CAS DE L'HELICOPTERE ANTI-CHAR

La définition de l'interface équipage-système d'un hélicoptère anti-char capable de réaliser sa mission de jour comme de nuit constitue une excellente illustration des problèmes évoqués.

La mission anti-char est par elle-même, une mission difficile, particulièrement par les contraintes qu'elle impose à l'équipage du fait de la nécessité de voler à très basse altitude pendant la phase tactique. Les difficultés sont accrues lorsqu'il s'agit de réaliser la mission de nuit.

L'objectif de l'étude d'interface équipage-système d'un hélicoptère anti-char actuellement menée en France, est l'établissement d'un projet d'interface équipage-système intégrée, qui respecte les possibilités techniques actuelles ou prévisibles conformément au calendrier de développement de l'hélicoptère.

Cette étude préliminaire complète, en particulier, une série d'études à caractère plus technologique destinées à évaluer certains équipements. Pour la mener à bien il a été nécessaire de mettre au point une méthode de travail dont l'exposé fait l'objet des paragraphes suivants.

2 - METHODE DE TRAVAIL

2.1. PRESENTATION GENERALE

La méthode proposée est, au demeurant, susceptible de s'appliquer à d'autres études similaires ; c'est pourquoi elle sera, dans la mesure du possible, décrite dans son principe, le cas de la mission anti-char de nuit étant utilisé pour illustrer l'exposé lorsque cela s'avère nécessaire.

Elle consiste à faire alterner des étapes de réflexions avec des étapes de définition concrète dans le cadre d'un processus itératif.

Elle distingue cinq étapes principales (voir planche 1) :

La première étape est une étape d'analyse de la mission qui permet de définir les principales fonctions à remplir par l'équipage et de décomposer la mission en différentes phases caractéristiques.

La deuxième étape permet de dresser la liste des informations et commandes nécessaires aux postes d'équipage.

La troisième étape définit le type de présentation des différentes informations et les commandes à installer.

La quatrième étape définit une organisation des postes d'équipage concrétisée par des maquettages en vraie grandeur.

La cinquième étape conduit à une évaluation de la charge de travail des équipiers et peut amener un rebouclage à différents niveaux suivant la nature des difficultés rencontrées.

Les rebouclages internes aux différentes étapes n'apparaissent pas sur la planche 1.

La méthode proposée nécessite la mise en place d'un groupe de travail qui comprend des représentants des avionneurs, des équipementiers et des opérationnels utilisateurs. Cette organisation permet une confrontation directe efficace de différents points de vue tout au long de l'étude et plus particulièrement lors des opérations de synthèse.

2.2. DEFINITION PRELIMINAIRE

Afin de hiérarchiser ce qui peut être remis en cause lors des phases de synthèse et d'optimatisation de manière à assurer la progression cohérente vers la ou les solutions recherchées dans le cadre de la mission, il semble nécessaire de distinguer, avant d'entreprendre tout travail, trois notions fondamentales :

- Objectifs opérationnels de la mission :

  Ils sont définis par l'Etat Major et représentent les buts ultimes à atteindre. Un objectif opérationnel est impératif, il ne peut être remis en cause qu'en tout dernier lieu. Sa modification sort, normalement, du cadre de l'étude.

  EXEMPLE : Nécessité de ne pas dépasser un certain seuil de détectabilité.

- Procédures opérationnelles :

  Elles définissent les procédures envisagées pour satisfaire aux objectifs opérationnels en tenant compte de l'environnement au sens le plus large.

  EXEMPLE : Utilisation du relief pour réduire la détectabilité.
  Utilisation de la vision directe du paysage pour les désignations diverses.

  Une procédure peut être remise en cause pour différentes raisons au cours de l'étude. Sa modification nécessite l'accord de tout le groupe de travail.

- Moyens embarqués :

  Il s'agit de tout système ou sous-système embarqué, ainsi que des commandes ou informations disponibles aux postes d'équipage. Ils sont les premiers mis en cause lorsqu'une difficulté apparait. Ils peuvent être modifiés sans l'accord de tout le groupe de travail.

Examinons maintenant plus en détail les différentes étapes illustrées par la Planche 2.

2.3. PREMIERE ETAPE : DEFINITION DE LA MISSION

Il s'agit, dans un premier temps, de donner une définition précise de la mission.

Un certain nombre de contraintes ou options initiales doivent être définies à ce niveau, par exemple :

- le type d'armement,
- la présence d'un armement secondaire,
- le nombre d'équipiers etc...

On peut ensuite préciser le profil général de la mission. Il peut être intéressant à ce stade, de dissocier les cas de la mission de jour et de la mission de nuit qui sont susceptibles de différer en bien des points.

La participation de représentants des utilisateurs opérationnels est essentielle dans ce travail ; elle permet de décomposer la mission en différentes phases opérationnelles caractéristiques en précisant leurs objectifs et les procédures opérationnelles à adopter dans chacune d'elles.

La décomposition de la mission en phases distinctes a pour but de faciliter l'étude en permettant de mieux saisir les difficultés dans leur contexte opérationnel et de les situer les unes par rapport aux autres.

Elle permet, d'autre part, une analyse systématique et progressive de la mission et des moyens à mettre en oeuvre pour la remplir.

Il est important de préciser que la mission ainsi établie correspond à la mission telle qu'elle est envisagée au départ de l'étude mais qu'elle ne correspond pas nécessairement à celle qui sera finalement retenue : Les procédures opérationnelles initiales peuvent évoluer en fonction des réflexions entreprises et des problèmes rencontrés. Cette première définition de la mission permet toutefois d'amorcer les réflexions et les analyses : le processus itératif de la méthode est ainsi initialisé.

Dans le cas de la mission anti-char, sept phases ont été définies ; elles sont données ici à titre d'exemple :

- Initialisation
- Décollage montée
- Croisière
- Vol TB A (Vol très basse altitude)
- Vol tactique
- Stationnaire tactique
- Atterrissage

Il faut remarquer que ces phases ne correspondent nullement à une décomposition chronologique d'une mission type ; elles ont été sélectionnées pour leurs caractéristiques opérationnelles. En les agençant de différentes façons il est possible de reconstituer divers déroulements de la mission anti-char. Il est, d'autre part, nécessaire de décomposer chaque phase en plusieurs sous-phases qui permettent une analyse plus fine du dialogue entre l'équipage et le système. Les sous-phases correspondent à ce que l'on peut appeler "actions opérationnelles élémentaires."

Un exemple peut être donné en considérant, dans le cas de la mission anti-char, la phase de stationnaire tactique qui a été décomposée en six sous-phases :

- Observation
- Acquisition et poursuite d'une cible
- Préparation du tir
- Tir
- Evasive
- Compte rendu après tir.

Pour chaque phase et sous-phase, les objectifs et procédures opérationnels doivent être définis clairement.

Une analyse globale et rapide de la mission permet, également, de retenir les fonctions principales qu'aura à remplir l'équipage. Les fonctions définies doivent être suffisamment indépendantes dans leur satisfaction pour pouvoir être étudiées par des spécialistes différents, du moins dans un premier temps. Il faudra ensuite évidemment "intégrer" les réflexions des différentes équipes.

Dans le cas de la mission anti-char, les six fonctions suivantes ont été retenues :

- Pilotage
- Navigation
- Tir
- CM et CCM
- Contrôle général machine
- communication.

4 - DEUXIEME ETAPE : ANALYSE DE LA SATISFACTION DES FONCTIONS AU COURS DES DIFFERENTES PHASES DE LA MISSION.

On peut ensuite analyser la façon dont les différentes fonctions peuvent être réalisées au cours des différentes phases de la mission.

L'objectif de cette étape est de définir les moyens nécessaires pour remplir la mission. Pour ce faire, il faut inventorier les ordres et informations nécessaires à la réalisation de chaque fonction. Les ordres et informations sont les éléments du dialogue entre la fonction et le système d'une part, et entre la fonction et l'équipage d'autre part.

A ce stade de la méthode il faut analyser la satisfaction des fonctions sans chercher à distinguer ce qui fera l'objet d'une intervention de l'équipage de ce qui sera automatisé.

Au sein de cette première catégorie de moyens, on introduira dans la suite la distinction entre les informations et ordres fonctionnels et les informations et ordres de mise en oeuvre.

On peut également dresser la liste des équipements nécessaires, respectivement, à la génération des informations et à l'exécution des ordres définis dans le paragraphe précédent. Ces équipements, constituant le système d'armes, comprennent à la fois les équipements senseurs qui détectent et transmettent les informations et les équipements de commande qui transmettent et exécutent les ordres.

Les équipements constituant l'interface équipage-système ne sont pas inclus dans la liste précédente. Ils apparaitront progressivement dans le déroulement ultérieur de l'étude.

Au cours de cette étape l'analyse est menée fonction par fonction et pour chaque fonction phase par phase (voir Planche 2).

Dans un premier temps, aucune intégration des fonctions n'est réalisée. Cette intégration n'interviendra qu'à la fin de cette étape.

Pour la mission anti-char étudiée, les cas de jour et de nuit ont en outre, été dissociés. Le cas de jour, mieux connu, est étudié en premier lieu ; il prépare l'analyse du cas de nuit en mettant en évidence toutes les informations fournies par la vision oculaire directe de l'équipage.

Les considérations opérationnelles caractéristiques des phases successives de la mission peuvent faire apparaître des modes fonctionnels différents pour chacune d'elles.

<u>EXEMPLE</u> : Modes de navigation différents suivant la détectabilité de l'hélicoptère.

Dans certains cas la validation de ces modes peut faire l'objet d'expérimentations ou d'essais en vol.

Une première liste des informations et ordres fonctionnels nécessaires peut alors être établie.

Il faut préciser, à ce propos, que les informations et ordres fonctionnels se distinguent des informations et ordres de mise en oeuvre qui ne jouent pas un rôle actif dans la satisfaction de la fonction : les ordres et informations de mise en oeuvre interviennent en particulier lors de la mise sous tension des équipements puis sous forme de contrôle (test, alarme) qui ne modifient pas la réalisation des fonctions sauf en cas de panne. Ils ne sont définis qu'à la fin de cette étape.

La liste des informations et ordres fonctionnels est complétée en précisant le taux d'utilisation de chacun d'eux et le type d'équipement susceptible d'être utilisé.

Lorsque ce travail est réalisé pour chaque fonction prise séparément, un premier bilan des équipements nécessaires est établi en faisant la synthèse des différentes listes d'équipements. Il permet de faire apparaitre les points suivants :

- Intervention possible d'un équipement nécessaire pour une fonction dans la satisfaction d'une autre fonction.

  <u>PAR EXEMPLE</u> : Utilisation de la télémétrie de tir pour les recalages du système de navigation.

- Intéractions nécessaires et contradictoires de deux fonctions sur un même équipement.

  <u>PAR EXEMPLE</u> : Utilisation simultanée du système de vision nocturne pour le pilotage et la navigation.

- Etablissement d'une liste des informations et ordres de mise en oeuvre rendus. nécessaires par le premier bilan des équipements embarqués.

Les deux premiers points peuvent amener dans un premier temps à modifier certains modes. Dans un deuxième temps, ils peuvent amener une modification des procédures opérationnelles. Des expérimentations ou essais en vol peuvent être recommandés pour confirmer la validité des modifications proposées.

5 - TROISIEME ETAPE : ERGONOMIE DES INTERFACES

Il s'agit de réaliser un travail préparatoire de l'étape suivante qui sera l'organisation des postes d'équipage et leur maquettage en vrai grandeur.

Pour cela on établit une liste des informations et ordres fonctionnels ou de mise en oeuvre qui sont utilisés par l'équipage et qui doivent donc être présentés aux postes d'équipage.

L'automatisation d'un certain nombre de tâches associées aux ordres et informations doit être choisie à ce stade en fonction d'exigences et de contraintes diverses.

Un premier contenu du dialogue équipage-système apparait à ce niveau. Il est susceptible d'évoluer par la suite, notamment lors de l'évaluation de la charge de travail de l'équipage.

Chaque ordre ou information de l'interface équipage-système doit être défini à l'aide d'une liste de caractères ergonomiques.

Les caractères considérés sont différents pour les informations et les ordres.

Parmi les caractères utilisés pour définir une information on peut citer à titre d'exemple :

- Présentation de type analogique ou non,
- Corrélation à d'autres informations,
- Localisation.

Pour les ordres :

- Forme du signal
- Localisation.

A ce stade, ni la configuration de l'équipage (Tandem, côte à côte) ni la répartition des tâches entre les différents membres de l'équipage ne sont prises en compte comme cela sera fait à l'étape suivante dont le but est d'aboutir à un projet d'organisation des postes d'équipage.

6 - QUATRIEME ETAPE : ORGANISATION DES POSTES D'EQUIPAGE - MAQUETTAGE

La configuration générale des postes (tandem, côte à côte) pour l'hélicoptère anti-char) est une donnée de départ dans l'analyse. La méthode de travail ne prévoit pas sa remise en cause.

Une première répartition des tâches principales entre les membres d'équipage doit être donnée à priori mais ce dernier paramètre peut être modifié en cours d'étude en fonction, notamment, de l'évaluation de la charge de travail de l'équipage.

Les différents postes d'équipage, par exemple pilote et tireur, sont dissociés et étudiés chacun phase par phase et fonction par fonction. On observe un emboitage des rebouclages inverse de celui de l'étape deux ; il s'agit en effet d'évaluer ici les tâches que doit effectuer chaque équipier pour réaliser toutes les fonctions au cours d'une même phase de la mission (Voir Planche 2).

Ce travail permet d'éliminer des redondances dans chaque phase. Une synthèse est ensuite réalisée sur tout le déroulement de la mission.

C'est à ce stade que sont faits les choix technologiques relatifs à la présentation des informations et des ordres (type d'instrument, visualisation classique ou électronique, etc...).

Cette première orientation technologique doit prendre en compte les possibilités de multiplexage ou d'intégration de certaines visualisations et commandes.

Il faut également définir les procédures de dialogue équipage-système en mode dégradé. On entend ici par mode dégradé tout mode résultant d'une panne, tenant également compte de la sécurité du vol et de la suite à donner à la mission en cas de panne.

Cette panne peut résulter d'un organe d'interface proprement dit (panne de transmission) ou d'un équipement en soute (panne de génération ou d'exploitation).

Un support matériel précis est alors associé, pour chaque poste, aux différentes informations et commandes. Il permet de fixer des cotes d'encombrement estimées des différents équipements d'interface équipage-système.

Le regroupement physique de certaines informations ou commandes distinctes est possible s'il facilite de façon évidente la satisfaction d'une ou plusieurs tâches.

Enfin le maquettage en vrai grandeur peut faire apparaitre des contraintes dimensionnelles qui nécessitent un regroupement supplémentaire de certaines informations ou commandes. Ces regroupements seront effectués en s'efforçant de respecter le type de présentation défini auparavant.

Le maquettage est également très utile pour l'évaluation de la charge de travail qui fait l'objet de l'étape suivante.

7 - CINQUIEME ETAPE : ANALYSE DE LA CHARGE DE TRAVAIL

Ce travail nécessite une description très détaillée de toutes les tâches élémentaires réalisées au cours de chaque phase de la mission. Ces tâches peuvent nécessiter l'intervention d'un membre de l'équipage ou s'effectuer selon un processus autonome :

<u>EXEMPLE</u> : Initialisation d'une plateforme gyroscopique.

A chaque tâche est associée une durée estimée de réalisation compte tenu des équipements embarqués et des organes d'interface définis précédemment. A ce niveau les maquettages peuvent jouer un rôle important pour préciser les temps opératoires élémentaires de certaines tâches.

L'analyse de la charge de travail sur les aéronefs actuels justifie souvent une étude particulière, longue et coûteuse.

Dans le cas d'une étude préliminaire d'interface équipage-système, il appartient à l'équipe responsable d'estimer jusqu'à quel niveau de détail il est souhaitable de descendre.

Cette analyse peut être, évidemment, complétée ultérieurement, lorsque la définition de la machine et du système d'armes sera acquise.

Si la charge de travail d'un membre de l'équipage vient à dépasser des valeurs acceptables, il faut envisager des mesures correctives qui peuvent être de plusieurs ordres:

- 1 Modification d'interface
- 2 Modification de la répartition des charges
- 3 Modification des modes fonctionnels
- 4 Modification des procédures opérationnelles.

Un rebouclage est alors réalisé au niveau correspondant à la modification dans le processus d'analyse et de synthèse et tout le processus doit être repris à partir de ce stade. (Voir Planche 2)

Il est impératif avant d'envisager de telles modifications de s'assurer qu'une adaptation au sein de la phase de vol considérée de l'enchainement envisagé, des tâches élémentaires, ne peut pas résoudre le problème. Ces enchainements sont théoriques et ne rendent pas parfaitement compte de la réalité. Il est important de déterminer si un enchainement type étudié n'est pas seul responsable de la difficulté rencontrée.

CONCLUSION

Il est important en conclusion de rappeler les deux points suivants :

- Pour mener à bien une telle étude, il est primordial d'adopter une approche rigoureuse et systématique qui permette avant tout une conception intégrée des postes d'équipage : une conception qui prenne en compte, du début à la fin des travaux la totalité des contraintes opérationnelles et technologiques pour aboutir à une organisation vraiment optimisée. La méthode de travail présentée ici est une tentative dans ce sens.

- La mise en place d'un groupe de travail comprenant des représentants des équipementiers, des avionneurs et des opérationnels est un élément important du succès de la méthode de travail : il permet, en particulier, de faire évoluer les procédures opérationnelles de façon efficace en déterminant le meilleur compromis entre les exigences opérationnelles et les contraintes techniques.

Si efficace que soit la méthode proposée, elle ne saurait constituer à elle seule, la garantie du succès de l'étude. La qualité des données qui permettent d'initialiser les travaux est tout aussi essentielle ; il faut accorder, notamment, un soin tout particulier au choix de ces données et à leur formulation.

PRINCIPALES ÉTAPES DE LA MÉTHODE DE TRAVAIL

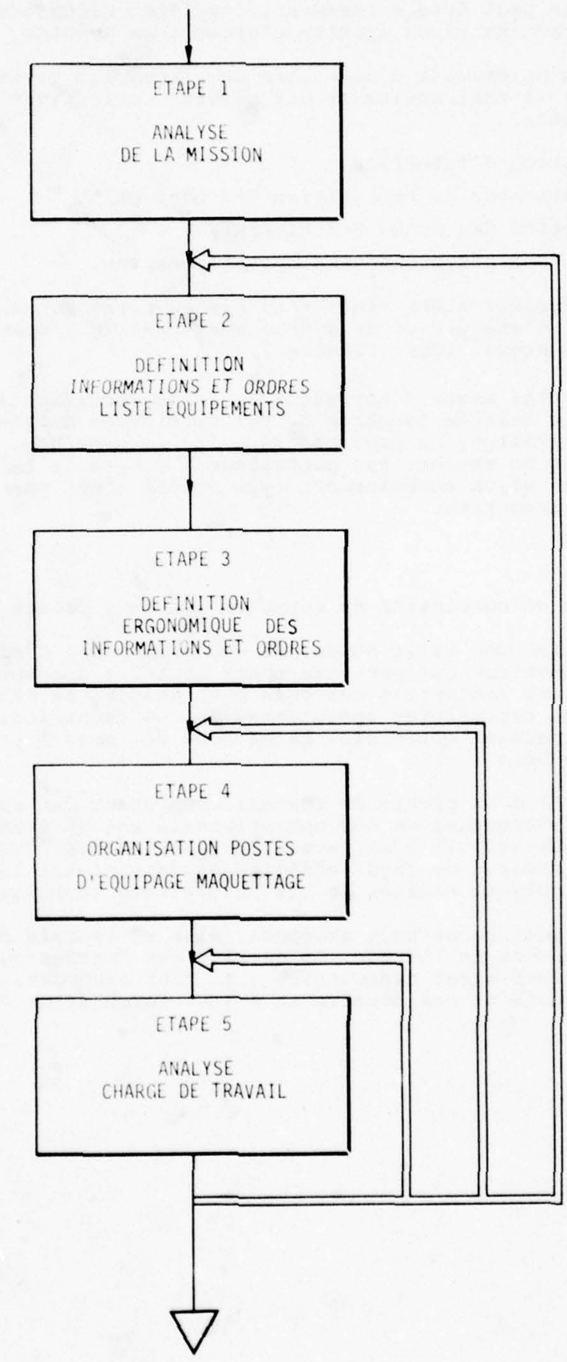

PLANCHE 1

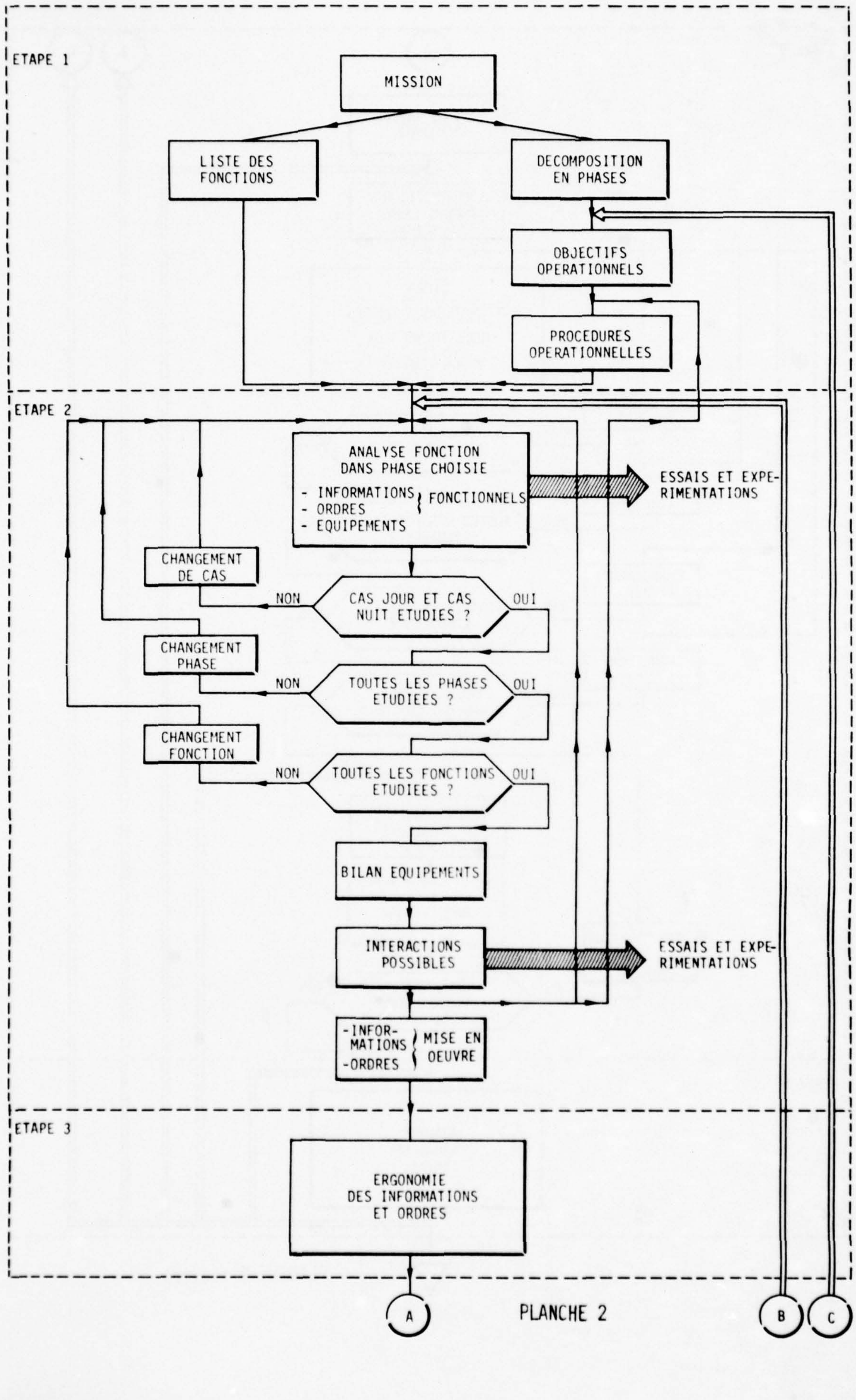
ETAPE 1
MISSION
LISTE DES FONCTIONS
DECOMPOSITION EN PHASES
OBJECTIFS OPERATIONNELS
PROCEDURES OPERATIONNELLES
ETAPE 2
ANALYSE FONCTION DANS PHASE CHOISIE
- INFORMATIONS
- ORDRES
FONCTIONNELS
- EQUIPEMENTS
ESSAIS ET EXPE-RIMENTATIONS
CHANGEMENT DE CAS
NON
CAS JOUR ET CAS NUIT ETUDIES ?
OUI
CHANGEMENT PHASE
NON
TOUTES LES PHASES ETUDIEES ?
OUI
CHANGEMENT FONCTION
NON
TOUTES LES FONCTIONS ETUDIEES ?
OUI
BILAN EQUIPEMENTS
INTERACTIONS POSSIBLES
ESSAIS ET EXPE-RIMENTATIONS
-INFOR-MATIONS
-ORDRES
MISE EN OEUVRE
ETAPE 3
ERGONOMIE DES INFORMATIONS ET ORDRES
A
B
C

PLANCHE 2

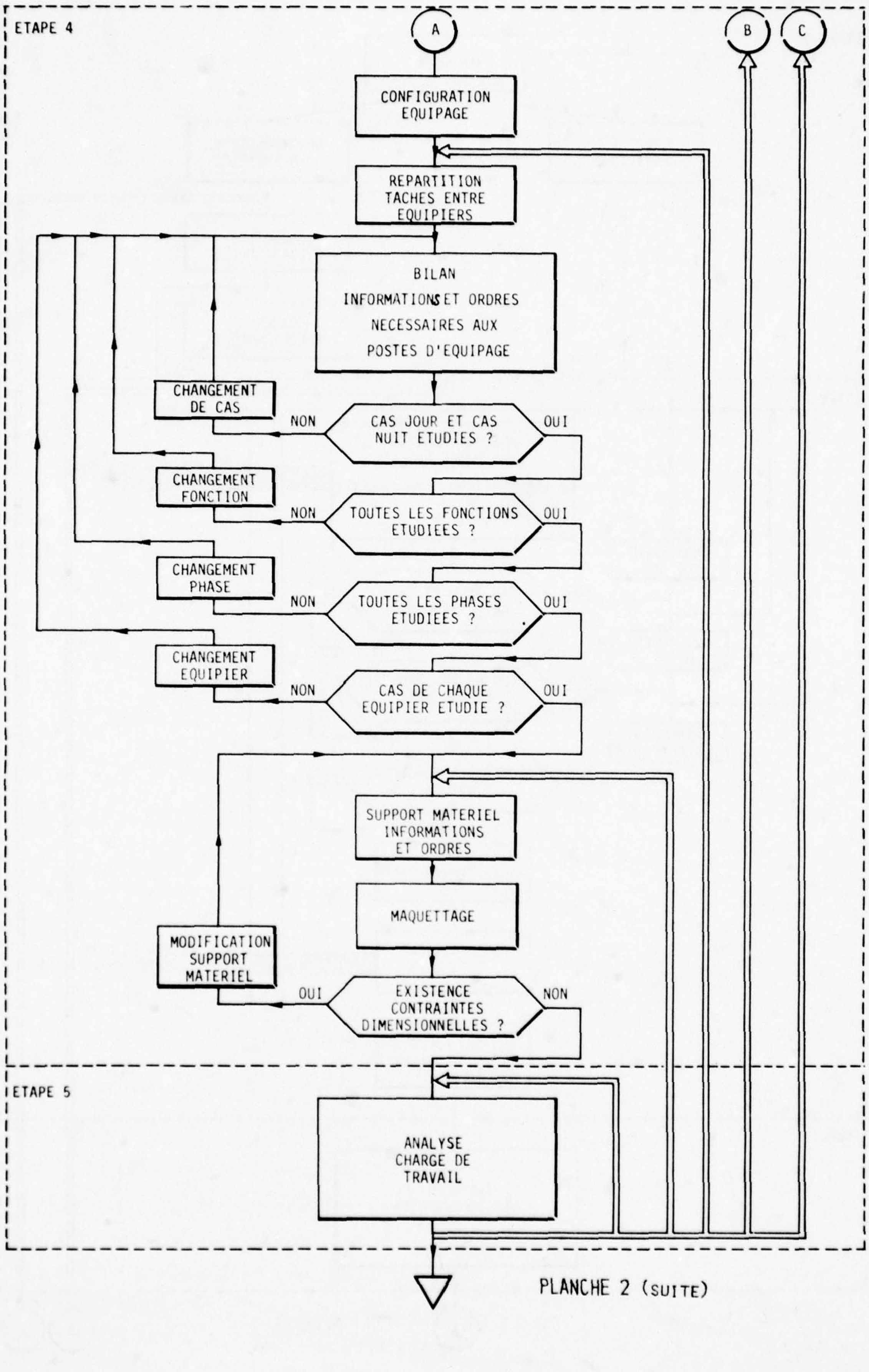

PLANCHE 2 (SUITE)

Project NAVTOLAND

(Navy Vertical Takeoff and Landing Capability Development)

By
Thomas S. Momiyama

Advanced Aircraft Development and Systems Objectives Office
Naval Air Systems Command
Washington, D.C. 20361, U.S.A.

SUMMARY

This paper describes Project NAVTOLAND, which is the U.S. Navy's integrated systems approach to improve the helicopter and fixed-wing VSTOL aircraft operational capabilities at sea and in tactical sites. The current capability is limited to generally 400-foot ceiling and one mile visibility to 200-foot and 1/2 mile weather minima due to elementary flight control systems or lack of precision approach and landing guidance. Inability to cope with ship motions limits the small air capable ship operations generally to Sea State 3. The NAVTOLAND project goals are zero ceiling and 1/8 mile visibility weather minima and Sea State 5 operation. An integrated development of the aircraft flight control and display systems to provide flying qualities with satisfactory level of pilot workload and the shipboard and tactical site installed guidance systems and visual landing aids to effect precision in touchdown will be applied toward improvement of the helicopter and AV-8 HARRIER operations and toward development of all-weather and rough sea operations of all future Navy and Marine Corps VSTOL aircraft.

INTRODUCTION

Six years ago an AGARD working group on "VSTOL Displays for Approach and Landing" observed, " . . . it is an unfortunate fact that no true VTOL aircraft has an effective poor weather capability, particularly into restricted sites, at the present time."(1)* At the present time, that is, in 1978, one must still observe that an effective poor weather capability still lies somewhere in the future of the VSTOL operations. While the U.S. Marines and the Royal Air Force together have logged some seven years of operation with the HARRIER, and the helicopter operation from the so-called Air Capable Ships has become a routine part of the fleet readiness, poor weather and rough sea operational capabilities of these VSTOL machines are still noticeably less than the all-weather capability of their carrier aircraft counterparts.

There are several reasons for this apparent "marking of time". One is the inadequate control system of the aircraft. The AV-8 HARRIER of the early 1970's vintage is still the only operational fixed-wing VSTOL aircraft besides the Russian YAK-36. The HARRIER's simple rate-damping stability augmentation system, which provides an excellent maneuvering quality for clear weather landing, causes excessive pilot workload in poor weather when the pilot does not have the aircraft attitude cues readily available in his visual field. Effecting all of the attitude-stabilizing, height-controlling, position-fixing and deck-motion-predicting functions manually and simultaneously demands the pilot's maximum sense of the real world. This is why the conversion flight, or transition form wing-borne to thrust-borne flight and the subsequent hover and landing must be accomplished in Visual Meteorological Condition (VMC) in the terminal area. Thus, the limited flight control system of the HARRIER dictates the current operational weather minima of approximately 400-foot ceiling and one nautical mile visibility.

The second problem is that the all-weather, precision landing guidance system has been an exclusive possession of the aircraft carriers. The helicopter carriers have been equipped with AN/SPN-35 precision approach radar but this voice-controlled system does not enable VSTOL conversion flight in Instrument Meteorological Condition (IMC). Nearly two hundred air capable ships of the U.S. Navy currently have no more than the TACAN navigation aid. As the AGARD report 594 pointed out, "there is a generally held opinion that operation of VTOL aircraft under poor weather conditions should be fundamentally more safe than Conventional Takeoff and Landing (CTOL) aircraft by virtue of the former's ability to fly slowly and stop if necessary". Beside being obviously unaware of the engineering complexity and human factors difficulty "to fly slowly and stop" the characteristically unstable VSTOL machines and of the generally more constrained VSTOL operational environment than that for CTOL, those who hold the above notion are ignorant of the fact that the stop-and-go flying in adverse weather actually requires more continuous and accurate guidance than the CTOL flying of constant speed and glide slope. While the CTOL final landing task is to maintain the aircraft control parameters in the conditions achieved at the commit-to-land "window", the VSTOL landing task in the same flight regime is a continuous and generally high workload maneuvering task about a "pin point" landing spot. In vertical landing, the typically small landing platform or site becomes obstructed from the pilot's field of view, even in helicopters. In this respect, the pilot of VSTOL aircraft, especially the streamlined high-performance ones, faces virtually blind landing even in clear weather. The importance of landing guidance system is evident in the fact that helicopters generally equipped with relatively more advanced stability augmentation system than the AV-8A are still constrained to weather minima similar to those for the AV-8A when landing on air capable ships which are not provided with approach and landing guidance systems.

*Numbers in parenthesis designate References at end of paper

Thirdly, of numerous VSTOL flight research programs to date, the most have been to demonstrate the feasibility of the particular vehicle concepts and only a handful of them have been dedicated toward obtaining data and exploring solutions of all-weather VSTOL operational problems. Among the latter category of programs are: Dornier Do31VSTOL transport with attitude command flight control system;(2) Canadair CL-84 with advanced displays;(3) U.S. Navy HOVVAC (Hovering Vehicle Versatile Automatic Control) system installed in a UH-IN helicopter;(4) U.S. Navy and FRG MoD joint flight tests of the VFW-Fokker VAK-191B for various VSTOL parameters;(5) Bell Aerospace X-22A variable stability research vehicle;(6) NASA Langley VALT (VSTOL Approach and Landing Technology) program with a series of CH-46 and CH-47 helicopters with programmable displays and variable stability flight control system;(7) and U.K. Royal Aircraft Establishment advanced display experiment with a Harrier for Sea Harrier operational capability development.(8) Of these VSTOL takeoff and landing problem researches, only the CL-84 and RAE Harrier programs have gone through at-sea trials. These research programs have explored several key segments of the multi-facet VSTOL terminal operation problems spectrum; but the results have not been systematically correlated or applied toward development of an efficient operational capability due perhaps to lack of a concerted program addressing such need.

Lastly, the VSTOL missions for the U.S. Navy are currently being formulated. The unique option of the VSTOL aircraft to operate in and out of remote tactical sites and numerous non-aviation ships no doubt can expand the Navy's repertoire of tactical capabilities. But the unique option can be exercised only if the VSTOL is equipped with an adequate all-weather take-off and landing capability to achieve its uniqueness.

Project NAVTOLAND

To close this VSTOL operational capability gap, the U.S. Navy has initiated Project NAVTOLAND, which stands for "Navy Vertical Take-off and Landing Capability Development".(9) The objective is to correlate and integrate the development of all systems and techniques, which are involved in enabling the pilot to fly VSTOL aircraft, both fixed wing and helicopter, into the Navy and Marine Corps ships and tactical sites.

NAVTOLAND PROJECT GOAL

Figure 1 illustrates the NAVTOLAND project goal of improving the current operational limitations of having to make conversion flight in VMC in terminal area to the all-weather, instrument-flight operational capability. The project goal of "zero ceiling" and 700-foot visibility was selected to correspond to the "obscured ceiling and 200-meter RVR (Runway Visual Range)" condition of ICAO Category IIIA operation. The ultimate "zero-zero" weather (ICAO Cat. IIIC) was not targetted because a completely automatic flying capability, which may be required for the zero-zero condition may not be possible or desired for all operational situations at costs imposed on the aircraft system acquisition. Actually, achievement of the NAVTOLAND goal would set a direct course for the zero-zero weather goal. A guidance system for the Cat. IIIA operation will be accurate enough for flight control coupling for automatic landing. The development of flight control and display software for Cat. IIIA operation would also produce basic control laws required for automatic flying, thus merely leaving hardware implementation, which must be done for each specific aircraft development anyway. Further, the high degree of manual landing performance attained will serve the purpose of the independent monitor necessary for automatic operation, as well as for emergency back up and, perhaps more importantly, for gaining pilot's confidence. One of the reasons for the U.S. aircraft carriers' not operating regularly in the automatic mode in bad weather is that the independent monitor system (AN/SPN-41 C-SCAN) is of one-fifth accuracy of the main system (AN/SPN-42). This is not adequate for the necessary parity check for safety and pilot confidence.

The project goal for rough sea operation is Sea State 5 which corresponds to the significant wave height band of approximately eight to 13 feet. The current operational capability of the Navy and Marine Corps helicopters and the AV-8A with respective operating ships is generally Sea State 3 (significant wave height band of four to six feet). In practice, operating limits are set, for example, in the SHIPBOARD OPERATING BULLETIN of various aircraft, in terms of ship motions, i.e., pitch, roll, and heave of a given type of ship. The Sea State is used here to indicate generalized operational capability situations

To determine or define what environmental limits should be used for a given operational requirement is not a simple operational analysis task. If an extreme environmental condition, in which a weapon system might be called to perform even once in its lifetime, is critical to the particular tactical or warfare mission accomplishment and is judged to be worth the expenditure of the money and effort, then that condition would be the target. How does one anticipate, let alone plan for, such an event? Hence, an all encompassing term "all weather" is normally and conveniently used. There has been much concern, particularly among programming and funding decision makers, as to just how much "all weather" is required and then how much is that requirement needed. The NAVTOLAND project environmental goals were derived through a brief analysis as represented in Figures 2, 3 and 4.(10) A survey of occurrences of weather and sea state conditions was made over some of the world's major sea lanes. In northern Atlantic and Pacific regions, except for a location off the Scandinavian coast Ocean Surveillence Vessel (OSV) MIKE, the current operational weather minima condition of 400-and-one occurs approximately 20% of the time in a year on annual average basis. This means that with the current VSTOL operational capability the fleet cannot operate 20% of the time in a year. The obscured ceiling and 1/4 mile visibility condition, the available data closest to the NAVTOLAND goal of zero ceiling and approximately 1/8 mile visibility, is shown in shaded blocks in figure 2. The data indicate that in these areas the fleet could reduce the non-operational situations down to about 6% if the NAVTOLAND systems capability were provided.

There is little difference in the frequency of occurrence between the 400 and one and 200 and 1/2 weather situations. It seems, therefore rather meaningless to set a goal of advancing from the 400 and one to 200 and 1/2 capability. On the other hand, there would be a significant potential gain in operational capability when one aims to improve to the NAVTOLAND advocated goal.

Figure 3 presents an overland example of weather situation, which might confront tactical site operations of the VSTOL aircraft. In this typical European highland area, the 300 and one weather can occur on an annual average of about 30%. In early morning hours, this weather situation characteristically increases to approximately 50%. Data were not available, during the analysis, for the NAVTOLAND goal situation. However, the 30-50% non-operational condition indicates a need for a substantial improvement of the current system capability.

A comparison of the annual average percent frequency of occurrence of the Sea State 3 and 5 conditions in world's sea lanes, known for high seas, is shown in Figure 4. The graph indicates a possible reduction of non-operational conditions from approximately 65% down to 15% by achieving the NAVTOLAND capability goal.

NAVTOLAND COMPONENTS

NAVTOLAND is a system integration project and the SYSTEM to be integrated is the vertical takeoff and landing operational CAPABILITY. Figure 5 is a pictorial representation of the NAVTOLAND capability components.

The major vehicular components, of course, are both the fixed-wing VSTOL aircraft and the helicopters. The NAVTOLAND project looks at the aircraft form the standpoint of those aircraft design parameters that affect its control in the vertical takeoff and landing flight regimes. The basic aircraft stability characteristics and the ground-effect induced thrust variation problems are very much specific to a given aircract type; the NAVTOLAND task is not to solve problems of a specific aircraft but to assess and attempt to generalize data as inputs into the flight control laws development. The out-of-the-cockpit field of view is a critical parameter in both the high performance stream-lined VSTOL and, surprising to many, the helicopters. Figure 6 shows that the Marine Corps steel-matted 92 feet by 92 feet tactical landing pad, in which a HARRIER can comfortably fit sitting on the ground, becomes totally obstructed from the pilot's view when the aircraft is at a nominal hover height for take-off and landing. The helicopters are not much better off as shown in figure 7. Through the larger field-of-view angles of the blunt fuselage shape and the capability to hover in the all-positive ground effect of the proximity of the ground, the helicopter pilot generally has at least some part of the landing area within his view. But some part is not always adequate for the precision maneuvering task at hand.

The landing platforms are on the aviation ships such as the aircraft carriers (CV), helicopter carriers (LPH) and amphibious assault ships (LHA), air capable ships such as cruisers, frigates, destroyers, etc., and in Marine Corps tactical sites.

Some NAVTOLAND hardware components are in the aircraft and others on the platforms. The guidance systems are split between the platforms and aircraft into sensors and receivers. In the aircraft, flight controls and displays are the principal pilot aids. Displays operate with the information received, decoded and sometimes additionally computed by the avionics. Signals from the avionics may input into the flight control as in the case of full or "split axis" automatic control modes. Appropriate electronic guidance sensors are installed on all platforms as are the visual landing aids (VLA). The helicopter recovery assist, secure and traverse (RAST) system is currently planned for installation in the air capable ships engaged in LAMPS operations.

There are two sensors which are not shown in figure 5 but are functional parts of the NAVTOLAND hardware components. One is the low speed sensor in the aircraft, whose output is displayed to aid the pilot's control of the aircraft. The other is the ship motion prediction technique with associated sensor and computer. The predicted information is fed into the guidance signal, displayed to the pilot and or inputted into the control system control law for appropriately timed automatic landing.

Functioning of some of those NAVTOLAND hardwares and interfaces among components obviously call for softwares and rather sophisticated computing capacity. Dedicated or shared computer use is necessary.

The human elements in the NAVTOLAND capability include the pilot, landing signal officer or enlisted man and operators of various subsystems. The pilot, of course, is not only the most important and influencing human element but more significantly the pilot is the focal point and the single "target" of the whole capability development. The pilot techniques and procedures to use the NAVTOLAND "system" constitutes a necessary and essential component.

NAVTOLAND "SYSTEM" FUNCTION

To achieve an integrated capability to aid the pilot in making a successful VSTOL take-off and landing operation, all components listed above must be developed as interfacing functions within a closed loop operation of this capability "system". Past VSTOL researches and the operational experience with the AV-8 HARRIER, pointed up the pilot's workload, especially in conversion flight to powered lift, as the primary limiting factor in the improvement of all-weather capability. Cooper and Harper related the pilot workload to aircraft control task in their definition of flying qualities: "those qualities or characteristics of an aircraft that govern the ease and precision with which a pilot is able to perform the tasks required in support of an aircraft role". Lamar and Neibor added: "From the (above) definition it can be seen that handling qualities involve those factors which affect the pilot workload (i.e., ease) and performance (i.e., precision) of the task".[11] To aid the pilot in "easing" the workload and achieving "precision" in the critical VSTOL landing performance, the NAVTOLAND component systems in toto can be considered within the aircraft flying qualities loop as envisioned by Cooper and Harper.(11)

Figure 8 shows the functional roles of the NAVTOLAND subsystems in the pilot's control loop for a task of landing maneuver, along the basic flying qualities factors of Cooper and Harper notion. The control loop is closed by pilot's receiving the information about the aircraft movement. In the NAVTOLAND flight regime, the aircraft movement is "filtered" through the landing platform before the pilot receives the motion information. The aircraft securing system, possibly required for rough sea operation from small ships, is a direct physical link between the aircraft and the platform.

The visual landing aids, electronic guidance systems and ship motion prediction system sense, translate and generate signals of the aircraft-platform motions and transmit to the information factor in the Cockpit Interface column, either through direct visual cues or through cockpit displays. These three subsystems increase the accuracy of the aircraft-ship movement information. The main control loop is tightened by pilot's receiving more precise information.

Both the electronic guidance systems and ship motion prediction system data must be transmitted through data link devices up to the aircraft. In the aircraft, either the NAVTOLAND-developed or -shared avionics receives, decodes and sometimes performs additional computations to the signals received from the ship. The processed data are then fed into symbol generator for appropriate format for display in the cockpit.

A very important pilot aid information in the powered-lift low speed and hover flights, the low speed sensor, derives data from the aircraft movement factor and sends the data to the information factor, or often to the display element for integration of the aircraft speed into other position and motion data.

It should be obvious even in a symbolic representation of the handling qualities factors relationship that all of the information coming from these various data derivation sources must be well coordinated and integrated when the pilot receives them out of the information factor block, because that link is the sole source of positive control guidance he receives before he makes his instantaneous action on the controls. In the NAVTOLAND version of the flying qualities factors chart, the (cockpit) field of view is added in the information factor block. As discussed in the introduction section of this paper, the field of view is very critical in pilot's acquisition of external visual cues.

The flight control system serves to smooth out the pilot's use of the controls against the basic flying characteristics of the aircraft, thus operating a control loop independent of the primary or information loop. The only exception is the automatic operation in which the main control loop is diverted at the avionics and fed directly into the flight control block, with the rest of the information loop becoming the secondary and monitoring role.

That the information loop and the flight control loop operate separately when the pilot is manually flying the aircraft and the ever-adaptive but also rapidly-degradable human pilot is the mixing ground for these two separate loops make this two-loop integration a major and difficult issue in the VSTOL operational capability development.

Finally, pilot's being the mixing ground for display and flight control loops means that the pilot would develop and operate certain strategies in the "mixing". The NAVTOLAND project will develop these basic pilot techniques including flight profiles, operational envelopes, auto/manual decisions, emergency procedures and so on.

In the discussions to follow, the NAVTOLAND project is described in terms of products to be anticipated, current developmental status or the state of the art of particular elements and critical developmental issues. The NAVTOLAND project has been in a preparatory phase of planning and piecemeal Exploratory Development to date. The formal project effort starts in the fiscal year 1979. Detailed descriptions of the project beyond this paper are found in the NAVTOLAND Program Plan documents. (12) (13)

NAVTOLAND CAPABILITY PACKAGE

The final products of the NAVTOLAND project will be a capability package, that is, a feasibility-demonstrated set of various systems and techniques manifested in design guidelines, specifications, operating manuals, system softwares and prototype hardwares. The basic and generic nature of the project products should allow flexible tradeoffs of the scope of coverage and degree of performance among component systems to match specific aircraft design characteristics, cost implications, mission needs, etc., while achieving the total capability level that is reasonably standard throughout the Navy and Marine Corps operations. The design and development of specific helicopters and VSTOL aircraft and VSTOL operating ships are the responsibilities of respective project managers and associated industry. The NAVTOLAND project will support these development by doing a concerted "homework" for incorporating vertical takeoff and landing operational capability.

VSTOL AIRCRAFT DESIGN - TAKE-OFF AND LANDING PARAMETERS

The aircraft design considerations with respect to stability and control requirements as augmented by flight control and pilot's response to display will be addressed by delineating the spectrum of operational situations with known control problem areas such as air turbulence and general wind-over-the deck effect, aircraft motion induced effect such as the side force problems and various thrust variation effects in the ground proximity. The pilot's field of view will be addressed in terms of what the pilot must see from his cockpit perspective, along with requirements imposed by flight profiles and other aircraft maneuvering procedures. What the pilot must see will be delineated in terms of visual cues for performance of specific subtasks.

The shortcoming of the current VSTOL handling qualities specifications and information such as MIL-F-83300,(14) AGARD Report 577,(15) and MIL-H-8501A(16) is that these documents are based on analytical and controlled flight research data and therefore do not necessarily provide design guidelines for successful operations in the reality of complex interaction of many factors as discussed in previous sections. Various segments of VSTOL terminal operations impose different problems in regard to aircraft control characteristics. For example, the VAK-191B flight research indicated a high sensitivity of the type and response characteristics of the flight control system to the manner in which the pilot conducts flight tasks in VMC vs IMC and also in different segments of approach and landing. A test pilot preferred the highly stability-augmented VAK-191B in IMC approach but preferred the force-feel augmented AV-8 in VMC landing maneuver. The height control in VSTOL mode is a function of thrust generated by the propulsive system. Therefore, the propulsive system becomes an inherent part of the flight control system in VSTOL operation. The required thrust to weight ratio and thrust response characteristics to meet all terminal operation situation must be spelled out for the aircraft designers.

The results of various on-going flight research programs, such as NASA Ames' VSTOLAND, NASA Langley's VALT, Navy's X-22A and RAE's Sea Harrier program must be centrally correlated for maximum benefit. In addition, specific VSTOL aircraft improvement and development programs such as AV-8A operational improvement, AV-8B, XFV-12A, XV-15 and Navy VSTOL development are and will be yielding information which must be generalized for use by the whole VSTOL design community. To accomplish these tasks would require an efficient, cross-agencies cross-disciplines and cross-nations cooperation. The NAVTOLAND project has begun the Navy contribution in this cooperation task. One of the more definitive and practical efforts to be started immediately might be a simulation research which incorporates all the factors and subsystems blocks in Figure 8 with adequate measurement and evaluation methodology for each block, including detailed delineation of the beginning TASK block and evaluation criteria of the end Performance block.

INTEGRATED FLIGHT CONTROLS AND DISPLAYS

The NAVTOLAND project views the flight controls and displays as two complementary subunits of an integrated pilot aid system in the control of the VSTOL aircraft. Figure 9 shows the postulated pilot workload curve as a function of flight control and display subsystems sophistication scales. (1)&(7) Toward the increasing end of both subsystems sophistication, inputs from guidance sensors are added to effect three-dimensional flight, i.e., flight path control. Through analytical work, simulations and flight tests, the NAVTOLAND project will define the acceptable pilot workload of loci of interfacing flight control and display capabilities. Different flight control display interface requirements in VMC and IMC will be accounted. Depending upon the aircraft system design selection, cost consideration, etc., an appropriate pair of flight control and display capabilities maybe selected by the specific aircraft designers following the specifications or Naval Air Systems Command documents called Air System Requirements (AR's) developed under this project. In addition, the integrated system software, i.e., control laws to program both the flight control and display subsystems and the necessary interface between them will be developed and flight proved with the testbed fixed-wing VSTOL and testbed helicopter in operational environment. A technique of integrating thrust and stability augmentation will be developed to effect an optimum power management and associated pilot and display roles. Interfacing of precision guidance systems with the integrated flight control and display concept will move the project further toward the automatic capability.

The state of the art in effecting automatic approach, hover and simulated vertical landing was demonstrated by Navy's HOVVAC system installed in a helicopter. (4) The X-22A project has also demonstrated similar capability and this project is continuing. The NASA Langley VALT program with the variable stability CH-47 research vehicle is continuing its actual automatic VSTOL landing experiments.

Figure 10 shows a result of the VALT program flight control and display interface flight research, which is typical of work needed in future to be performed under tactical operational situations. The traces of the aircraft, with the pilot following the two classical modes of display, i.e., FLIGHT DIRECTOR and SITUATION (FLIGHT DIRECTOR-off), indicate a definite advantage of the FLIGHT DIRECTOR display. The data also shows that the ATTITUDE COMMAND control system is a requirement (validating previous research findings and pointing up the HARRIER operational problems) in conjunction with the FLIGHT DIRECTOR display. (17). The FLIGHT DIRECTOR display, however, is known to cause pilot's tendency to become fixated on symbols and also pilot's annoyance or often loss of confidence due to lack of SITUATION information. NAVTOLAND pilots and engineers are currently evaluating a NASA Langley display concept called VECTOR-PREDICTOR (18) as a possible solution to this FLIGHT DIRECTOR problem.

Figure 11 shows the VECTOR-PREDICTOR concept in a vertical position or altitude control symbology. At the top, the aircraft is below the desired altitude and in horizontal flight. In the middle, the pilot has added sufficient power to climb toward the reference altitude as the VECTOR symbol predicts the eventual on-altitude outcome. At the bottom, the aircraft is still below the path and climbing but the VECTOR symbol predicts an eventual overrun unless power reduction is made.

There is a need for a concerted effort in defining requirements for and developing an integrated advanced display set for the helicopters. The "to fly slowly and stop, if necessary" syndrome has contributed toward the continued use of standard instruments, some of which are functional only in CTOL flight regimes; toward piecemeal, non-human-engineered add-on installations; and the solely head-down displays despite the helicopter pilot's need to stay "head-up" in the largely ground proximity flying. An eager suggestion for solution by the display technological community is the head-up display (HUD). The problem of course is the capability of the helicopter to fly in all directions and therefore to approach in headings optimum from the wind direction standpoint makes the narrow field of view and orientation of current and projected HUD viewer incompatible. The standard follow-on suggestion of the helmet-mounted display appears to have some weight and ocular problems which need to be investigated. Peripheral displays in edges of the cockpit field of view may work well. For development of an integrated helicopter display, helicopter-peculiar parameters, dynamics and respective display formats should be examined. The current Navy-NASA Langley joint research program using the VALT variable stability CH-47 helicopter, as mentioned previously, is exploring display flight control interface with ship motions in the program.

LOW SPEED SENSOR

The low-speed sensor, which is not currently available either in the AV-8A or fleet operational helicopters, has become one of the regular topics whenever the VSTOL and helicopter pilots and flight test engineers discuss operational problems. This requirement for detecting airspeed below 40kt, which is the lowest reliable limit of the currently operational air-speed indicator, has to do with the flight envelope warning. Kolway explains, for example, the UH-1N sideslip limits corresponding to 35kt sideward speed and rearward speed limit of 30kt can be safely approached with an adequate low-speed sensor, thereby permitting the pilot to achieve the maximum capabilities of the helicopter in hover and low-speed regime. (19) The recent HARRIER pilots safety symposium also raised the same issue by concluding that the lack of an accurate indication of flight envelope airspeeds among other things, precludes the realization in IMC of full VSTOL potential achievable in VMC. (20)

There are a few low-speed sensors in experimental stage. The accelerated development of such device appears warrented and would contribute toward improved task performance in the handling qualities loop. Another approach to provide this information is to use the range/speed sensing portion of the precision landing guidance system in conjunction with wind information when such systems are installed on the landing platform.

AIR TRAFFIC CONTROL, APPROACH AND LANDING GUIDANCE SYSTEMS

The development or structuring of the Air Traffic Control (ATC) or inbound navigation, approach and landing guidance systems and their integration for the VSTOL terminal operations is another subintegration task within the NAVTOLAND project. Since Navy operations involve many different terminals, aircraft (CTOLs and VSTOLs) and systems for ATC, navigation, approach and landing, it is essential that the VSTOL terminal guidance systems packaged by NAVTOLAND use or be compatible with existing and planned systems.

The chart of the VSTOL guidance systems (figure 12) differs from that of the carrier counterpart by the specially identified sector of transition, precision hover and landing. In contrast to the absolute need of the CTOL aircraft to be stabilized on the precise speed and glide path in approach before getting to the commit-to-land "window", the most critical precision maneuvering task for the inherently unstable VSTOL machines begins at this just-off-the-touchdown position. In order to achieve all-weather operational capability, a precise landing guidance system must be developed to tighten the aircraft control loop from the current "seaman's eye" VMC work to a totally new experience of IMC maneuvering in close. The IMC/VMC transition would occur instant before the touchdown, requiring perhaps more stringent technique of using HUD or of achieving stable aircraft attitudes, etc. It is important to note that currently there exists no operational landing system, shipboard or shore-based, which can provide the precision guidance for VSTOL within about 0.2 mile of the touchdown point. Installing an existing so-called landing system aboard the air capable ship, therefore, would provide the VSTOL merely an approach aid. On the other hand, for such approach guidance systems which are operational elsewhere, aircraft are already equipped with the relevant avionics. For example, the ARN-28 receiver/recorder group is about to be installed in the U.S. Marine helicopters and possibly in the AV-8's for operation of the TPN-30 MRAALS (Marine Remote Area Approach and Landing System), a compact (100-plus lb) scanning beam microwave device. The UK MADGE (Microwave Aircraft Digital Guidance Equipment) also belongs in this category. It would be more efficient if such an "approach" system capability could be extended to touchdown, rather than to develop a new system with an inevitable aircraft payload increase.

One such "extension" concept under NAVTOLAND evaluation is depicted in figure 13. This approach uses the MRAALS or the forthcoming U.S. MLS (Microwave Landing System) for angular position coordinates sensing and the standard 4.3 GHz radar altimeter modified as a transponder to obtain the necessary precision DME capability (±1 foot accuracy).

Several other promising concepts have been identified and some analytical and evaluation work have been conducted. They include other microwave scanning beam system extensions, microwave-electro-optical hybrid, and an independent K-band FM/CW or pulse radar transponder system. An electro-optical system (e.g., laser tracker) suffers from inability to penetrate through certain fog situations. Detailed discussions of these candidates with respect to the NAVTOLAND program goal appear in Miyashiro's paper.(21)

The VSTOL and helicopters can be anticipated to operate within the next decade from some 200 to 300 ships of various types and tactical sites in different surroundings. The approach and landing profiles and procedures may be dictated by these operating conditions. Maximum standardization will be strived in the NAVTOLAND development and installation of the guidance systems for pilot's operational ease and safety. It is important for the guidance system development to explore whether or not and what degrees of steep, curved (both in plan and elevation) or stepped profiles may be needed for various reasons, e.g., minimum time, maximum descent rate, obstacle/enemy clearnnce, etc. Of course, these situational variations impose additional requirements to display parameters and to flight control laws, if coupled to guidance signals.

VISUAL LANDING AIDS

The Visual Landing Aids (VLA) include:

Dedicated displays or indicators for specific flight guidance parameters such as glide slope, line up, attitude, etc. The well-known "meatball" aircraft carrier optical landing system is an example.

Deck markings and lightings to enhance the pilot's perspective of the landing platform, e.g., white flood lighting, deck edge and centerline markings and lights.

Various natural cue elements which singularly or collectively give the pilot some secondary position, speed, attitude and other cues, such as grass on the runway, sea surface.

All of these three VLA elements must be organized so that they guide the pilot and do not present an inefficient clutter or not confuse the pilot.

Figure 14 depicts the general arrangement of the currently operational visual landing aid assortment aboard a typical air capable ship. The clutter factor is evident in this piecemeal and quick-patch evolved configuration. The "Drop-Line" lights line-up aid and the Glide Slope Indicator are remnant of the aircraft carrier VLA concept which, again, emphasizes precision in approach. The precision guidance in final landing, essential for VSTOL is addressed only by the "analog" judgement of the signal man.

Challenges which face the VSTOL VLA development are not insignificant. The lower weather minima of the project goal call for high VLA performance at visual threshold. That is, the VLA must be sensitive and flexible enough for the corresponding critical flying tasks and VSTOL-peculiar flight paths and above all must be comprehensive for the pilot's high mental workload state. The cockpit field of view limitations and precision maneuvering cue requirements demand close interface with and/or direct use of electronic guidance signal in the VLA design. These high performance VLA requirements would make the VLA even amenable for use as an adequate independent monitor for automatic landing.

Figure 15 illustrates the current VSTOL visual cue requirement study matrix which is being completed through a series of VSTOL/helicopter pilot workshops. The NAVTOLAND project will define the necessary VLA package configuration which is:

Integrated within the total visual scene of the pilot in all of the applicable segment in approach and landing;

Useable for each class of ships and tactical sites and respective VSTOL and/or helicopter operations;

Standardized across aircraft types, ships and tactical sites as much as possible.

Such VLA packages will consist of:

Existing lighting and optical systems validated as parts of the package (existing systems which are found to be inadequate, confusing or non-contributing would be eliminated);

Additional lighting schemes and other visual cue enhancement devices;
Other optical devices which may be coupled to non-VLA guidance and control sensors such as electronic guidance sensor, aircraft control systems or ship motion sensors.

SHIP MOTION INFORMATION

The ship motion information is required for VSTOL terminal operation in two forms, i.e., real time motion data and motion prediction or forecasting. The real time motion information is required for stabilizing various ship-installed guidance signal sources. No new technology development is required for this purpose; existing ship motion data in conjunction with any required additional data which can be obtained through future shipboard trials, tow tank research or computer simulation will adequately meet the project needs. The technique for forecasting ship motion lull period for pilot aid and guidance system (decision making) coupling is at an early stage of development. There seems to be a consensus that the high pilot workload and flight control system saturation in the VSTOL landing maneuvering flight do not permit deck chasing. Currently, only the pilot with his "seaman's eye," can judge for himself when a lull in ship motion may occur and attempt to land during that lull. For guidance system controlled landing, including automatic mode, the control loop must be rapidly tightened from the approach mode (necessarily stabilized against ship motions), through progressively shorter periods of motion prediction and compensation, to arrive at real time deck chasing at the moment of touchdown.

Several experimental schemes for ship motion prediction are under study. In one promising concept ship motion history is processed in mathematical filters. This approach applied in limited actual data indicated the possibility of forecasting ship motion within a one-cycle time frame of 7 to 10 seconds. Figure 16 shows an example of the computation. Additional computations are being conducted to verify merits of this scheme.

The use of artificially derived and displayed ship motion information by the pilot as a total information source in complete darkness or adverse weather or as a quantified aid to his seaman's eye will be a new experience and will have to be integrated into his control loop. One of the important tasks is to determine and develop method (location, symbology, dynamics and integration) of displaying such ship motion information to the pilot.

AIRCRAFT SECURING SYSTEM

If the aircraft has no adverse thrust problems and has adequate stability augmentation system, as in the case of the helicopter, a rather brute but quite effective mechanical hauldown device such as shown in figure 17 can be used. Some past shipboard tests and analytical data indicate that in Sea States beyond 3, it would be extremely difficult to hold the aircraft on the deck in ensuing large ship motions. The NAVTOLAND project will monitor this LAMPS RAST (Recovery Assist, Secure and Traverse) system development while studying shipboard securing requirement for the fixed-wing VSTOL vis-a-vis the respective operating ship sizes and motion characteristics.

AIR WAKE INFORMATION

The air wake and WOD(wind-over-the-deck) are major factors in affecting handling qualities in the close-proximity terminal operation and major operational envelope limiting parameters. Yet, little is known about the characteristics of the air flow in the vicinity of small air capable ships. The operational limits and pilot's experience in coping with adverse effects of air wake turbulence are obtained by a process of trial and error. AGARD Report 594 also warned about particular problems for VSTOL aircraft in approach and landing under high cross winds and recommended theoretical and flight investigation in this area to be carried out as soon as possible. [1]

The Navy has initiated a wind tunnel model test and analysis of a mathematical model of the turbulence. A laser doppler remote wind measurement device has been tested aboard a carrier and is currently being proposed for additional tests.

While a consensus exists that detailed knowledge of ship air wake is necessary, how one might use such information for VSTOL operations does not yet seem to have been thought out. For example, how would a detailed mapping of turbulence in the landing area help a pilot who must continuously react to a gust as it hits the aircraft in "real time"?

An inconclusive but interesting finding of the NASA Langley VALT flight research is that the high gain flight control system, as was available in the CH-47 variable stability vehicle, appeared to "cancel out" air turbulence effects. To what extent the severe turbulence of the air wake in the shipboard environment can be dealt with the flight control system gain is not yet known.

SIMULATION FOR NAVTOLAND DEVELOPMENT

It was suggested earlier that an overall aircraft control loop simulation might be initiated. Such a general purpose or total system simulation is needed for specific type aircraft development anyway and the NAVTOLAND project can "piggy-back" on it to develop and test the total NAVTOLAND package capability. A number of different kinds of other simulation efforts are needed also to accomplish the development of various NAVTOLAND sub-elements. These includes: mathematical modeling for quantitative, comparative analyses of candiate system concepts through prototype development; fixed base man-in-the-loop simulation for broadbase comparison by pilot's evaluation of competing concepts through prototype development; and moving base, high fidelity man-in-the-loop simulation for evaluation and development of prototype systems and associated pilot techniques and flight procedures. These simulation facilities will be used in support of four functions: problem analysis and concept evaluation; prototype hardware development; flight test; and pilot techniques/procedures development.

NAVTOLAND PROJECT PLAN

Figure 18 is an overview of the NAVTOLAND project. The effort to date has consisted of a number of research and exploratory development programs in various supporting technologies for feasibility investigations. The development of subsystems prototypes is planned to begin fiscal year 1979. These prototypes will be installed in one VSTOL and one helicopter testbed aircraft and flown into and out of testbed VSTOL operating ships and tactical sites equipped with respective ship or ground units of the NAVTOLAND pilot aid subsystems. The feasibility demonstrated NAVTOLAND capability package will be disseminated to the VSTOL aircraft development offices and industry in the form of design guidelines, specifications and regulations as well as installed in the fleet as a standardized set of aircraft guidance and control systems on the respective VSTOL operating ships and tactical sites. Figure 17 also shows how various simulation efforts are used to complement the final flight proof of the NAVTOLAND package capability.

Figure 19 illustrates the "two pronged" approach of the NAVTOLAND capability to ensure that the capability goal achieved will be a cost-effective, technically logical extension of interim capabilities which the various helicopter and VSTOL aircraft developmental and operational communities are evolving. The NAVTOLAND project office will monitor and participate in these interim activities by timely inputs of project results as they become available as well as judicious midcourse reorientations to ensure minimum retrofit of existing capabilities.

The NAVTOLAND project master schedule is shown in figure 20. Various subsystems advanced development models will be individually developed and tested before the total capability package demonstration. The simulation work will serve to integrate and monitor subsystems development during the project to maintain the total system perspective.

CONCLUSION

The NAVTOLAND project will improve the all-weather and rough sea operational capability of the VSTOL and helicopter fleet, that is, some two hundred air capable ships in the U.S. Navy, to the level heretofore possible only with the sophisticated aircraft carrier systems.

The VSTOL terminal operational capability data produced by the NAVTOLAND project should be applicable universally to any VSTOL and helicopter operation at sea or on land. NATO-wide collaborative data and expertise exchange and developmental programs in the areas addressed by Project NAVTOLAND would be productive for mutual benefit.

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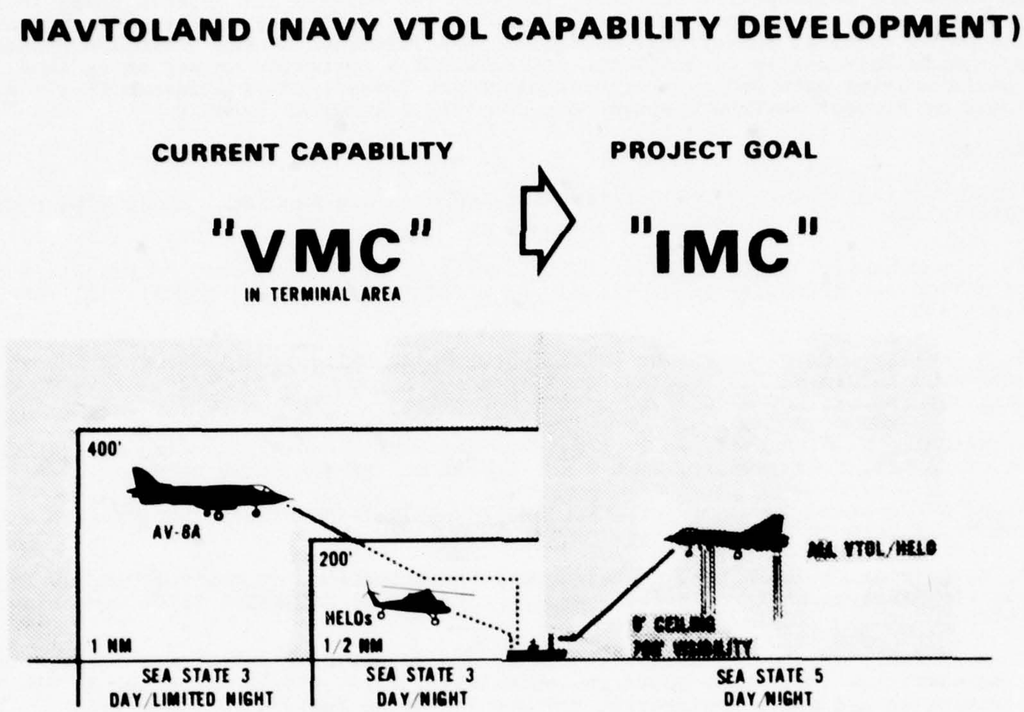


Figure 1

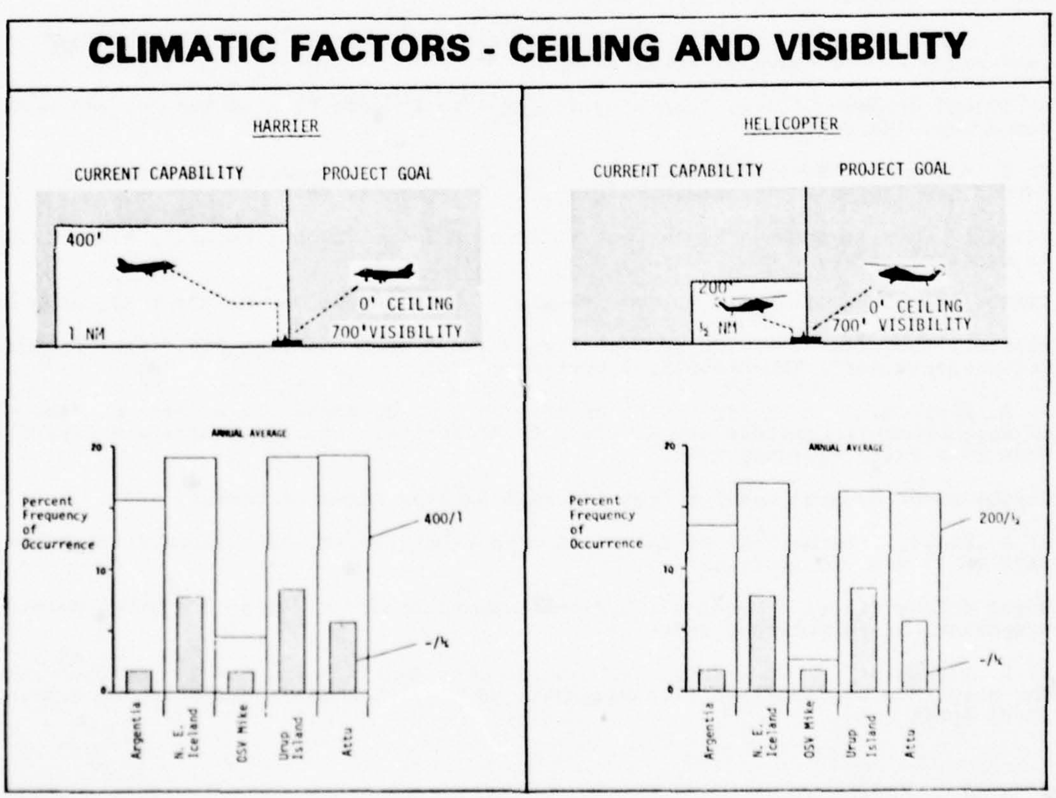
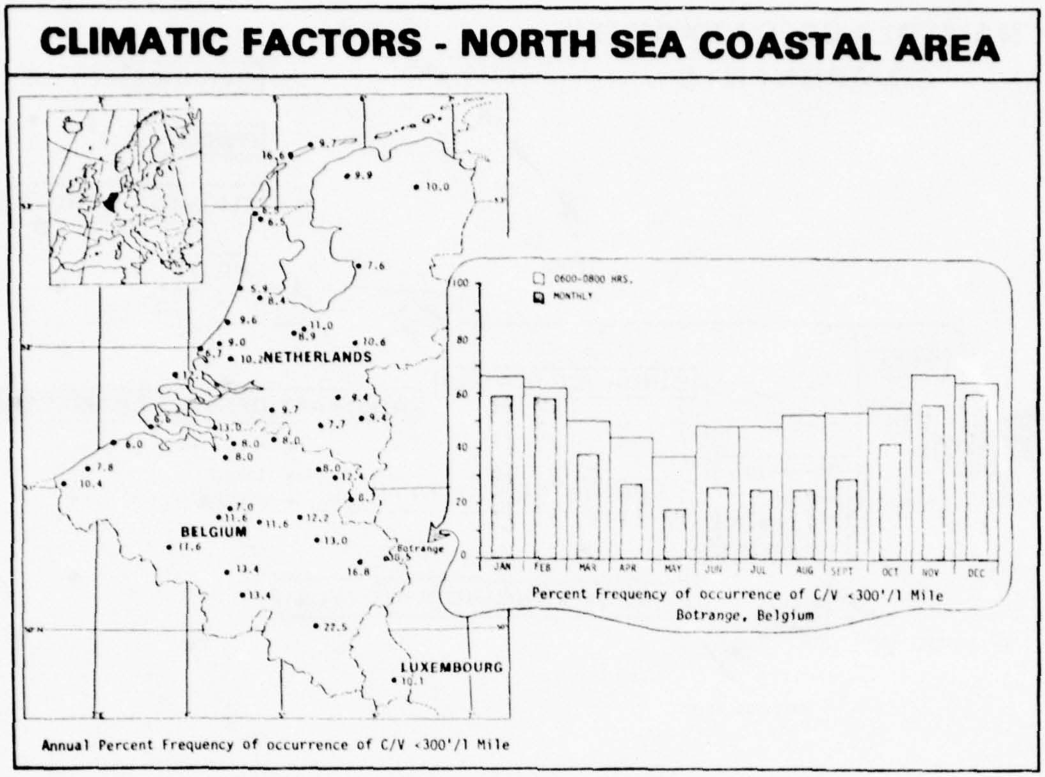


Figure 2



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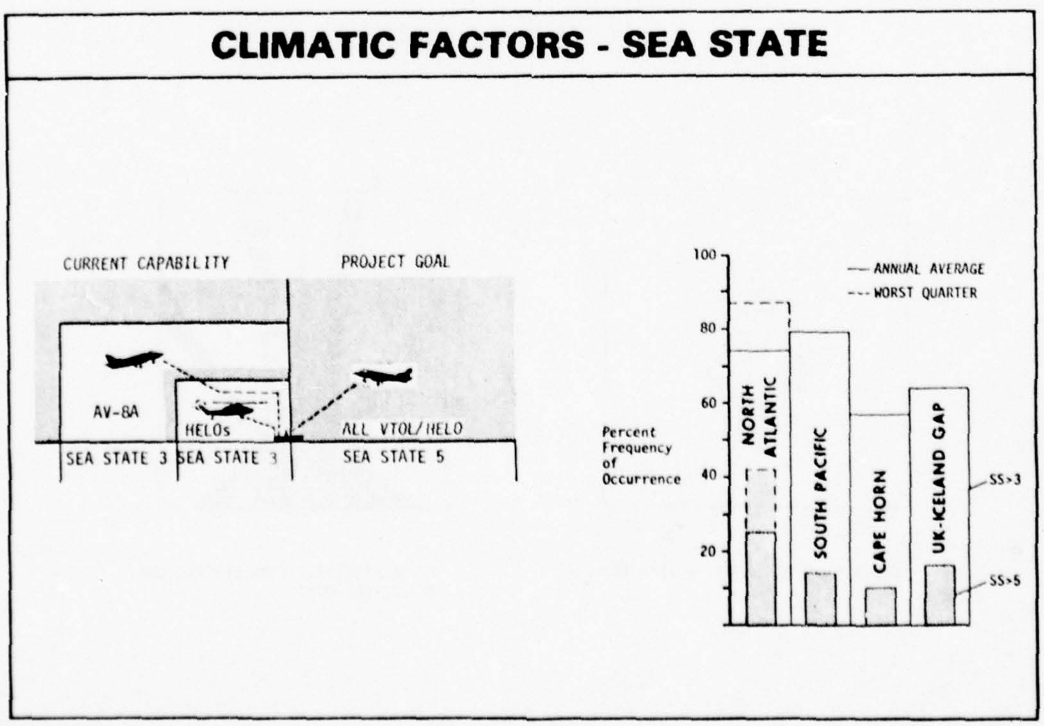


Figure 4

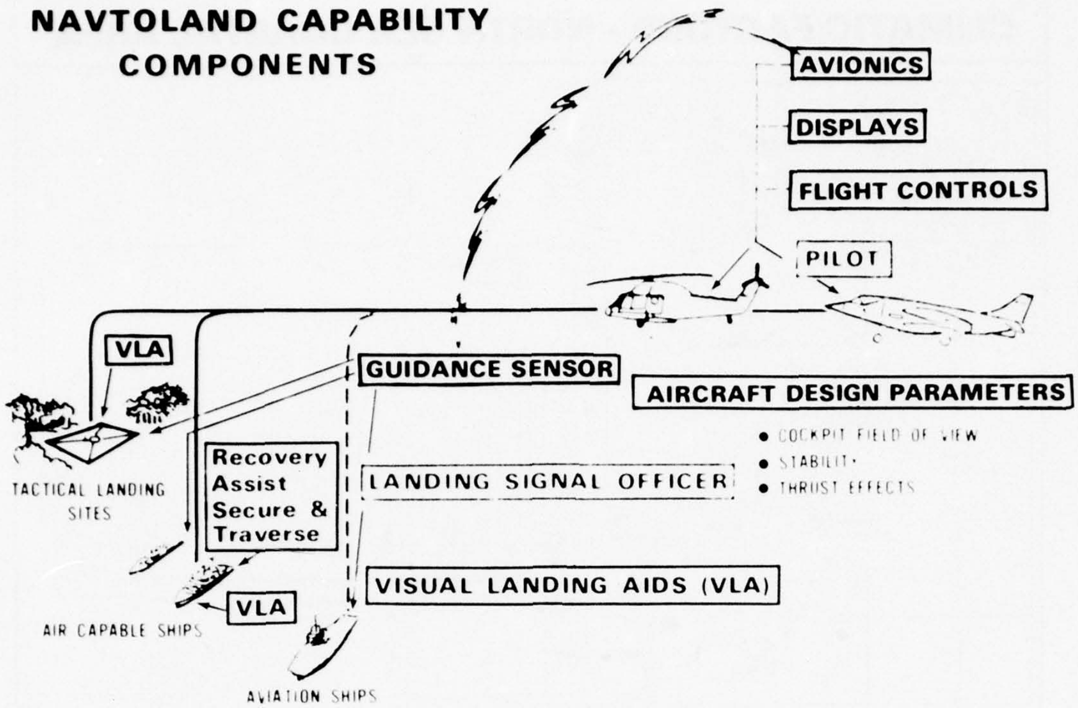
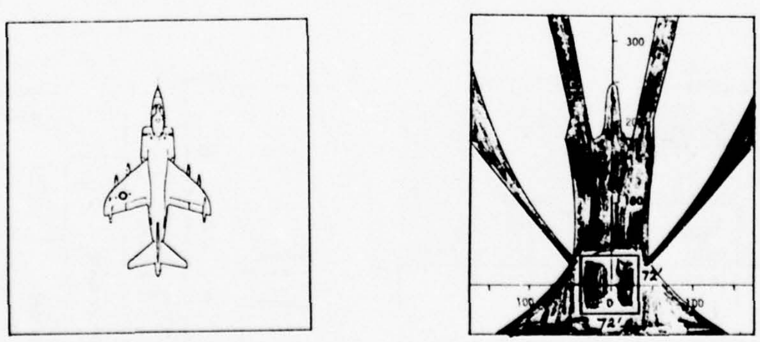


Figure 5

COCKPIT FIELD OF VIEW PROBLEM

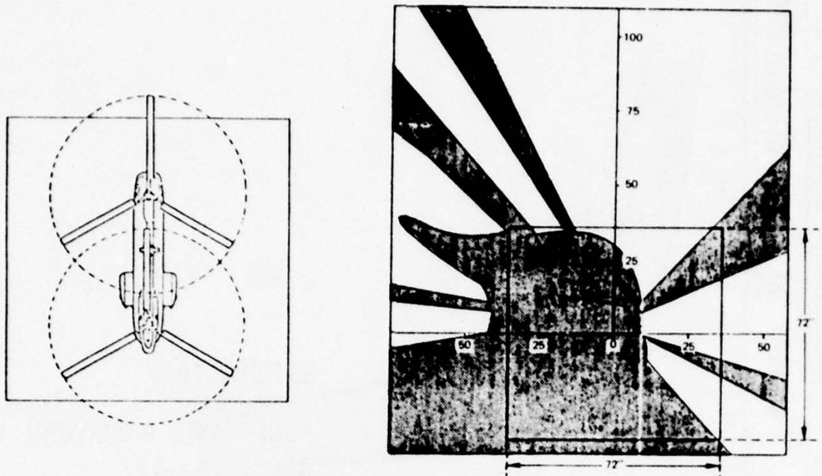


AV-8A AIRCRAFT ON 72' x 72' PAD

AV-8A PILOT'S VISION OBSTRUCTION
AT 50 FOOT HOVER

Figure 6

COCKPIT FIELD OF VIEW PROBLEM



CH-46D HELICOPTER ON 72' x 72' PAD

CH-46D PILOT'S VISION OBSTRUCTION AT 10 FOOT HOVER

Figure 7

FLYING QUALITIES FACTORS AND NAVTOLAND SYSTEMS

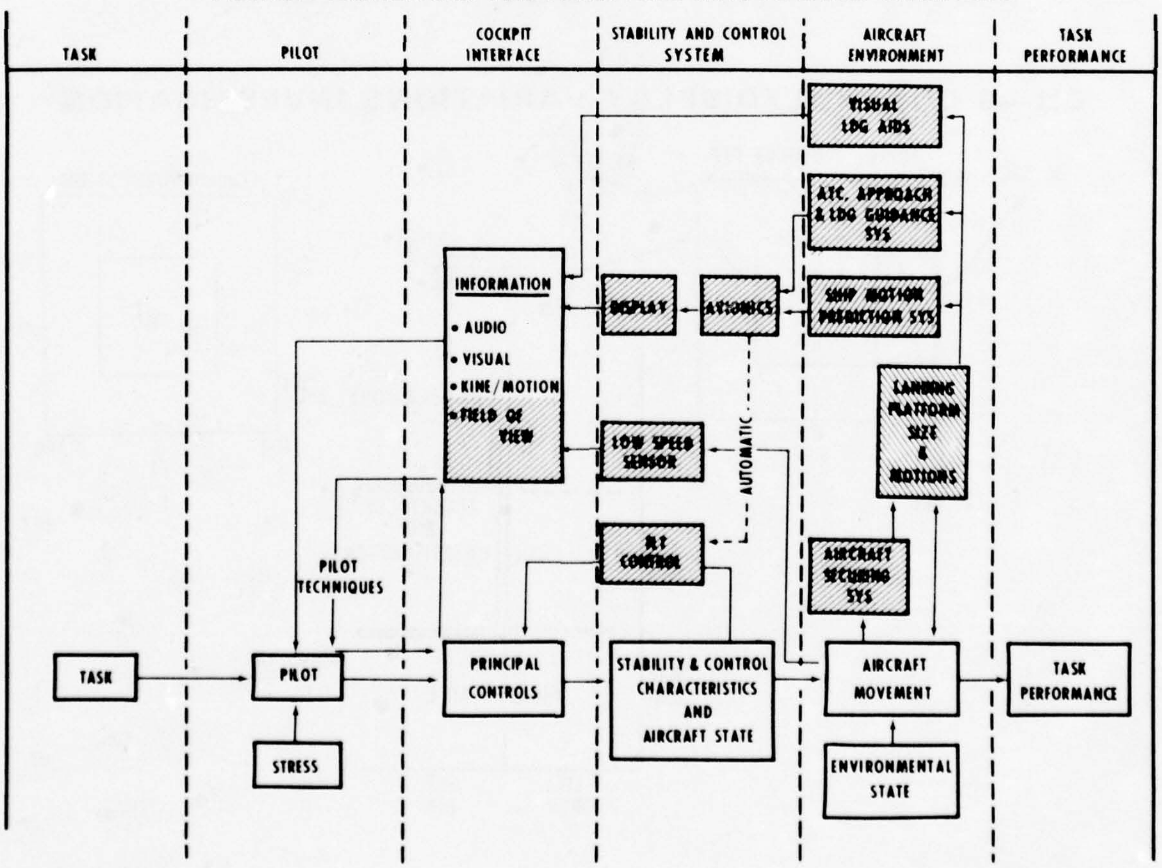


Figure 8

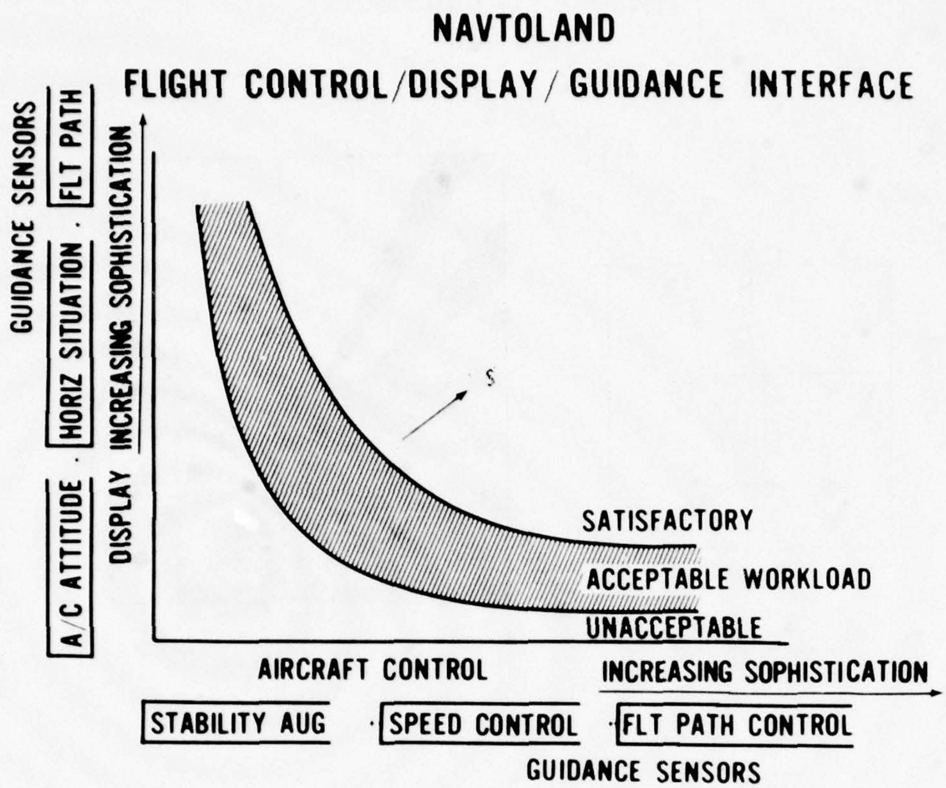


Figure 9

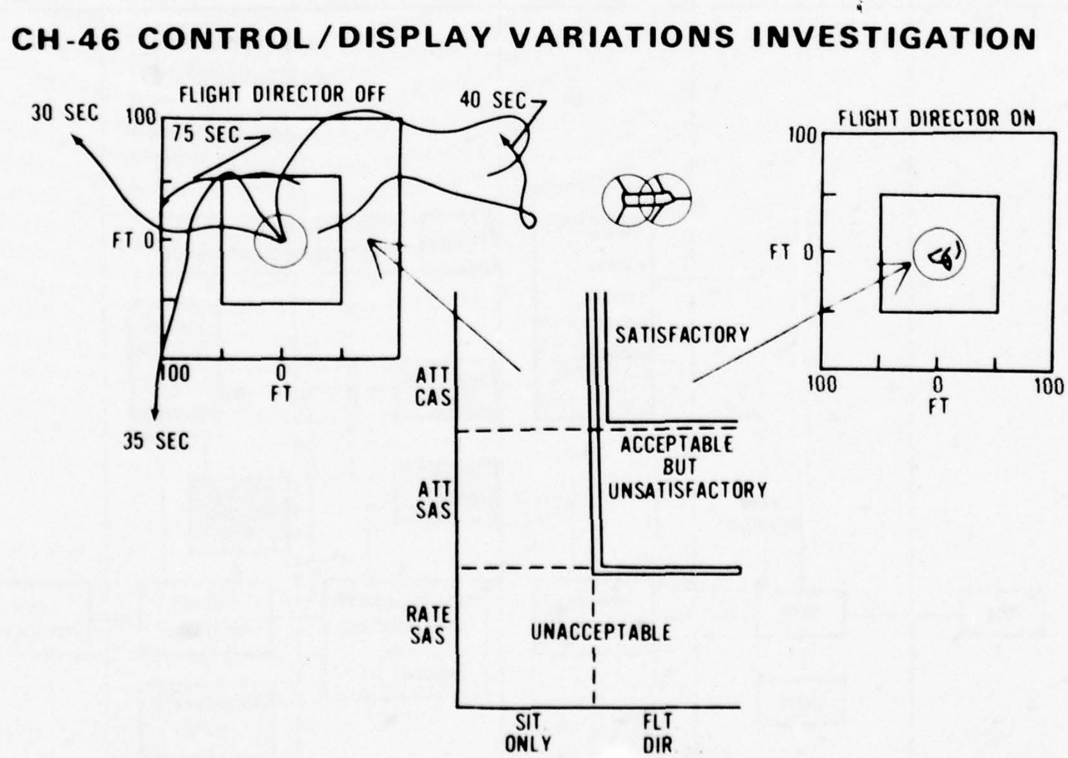
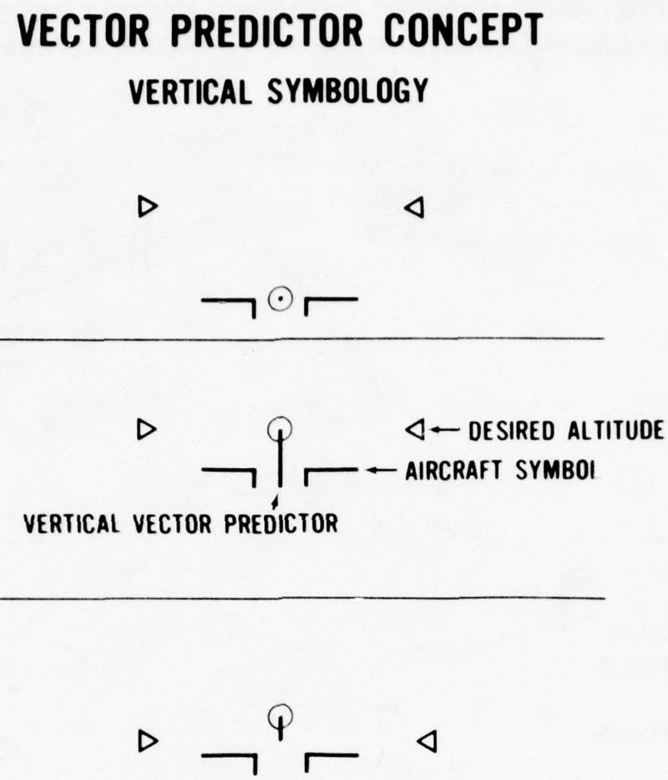


Figure 10



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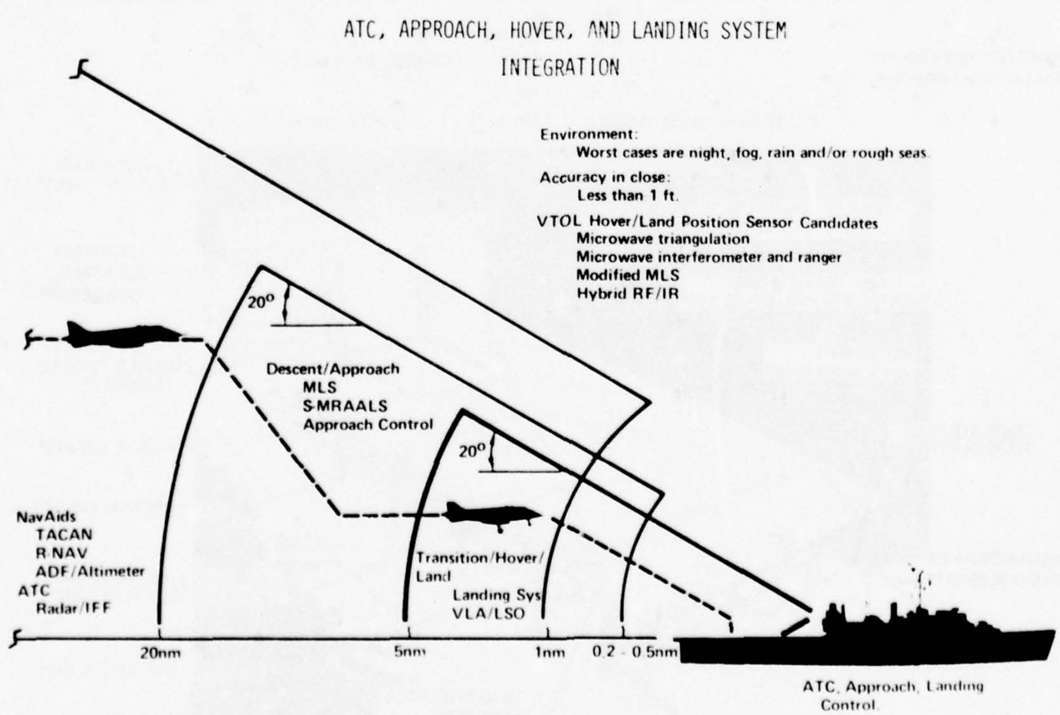

Figure 12

NAVTOLAND
PRECISION LANDING GUIDANCE SYSTEM CANDIDATE
Ku-BAND SCANNING BEAM WITH RADAR ALTIMETER/DME

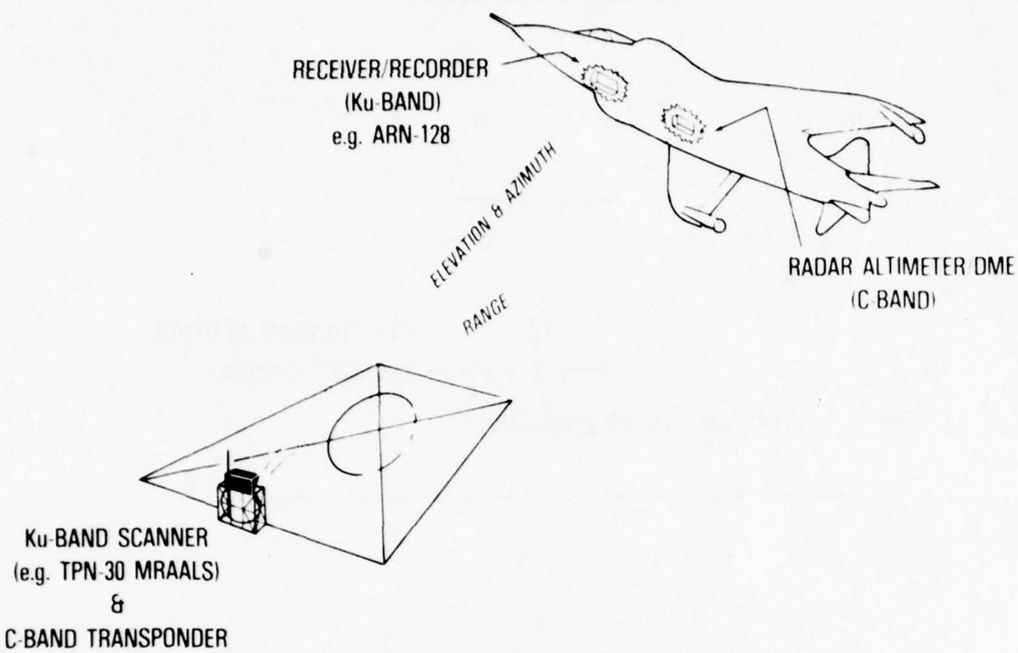

Figure 13

AIR CAPABLE SHIP VISUAL LANDING AIDS GENERAL ARRANGEMENT

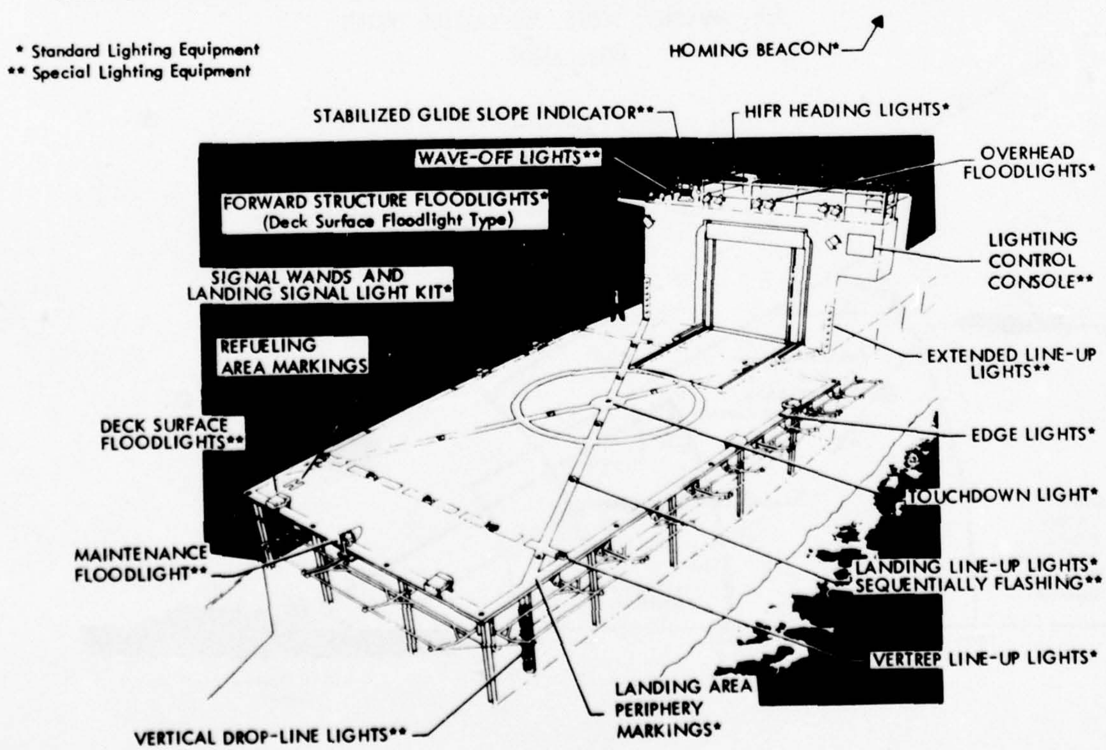

Figure 14

Pilot Information Requirement Matrix

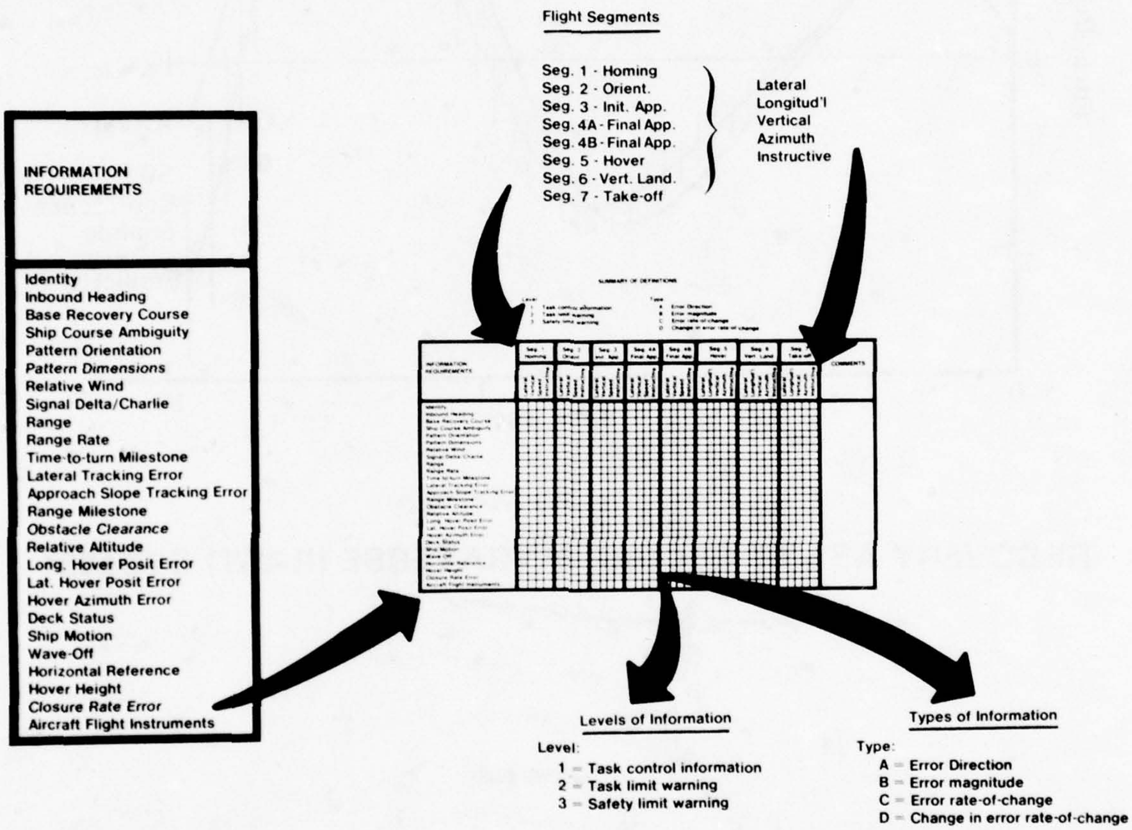

Figure 15

SHIP MOTION FORECASTING

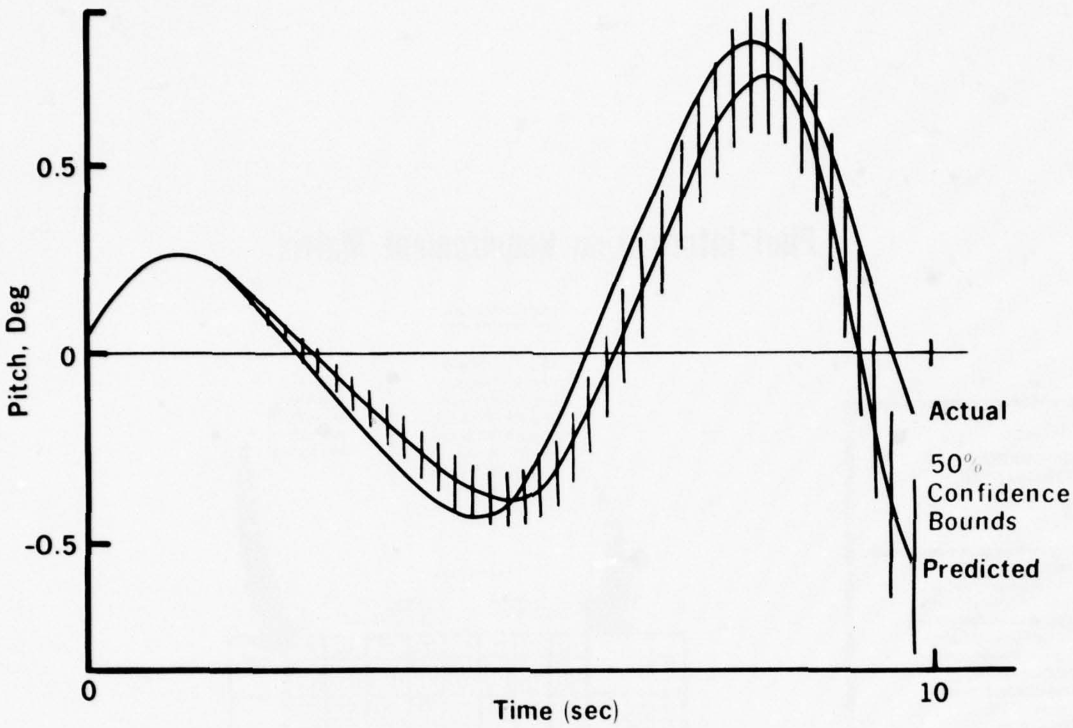

Figure 16

RECOVERY ASSIST SECURE & TRAVERSE (RAST) SYSTEM

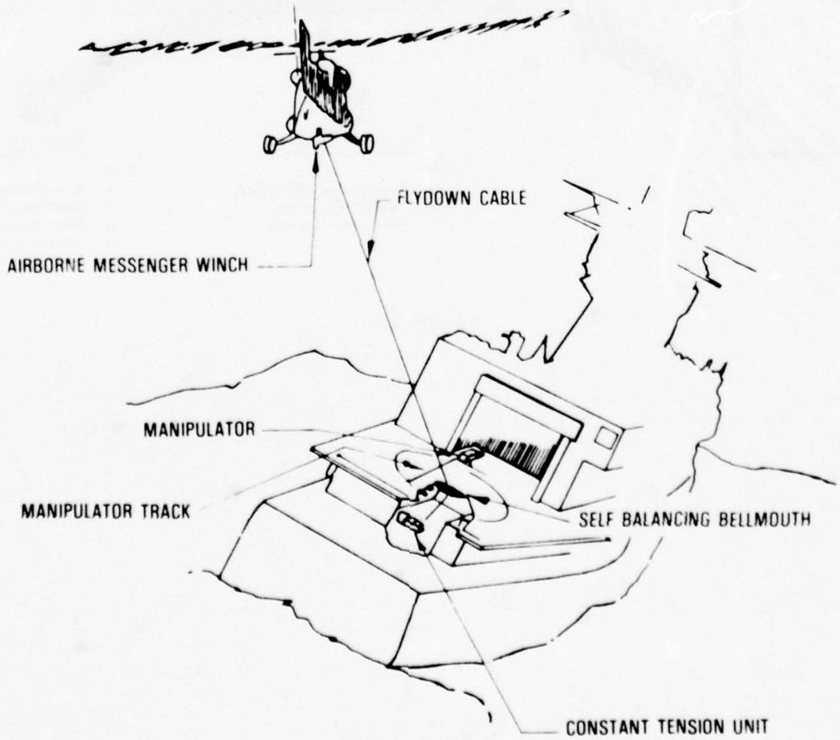

Figure 17

NAVTOLAND PROGRAM

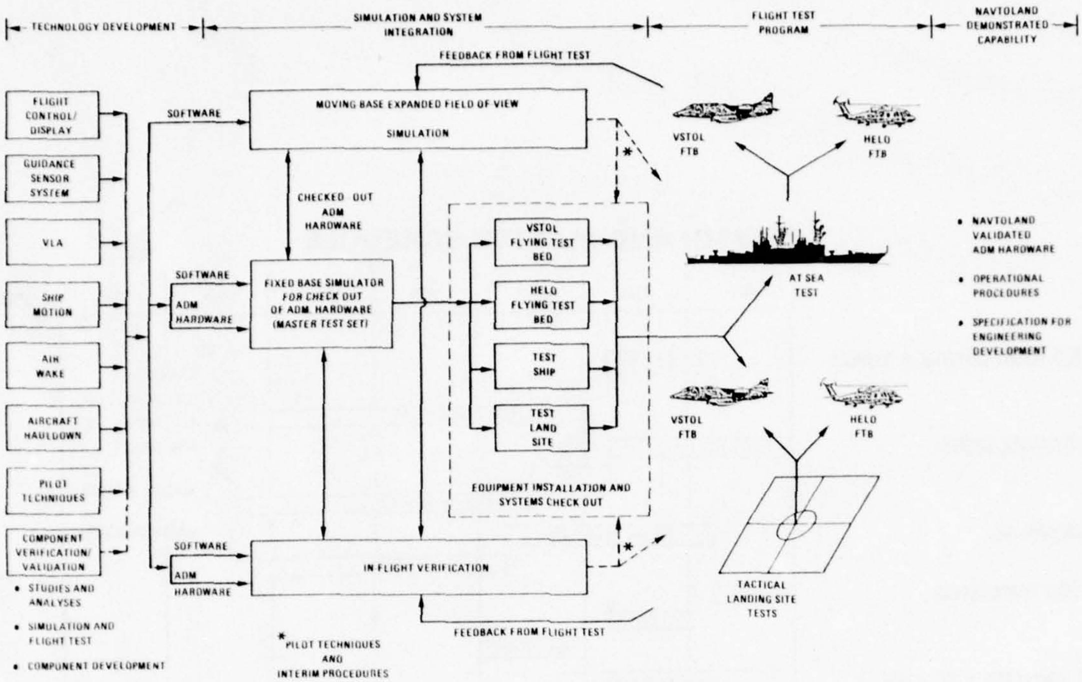

Figure 18

NAVTOLAND METHODOLOGY

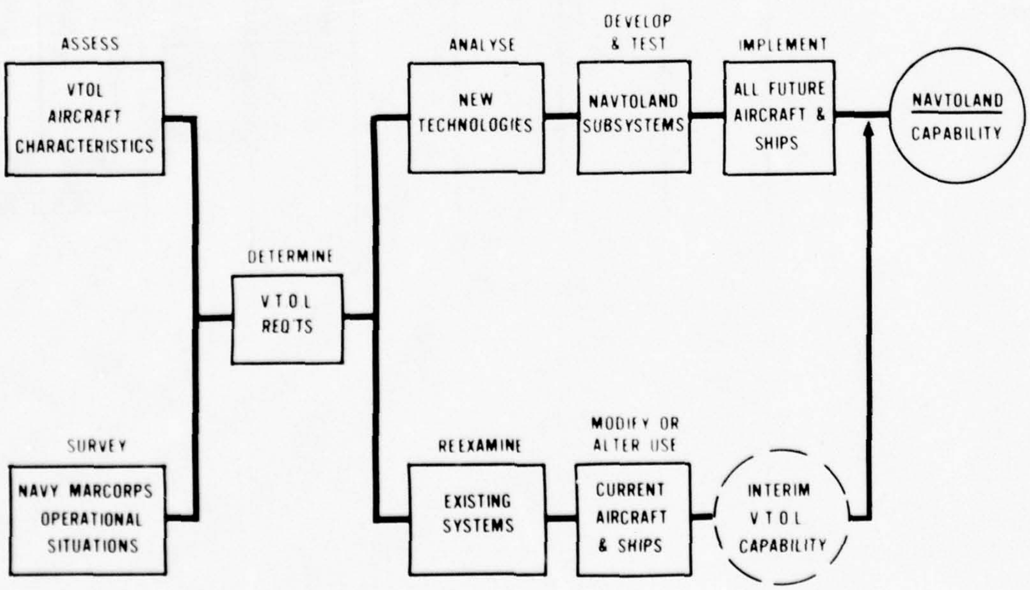

Figure 19

NAVTOLAND MASTER SCHEDULE

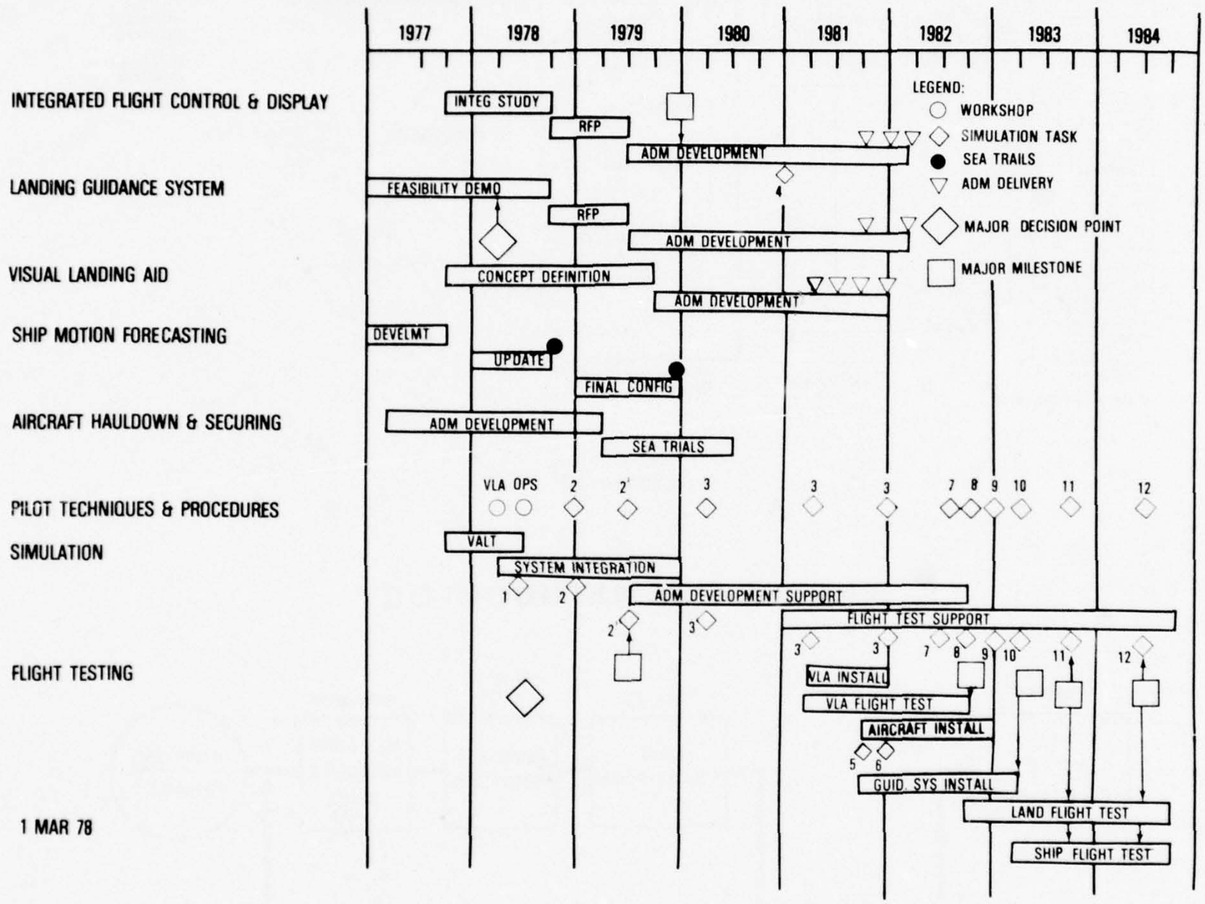

Figure 20

GCU, THE GUIDANCE AND CONTROL UNIT FOR ALL WEATHER APPROACH

by

Hartmut Böhret
Bodenseewerk Gerätetechnik GmbH
D7770 Überlingen
Germany

1. Summary

Making use of the properties of microwave landing systems, the problems of terminal area capacity, noise impact, obstacle clearance and adverse weather conditions can be overcome by application of adequate flight guidance.Utilizing the SETAC-MLS, the guidance and Control Unit GCU developed by Bodenseewerk and sponsored by the German Ministry of Defense, demonstrated in flight test the improvements of future landing procedures. The short-captured steep approach paths generated by the GCU can be flown manually with the flight director instrument due to the high accuracy of signal processing by means of Kalman filter techniques. The paper will present the technical equipment and discuss the flight test results.

2. Introduction

Radio based landing systems are used to assist the pilot during approach and landing. A safe landing under poor visibility conditions is not possible without these landing aids. During decades, the instrument landing system (ILS) in civil aviation and the ground controlled approach (GCA) in military aviation have been the only means available. The disadvantages of both ILS and GCA led to great efforts in the development of new landing systems based on microwave techniques.
Because of necessary terrain smoothing, the ground equipment of ILS is so expensive, that only great airports can be supplied. GCA needs a well-trained ground service for the data transfer and there are no electrical signals airborne available which could be fed to the autopilot to perform an automatic landing as far as touch down.

A further disadvantage of both ILS and GCA is the missing flexibility. The commanded flight path is rectilinear and fixed in centerline with an elevation of about 2.5^{o}. This angle has to be so small to ensure, that the admissible rate of descent is not exceeded near the ground by fast fixed wing aircraft.

The most important improvements of microwave landing systems are expected with respect to

a) Operational benefits (lower minima, reduced delays, capacity increase, standardization, signal availability, new operational capabilities, reduced workload)

b) Technical benefits (improved performance, higher guidance flexibility, wheel crossing height compensation)

c) Environmental benefits (noise reduction with vertical and horizontal segmented or curved approaches)

d) Safety benefits (precision approaches, obstacle clearance, improved performance, improved crew confidence, flare guidance, auxiliary safety data)

e) Military benefits (improved mission capability, expanded operational applications, ground fire avoidance, elimination of GCA personal hazards).

Most of the benefits mentioned above only turn to account by processing the original MLS data (angle deviations from the preselected glidepath and DME) in a guidance and control computer.
Since 1970 a program has been sponsored by the Ministry of Defence, parallel to the development of the military MLS SETAC, in order to improve the operational facilities of microwave landing systems. This program led to the development of the Guidance and Control Unit GCU by Bodenseewerk.

In 1973 the first flight tests with an analogue signal processing of steep gradient curved approaches successfully were performed but disclosed the limits of that technology. After demonstrating the improvements in guidance precision with a digital general purpose computer in 1976, the GCU as a special purpose computer came into its development phase. An experimental version of the GCU, consisting of data processor, memory and interface modules for input/output data transfer was flight tested in the last months using the MLS SETAC and a Dornier Do 28 Skyservant aircraft.

3. MLS SETAC

In Germany, during the last years, the approach and landing system SETAC was developed for military service.
The evaluation of signals transmitted from the mobile ground stations is performed in the TACAN (or MITAC) airborne set and in the SETAC supplement.Within a great coverage volume, the SETAC system provides the pilot with all essential information during approach and landing:

- deviation from selected landing course
- deviation from selected glide path
- precise distance to runway stop end
- precise distance to touch down point
- absolute elevation angle
- sector azimuth with reference to centerline
- omnidirectional azimuth for finding the approach sector and for overflight guidance
- azimuth angle of centerline with reference to north

These eight output signals are available to users equipment (displays, autopilot, flight director, GCU) in a digital serial data format.

The serial data channel is insensitive against induced failures and easily can be supervised by parity checks. For this reason there is no loss on accuracy of the SETAC-signals by airborne data transmission.

4. Guidance and Control Unit GCU

Using the MLS in the same manner as the conventional ILS, most of the MLS benefits dont become effective. The angle deviations of the preselected approach path are producing the following disadvantages:

- different localizer intercept procedures depending on distance (long straight-in final for stabilizing the aircraft) (fig.1).
- increasing sensitivity of the localizer and glide path deviations during approach (instability of pilot or autopilot due to the cone effect) (fig.2)
- overshooting the glide path at steeper elevation angles or segmented approaches (fig.3)

The Guidance and Control Unit GCU solves these problems by means of

- signal filtering
- adaption of the signal sensitivity
- coordinate transformation
- generation of curved flight paths
- wind estimation

as it is shown in the block diagram (fig.4). Flight director, autopilot and cross-pointer instrument are supplied with analog control corrections to perform curved approach pathes for short capture and steep descent (fig.5, fig.6).
The parameters of these approach paths are preselectable within a wide range.

4.1 Signal filtering

The accuracy of the MLS data depends on the distance of the aircraft to the transmitter multipath effects, weather conditions (rain, snow) and airborne antenna location. These influences extend from a small augmentation of signal noise up to short period signal black out.
For the generation of curved flight profiles and for coordinate transformation of the MLS data, the signal errors must be kept as small as possible. Therefore all SETAC data are passed through Kalman filters which have two different modes to reduce or to eliminate these errors. One mode is the normal filter mode where the r.m.s. value of the signal noise can be reduced by the factor 5 to 10 depending on filter gains. As long as the difference between the actual measurement and its predicted value is above a defined level, the filter algorithm ignores the measurements and uses the prediction mode only. In this mode the false input signal is not used and the output signal proceeds according to the process-model. By this way a signal loss can be suppressed completely and therefore, additionally to its normal task, the filter has a monitor function for supervising the MLS data.

4.2 Coordinate Transformation

For flight guidance along complex approach paths, the position of the aircraft within the terminal area has to be known exactly in orthogonal coordinates. Since the MLS data because of its physical origin are delivered as angles and distances with respect to the ground transmitter stations, a coordinate transformation has to be performed.
The result of this transformation is the height above runway level (H), the distance to the azimuth transmitter projected on centerline (X), the distance to the elevation transmitter projected on centerline (R), and the distance perpendicular to the centerline (L). The origin of the coordinate frame is for L and X the location of the azimuth antenna and for H and R the point on centerline across the elevation antenna. The height above runway level (H) would be very noisy at great distances by using only MLS data for calculation. Therefore the barometric altitude is additionally taken for complementary filtering, where the high frequency signal parts are used from the barometric sensor, and only the low frequency signal parts are used from MLS data. The GCU has a built-in barometric sensor in the form of a piezoceramic element, and

therefore does not depend on other airborne equipment.

4.3 Approach profiles

Flight tests have shown, that segmented flight paths for achieving steeper approach angles have decisive disadvantages. The transition from the horizontal flight to an angle of for instance 6 degrees, and in particular the transition near the ground from 6 degrees back to 2,5 degrees, results in increased pilot workload and in overshooting of the aircraft. Therefore these approach paths have to be curved. Fig.5 shows the principle of the curved elevation flight paths, generated by the GCU.

The aircraft approaches from a larger distance at constant but selectable altitude and intercepts in the outer section, as for the ILS, a nominal path which is rectilinear and which has an inclination of 2 degrees. The overshooting during transition to the steep approach is avoided by the fact that the inclination of the nominal path is increased with a smooth curve to a preselectable value which is admissible for the respective aircraft. This value is maintained throughout the main part of the approach which is rectilinear aigain.
In order to reduce the sink rate to a safe level near the ground, the path angle is reduced by means of a smooth curve in the last part of the final.

In azimuth (fig.6), the aircraft intercepts in the outer part of the approach a flight path which has a bias angle with respect to the center line. This bias angle is selectable from plus to minus 19 degrees with steps of one degree.
Between 7 and 3 miles to the azimuth antenna on the runway stop end, the aircraft is brought back to the centerline in a smooth s-turn flight path.

The flight guidance along curved flight pathes in the final approach only can be performed safely by the computation of precise steering commands for the pilot or the autopilot. Therefore, the output data of the GCU are not only the vertical and lateral deviations ($\Delta H, \Delta L$) of the aircraft from the nominal flight path. Additionally command signals for pitch angle (θ), pitch rate (q), bank angle (ϕ) and heading (Ψ) which are variable during approach, are generated.
These command signals are compared with the actual aircraft data and the difference is fed to the flight director computer and to the autopilot in order to avoid deviations from the curved flight path elements.

4.4 Standard Intercept Procedures

By using the MLS signals in the conventional ILS data format some problems will arise when the azimuth flight path is intercepted (fig.1).
The GCU data format, however, enables standard intercept procedures which are independent from distance and wind conditions.

When the aircraft is flying on base leg with its intercept heading Ψ_o, the pilot gets the exact information when he has to turn to final, in order to meet the azimuth flight path without considerable deviation. The beginning of the final turn is displayed at the flight director instrument, the crosspointer instrument and at the control panel of the GCU. At the control panel, the pushbutton lights are switched over from the armed mode (ARM), which is amber during the base leg, to the capture mode (CPTR), which is green during the final turn. The end of the final turn is displayed by the switch-over from the capture mode to the approach mode (APPR), likewise in a green coloured light.

At the flight director instrument, the beginning of the final turn is indicated at the same time by a turn command, which enables the pilot to perform this procedure easily.
If the aircraft is not equipped with a flight director, the final turn can be flown as well with the crosspointer instrument. The sensitivity of the crosspointer instrument is continously processed during the base leg in such a way, that at the very beginning of the final turn the vertical bar has reached ist two dot deflection. A standard turn with a yaw rate of 3^o/sec initiated at this time will bring the aircraft exactly to the approach profile.
This procedure can be performed from far to very short distances to touch down. For the starting point of the final turn the actual wind conditions are taken into account and therefore the approach path can be met co precise that a long stabilizing phase is not necessary.

4.5 Hardware Aspects

The experimental model of the Guidance and Control Unit GCU consists of a digital computer unit and a control panel (fig.7). The computer is mounted within a size of 1/2 ATR short with an external power supply unit added at the back side of the case.
The computer has a modular structure where the several modules are corresponding with the central processor unit by a parallel 16-bit bus system.
The use of C-MOS technique combines short access time with low power consumption.
Only half the case volume is covered by the descrete central processor unit and the data memory of 4 k PROM (programmable read only memory) and 1/4 k RAM (random access memory).
The other half is needed for interface modules to communicate with airborne sensors and displays due to the fact, that even modern aircraft are equipped with all different types of electrical data formats, e.g. analog AC-DC and digital serial-parallel.

The GCU has already been flight tested in a Dornier Do 28 Skyservant. In the near future it will also be tested in a Transall C160 transport aircraft and a Bell UH 1D helicopter. For these aircraft the following interface modules were used:

A/D-converter for the analog DC-signal of the barometric height sensor and for self-test of the GCU by wrap-around of the output signals.

D/A converter for analog converting of the output signals to drive the flight director computer and the crosspointer instrument.

D/D converter for the parallel digital input signals from the control panel and the mechanical digits which determine the parameters of the approach profiles.

D/D converter for the parallel digital output signals for flags and pushbutton lights of the GCU control panel.

D/D converter for the serial to parallel convertion of the digital input signals from the airborne SETAC receiver.

AC/DC converter to convert the AC-signals from the airborne gyro platform into DC-signals.

DC-interface module for adaption of different DC signal levels.

Teletype-interface for communication with the central processor and for program modification in the development phase.

RTC/PASS module for real time clock and programmable automatic supervisor system.

Fig. 8 shows the modular structure of the GCU hardware. Due to the fact, that the great number of interface modules is decisive for the failures rate of a digital computer, redundancy or failure self-detection has to be provided. The GCU was developed for operation on a Cat II environment and failure selfdetection by software and hardware means was applied.

4.6 Failure Self-Detection-Procedures

The GCU's failure self-detection consists of three different elements:

- built-in test program (BITE)
- inflight test routine
- programmable automatic supervisor system (PASS).

The built-in test programm can be started by pressing the TEST-button and a time of 10 seconds is required to carry out the following tests:

- CPU test

A complete check of the instruction repertory is performed. A test program is worked out by presetting constant input values. By utilizing all instructions available in the processor, the comparison with known nominal values results in failure detection of the processor hardware.

- ROM test

The contents of the read-only memory (ROM) are verified by parity-check in the columns

- RAM test

The working storage (RAM) has to be checked in a different manner than the ROM, since the contents of the RAM change continously and are therefore not known. Memory cells, address registers. decoder network, read and write amplifier and switch over unit for read/write are checked by software means using complementary input values.

- Interface Test

A/D and D/A-channels are tested by wrap-around procedures of ramp-functions by using software and the crosspointer instrument for monitoring.
D/D-channels are tested by using the control panel lights as a monitor. The serial dataway and the serial-parallel convertion are tested by comparison of constant test-values of the SETAC-dataword with its nominal values. The constant test values are generated by the SETAC airborne receiver in its own test mode. During the BITE-loop, the commanded flight pathes in azimuth and elevation are processed as a function of distance and can be monitored.

Since the BITE program with an endurance of 10 seconds only can be performed when the aircraft is located outside of the terminal area, the confidence level of the GCU is further augmented by an inflight test program, which is worked down during the program intervals available between the computing cycles. This inflight test program has only a reduced capacity, but the essential elements as CPU-test, A/D-D/A wrap-around and monitoring of the power supply unit are implemented. The serial dataway is continously supervised by parity checks.

In contrast to these software means the programmable supervisor system (PASS) ensures by an external self-contained hardware unit the regular program run. It operates in principle as a watch dig timer. Unless a reset instruction comes in time to the supervisor from the processor, a flashing signal is given to the fail-lamp at the control panel and all output signals can be disconnected from autopilot, flight director and crosspointer instrument. The pilot is additionally informed by setting the respective instrument flags.

5. Flight Test Results

The flight tests of the GCU 70 were performed with the SETAC microwave landing system available at the northern runway of the German Air Force test center in Manching. At first, a Bodenseewerk-owned Do 28 D Skyservant was used, equipped with a SETAC airborne receiver, a flight director and the Guidance and Control Unit GCU. The following stages of flight testing the GCU will be the installation in a Transall C 160 and a Bell UH 1D helicopter. With the Skyservant, the demonstration flight series consisted of 138 approaches which were curved in azimuth and elevation and which were all flown manually with the flight director.
To simulate the later Transall tests all approaches were carried out with a speed of 120 knots so that the bank angle command signals could be adjusted for this speed. In azimuth all bias angles from 0 to 19 degrees were tested, whereas in elevation most of the approaches had a maximum path angle of 5 degrees due to the high approach speed. At lower speeds, path angles up to 7 degrees could be flown with the Do 28.

The intercept distance normally was about 10Nm, but varied in some cases from 5 Nm up to 18 Nm, in order to prove the accuracy of the standard intercept procedure at all distances and under adverse wind conditions.

Fig.9 shows the flight record of 4 successive approaches in azimuth with different intercept angles. The overshooting of the aircraft at the end of the intercept turn is less than 100 m, and the aircraft follows the commanded flight path even at the curved segments with a maximum deviation of 20 m.This accuracy could be obtained even by unexperienced pilots up to bias angles of 10 degrees. At greater angles, the final approach in centerline direction, which began at a distance of 1.5 Nm to runway threshold, seemed to be too short. Some pilots asked for a better information about the nominal value of the course angle, especially during the path segment between 7 and 3 Nm where the difference between actual course and centerline course comes up to a maximum of 50 degrees.

Because of the shooting activities in the Siegenburg range, which is located only a few miles southerly of the extended runway centerline, approaches in centerline direction could be flown only during few hours per day. Therefore approaches with bias angles greater than 10 degrees had the great operational advantage to avoid this range, and to be not subjected to any time restrictions.

Fig.10 shows the time history of a curved approach in elevation. The aircraft intercepts the curved profile at a distance to touch down (R) of 10 km at a height (H) of 600 m. The height deviation (ΔH) is less than 5 m, even at the curved segments. The pitch angle (θ) is following the commanded flight path angle (γ_c), a result of constant angle of attack and constant indicated airspeed.

The elevation profile, as the azimuth profile, easily could be flown very accurate manually with the flight director. Only the power setting at the curved path segments demanded some practice because there was no fast-slow indication at the flight director. Because of a remarkable loss of elevation signal quality at azimuth angles greater than 15^o no curved elevation profiles should be flown operationally at this azimuth angles without signal smoothing by an inertial navigation system.

The flight data available on board were recorded with a magnetic tape recorder and about 50 approaches were surveilled additionally ba a radar ground station and plotted on a map (fig.9). For the evaluation of the flight data with respect to flight path accuracy a lot of time has to be spent because of the great data amount.
A final result cannot be given at this time but all 138 approaches were carried out under simulated IFR conditions down to a decision height of 50 ft and not a single one had to be interrupted due to a failure of the GCU.

6. Conclusions

The Guidance and Control Unit was developed as a supplement for microwave landing systems to increase their operational capability by means of:

- signal filtering and signal monitoring
- adaption of the signal sensitivity
- coordinate transformation
- generation of curved flight paths
- wind estimation

The GCU was realized as a modular digital signal processing system with high reliability by application of failure self-detection means.
For preparing the flight tests in a military Transall C 160 and a UH 1D helicopter, the Guidance and Control Unit first was installed in a company-owned Do 28 Skyservant. 138 landing approaches, which were curved in azimuth and elevation, were performed at the microwave landing system SETAC to prove the pilot's capability of flying steep, short-captured S-turn flight paths manually with the flight director. This task was solved by six pilots at the first attempt due to the precise flight

director steering signals computed by the GCU.

A final assessment will be made after the complete evaluation of all flight data, including the Transall and UH 1D tests. The first review shows, however, that a high flight path guidance accuracy has been obtained, which is the basic requirement of steep, short-captured approaches unter worse weather conditions.

7. References

1	RTCA Document Do 166	:	Microwave Landing System Implementation
2	G.Schänzer, H.Boehret	:	Integrated Flight Control System for Steep Approach and Short Landing AGARD - CP 137, 1973
3	G.Schänzer	:	The Effect of Gust and Wind Shear for Automatic STOL Approach and Landing AGARD - CP 140, 1973
4	G.Schänzer	:	The Influence of Microwave Landing Systems on Guidance and Control AGARD - CP 1976
5	H.Drtil, W.Meyer	:	Failure Self-Detection in Digital Flight Guidance Systems AGARD - AG 224, 1977

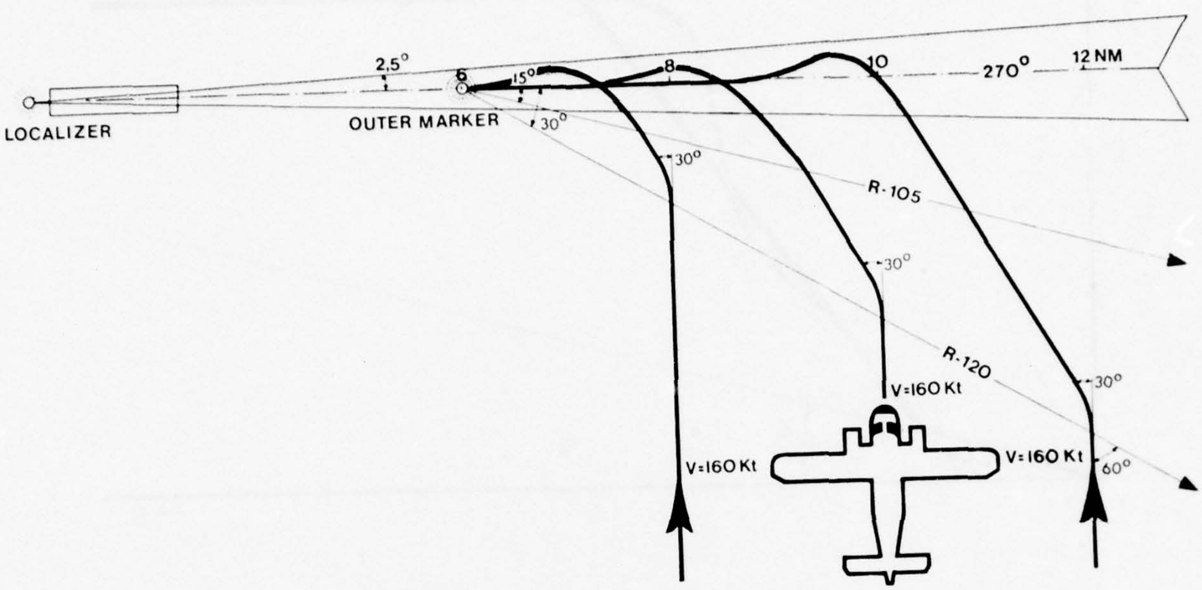

Fig. 1: Conventional ILS Localizer Intercept

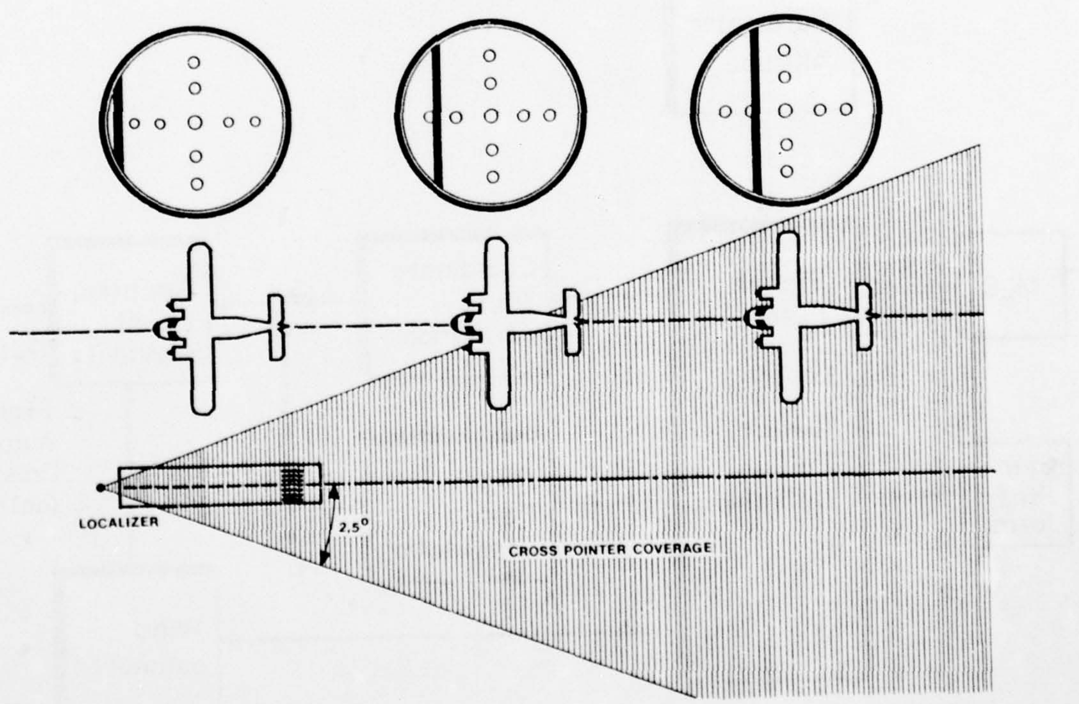

Fig. 2: Cone Effect

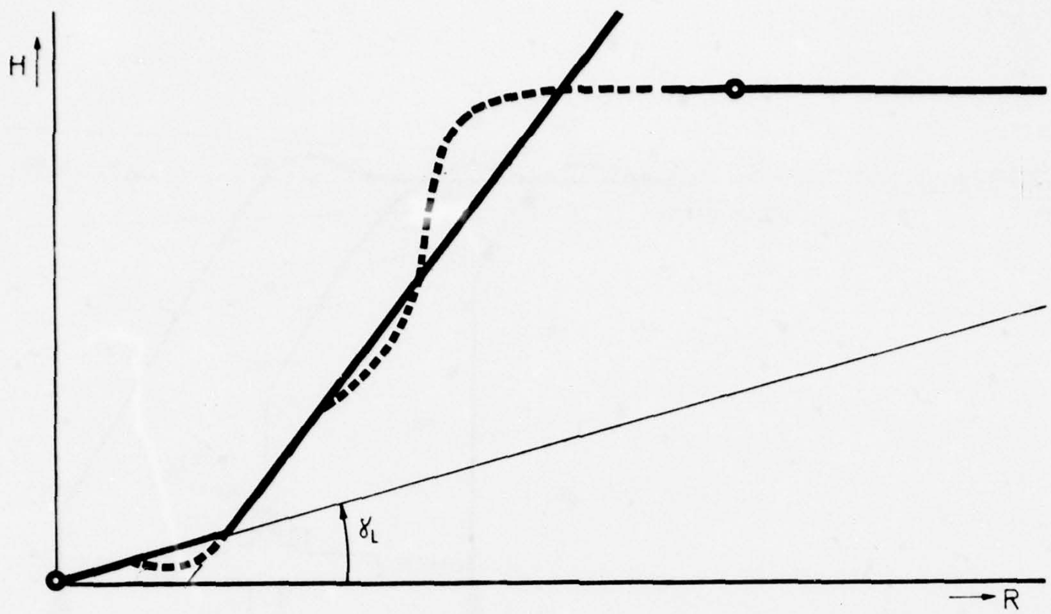

Fig. 3: Overshooting at Steeper Glide Slope Angles

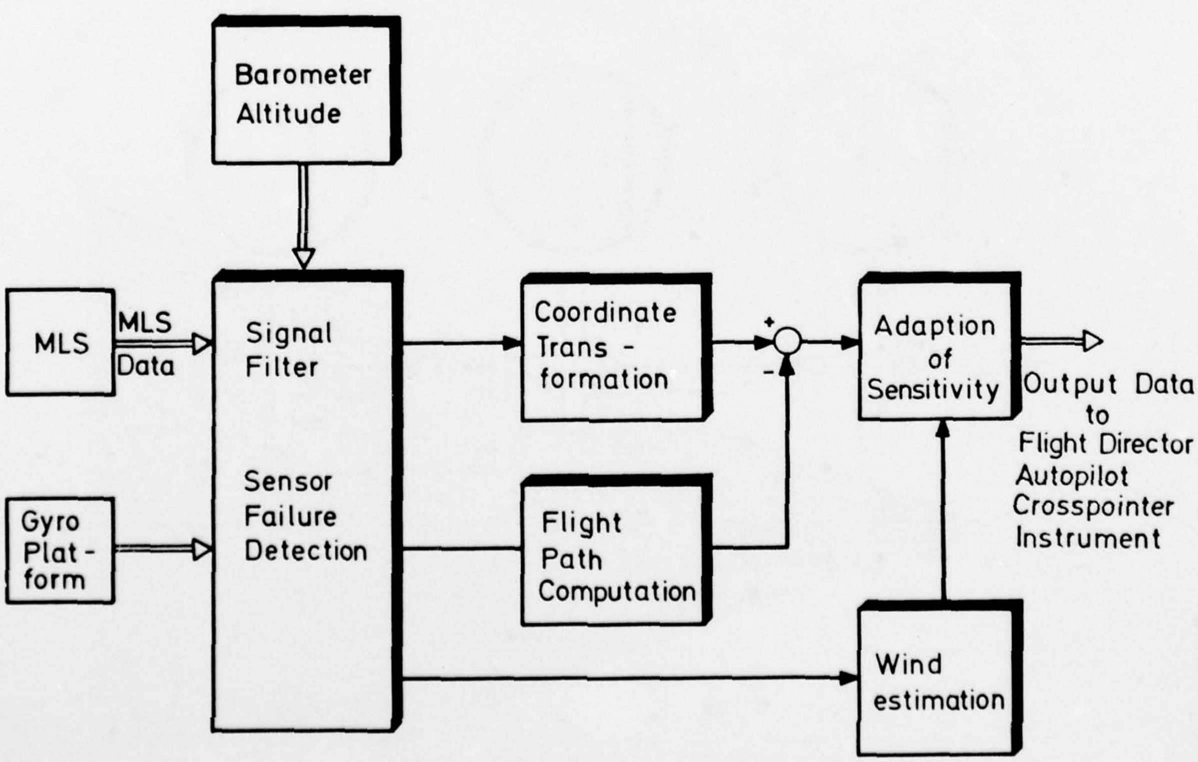

Fig. 4: GCU Block Diagram

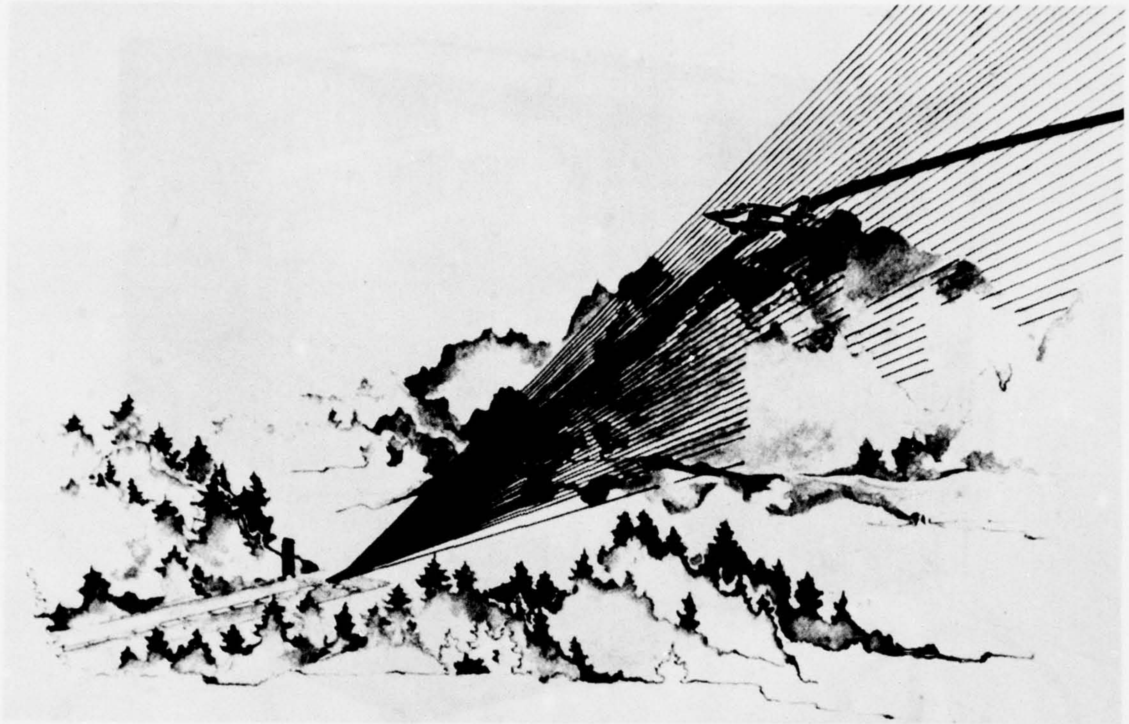

Fig. 5: Curved Elevation Profile

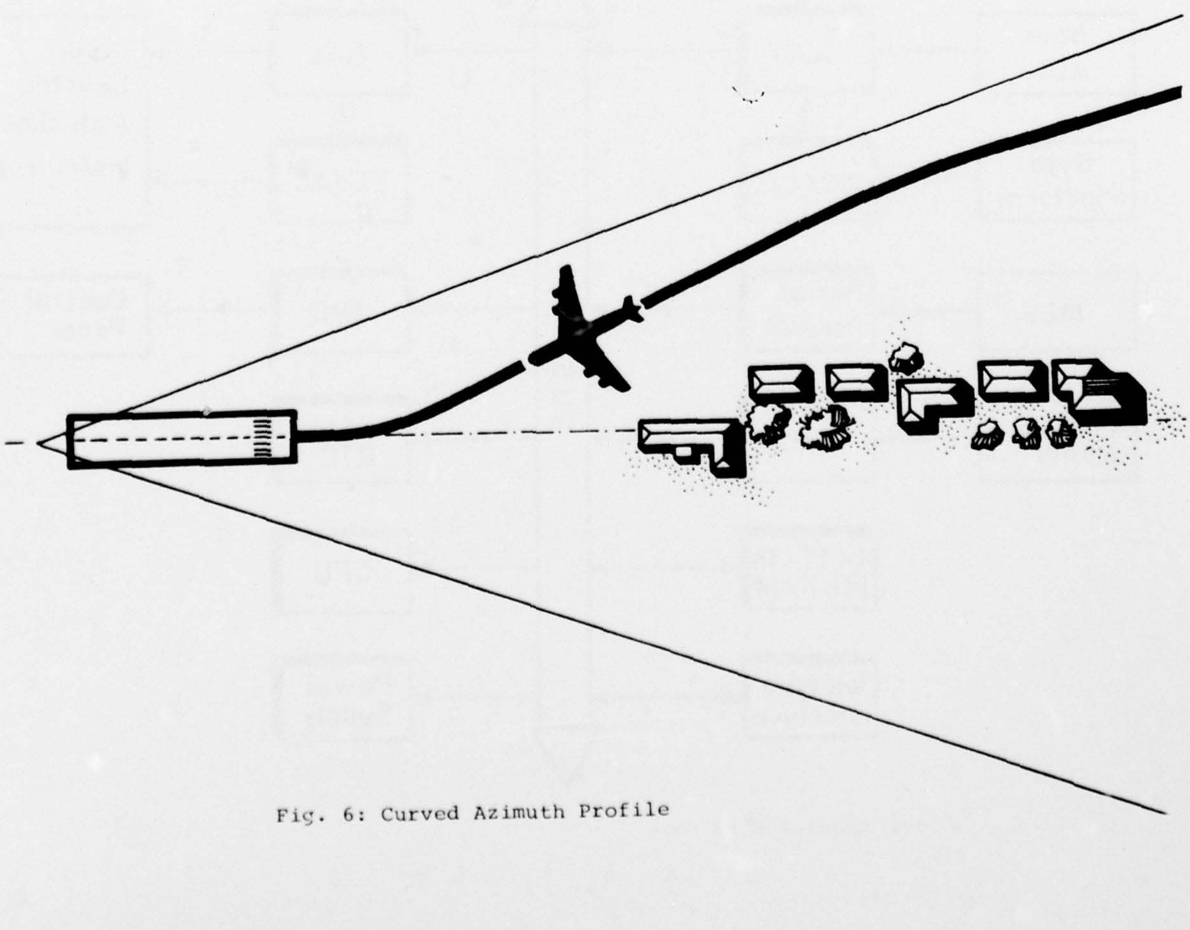

Fig. 6: Curved Azimuth Profile

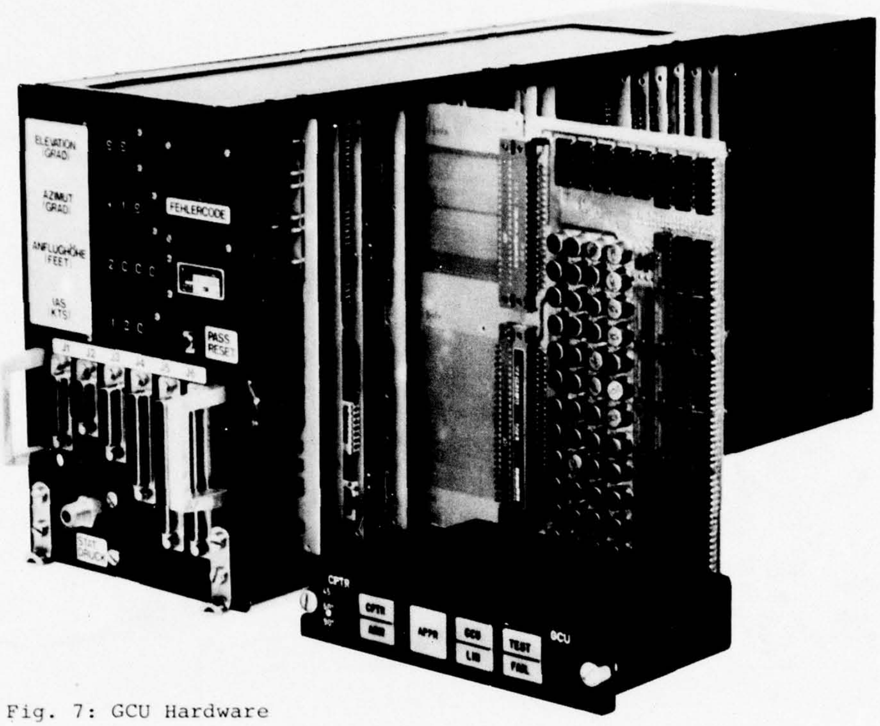

Fig. 7: GCU Hardware

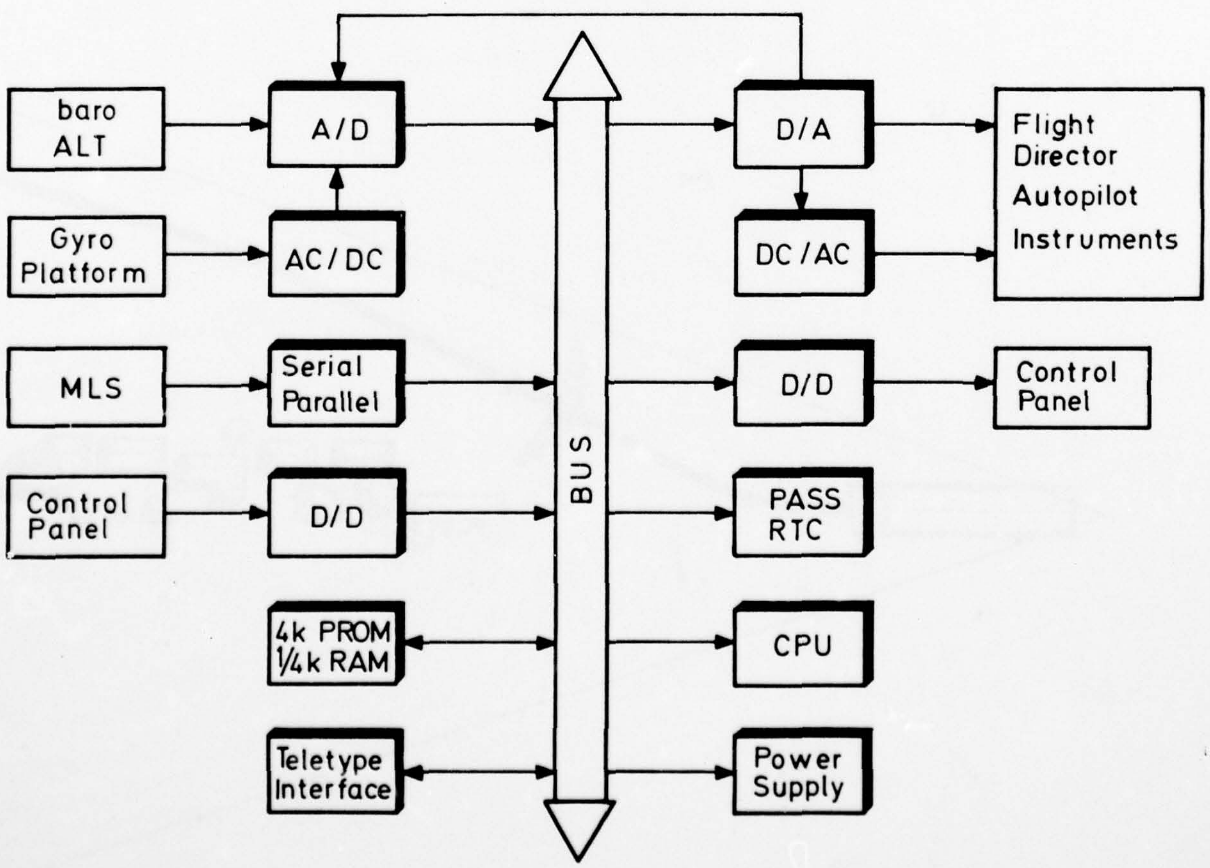

Fig. 8: GCU Modular Structure

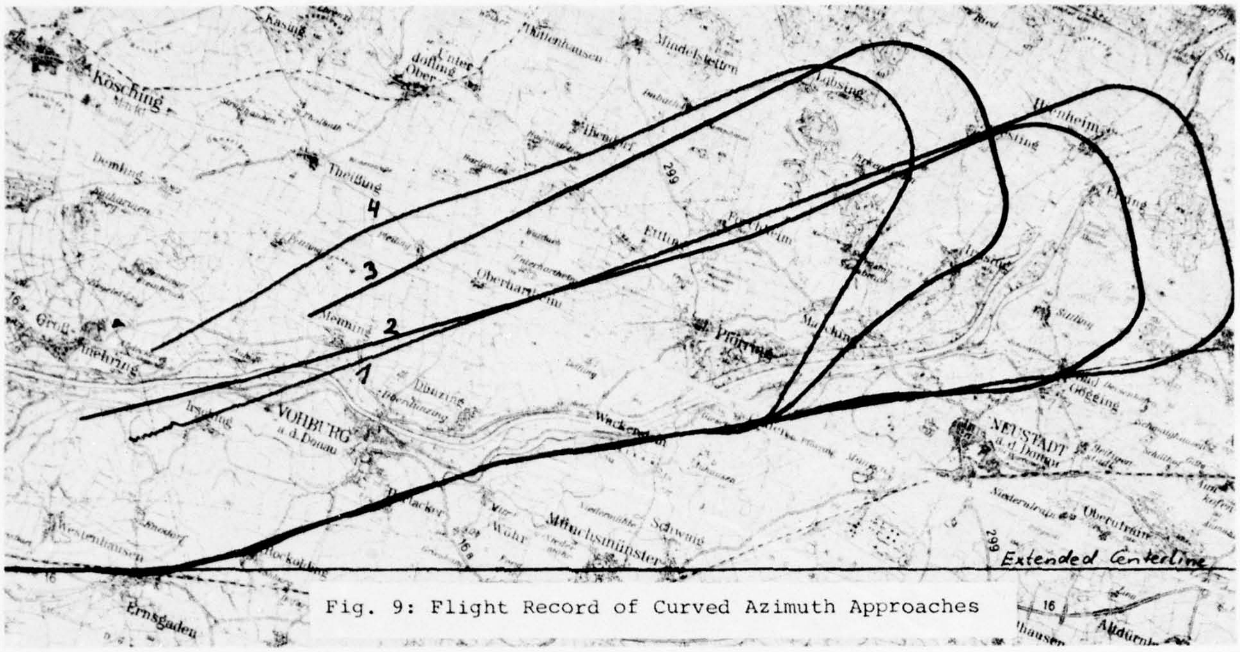

Fig. 9: Flight Record of Curved Azimuth Approaches

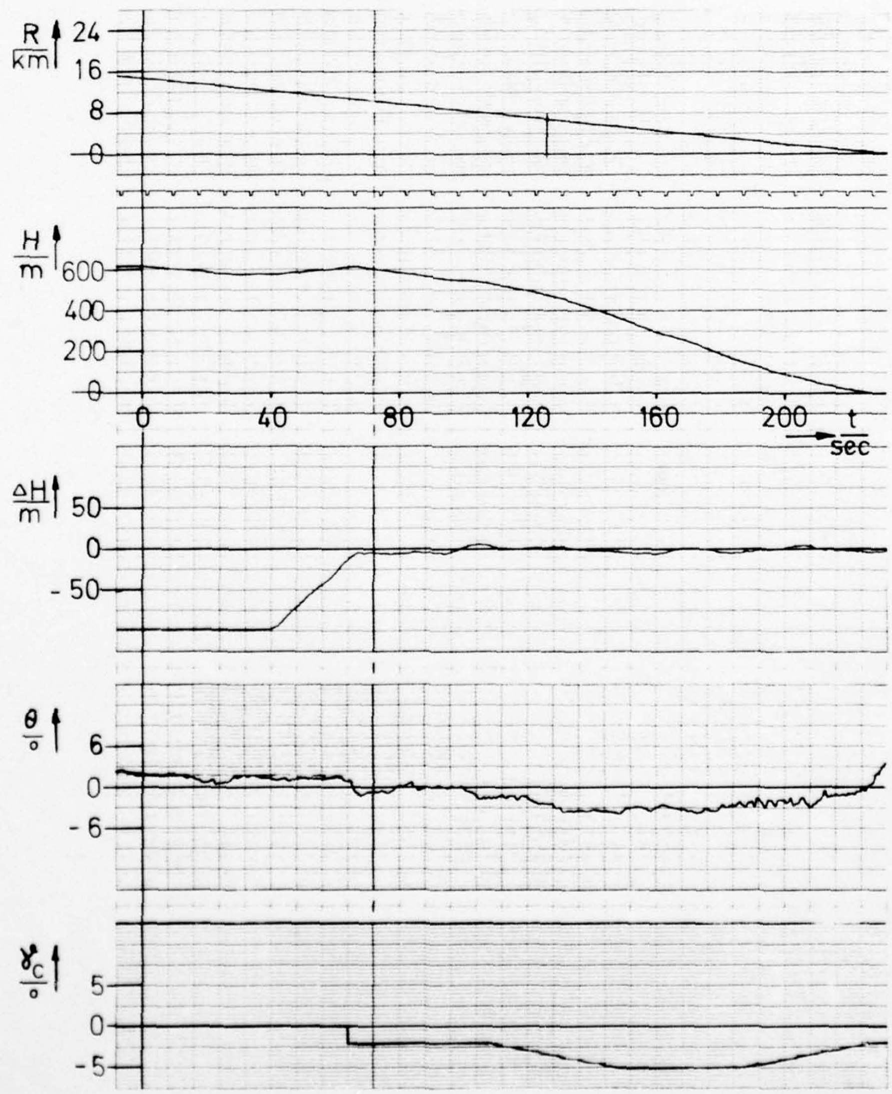

Fig. 10: Flight record of a Curved Elevation Approach

SIMULATION AND STUDY OF V/STOL LANDING AIDS FOR USMC AV-8 AIRCRAFT

W. E. BODE*, R. A. KENDRICK** AND E. J. KANE***
MCDONNELL DOUGLAS CORPORATION (P.O. BOX 516)
ST. LOUIS, MO. 63166

SUMMARY

In seven years of deployment with the United States Marine Corps (USMC) the AV-8A Harrier has decisively demonstrated the effectiveness of vectored thrust V/STOL. Flexibility in basing, rapidity of response and minimal turn around time give it a significant close air support advantage over conventional aircraft. The present AV-8A is cleared for operations day and night from forward sites and ships in weather minima down to 81 m (300 ft) ceilings and 1.4 km (3/4 nm) visibility. The Marine Corps operational doctrine, however, requires that close air support be available around the clock and in poor weather. The goal, then, is for operation in minima as low as 30 m (100 ft) ceiling and 0.45 km (1/4 nm) visibility.

A study was performed to find ways to enable the AV-8A to operate from ships and remote sites in this environment. At the outset a team consisting of Naval Air Systems Command, the 2nd Marine Air Wing, USMC Headquarters, and McDonnell Aircraft Company (MCAIR) established the ground rules and operational requirements.

FIGURE 1
STUDY GROUND RULES

Operational
- A Weather Minimum of 30 m (100 ft) Ceiling and 0.46 km (¼ nm) Visibility is the Established Goal
- Emphasis on Forward Site Night VFR Operations is Required
- *Takeoff Aids Require No Additional Study*

Mission Related
- *The Mission Scenario Shall be a Ship Takeoff to a Remote* Site Landing and Alert Status Followed by a Remote Site Takeoff and Air-to-Ground Mission Terminating in a Ship Landing
- The Forward Site is Defined as a *113 m (372 ft)* by *113 m (372 ft)* Clear and Flat Area Surrounded by 15 m (50 ft) Obstacles with a 22 m (72 ft) Center Landing Pad
- Two Aviation Capable Ships Including the LPH Shall be Addressed
- Ship Landings Include Both Stern and Bow Approaches

Pilot Related
- Pilot Workload Shall Not Exceed the Present AV-8A Requirement
- Transition to Jetborne Flight Shall be Accomplished at Reasonable Altitudes Prior to 30 m (100 ft) Breakout to a Safe Transition
- *Longitudinal Deceleration Shall Not Exceed 0.15g Maximum on* the Glide-Slope Based on Human Factor and Energy Management Considerations

Equipment
- A Common Ship/Site Sensor Suite is Required for Minimum Aircraft and Training Impact
- Added Sensors Shall be Designed for Maximum MTBF and Minimum Maintenance Hours Per Flight Hour
- *Where Practicable, Radiating Aids Having Low Electromagnetic* Radiation Profiles Shall be Selected to be in Accord With Emission Control Philosophy

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The study included motion base simulations, flown by USMC Harrier pilots, in which 600 low speed IFR approaches to a forward site and ship were made. Glideslope angles from 3° to 9° were simulated and flown both head up and head down. An attitude hold autopilot was evaluated and flight director steering was studied. The effects of crosswind, turbulence, sea state and system errors were included in the simulation. Visual landing aids for ships and forward sites were devised and aircraft and ground/ship based equipment selected. At the heart of the study was the evaluation by the pilots of the degree of improvement offered by the simulated landing aids.

The effect on the aircraft and on the forward site and ship of lowering the operational minima was established, as seen in Figure 2.

TACAN and lights at the forward site and improved lights and a glideslope indicator for the ship will enhance operations in the current AV-8A weather minima. Operation at 61 m (200 ft) and 0.9 km (1/2 nm), however, requires the addition of an all weather landing system (AWLS) and flight director to the aircraft and a radiating AWLS ground subsystem at the forward site and ship. An attitude hold autopilot is also required if the wingborne to jetborne transition must be made without external visual references.

Operation at 30 m (100 ft) and 0.45 km (1/4 nm) requires the initiation of thrust vectoring for transition while IFR. Airspeed is maintained to provide aircraft stability and effective flight controls. The attitude hold autopilot is required during this phase to reduce pilot work and the chance of pilot disorientation. After breakout the transition to jet-borne flight is completed when external visual references are available.

A 3° glideslope profile is recommended for ship landing and is the desired glideslope for forward site operation. But since mountainous terrain or other obstacles may require steeper glideslopes, the selected equipment should have variable glideslope capability. Zero/zero weather minima were not a study requirement but were included to bound the effort.

*BRANCH CHIEF-ELECTRONICS
**UNIT CHIEF-ELECTRONICS
***SENIOR HUMAN FACTOR ENGINEER

FIGURE 2
LANDING AIDS SUMMARY

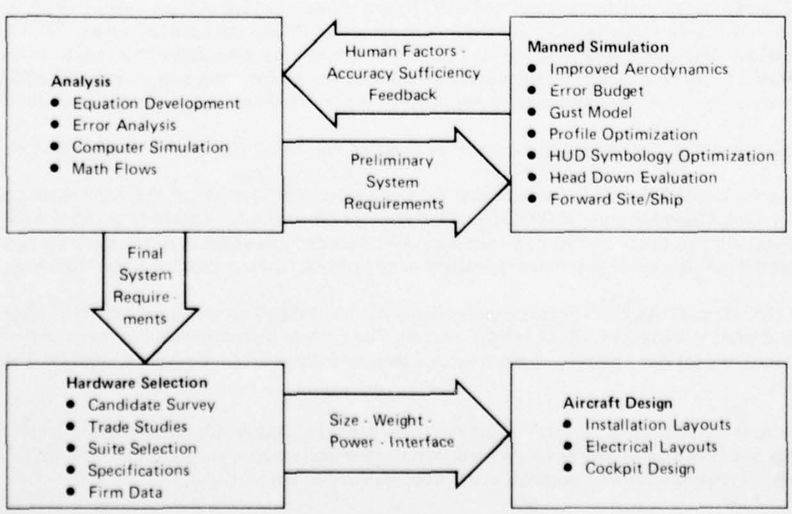
	122 m and 1.8 km (400 ft and 1 nm)	91 m and 1.4 km (300 ft and 3/4 nm)	*61 m and 0.9 km (200 ft and 1/2 nm)	30 m and 0.46 km (100 ft and 1/4 nm)	Zero/Zero
Installed Weight Increase			21 kg (47 lb)	56 kg (124 lb)	>115 kg (250 lb)
Aircraft Operation	**Transition VFR, No Additional Landing Aids**	**Transition VFR, CCA Talkdown For Ship, TACAN Penetration For Shore**	**Transition VFR, Terminal Guidance Using AWLS. HUD Command Steering from FDC**	**Transition IFR, Terminal Guidance Using AWLS; HUD Command Steering From FDC; Autopilot**	**Automatic Landing**
Aircraft Equipment	Present Harrier with Radar Altimeter and TACAN		Add Flight Director and AWLS	Add Attitude Hold Autopilot	Add Coupled Autopilot (Redundant) Touchdown Aids Replace Actuators
Forward Site	Night Operation; TACAN, Lights		Lights and AWLS		Lights, AWLS and Touchdown Aid Transmitter
Ship	Lights and Improved GSI		Improved GSI, Lights and Stabilized AWLS		Lights, Stabilized AWLS and Touchdown Aid Transmitter

*Potential Autopilot Requirement

STUDY APPROACH

Data were collected by interviewing government and industry sources and submitting questionnaires to AV-8A pilots. Meanwhile, ship and forward site configurations were examined in terms of approach profiles and visual landing aids, and electronic aids were evaluated for the ship, forward site and aircraft. The autopilot figured prominently in the studies since it was the most complex element of the effort.

FIGURE 3
STUDY APPROACH



Landing profiles were developed and simulations performed to identify candidate airborne and surface mounted landing aids. Electronic and visual landing aids for the ship and forward site were selected and simulated to assess their contribution to AV-8A IFR operations. An autopilot to supplement the AV-8A Stability Augmentation System (SAS) was included in the simulation of IFR operations.

DATA COLLECTION

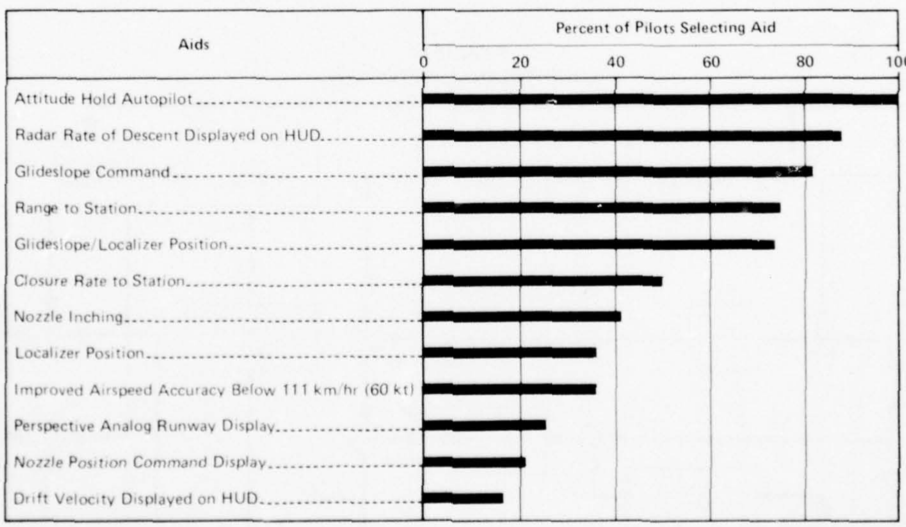
A literature survey provided a data base and pinpointed the government agencies and companies engaged in landing aids work. The subsequent government and industry survey consisted of 32 meetings and numerous informative conversations. Discussions with the Naval Air Development Center, Naval Electronic Center, Naval Weapons Center and Air Force Flight Dynamics Laboratory yielded data on advanced concepts which were used to postulate candidate landing aids and procedures for evaluation by AV-8A squadron pilots. The VTOL visual landing aids evaluation program at the Naval Air Test Facility provided useful inputs to the site and ship lighting analysis.

Our discussions with Harrier pilots revealed the importance of flight trend information. It was also apparent that the mere display of additional situation information would not be sufficient for IFR transition in the 30 m (100 ft) and 0.46 km (1/4 mile) environment. A pilot survey early in the study sought the causes of the heavy workload encountered in low visibility landings and solicited recommendations for reducing it. Information was collected from 23 Harrier pilots, including RAF exchange pilots, whose combined experience in the AV-8A totalled 7,582 hours and included:

- 242 IFR ship launches and recoveries.
- 921 VFR ship launches and recoveries.
- 2,703 IFR shore launches and recoveries.
- 21,540 VFR shore launches and recoveries.

The pilots selected certain aids which they felt would be required for operations at 30 m and 0.46 km. All of them selected an attitude hold autopilot to reduce workload and permit transition to jetborne flight in the clouds. The selection by the pilots of radar

FIGURE 4
PILOT SELECTION OF AIDS REQUIRED FOR RECOVERY
- 30 m Ceiling (100 ft)
- 0.46 km Visibility (1/4 nm)



descent rate for display on the HUD was attributed to the inaccuracy of the inertially derived descent rate display on early AV-8A's. Later AV-8A aircraft, however, have a separate, dedicated pneumatic vertical speed transducer which provides acceptable data to the HUD.

Command steering was selected by over 80 percent of the pilots to reduce the workload and permit precision approaches. Situation data (glideslope/localizer position and range to station) was selected by about 70 percent of the pilots for use as a crosscheck and to assist in locating the site or ship after breakout. Of the remaining aids, only display of closure rate and nozzle inching (vernier control of nozzle position implemented on the throttle or stick) were subsequently assessed in simulation. They were eliminated as candidate aids.

The survey of AV-8A pilots included workload ratings for current operations. In addition, estimated ratings were given for proposed aids and procedures. Acceptable workload was judged to have been attained when an attitude hold autopilot and an improved HUD with command steering were added to the envisaged airplane.

THE PROPOSED AUTOPILOT

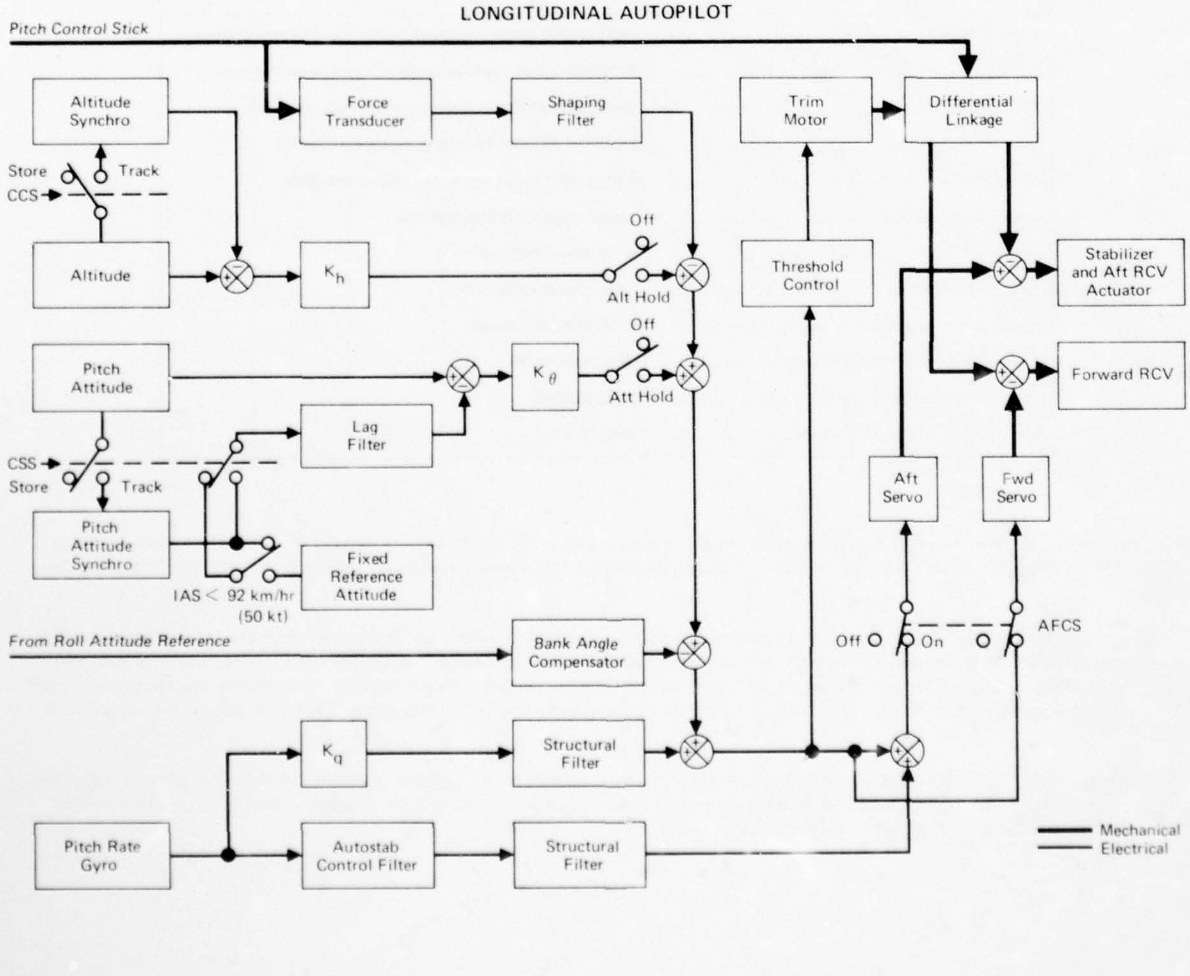
An autopilot was devised based on the pilot survey. It was mechanized in the flight simulator and its ability to reduce workload and thus permit operations in the study weather minima was evaluated. The limited control authorities of the AV-8A (8 to 35%) were retained in the simplex mechanization. Higher authority, although more desirable, requires redundancy which would add significant weight, increase development cost, and complicate AV-8C retrofit.

FIGURE 5
AUTOPILOT OPERATIONAL MODES

Axis	Mode	IAS Range - km/hr (knot)		
		−56 To 93 (−30 To 50)	93 To 296 (50 To 160)	296 To RCV Off (160 To RCV Off)
Pitch	CSS and Rate Command	X	X	X
	Fixed Attitude Hold	X		
	Attitude Hold		X	X
	Bank Angle Compensate		X	X
	Automatic Trim	X	X	X
Roll	CSS and Rate Command	X	X	X
	Zero Attitude Hold	X		
	Attitude Hold		X	X
Yaw	Cancelled Rate Damping	X	X	X
	Turn Coordination		X	X

The longitudinal channel of the autopilot uses control signals, derived from pitch and pitch rate sensors, to drive the forward and aft pitch RCV actuators and the stabilizer for pitch attitude hold. This channel includes bank angle compensation and automatic pitch trim. Automatic pitch trim is required to ensure that the full autopilot authority is available to the control surfaces and RCV's, and to prevent disengage transients.

FIGURE 6
LONGITUDINAL AUTOPILOT



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THE GUIDANCE AND CONTROL OF HELICOPTERS AND V/STOL AIRCRAFT AT --ETC(U)
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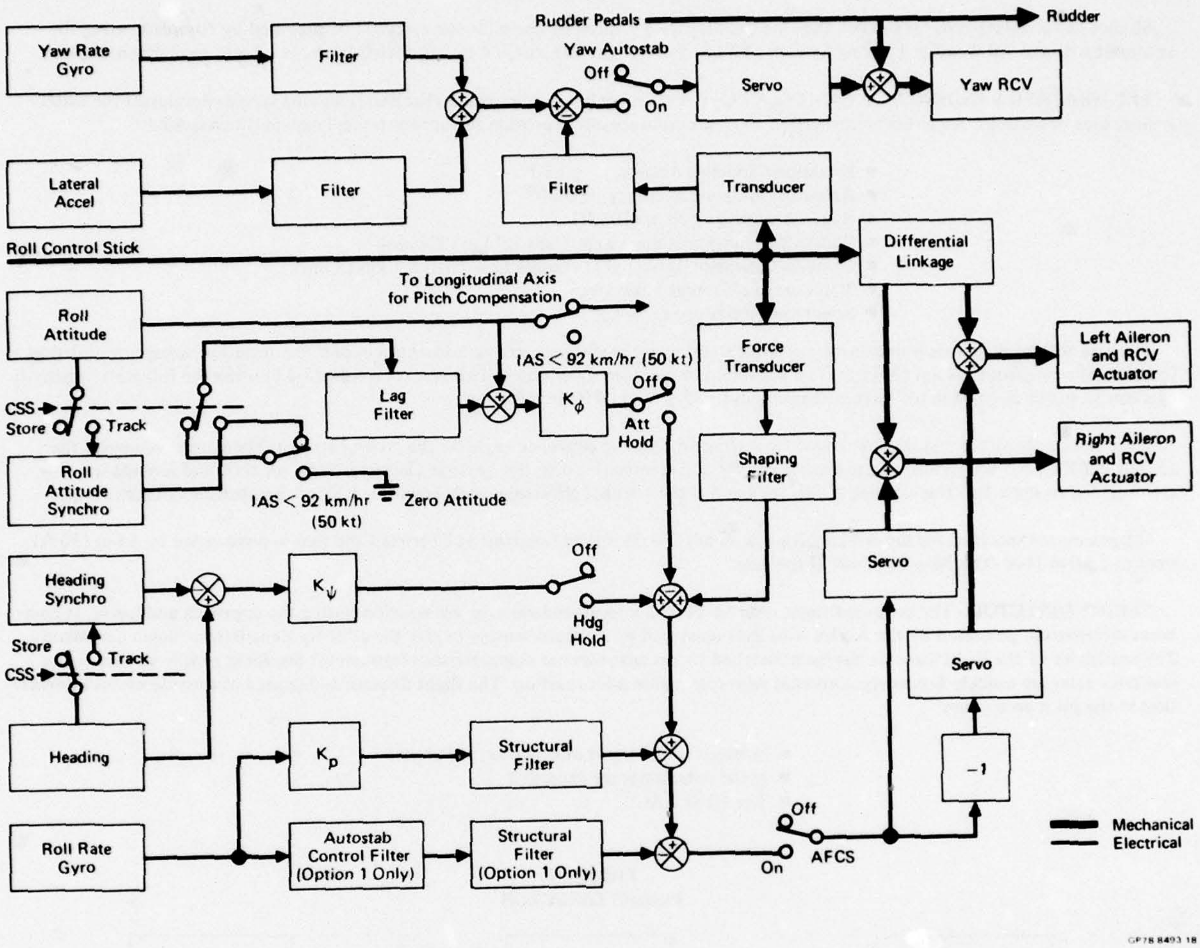
The lateral directional channel of Figure 7 drives the yar RCV (jetborne only) and the roll RCV's. Roll rate and yaw rate are used to provide damping. Sensed lateral acceleration and a roll to yaw interconnect provide turn coordination. Roll attitude is combined with roll rate to provide roll-attitude-hold signals to the ailerons and roll RCV's.

FIGURE 7
LATERAL AUTOPILOT



ATTITUDE HOLD - The attitude hold mode is pilot-selected for control of aircraft pitch and roll attitude. It can be engaged at any airspeed below 460 km/hr (250 kt) to lighten pilot workload, particularly in crosswinds and turbulence. At speeds below 92 km/hr (50 kt) it will return the aircraft from any attitude to a preset reference attitude. In pitch, this fixed reference can be set between 0 and 10 degrees. In roll it is zero degrees. (At these very low speeds zero roll must be quickly restored when the control stick is centered.)

Bank angle compensation is a crossfeed control loop between the pitch and roll axes which commands pitch changes to maintain altitude during long, steady, banked turns. This loop is part of the roll attitude hold mechanization.

CONTROL STICK STEERING (CCS) - The CSS mode enables the pilot to switch out the attitude hold loops by applying control stick force above defined breakout levels. Thus, the pilot can establish a new attitude without disengaging the attitude hold mode.

YAW CHANNEL - The yaw channel, which provides rate damping and turn coordination, is not a separate mode. The rate damping consists of a yaw rate feedback processed through a high pass filter which cancels the steady state signal. This feedback is combined with a lateral acceleration feedback and roll-to-yaw interconnect to improve turn coordination at wing-borne to jet-borne transition speeds. Operation of the yaw autopilot is restricted to the powered lift flight regime.

The present AV-8A has a stability augmentation system (SAS) which provides turn coordination in the yaw axis and rate damping in all three axes. The SAS operates through the ailerons and horizontal tail and the associated reaction control valves (RCV). The yaw SAS is mechanized through the RCV systems only (the rudder is not powered). Thus, the yaw SAS is effective only during jetborne flight. The forward pitch RCV is manually controlled and does not operate in the SAS mode. Yaw rate and lateral acceleration are sensed and processed through the appropriate control filters (yaw rate is washed out) to generate damping and turn coordination signals. A roll-to-yaw-RCV-interconnect signal is used to generate roll-proportional yaw commands for turn coordination. The current SAS is functional only when the aircraft is below 460 km/hr (250 kt) and either the landing gear is down or the flaps are below 45°.

GROWTH POTENTIAL - Altitude and heading modes can be mechanized to provide pilot relief during the cruise portion of a flight. Heading hold would consist of a control loop closed through a heading feedback which would be compared with a pilot selected heading to determine the heading error. Roll commands would then be generated in proportion to the heading error. Altitude hold would operate through the horizontal tail surface to correct the altitude error by pitching the aircraft.

PROPOSED LANDING AIDS

Studies of candidate systems showed that the capabilities indicated by the pilot survey could be provided by complementing the autopilot with the All Weather Landing System (AWLS) and a flight director and by improving the head up and head down displays.

ALL WEATHER LANDING SYSTEM - The AWLS is a scanning beam, microwave (Ku-Band) landing system developed for USMC remote area operations. An airborne subsystem and a ground subsystem provide the accuracies and capabilities required:

- Elevation Guidance Accuracy: $\pm 0.10^{\circ}$
- Azimuth Guidance Accuracy: $\pm 0.20^{\circ}$
- Range Accuracy: ± 30 m (100 ft)
- Course Softening at Ranges Less Than 1.4 km (3/4 nm)
- Obstacle Clearance Activated at Ranges Less Than 5.5 km (3 nm)
- Man Portable Ground Subsystem
- Selectable Glideslope (3° - 12°)

A course softening feature prevents oversensitive steering as the distance to touchdown decreases; the deviation signals are scaled as a function of range from 1.4 km (3/4 nm) to a minimum range from touchdown. This is accomplished by limiting the full scale sensitivity to a course width of ± 144 m (473 ft) in azimuth and ± 33.5 m (110 ft) in elevation.

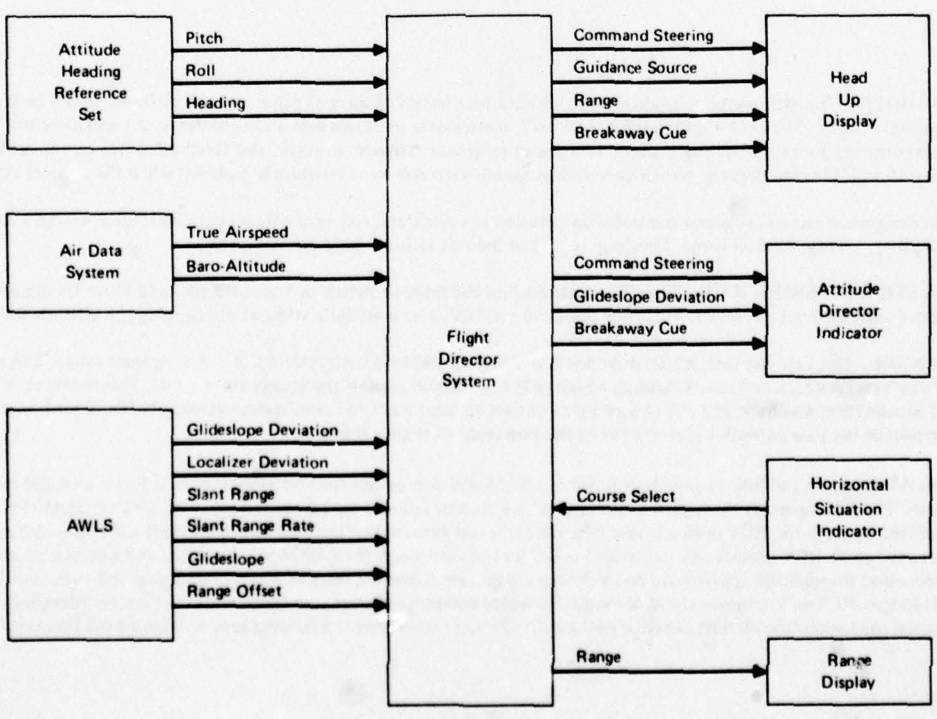
The obstacle clearance feature is activated by setting an obstacle clearance angle on the ground station. Thereafter, whenever the aircraft is within 5.5 km (3 nm) of the ground station and is intruding into the obstacle clearance zone, an aural and a visual warning are provided to the pilot. The warning is also activated if the selected glideslope angle is within 1.5° of the obstacle clearance angle.

The accuracies specified for the AWLS permit a 46 m (150 ft) ceiling breakout at a forward site that is surrounded by 15 m (50 ft) trees or a 30 m (100 ft) ceiling breakout at the ship.

FLIGHT DIRECTOR - The proposed flight director provides command steering information during the approach and hover. It combines information generated by the AWLS with data measured by onboard sensors to give the pilot fly right/left/up/down commands. The sensitivity of the flight director has been matched to aircraft response characteristics throughout the flight profile so that errors and error rates are quickly but safely corrected with reasonable pilot reaction. The flight director is designed to provide useful information to the pilot even when:

- Portions of the input information are absent
- Input data limits are exceeded
- The HUD fails

FIGURE 8
FLIGHT DIRECTOR



The flight director has self monitoring capability and indicates to the pilot if it is no longer providing useful information. It has separate sections for lateral and vertical steering and is programmed to permit glideslope selection without disturbing the basic deviation correction.

Inputs to flight director are provided by the Attitude and Heading Reference Set (roll, pitch and azimuth) and the air data system (altitude rate and true airspeed). The flight director provides outputs to the HUD and head down displays.

HEAD-UP DISPLAY (HUD) - The HUD symbology evolving from this investigation was derived by modifying the AV-8A V/STOL mode symbology to give an improved display of attitude, to display the flight director functions, and to display range to the guidance source.

Pitch attitude is scaled 3:1 to the outside world and displayed at 10^{o} intervals. This scaling and interval were selected by the pilots during simulation; being offered scalings of 1:1, 3:1 or the present AV-8A 5:1 scaling, they unanimously selected 3:1. The 1:1 scaling with display intervals of 5^{o} provided good resolution of attitude but was too dynamic to interpret except under constant or low pitch rate conditions. The 5:1 scaling on the other hand (display intervals of 30^{o}) did not provide a good cue of attitude. The 3:1 scaling was a good compromise between the dynamics of the 1:1 display and the lack of attitude resolution of the 5:1 display.

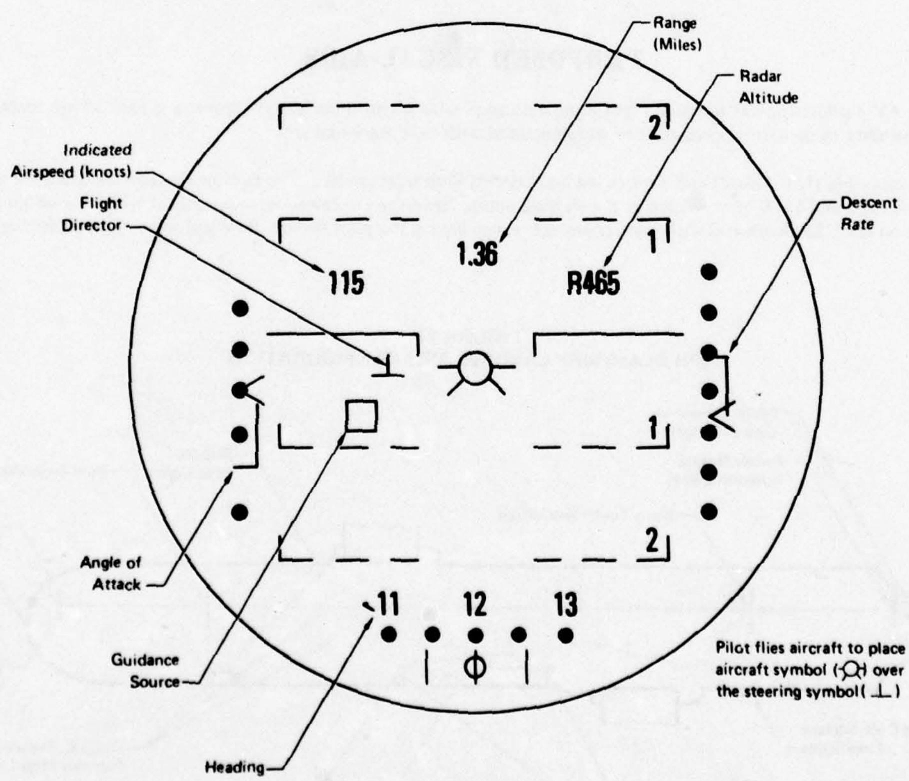
The flight director symbol is the inverted T and is constrained to a 5^{o} radius circle about the airplane symbol. The pilot's task is to keep the airplane symbol superimposed on the flight director symbol.

The range display is provided by signals from the AWLS. This display was assessed by the pilots in the simulation as being essential for providing rate trends and as an event cue to signal glideslope entry, nozzle change points, etc., while on the IFR profile.

The guidance source marker (or situation) symbol is a 0.5^{o} square positioned by signals from the flight director. This symbol is constrained to the HUD total field of view. If the guidance source is out of the HUD field of view, the symbol is parked at the corresponding HUD boundary.

A breakaway cross displayed over the miniature airplane symbol warns the pilot that he is either below 21 m (70 ft) or will descent through 21 m (70 ft) in 2 seconds.

FIGURE 9
LANDING AIDS V/STOL SYMBOLOGY



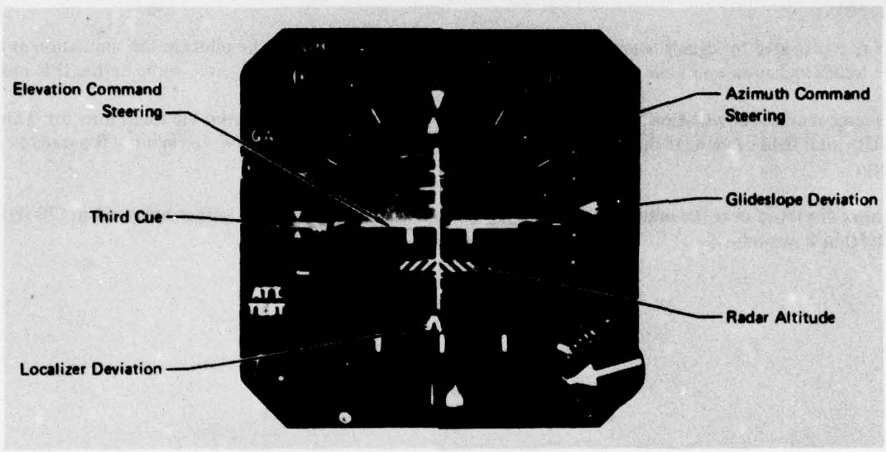
HEAD DOWN DISPLAY - A three-cue Attitude/Director Indicator (ADI), originally developed for helicopters, was used to evaluate the usefulness of a third cue during head down IFR approaches. Azimuth and elevation steering commands are the two principal cues.

Initially, the third needle was used to display speed error from the 213 km/h (115 kt) nominal. The pilots, however, did not find this useful, commenting that they preferred to derive speed error or trend from the airspeed indicator. They suggested that descent rate might be more useful. Display of descent rate did prove to be useful. However, its addition to the ADI does not merit the development involved because it can be derived from the existing VSI (vertical speed indicator).

An analog display of altitude was also evaluated on the ADI. This display was found by the pilots to be useful in providing altitude trend information. However, once again, it does not merit the development of a new indicator.

The three-cue ADI is a two-axis unit that does not provide heading. It was therefore difficult for the pilots to maintain lateral steering head down, even with the display of command steering. For this reason it was recommended that the three-axis ADI presently in the AV-8A be retained.

FIGURE 10
THREE-CUE ADI

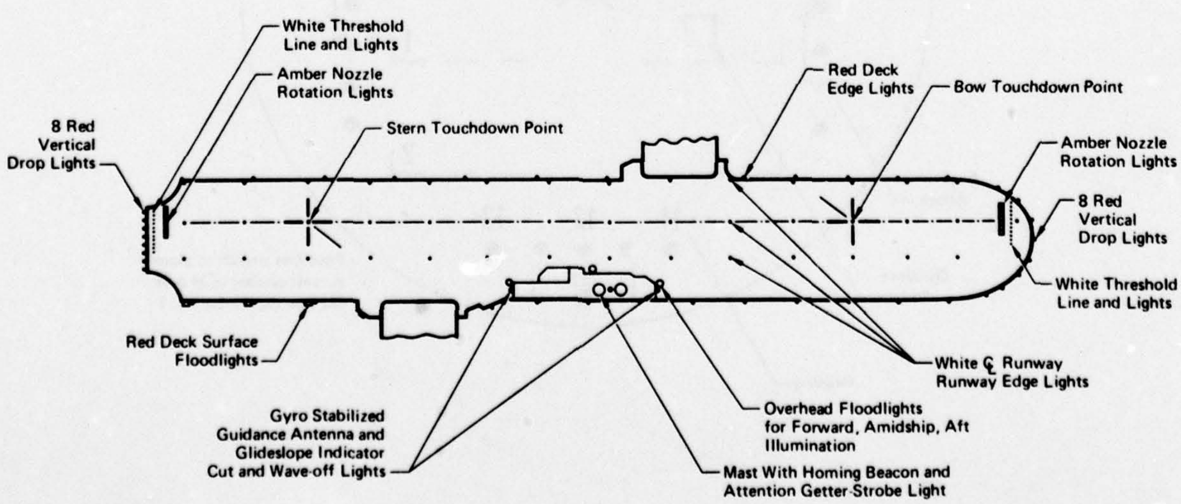


PROPOSED VISUAL AIDS

After breakout, AV-8 pilots convert to visual flight using natural cues to perform the air taxi, hover and touchdown maneuvers. At night and in low visibility these natural cues must be supplemented with external visual aids.

SHIP - An LPH class ship (USS Guam) and a proposed Sea Control Ship were studied. The lighting systems specified for amphibious assault class ships (LPH-2 and LHA-1) were used as the starting point. The results obtained were combined with data obtained from AV-8A operations on the USS Guam and with pilot comments taken during the pilot survey. Bow and stern approaches were studied in sea state 3.

FIGURE 11
LPH CLASS SHIP LANDING AIDS CONFIGURATION



Our recommendation for the LPH class and sea control ships is to increase the lights recommended in "Helicopter Visual Landing Aids Service Bulletin" (NAEC-NE-724) by adding:

- Drop Lights
- Runway Edge Lights
- Nozzle Rotation Lights
- Deck Edge Floodlights
- Gyro Stabilized Glideslope Indicator (GSI) with pulse coded glideslope deviation information
- Threshold Lights
- Touchdown Point
- Cut and Wave-off Lights
- Attention Getter Strobe

The optimum color and location for the added lights should be verified by simulation and flight tests. We did find that the best GSI was a Fresnel lens system.

A landing signal officer (LSO) is required for all shipboard AV-8A landing operations. His primary function is to provide status information to the pilot. This information is supplied by radio link or with lights situated alongside the GSI. The LSO observes the landing and commands a "wave-off" when necessary. Red "wave-off" lights are situated in vertical banks, along with green "cut" lights, on either side of the GSI.

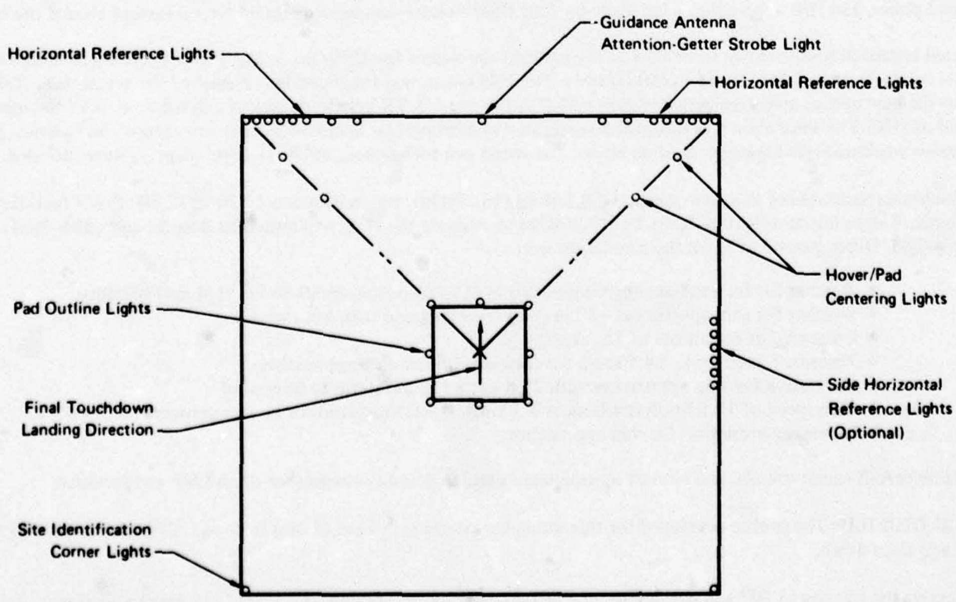
To help the LSO judge aircraft position, a three-color light has been installed on the AV-8A aircraft nose gear. This aid, which is turned on by the pilot during the approach, has been extremely useful to the LSO for aircraft attitude and position information. The lens on this light is similar to that of the GSI; only one of the three colors (green, amber or red) is perceived depending on the viewing aspect. When the aircraft is on the command path and at the proper attitude, the LSO will observe an amber light.

FORWARD SITE - Pilot comments plus testing performed by the Naval Air Test Facility indicated that natural cues for night operations ashore are plentiful and of high quality. Operation from an austere landing site, however, may require augmenting these natural aids.

Many lighting schemes and arrangements were studied which could provide the pilot with sufficient hover and descent cues. One obvious scheme would be simply to configure the forward site with the shipboard light complement developed in this study. The use of the complete ship's lighting scheme at the forward site, however, would require obstacle removal, impose a severe logistics penalty and require extensive site enlargement. The forward site is 113 m (372 ft) square surrounded by 15 m (50 ft) obstacles. It does not permit the use of the complete lighting package used at large facilities. However, complete lighting installations were studied and the results combined with those of applicable portions of previous studies performed by the Naval Air Test Facility. Thus, our forward site visual landing aids study sought the minimum lighting consistent with operational safety and remaining within the confines of the site.

The lighting aids for converting from instrument flying to visual flying must be elevated into the pilot's forward field of view so that he can find the site. During vertical descent to touchdown they must also be in his lateral field of view. An LSO at the forward site provides the pilot with wind direction and speed, atmospheric conditions, obstacle briefing, glideslope deviation alert, emergency assistance, etc. The best color for the visual aids remains to be determined.

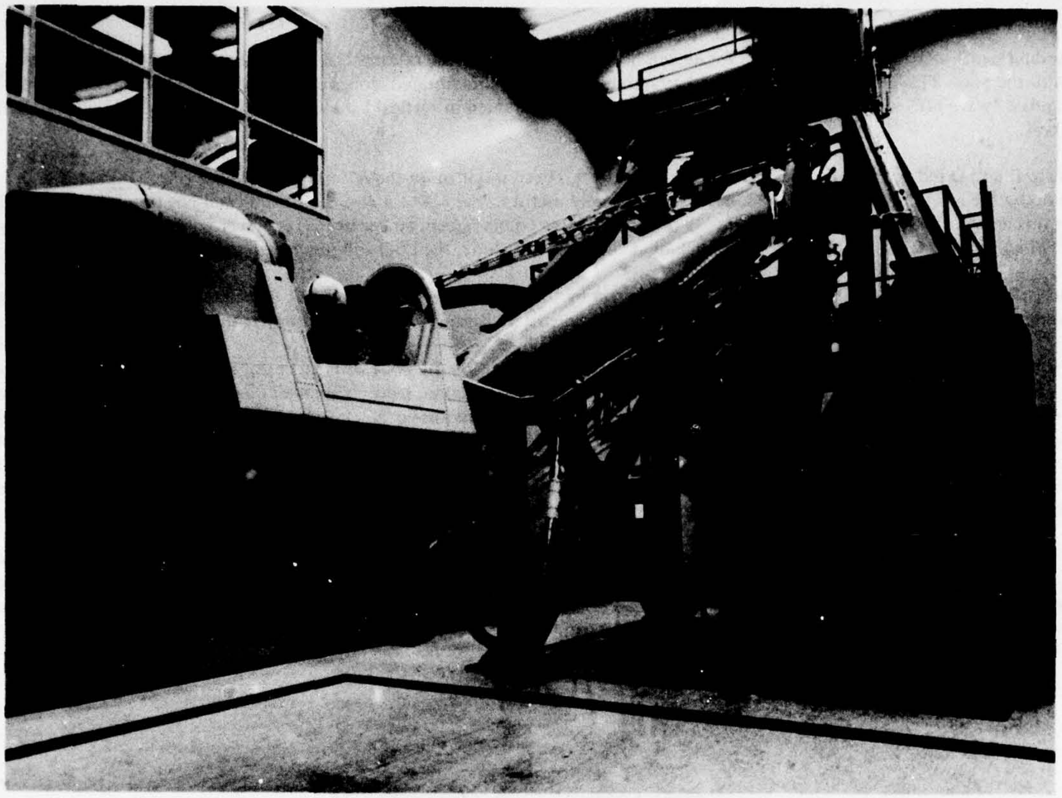
FIGURE 12
VISUAL LANDING AIDS - FORWARD SITE



SIMULATION

The survey had identified areas where improvements were needed and the study had identified ways to implement them. The potential solutions were mechanized in the MCAIR motion base simulator and manned simulations were conducted in two phases over a two year period. The first phase evaluated 3°, 5° and 9° glideslope profiles, several glideslope indicators, ship lighting schemes, and airborne landing aids. USMC pilots flew 280 approaches.

FIGURE 13
MCAIR MOTION BASE SIMULATOR



In the second phase, 336 IFR approaches were made by four USMC pilots who were qualified for operations aboard the USS Guam.

The simulated approaches were flown to models of the forward site and of the USS Guam. The forward site was a square clearing 113 m (372 ft) on a side, surrounded by 15 m (50 ft) trees. The USS Guam was simulated by a model of the actual ship, with drop lights added to the bow and an attention-getter beacon added to the mast. A TV system projected a dynamic view of the models into the pilot's field of view. The simulation was designed to evaluate the piloting task in crosswind and turbulence and to assess the effectiveness of various workload relief systems. System errors, crosswind and turbulence, and head down displays were included.

Most approaches were initialized at a gross weight of 6,895 kg (15,200 lb), which included 1,270 kg (2,800 lb) of fuel. Each pilot also flew approaches at an initial weight of 7,801 kg (17,200 lb) to evaluate the ability to arrest the descent and either land or go-around at the heavier weight. Other ground rules for the simulation were:

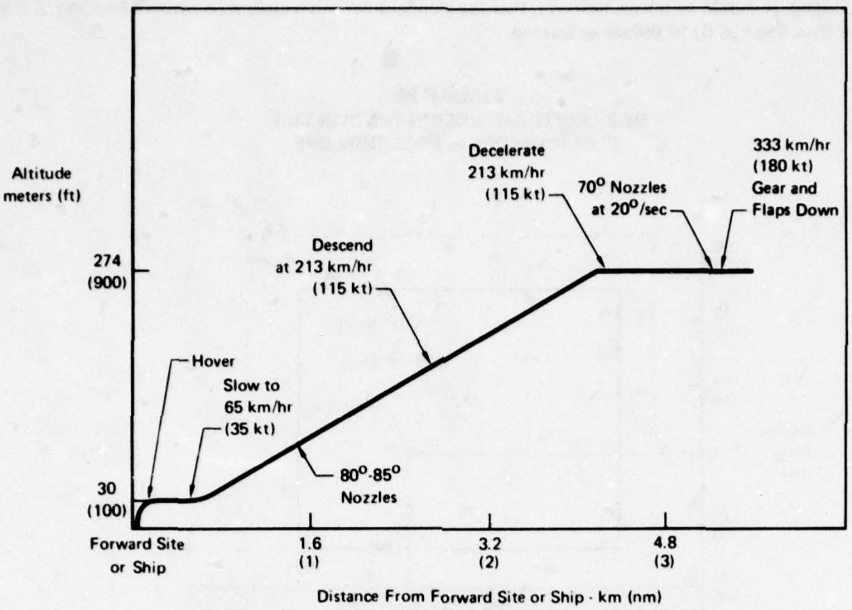
- Weather for forward site approaches - 46 m (150 ft) ceiling and 0.46 km (1/4 nm) visibility.
- Weather for ship approaches - 30 m (100 ft) ceiling and 0.46 km visibility.
- Crosswind at forard site of 18.5 km/h (10 kt)
- Turbulent gusts up to 18.5 km/h for forward site and ship approaches
- Sea state 3 for ship approaches with 27.8 km/h (15 kt) sterm to bow wind
- Ship speed of 18.5 km/h resulting in 9.3 km/h (5 kt) headwind for bow appraoches
- No steady crosswind for ship appraoches

Clouds, engine/aerodynamic sounds, and aircraft motion were simulated and the assistance of an LSO was provided.

APPROACH PROFILE - The profile developed for this simulation starts at 7.4 km (4 nm) from the AWLS source at 333 km/h (180 kt) with gear and flaps down.

The pilot rotates the nozzles to 70° and decelerates at constant attitude to establish 213 km/h (115 kt) at glideslope entry. He maintains this speed on the glideslope at 8° nominal AOA, and controls glideslope position (vertical vector) via the throttle. At 1.7 km (0.9 nm) he selects hover stop (81° nozzles) and begins deceleration to jetborne flight, still maintaining 8° AOA to breakout. Breakout occurs at 46 m (150 ft) altitude at the forward site and 30 m (100 ft) at the ship. The desired level-off altitude is 30 m and, on approaches

FIGURE 14
APPROACH/LANDING PROFILE



with an operational flight director, the displayed steering commands produce level-off at that altitude. At the completion of the level-off maneuver, the desired airspeed is about 65 km/h (35 kt).

Approximately 100 measurands were recorded at each 61 m (200 ft) interval along the 7.4 km approach. The mass of data obtained on more than 600 approaches was reduced to manageable proportions using MCAIR computer programs to convert raw data to meaningful parameters for dispersion analysis and analysis of variance. The pilots provided valuable qualitative information in their post flight debriefings and responses to written questionnaires:

FIGURE 15
PILOT COMMENTS

Displays:

Head Up
- Command Steering is Very Useful and Probably Required for IFR Landing Approaches
- No Problems in Flying Flight Director
- Everything Necessary to Fly Safely in the Weather is Presented
- 3:1 Pitch Scaling Displayed at 10° Intervals is the Preferred Attitude Display (Unanimous)
- 1:1 Scaling With 10° Pitch Bars is Too Dynamic
- 5:1 Scaling With 30° Pitch Bars is Too Gross

Head Down
- The Simulated Head Down Display is Sufficient to Fly the Approach
- Greatly Increased Workload
- After Training and Pilot Confidence is Established Will be Satisfactory Back-up to the HUD
- Undesirable Without Heading on ADI

Profiles:

3° Glideslope Head Up
- Easy to Fly Under All Conditions With Flight Director
- Light to Moderate Workload
- Adequate in All Aspects With Flight Director
- Adequate Response Time Available After Breakout

5° Glideslope Head Up
- Satisfactory on Approach
- Workload Increases Significantly After Breakout
- Marginal Time Available After Breakout
- No Big Problems With Adequate Pilot Training

3° Glideslope Head Down
- Difficult to Fly
- High Pilot Workload Tapering Off With Experience
- Cannot be Flown as Precisely as Head Up

Autopilot:
- The Autopilot is Highly Desirable for Reducing the Pilot Workload During the Approach
- Attitude Hold and Control Stick Steering Provide More Time for Instrument Scan in Head Down Recoveries
- Transitioning From Wingborne to Jetborne Flight in IFR Requires Automatic Control of Sideslip

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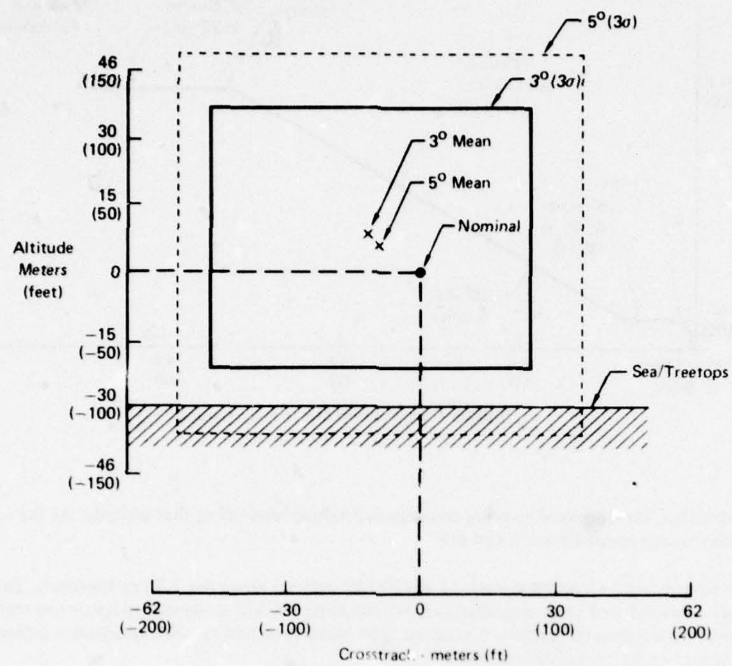
3° GLIDESLOPE vs 5° GLIDESLOPE PROFILE - Airspeed control on the two glideslopes presented no problems. Therefore, the major criteria for comparing the quality of 3° and 5° glideslope profiles were vertical error dispersion and control of angle of attack (AOA). The 3σ dispersion at breakout on the 3° profile is 9 m (30 ft) above the sea or treetops, but for the 5° profile it is below these levels for the minimum ceiling. It should be noted, however, that the standard deviation analysis is conservative and that in no case did an approach come closer than 8 m (26 ft) to the sea or treetop.

FIGURE 16
BREAKOUT DISPERSION WINDOW (3σ)
3° GLIDESLOPE vs 5° GLIDESLOPE



Pilot use of the throttle and stick was recorded on all approaches. Since the throttle is the primary means of controlling sink rate, the increase in throttle operation which occurred on the 5° glideslope indicates that pilot workload is increased and that desired altitude is more difficult to maintain. This increase in pilot workload on the higher angle glideslope was further substantiated by the doubling of longitudinal stick usage on the 5° profile.

On both profiles angle of attack was controlled very well. The mean was slightly below the 8° nominal AOA on the 3° profile and slightly above it on the 5° profile.

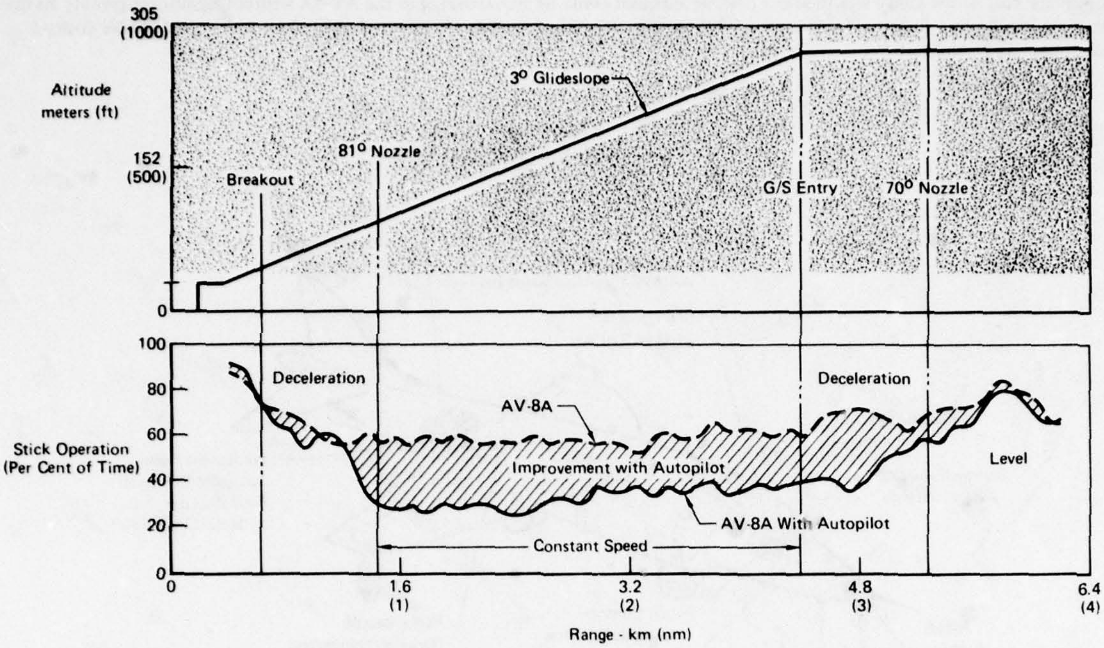
Nominal crosstrack error along the profile was slightly greater for the 3° profile than for the 5°. This was attributed by the pilots to the intense concentration required for steep approaches. Since localizer control is no more difficult for 5° profiles than for 3° profiles this concentration results in a smaller crosstrack error, while workload is increased considerably. In no case was the crosstrack error of unmanageable proportions, and recovery from the dispersion after breakout was accomplished in every simulated run.

The increased workload and performance errors associated with the 5° and 9° glideslope resulted in our recommending the 3° glideslope profile. But should higher angle glideslopes be required for obstacle clearance, glideslopes up to 9° are feasible although they would require higher ceilings and more pilot training.

AUTOPILOT - Because we thought that differences in autopilot experience among the four USMC pilots could bias the simulation results (all had previous inflight experience with either the A-4 or F-4 autopilots) we conducted a brief training session to familiarize them with the simulated autopilot.

The autopilot was mechanized so that attitude hold could be disengaged by exceeding 68 grams (2.4 lb) longitudinal force or 91 grams (3.2 lb) lateral force on the control stick. Whenever one of these forces was exceeded, the event was recorded. These data were used to prepare a graph of stick usage on the 3° glideslope.

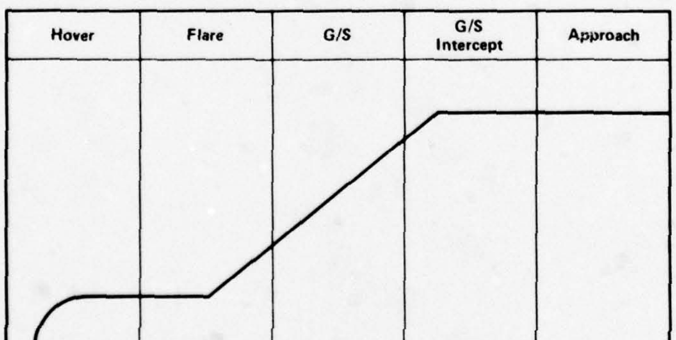
FIGURE 17
LONGITUDINAL STICK OPERATION DURING APPROACH



This plot is an effective measure of pilot usage of attitude hold and was used to assess the workload effect of attitude stabilization. As can be seen from the figure, some phases of the landing approach benefit more than others. During level deceleration after the nozzles are rotated to 70°, pitch attitude hold becomes more useful as shown by the decrease in stick motion. During the constant velocity flight which follows, pitch attitude hold becomes more effective down to the 81° nozzle select point where the pilot begins to take over manual control. The large nozzle and throttle operations required for completion of transition overpower the autopilot authority at this point causing the pilot to compensate. The conclusion drawn is that during constant velocity descent, the autopilot provides pilot relief and improved handling characteristics allowing him to set up and prepare for transition from 213 km/h (115 kt) to 65 km/h (35 kt) at breakout.

Detailed debriefings were conducted following the simulations. The pilots were unanimous in their belief that the selected landing aids would permit the AV-8A to operate in IFR conditions from forward sites and ships.

FIGURE 18
SUMMARY OF WORKLOAD

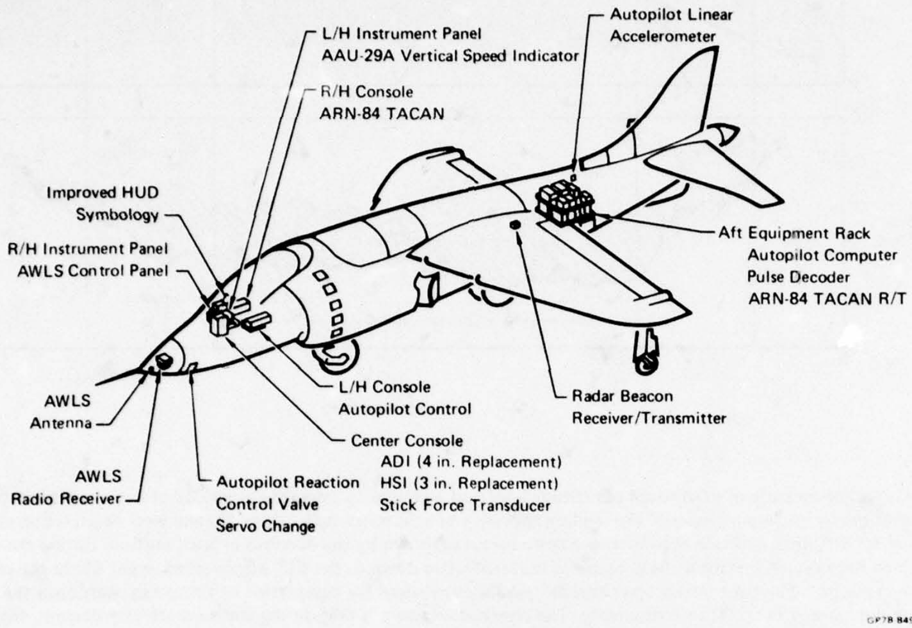


	Hover	Flare	G/S	G/S Intercept	Approach
Without Landing Aids	Moderate to High	Very High	High	Moderate to High	Moderate
With Landing Aids	Low to Moderate	Moderate	Low	Low to Moderate	Low

AIRCRAFT INTEGRATION

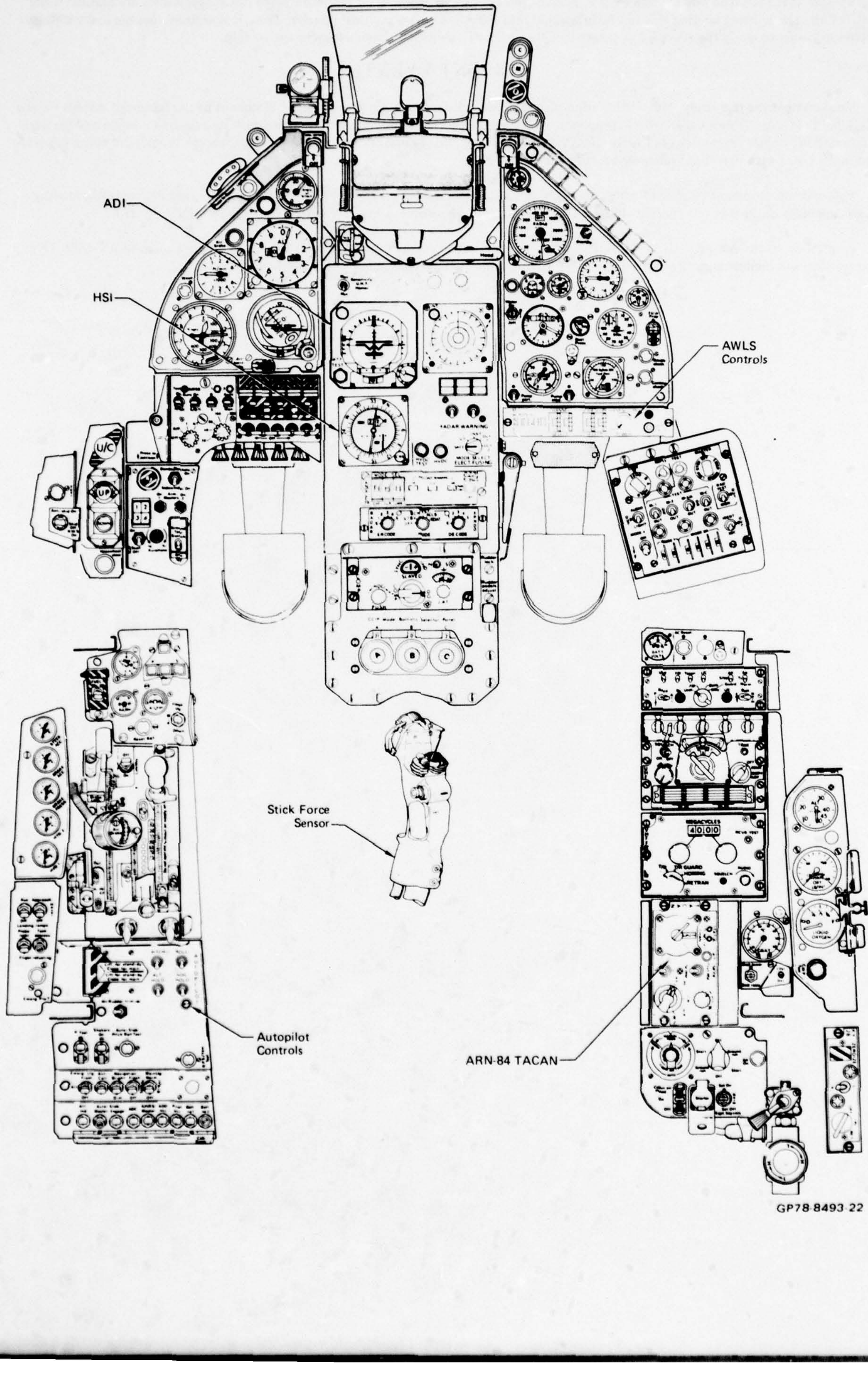
A primary goal of the study was to select landing aids that could be incorporated in the AV-8A without significant penalty to the combat payload/radius. Space, aircraft power, cooling, etc., can accommodate the selected equipment and a comfortable cockpit arrangement can be laid out.

FIGURE 19
AV-8A HARRIER LANDING AIDS



Installation of the selected landing aids in the aircraft adds approximately 56 kg (124 lb) to the operating weight empty (OWE) of the aircraft. The added weight reduces the combat radius of the aircraft by only a few kilometers.

FIGURE 20
LANDING AIDS COCKPIT ARRANGEMENT



CONCLUSIONS

We found that 3° profiles can be safely flown to a forward site or ship in IFR conditions with turbulence and crosswinds and that operational minimum ceilings are 46 m (150 ft) for the forward site and 30 m (100 ft) for ships. An attitude hold autopilot is required to operate at the reduced minima and a flight director presentation is required on the HUD. Head down approaches are feasible if the HUD fails. The selected landing aids can be integrated into the AV-8A with minimal penalty. Thus, it is entirely feasible and within the state-of-the-art to equip the AV-8A to operate at night or in IFR conditions from a remote site or ship.

RECENT EVENTS

Since completing this study, MCAIR has been placed under contract to begin prototype development of the advanced AV-8B for the U.S. Marine Corps. A new supercritical composite wing with positive circulation flaps, combines with redesigned air inlets and fuselage-mounted lift improvement devices to give the AV-8B twice the payload radius of the AV-8A. Two prototype models are being fabricated in St. Louis with first flight scheduled for November 1978.

Full scale development go ahead for the AV-8B is anticipated for January 1979. The AV-8B will incorporate the complete landing aids capability defined in our study including AWLS, attitude hold autopilot, improved HUD displays and flight director.

In addition to the AV-8B activity, MCAIR is integrating advanced avionics in the AV-8A aircraft to convert them to AV-8C's. This conversion will include an attitude hold autopilot derived from that defined in our landing aids study.

IMPLEMENTATION OF FLIGHT CONTROL IN AN INTEGRATED GUIDANCE AND CONTROL SYSTEM

H.J. Bangen   Dornier System
W. Hoffmann   Friedrichshafen
W. Metzdorff   FRG

SUMMARY

To meet increasing helicopter mission requirements a need exists for new additional command and control equipment. In order to reduce system complexity and cost an integrated modular signal data processing system has to be used. Verified by flight tests an increase of system performance is achieved by implementing nonlinear control laws and high control authority in the flight control system. The hardware and software technology is presented as it is required to solve the control problem in the integrated helicopter guidance and control system.

0. INTRODUCTION

The German Army has extended their requirements to the night and adverse wheather capability for a new generation of antitank and transport helicopter. Figure 0.1 shows the performance regions subdivided in mission profiles for which new technologies are required.

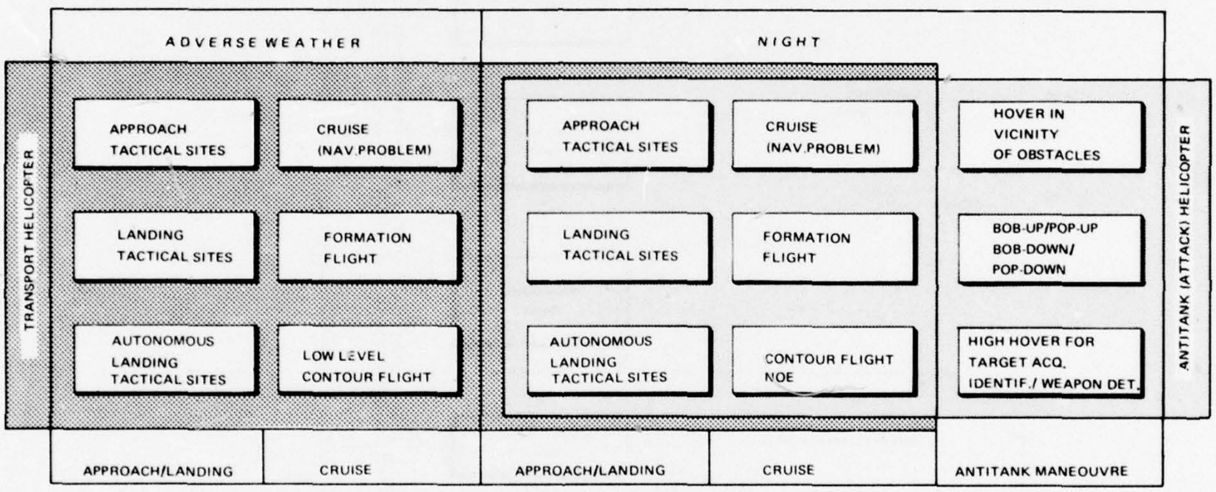


Fig. 0.1 Performance Region of Antitank and Transport Helicopter

The technical solution for the night capability requirement is not just a night vision system which will be added to already incorporated guidance and control-/avionic subsystems. The degradation of visual cues by the night vision system requires a substitute in the form of better handling qualities to flight control and housekeeping management functions.

As a consequence, the higher mission requirements lead to a marked increase in the number of systems onboard. If implemented by conventional means, an extremly large increase in complexity and cost will occur in all phases: development, production and mainly in the operational phase. In order to overcome some difficulties the German Ministry of Defense sponsors a Helicopter Guidance and Control Program (HSF). The goal is the increase in mission capability and configuration flexibility and the reduction of complexity and cost by new hardware and software structures and technologies.

1. SYSTEM ARCHITECTURE

The conventional architecture of helicopter equipment (avionics, flight controls) has been an amalgamation of nearly autonomous subsystems for all different functions, each subsystem having independent sensors, processors, effectors (displays/controls). The new system architecture comprises

- an integrated sensor system
- an integrated data acquisition/processing/distribution system
- an integrated effector system (actuators, displays).

The hardware is locally distributed all over the aircraft. System functions will be mechanized by application software. In order to reduce the complexity and the cost, a higher degree of commonality (standardization) among all line replaceable units (LRUs) of the integrated data processing system has to be attained Fig. 1.1.

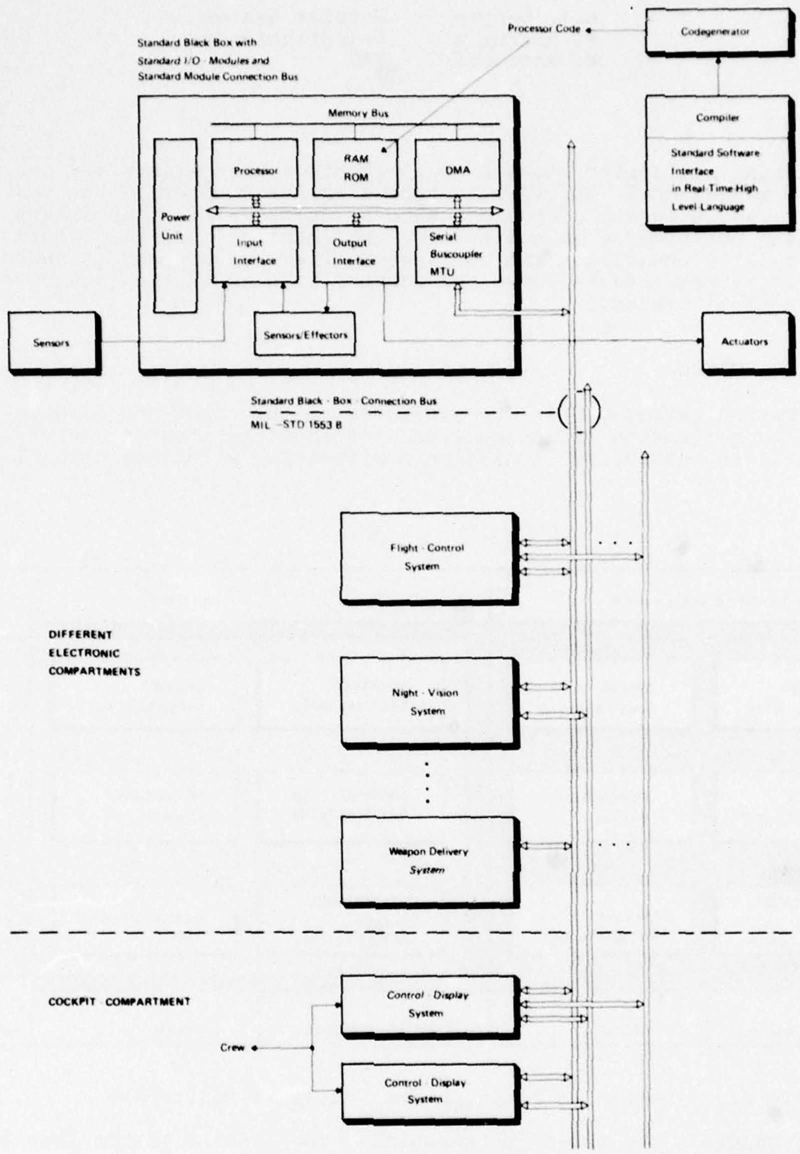


Fig. 1.1 Digitally Integrated Architecture of Avionics and Flight Control System

MIL-STD-1553 as a standard interface and protocol for data distribution between LRU's is the step towards commonality among LRU's in a distributed system. This data high way is a tool for functional partitioning and flexible modular integration at the LRU-level [1.1].

A second but consequent step with the goal of higher commonality is comparable to the MIL-STD-1553 approach defining a standard for the data way on the LRU-level. A parallel data bus has proven to be an appropriate solution. Neither the serial bus nor the module connection bus restricts the technological innovation, as they are interface definitions. The example of the serial bus coupler shows, that the interface definition make possible a higher integration of hardware and that all boxes may use the same module. By this way there is an increase of common modules in the system and a reduction of cost.

The definition of two hardware interfaces lead directly to a third interface definition - software: Using a real time high level language a structured programming of application software is possible. The translation is done by a compiler and special code generator to the special processor code. A candidate system may be PEARL [1.3]. This language describes also the hardware up to registers by software. Since the hardware is well structured the compiler generates an effective code required to real time application.

2. CONTROL FUNCTION

2.1 Mission Requirements

Regarding flight characteristics the military helicopter can be designed for the following tasks:

1. transport task
2. platform task for antitank and antisubmarine warfare.

The 1st task requires long distance flight including start and landing. Thereby the partially unstable flight conditions can be compensated by the pilot and stable flight performance is obtained. To reduce pilot workload additional damping and an autopilot function for the long distance flight can be included.

In the platform task angular and transitional position of the helicopter has to be accurately maintained in the presence of external disturbances. In order to minimize flight hazards during the mission the pilot has to be able to set flight conditions with adequate speed. Simultaneously, precise control on a curved flight path is required during the nap of the earth flight. Thereby attitude angles up to 60° and loads up to and above 2 g are encountered. Contrary to the transport task large and fast changes of the flight conditions have to be executed.

Since the control accuracy is inversely proportional to the cut-off frequency of the control loop and since man in a control task can process only frequencies up to 0.3 - 0.5 Hz, the loop gain for the platform task is not sufficient sometimes even under good visibility conditions. Under bad weather conditions open loop gain will be further reduced. Substitution of lost information by display system and visual aids will not suffice since these systems limit man's adaptability. Supporting man by a control system designed for the transport task is inappropriate. Instead, one has to start from the more demanding platform task which includes the transport task.

2.2 Structure of Control Law

Since mission requirements are partially contradictory to helicopter flight characteristics, it is necessary to change helicopter dynamics over the total range of flight conditions by means of a control system. If this requirement has to be satisfied in the presence of large disturbances and if the control system is not permitted to reduce natural flight performance and natural control range, the so called manoeuvre demand-system has to have large authority on control surfaces. For technological reasons, a mechanical control system with superimposed electrical control is not advisable and the manoeuvre demand function should be designed as a redundant digital Fly-By-Wire-System. The suitability of this approach for the helicopter has been confirmed by various test programs [2.1].

If the manoeuvre demand system is mechanized in digital Fly-by-Wire technology one should take full advantage of the possibilities of digital signal processing and should not only use it as an additive control system. Flight tests have shown that the manoeuvre demand system must not change the characteristics of the natural control system. In order to obtain precise flight path guidance it is appropriate to use a helicopter referenced coordinate system. Pilot's commands have to be decoupled and rotational and translational damping has to be provided. Additional pilot corrections should not be required in order to obtain a flight free of angular accelerations. The functional requirements for a manoeuvre demand system are satisfied by a controller structure based on the natural aerodynamic characteristics. Nonlinear flight characteristics are mechanized in the controller in their inverse form. The basic controller structure is shown in fig. 2.1. Block W and disturbance feed forward in connection with RD and RW constitute the nonlinear inverse flight characteristics mentioned above. They also reduce interaxis cross-coupling. In addition, a proportional integral control algorithm (P.I.) is utilized to minimize stationary residual crosscoupling terms and disturbances. Trimming is not necessary since this proportional integral algorithm uses 4 steady state conditions being mutually dynamically perpendicular to each other, corresponding to the 4 control axes (pitch, roll, collective, yaw). The integrator provides automatic trimming over the full range of flight conditions, thus minimizing pilot controls. The 4 orthogonal controlvariables X_R are determined in the block RW. The block called adaptation of control command has been added. With this block the control response is being modified. In addition controller sensitivity can be altered and new meaningful command functions like coordinated turns can be obtained.

In the HSF-experimental system, compensation of computer dead time was obtained by feeding back the actuator command η. In this way, in spite of the low sampling frequency, the total stability range could be enlarged by variation of parameters.

Controller structure used in the experiment differs from other digital flight controller systems [2.2]. These use separate trim-computation to obtain the stationary part of the actuator command. Linear feedback is used to obtain the control signal varying about the trim flight condition.

The controller structure derived from the aerodynamic characteristics offers many advantages for operational useage. The controller can be integrated and tested in a step by step fashion. Basic selection of parameters is made from a simplified linear model.

Flight tests are based on these parameter values. The nonlinear terms are being added successively, whereby the effect of each term can be observed exactly. Each single controller parameter can be interpreted by its physical meaning, such that the effect on flight performance can easily be estimated.

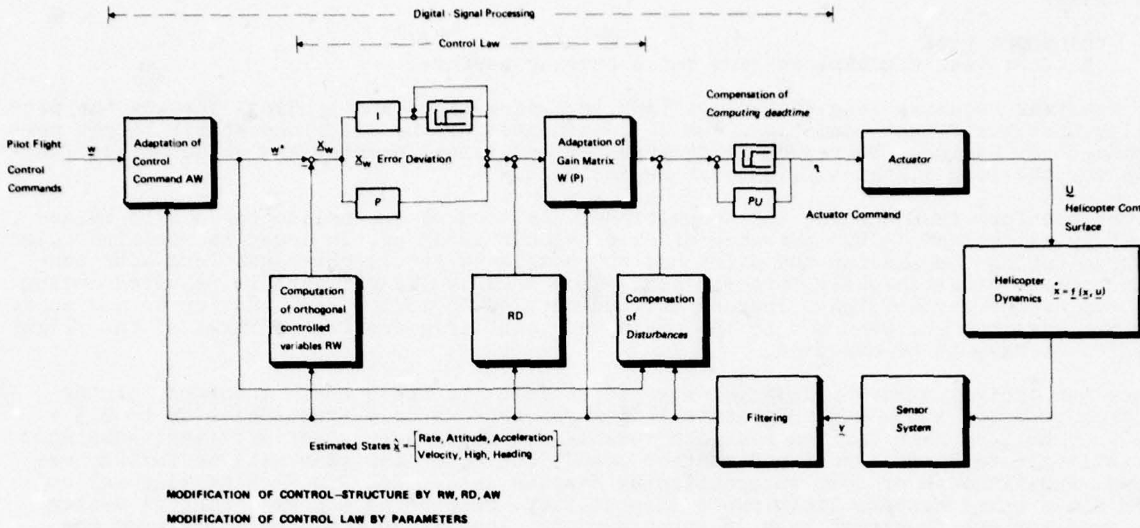


Fig. 2.1 Structure of HSF Manoeuver Demand System

2.3 Flight Results

After qualitative testing of different demand functions the system most suitable for the platform task was further investigated. The pilot maintains direct access to the rotational and translational movement of the helicopter. He commands the 3 rotational velocities. In addition he has direct command of lift. In the absence of inputs the controller maintains attitude (θ, ϕ) and heading.

Fig. 2.2 a shows a commanded change for forward velocity from 90 kts to 60 kts. For this speed variation only a large pitch command W_1 is necessary to start the change, one command is required to end it. Except for a small correctional input there is no command required in the roll and yaw axes. Collective control is obtained as in a conventional system. Fig. 2.2 b shows the corresponding commands of a conventional mechanical 1 : 1 control system of the same helicopter. Continuous pilot inputs are necessary to maintain the speed of 60 kts.

Fig. 2.3 shows the frequency distribution of attitude and heading angles during controlled and uncontrolled hovering.

The example of the rate demand system with attitude and heading hold shows that flight path guidance and attitude accuracy can be increased without reduction of accuracy and speed of control inputs. The tests showed on a qualitative basis that neither a rate demand nor an acceleration demand-function satisfies the requirements of the platform task. Furthermore a combination of different demand-functions has to be used. A system of this type is presently mechanized.

It should be mentioned that a manoeuvre demand system does not require continuous inputs for fine control and trimming. These new system characteristics cause difficulties to the pilot if he is not given sufficient opportunity to fly the system to get acquainted with the new flight characteristics.

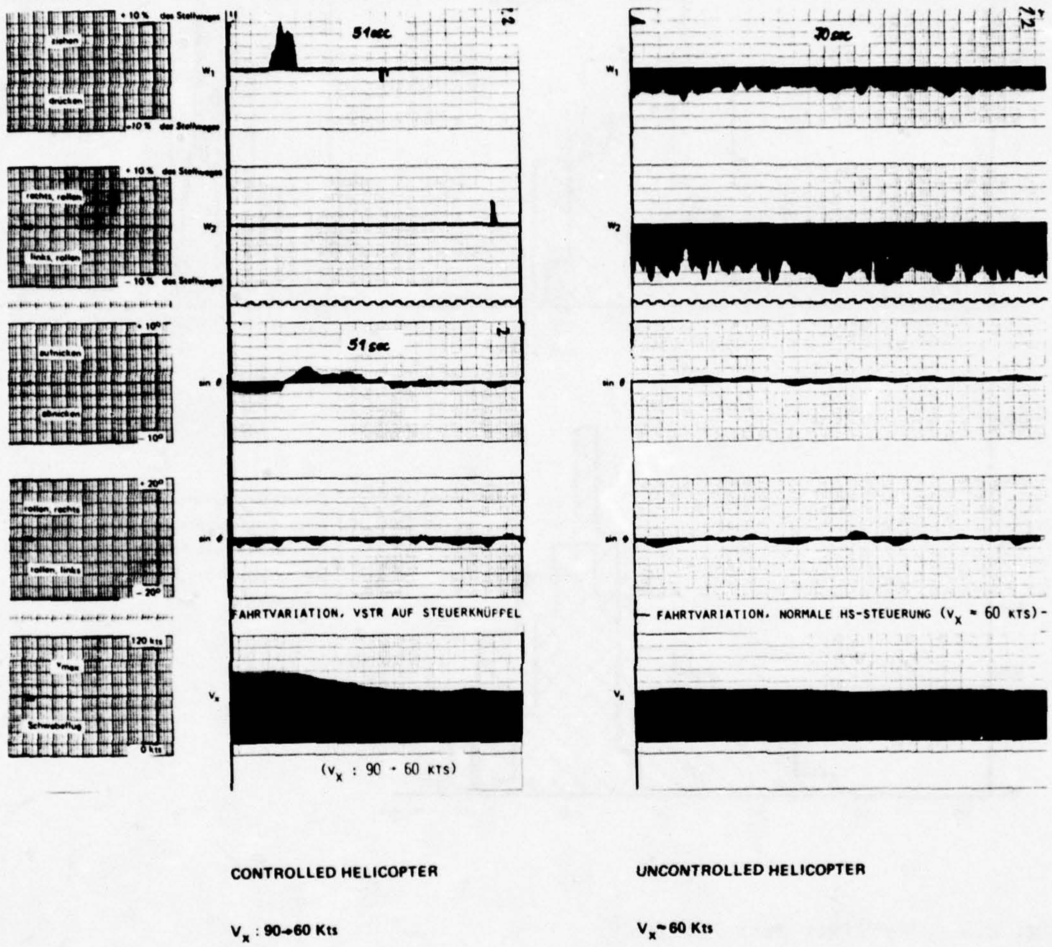


Fig. 2.2 HSF-Flight Test Results

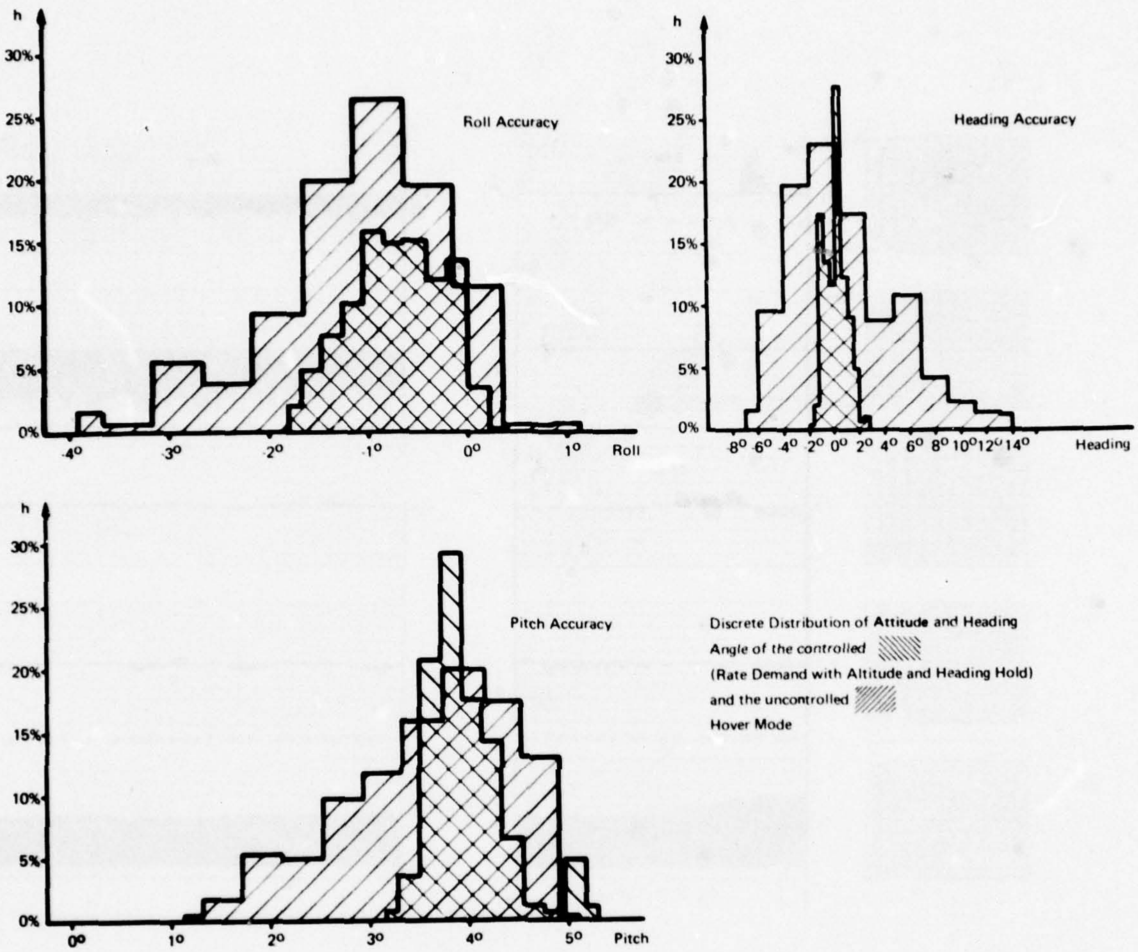


Fig. 2.3 HSF-Flight Test Results

3. SYSTEM MECHANIZATION

3.1 Simplex Control Function

The command and control system as described and flight tested was mechanized by means of a non-redundant digital signal processing system. Redundancy was not used in the system during the first part of the tests since the test helicopter BO 105-S3 has a safety-back-up system and the test flights were performed under simulated operational conditions.

Two digital processors are used to mechanize the manoeuvre demand functions. This enables multiple useage of sensor functions as required in an integrated avionic system and installation of electronic equipment at separated locations (Fig. 3.1). The 1st processor provides the basic Fly-by-Wire computations. The signals from the flight data sensors (rate, attitude, acceleration) and pilot inputs are processed to provide control of actuators and of the helicopter safety system. The 2nd processor is used to expand the functions of the basic computer. Using information from additional sensors (doppler, velocity, airdata, heading and altitude sensors) permits switching between different control functions, modifications of control parameters, and additional control functions (e.g. heading hold) without affecting system computation time. Cockpit equipment is directly connected to the processors in the electronic compartment.

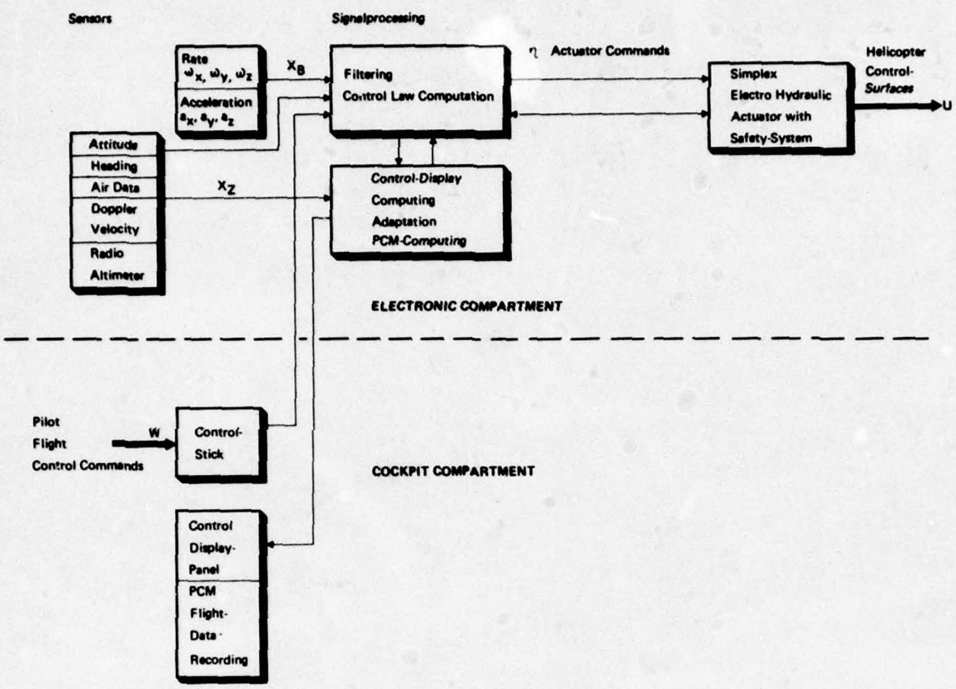


Fig. 3.1 HSF Simplex FWB Control System

The control system has 4 mutual orthogonal steady state conditions which can be altered by the control inputs w. If one of the control commands exceeds a certain value, automatic change-over will occur in this axis, for instance from a rate command system to an attitude command system. The command value for the attitude command system will be the angle present in the system at the moment of change-over. As one can see from the simplified equations

$$|W| > \varepsilon \quad U(n) = p_1\,[w(n) - \omega(n) + x(n)]$$

$$x(n+1) = x(n) + I_1\,[w(n) - \omega(n)]$$

$$\phi_o = \phi(n)$$

$$|W| < \varepsilon \quad U(n) = p_2\,\omega(n) + p_3 \sin\,[\phi_o - \phi(n)] + x(n)$$

$$x(n+1) = x(n) + I_2 \sin\,[\phi_o - \phi(n)]$$

there can be steps in the actuator commands when changing from one control mode to the other. Using a decaying smoothing function, disturbances can be minimized without adversely affecting stability. This fundamental system capability permits in case of transmission errors switching between different control laws without changing the total system function.

The P.I. control function achieving automatic trimming does not tolerate control commands with bias errors. Since analytical determination of constant bias values is not practical, bias values of the accelerometer- and rate-gyro signals are automatically measured and stored during the preflight phase such that automatic compensation during flight is possible.

The sampling frequency of the control processor is 20 Hz and of the 2nd processor is 10 Hz. These update rates are sufficient for helicopter stabilization purposes. These sampling frequencies cause problems in connection with rotor rotation. Using a low pass filter for the 28 Hz rotor disturbance leads to lower loop gain to maintain stability. Since the required control function does not permit this, the 28 Hz disturbance is compensated by introducing an additional sampling process [3.1].

The simplex Fly-by-Wire control system used in the 1st test series was tested toget with a navigation function and a sight system. Fig. 3.2 shows the equipment of the e tronic compartment.

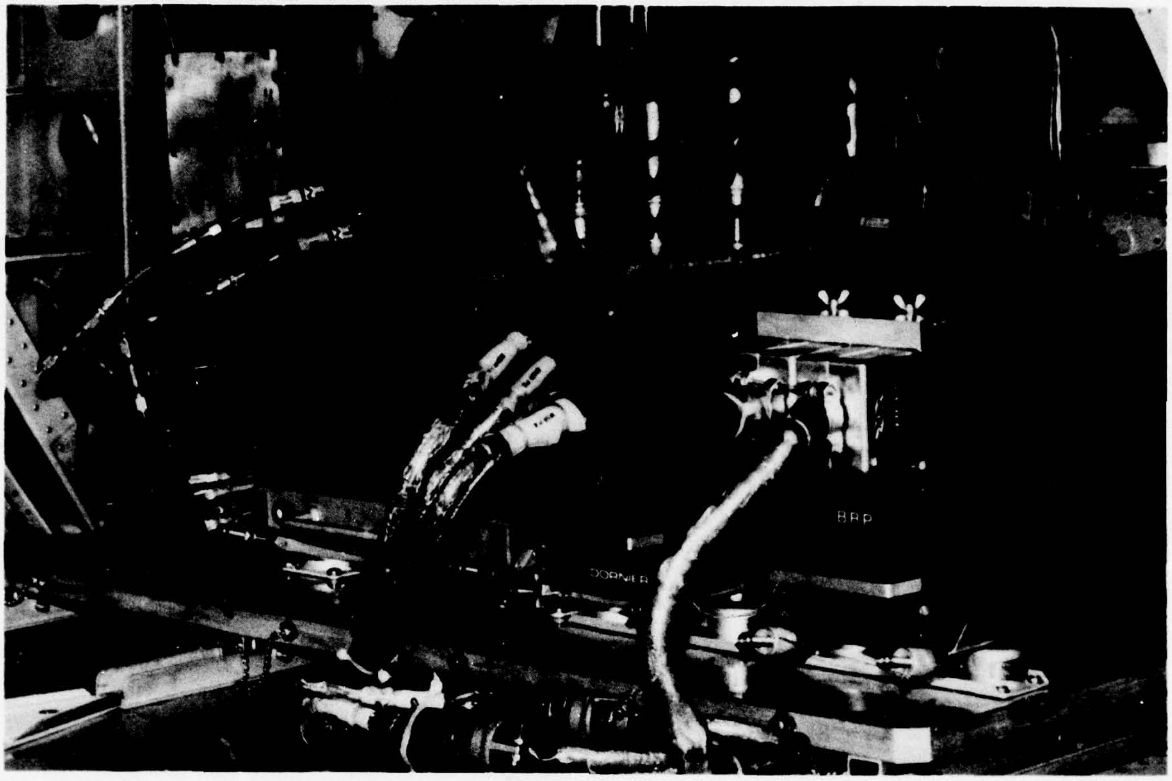


Fig. 3.2 HSF Signalprocessing Units in the Electronic Compartment

3.2 Redundant FBW Control System

During the 2nd test series the described control command function will be mechanized as a redundant system. Since the electrical and hydraulic supply systems of a helicopter are only double redundant, also the actuators of the redundant Fly-by-Wire system will only be double redundant. Triple redundancy will be employed for the signal processing part (processors, sensors). Using automatic failure self detection, a Fly-by-Wire system can stand two failures of the same type and provide an MTBF which meets the requirements of the mission.

Redundancy requires additional equipment. Based on technological progress and experience with standardized signal processing modules and standardized signal data transmission technique (parallel-data-bus; serial-data-bus MIL-STD-1553), one can simplify the configuration of Fig. 3.1 to Fig. 3.3. The electronics for the command and control function will be contained in separate units.

G_4, G_5, G_6 are units used in the cockpit area for helicopter control and other operational control and display functions. G_1, G_2, G_3 process signals of flight data sensors and control each one of the double redundant electro hydraulic actuators. Using a triplex data bus to interconnect all electronic units, wiring can be minimized in comparison to point to point wiring.

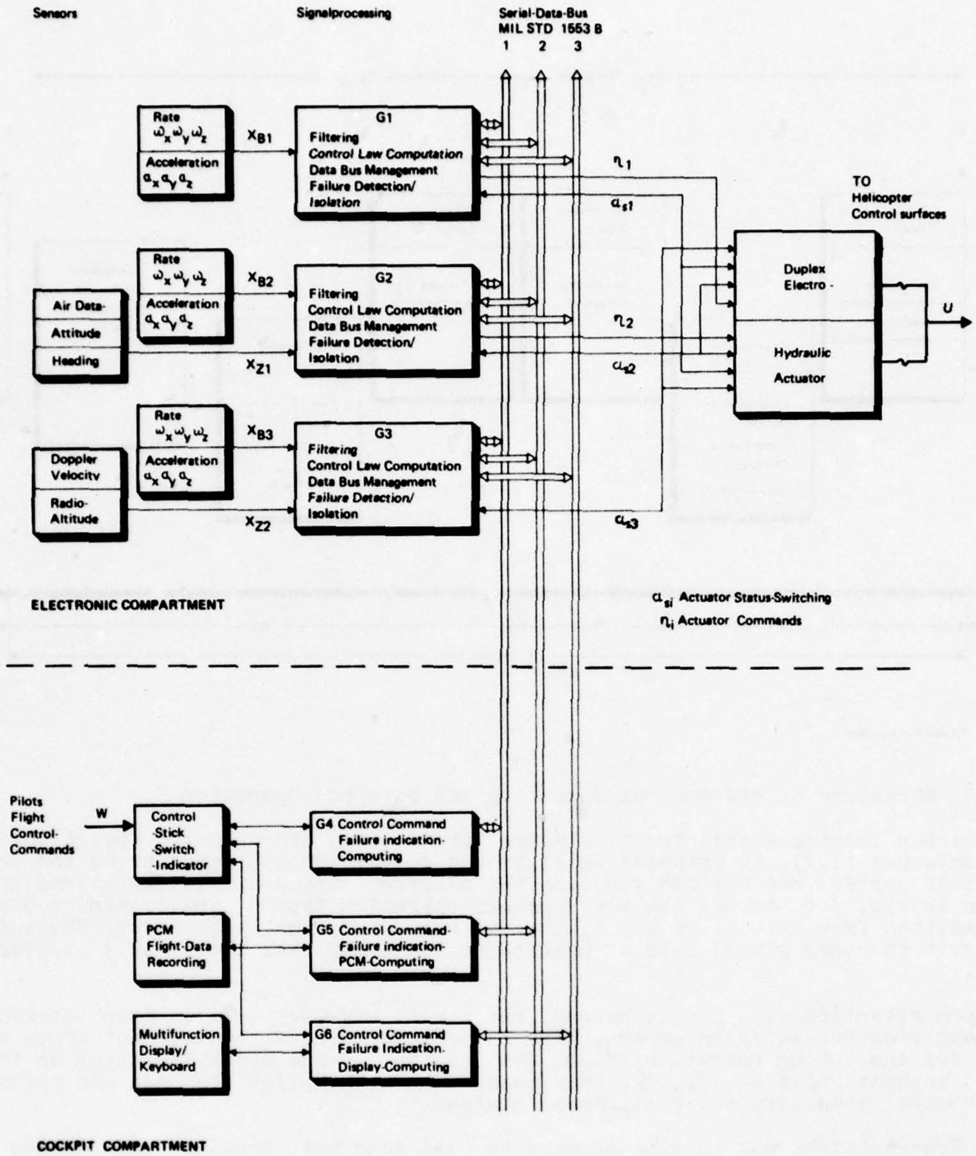


Fig. 3.3 HSF-Redundant FBW Control System

The safety critical redundant command and control system places the highest demands on computational rates of the units and transmission capacity of the serial busses. To reduce system complexity symmetrical structures were selected to implement the command and control function in hardware and software. The control algorithm is contained in units G_1, G_2, G_3. Fig. 3.4 shows timing and synchronization. Non interacting system synchronization is obtained by means of data transmission on the serial busses. Thereby one of the clocks initiates the data transfer which is monitored by the other two clocks.

After the 1st bus operation a synchronized condition exists such that the output of the selected actuator commands can occur almost simultaneously with the input of the sensor values. During the 2nd bus operation the processed sensor data are being exchanged. During the 3rd bus operation control commands are being transmitted from units G_4, G_5, G_6. Subsequently each unit can perform an autonomous selection. The selected information is used for a nonlinear filter computation to determine an estimated value for the state variables and to perform the control law computation. The other procedures are interruptable auxiliary computations as unit self test, signalprocessing, and transmitting of non-redundant flight state variables.

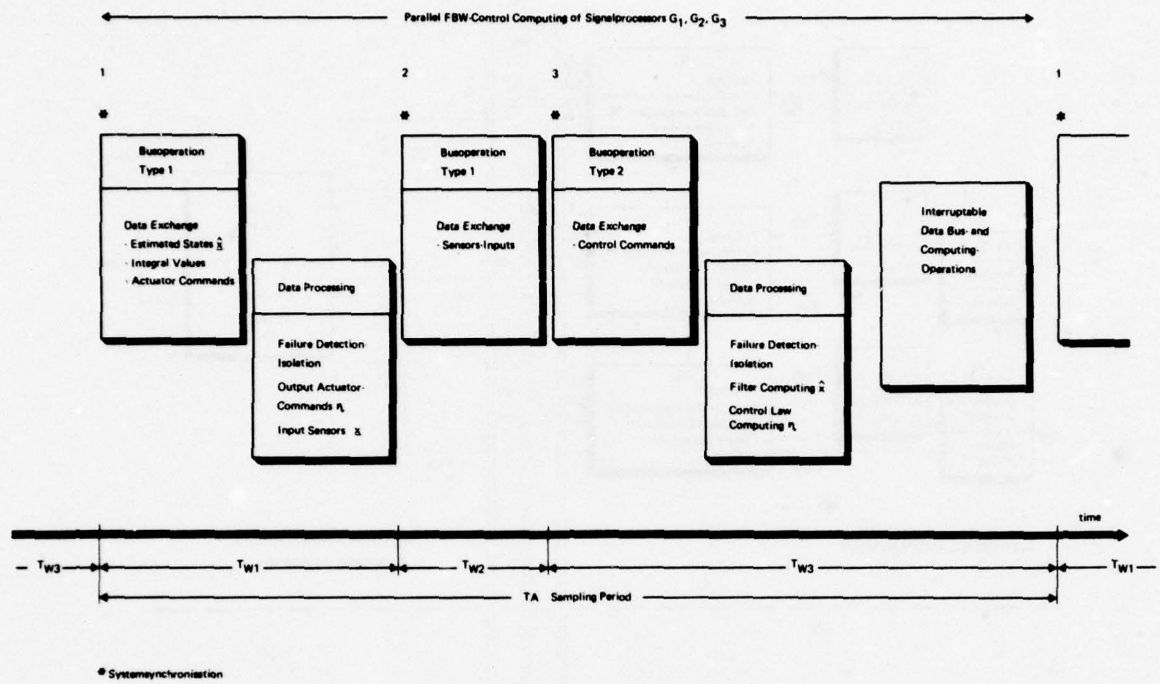


Fig. 3.4 Structure of FBW Control Computing and Data Bus Operation

To reduce bus loading global transfer modes (broad-cast) are used for operation on the serial-data-bus [1.1], as proposed in [1.2] and successfully tested during the 2nd part of the test series. For the FBW function two different types of bus operations are used as shown in Fig. 3.5. As one can see from bus operation type 1, the transmitted message is transmitted from unit G_1 to bus 1, then to bus 2 and finally to bus 3. The units G_2, G_3 transmit the same global type of message sequentially over bus 2, 3, 1 or over bus 3, 1, 2.

Since upon reception of a global message the serial bus coupler issues an interrupt and also identifies transmission errors, only information transmitted without error will be cleared for the voting operation [3.2]. The 2nd type of bus operation picks up the data of the 3 cockpit units G_4, G_5, G_6. This ensures that the pilot commands are transmitted to the control computers via 3 different routes.

For the FBW-functions bus loading amounts to 2.46 msec per sampling cycle. Since the control computation needs only a refresh rate of 20 Hz, the serial busses 1, 2 and 3 are occupied only about 5 % of the time. The remaining transmission capacity can be used to connect other units in an integrated system. The effectivity of the utilized bus protocol is caused by using global operations in place of local operations as initially proposed in MIL-STD-1553B. Units G_4, G_5, G_6 include display controllers and PCM data output. By transmitting state variables and voting conditions as global messages a separate data transfer with subsequent voting operation is not required. Each unit connected to the serial bus can monitor the total data transfer on the bus and can evaluate the information according to its specific requirements.

Integration of the redundant system has been started. Flight tests and EMI-environment will show whether the selected bus protocol will enable reliable system operation. Low bus loading as mentioned above will provide ample room for system modifications. Using the standardized hardware and software structure, modifications can be implemented in a very cost effective manner.

	Message Number	Bus-Transfer			Message Formats		
		Bus 1	Bus 2	Bus 3	Message Type g_i global l_{ig} local Transfer	Message time 2,46 msec	Datawordstructure
redundant Busoperation Type 1	1	g^1_1	g^1_2	g^1_3	$g^1 \hat{=} \{\text{CW, 14DW}\}$	$\hat{=}$ 300 μsec	$g^1 \hat{=} g^2 \hat{=} g^3 \hat{=}$ CW – Commandword + 14 DW: – 2 Failure Statuswords – 4 Integral Values – 4 Estimated Attitude ($\sin\vartheta$, $\cos\vartheta$, $\sin\psi$, $\cos\psi$) – 4 Actuator Commands
	2	g^2_3	g^2_1	g^2_2	$g^2 \hat{=} \{\text{CW, 14DW}\}$	$\hat{=}$ 300 μsec	
	3	g^3_2	g^3_3	g^3_1	$g^3 \hat{=} \{\text{CW, 14DW}\}$	$\hat{=}$ 300 μsec	
redundant Busoperation Type 1	4	g^4_1	g^4_2	g^4_3	$g^4 \hat{=} \{\text{CW, 14DW}\}$	$\hat{=}$ 300 μsec	$g^4 \hat{=} g^5 \hat{=} g^6 \hat{=}$ CW – Commandword + 14 DW: – 2 Failure Statuswords – 3 Rate (ω_x, ω_y, ω_z) – 3 Acceleration (a_x, a_y, a_z) – 4 Actuator – 2 Changing words
	5	g^5_3	g^5_1	g^5_2	$g^5 \hat{=} \{\text{CW, 14DW}\}$	$\hat{=}$ 300 μsec	
	6	g^6_2	g^6_3	g^6_1	$g^6 \hat{=} \{\text{CW, 14DW}\}$	$\hat{=}$ 300 μsec	
redundant Busoperation Type 2	7	l^7_{14}	l^7_{25}	l^7_{36}	$l^7 \hat{=} \{\text{CW, SW}\}$	$\hat{=}$ 80 μsec	$l^7 \hat{=} l^9 \hat{=} l^{11} \hat{=}$ CW – Commandword DW – Dataword SW – Statusword
	8	g^8_4	g^8_5	g^8_6	$g^8 \hat{=} \{\text{CW, 6DW}\}$	$\hat{=}$ 140 μsec	
	9	l^9_{34}	l^9_{15}	l^9_{26}	$l^9 \hat{=} \{\text{CW, DW, SW}\}$	$\hat{=}$ 80 μsec	$g^8 \hat{=} g^{10} \hat{=} g^{12} \hat{=}$ CW – Commandword + 6 DW: – 2 Control-Statuswords – 4 Controlinputs (w)
	10	g^{10}_4	g^{10}_5	g^{10}_6	$g^{10} \hat{=} \{\text{CW, 6DW}\}$	$\hat{=}$ 140 μsec	
	11	l^{11}_{24}	l^{11}_{35}	l^{11}_{16}	$l^{11} \hat{=} \{\text{CW, DW*, SW}\}$	$\hat{=}$ 80 μsec	
	12	g^{12}_4	g^{12}_5	g^{12}_6	$g^{12} \hat{=} \{\text{CW, 6DW}\}$	$\hat{=}$ 140 μsec	

$\{ l^n_{ig}, g^n_i \}$ i: source; g: destination; n: message number

Fig. 3.5 Serial-Data-Bus Organisation to Redundant FBW Control Function (MIL-STD-1553B)

4. LITERATURE

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[2.1] A. Simpson, H. Hitch, Active Control Technology, Aeronautical Journal. 6.77, p. 231-246

[2.2] W. Kubbat; Evaluation of a Digital Helicopter Control System, Paper 33 - 3. European Rotorcraft and Powered Lift Aircraft Forum 9.77. AIX-EN-Provence

[3.1] H.J. Bangen, Über die Tilgung periodischer Störungen bei digitaler Meßwert-verarbeitung; Ingenieur-Archiv 45 (1976) 403-412

[3.2] MBA-Modulbeschreibung DO 900-340; Ankopplung von MUDAS-P an den seriellen Bus nach MIL-STD-1553, 7.77

STABILISATION DES SYSTEMES ELECTRO-OPTIQUES
SUR HELICOPTERES

par
Dominique de Ponteves
Adjoint à la Direction des Etudes
Société de Fabrication d'Instruments de Mesure
13, avenue Ramolfo Garnier
91301 Massy
FRANCE

ABSTRACT

The performances of an electro-optical sensor on board of an helicopter are directly related to stabilization quality.

It can be demonstrated that the effect of a sine-wave stabilization error on the system transfer function is described by a BESSEL Function.

Developed under DTAT (French Army Technical Services) sponsorship, the APX-BEZU M 260 gyro-stabilized sights were chosen for helicopters in a number of countries for surveillance and aiming of anti-tanks missiles such as AS 11 or similar.

More recently, the APX M 397 sight, together with an I.R. goniometer, performs weapon aiming for HOT missile.

In these systems, stabilization is implemented by a mechanical relation between a mirror and a conventional gyro.

The development of I.R. sensors needing large optical aperture has led SFIM to design and implement other stabilization techniques and new pick-ups optimized for these problems.

The GAM 1 gyro, developed by SFIM for this purpose, is a two-axis tuned gyro. It can be used either as a free gyro or as a rate gyro. The gyro performances allow the stabilization of the most performing electro-optical sensors with a MTF degradation of not more than 15 %.

Under STAé (French Air Force Technical Services) sponsorship, SFIM currently develops an electro-optical system combining : a TV camera with a very high focal length, a laser range finder and a twin-focal FLIR. Preliminary flight tests have been performed during 1977-1978 period, on an Alouette 3 helicopter, within the frame of a feasability research program for a weapon aiming system using 8-13 microns band, under STET (French Tactical Missiles Technical Service). They demonstrated the advantages of this technique.

SFIM has also developed simulation and test facilities allowing prediction and verification of system performances in real operational environment.

RESUME

Les performances d'un capteur électro-optique embarqué à bord d'un hélicoptère sont directement liées à la qualité de la stabilisation. On montre que l'influence d'un résidu sinusoïdal sur la fonction de transfert du système est donnée par une fonction de BESSEL.

Etudiés sous l'égide de la DTAT, les viseurs stabilisés APX-BEZU M 260 équipent les hélicoptères dans de nombreux pays pour l'observation ou le tir de missiles anti-char de la génération de l'AS 11.

Plus récent, le viseur APX M 397, associé à un localisateur infra-rouge, permet la conduite de tir du HOT.

La stabilisation de ces systèmes est effectuée par liaison directe d'un gyroscope classique à un miroir.

Le développement des capteurs fonctionnant dans l'infra-rouge, qui nécessitent des pupilles de plus grandes dimensions, a conduit la SFIM à étudier et à réaliser des techniques de stabilisation différentes ainsi que les capteurs les mieux adaptés à ces problèmes.

Le gyroscope GAM 1, spécialement développé par la SFIM, est un gyroscope deux axes à suspension souple. Il peut être utilisé en gyroscope libre ou bouclé en gyromètre. Ses performances sont compatibles avec la stabilisation des capteurs électro-optiques les plus performants sans dégradation de MTF supérieure à 15 %.

Sous l'égide du STAé, la SFIM développe actuellement un système électro-optique comprenant une caméra de télévision équipée d'une focale très longue, un télémètre laser et une lunette thermique à deux champs. Les essais préliminaires effectués dans le cadre d'une étude de faisabilité d'une conduite de tir 8-13 microns, menée sous l'égide du STET, et au cours d'essais en vol en 1977 et 1978 sur Alouette III ont montré tout l'intérêt de cette formule.

La SFIM a développé également les moyens de simulation et de test qui permettent de prévoir et de contrôler les performances d'un système dans l'environnement réel prévu pour son utilisation.

L'optique n'aime pas les vibrations.

Il suffit d'avoir essayé de faire des photographies avec une longue focale pour se rendre compte que la stabilité d'un instrument doit être d'autant plus grande que le grossissement ou la focale utilisée sont eux-mêmes plus importants.

Le développement très important des missiles anti-char et les performances des plate-formes d'observation que constituent les hélicoptères ont incité la D.T.A.T. à entreprendre dès 1960 l'étude d'un matériel qui en série dès 1965 est aujourd'hui encore commandé pour équiper les hélicoptères armés de la première génération de missiles anti-char : AS 11, AS 12, viseur APX BEZU 260, dont les caractéristiques sont :

masse du viseur complet : 33 kg

bioculaire biobjectif

simulateur

champs : 100 ₥ à G x 10
400 ₥ à G x 2,5

changement de champ < 0,1 s

anneau : Ø 5 mm

mise au point : -5 +2 dioptries

amplitude de balayage : ±27° précession } Gisement
±30° rotation
±30° précession) Site

Les différents constituants du viseur :

suspension

galiléique

gyroscope

capot (essuie-glace)

lunette

Montage sur Alouette 2.

La séquence de tir est la suivante :

recherche

détection

centrage dans le cercle lumineux *

changement de grossissement *

retour au grossissement 2,5

tir (guidage manuel)

Le viseur a également été monté sur vedette (*, *) ; la différence avec le montage hélicoptère consiste à asservir l'ensemble du poste de tir en gisement (*).

Plus récent, le viseur 334 résulte d'une analyse de la valeur effectuée sur le viseur 260.

Caractéristiques :

masse : 18 kg

monoculaire monobjectif

simulateur

champs 100 ₥ G x 10
400 ₥ G x 2,5

changement grossissement 0,1 s

anneau : Ø 5 mm

mise au point : -5 +2 dioptries

amplitude balayage : ±120° gisement
+ 30° } site
- 20°

correction déversement image

vitesses 2 gammes 0,1 50 ₥/s continu
50 à 500 m/s paliers

alimentation : 200 W
115 V 400 Hz 30 VA *

Le viseur équipe un grand nombre d'hélicoptères légers : GAZELLE, BELL SIKORSKY, ou moins légers : SUPER-FRELON, 330, etc ... ou d'avions : ATLANTIC, ...

Plus récent, le viseur APX 397 associé à un localisateur infra-rouge, permet la conduite de tir semi-automatique du missile HOT (Euromissile).

La stabilisation de tous les viseurs que nous venons de citer est assurée par une suspension élastique et le miroir de tête qui est lié mécaniquement à un gyroscope classique dans le rapport 1 en gisement et dans le rapport 1/2 en site.

Le gyroscope étant relativement chargé, malgré des efforts particuliers pour alléger les miroirs (suivant un procédé mis au point par l'APX), la suspension doit être bien adaptée à l'environnement hélicoptère pour éviter la nutation du gyroscope (50 Hz pour le 260, 80 à 90 Hz pour les 334 et 397).

Les performances globales de pointage sont de l'ordre de 0,1 m̸ pour la configuration hélicoptère la plus favorable. La dégradation de cette performance pour des visées en évasive dépend d'autres paramètres que de la stabilisation proprement dite et notamment de la conception du système de compensation de la parallaxe : tout cela sort du cadre de notre exposé.

Nous avons volontairement distingué pointage et stabilisation.

Revenons sur cette notion, et essayons de fixer les ordres de grandeur.

Notons tout d'abord que la précision est un critère presque aussi mauvais pour qualifier une stabilisation que le pouvoir séparateur l'est pour décrire les qualités d'un instrument d'optique ; si la précision angulaire est fondamentale en effet pour certaines applications dans lesquelles le dispositif d'observation a en outre une mission de visée et/ou de guidage, elle peut être secondaire dans le cas où l'opérateur se consacre exclusivement à l'observation, sous réserve que les résidus aient des fréquences suffisamment basses.

STABILISATION ET FONCTION DE TRANSFERT

Les phénomènes liés à la détection sont complexes ; leur étude a mis en évidence le rôle très important des contrastes. Par ailleurs, la fonction de transfert de modulation (F.T.M.), qui donne l'atténuation du contraste en fonction de la fréquence spatiale est l'un des meilleurs critères permettant de définir les performances d'un instrument d'observation. La F.T.M. de la chaîne sera le produit de la F.T.M. de la chaîne statique par la F.T.M. de la fonction stabilisation.

Nous nous proposons donc d'étudier la F.T.M. de la fonction stabilisation. Pour cela, il suffit de calculer la répartition du signal correspondant à l'image d'un point, en limitant le temps à la durée d'intégration.

Celle-ci sera par définition le temps le plus grand pendant lequel la chaîne complète peut être considérée comme linéaire ; en fait, ce sera l'ensemble des espaces de temps le plus petit nécessaire pour donner deux "perceptions" différentes. Le signal correspondant à l'image d'un point est obtenu en intégrant les éclairements pendant cet espace de temps.

La F.T.M. en est la transformée de FOURIER.

On voit l'analogie entre les aberrations transversales d'un système optique et les résiduelles de stabilisation à fréquence élevée.

On retrouve également le fait que la dégradation des qualités du système décroit avec la fréquence dès que celle-ci est inférieure à l'inverse de la durée d'intégration. On est donc amené à déterminer avec précision le spectre des défauts de stabilisation.

REPONSE DE LA PLATE-FORME AUX SOLLICITATIONS EXTERIEURES

On peut procéder pour déterminer le filtrage que donne la stabilisation de la façon suivante :

- on relève les accélérations linéaires et angulaires en des points du véhicule voisins des fixations du dispositif d'observation ;
- on mesure par ailleurs le filtrage introduit par la stabilisation : pour cette opération, on place la plate-forme sur un support vibrant et on mesure les résidus en fonction de la fréquence à l'aide d'un autocollimateur électro-optique de bande passante suffisante;
- en faisant le produit des deux données précédentes, on obtient le spectre des défauts de stabilisation ;
- on en déduit la répartition des éclairements correspondant à l'image d'un point (la densité d'éclairement étant inversement proportionnelle à la vitesse), et la fonction de transfert par transformée de FOURIER.

Il sera en particulier intéressant de calculer la F.T.M. au voisinage de la fréquence limite de l'ensemble de la chaîne qui n'est pas nécessairement celle du dispositif d'observation.

INFLUENCE D'UNE RESIDUELLE A FREQUENCE ELEVEE

Pour préciser les ordres de grandeur, on a pris le cas d'une résiduelle à une fréquence nettement supérieure à l'inverse de la durée d'intégration.

On montre que la F.T.M. était égale à la fonction de BESSEL J_o (Z).

En effet, le résidu sinusoïdal peut être mis sous la forme :

$$x = x_o \sin \omega t$$

Les photons se répartissent dans l'image de manière inversement proportionnelle à la vitesse :

$$\frac{dq}{dx} = \frac{1}{\pi x_o} \frac{1}{\sqrt{1 - \left(\frac{x}{x_o}\right)^2}}$$

La MTBF est la transformée de FOURIER de l'image d'un point :

$$F(Z) = \frac{1}{\pi} \int_{-1}^{+1} \frac{\cos 2x}{\sqrt{1 - x^2}} dx = J_o (Z)$$

Pour $Z = \pi$, ce qui correspond à une période spatiale égale à l'amplitude crête à crête du défaut de stabilisation, le contraste est inversé et égal à -0,3.

Le contraste est nul pour Z = 2,4.

Il est seulement égal à 0,5 pour Z = 1,5
et à 0,9 pour Z = 0,6

Ce dernier cas correspond à une période spatiale cinq fois plus grande que l'amplitude du défaut de stabilisation :

Pour éviter toute perte de contraste supérieure à 10 %, l'amplitude d'un résidu sinusoïdal ne doit pas être supérieure au cinquième de la résolution de la chaîne statique.

Bien entendu des résidus plus importants pourraient être tolérés si leur fréquence était suffisamment basse par rapport à la cadence image.

VIDICON ET CAPTEUR PONCTUEL

Dans le cas d'un vidicon, la durée d'intégration est sensiblement l'inverse de la cadence image. Elle est, en général, du même ordre de grandeur que la durée d'intégration de l'oeil (0,05 à 0,1 seconde).

Dans le cas d'une caméra utilisant un capteur unique et un balayage, la durée d'intégration est égale au produit de l'inverse de la cadence image par le rapport entre l'angle solide du capteur et le champ total.

Elle est donc beaucoup plus petite que dans le cas précédent. De ce fait, les résiduelles de fréquences basses et moyennes auront une influence réduite sur la fonction de transfert au niveau de la formation d'une image unique.

Mais il apparait une distorsion fine dont l'amplitude est égale à celle des résiduelles et comme ces distorsions ne sont pas en phase d'une image à l'autre et que plusieurs images sont en général utilisées pour constituer ce que nous avons appelé une "perception", l'ordre de grandeur de la baisse de contraste au niveau de la chaîne complète sera finalement le même.

Ces considérations font néanmoins apparaître l'intérêt qu'il pourrait y avoir à mettre en mémoire une image et à la visualiser pendant un temps nettement plus long que celui qui a été nécessaire à sa formation, pour compenser les imperfections d'une stabilisation, les petites distorsions dans l'image pouvant être moins gênantes vis-à-vis de la détection qu'un abaissement de la F.T.M.

Le développement des capteurs fonctionnant dans l'infra-rouge, qui nécessitent des pupilles de plus grandes dimensions, à conduit la SFIM à étudier et à réaliser des techniques de stabilisation différentes ainsi que les capteurs les mieux adaptés à ces problèmes.

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Sous l'égide du STAé, la SFIM développe actuellement un système électro-optique comprenant une caméra de télévision équipée d'une focale très longue, un télémètre laser et une lunette thermique à deux champs. Les essais préliminaires effectués dans le cadre d'une étude de faisabilité d'une conduite de tir 8-13 , menée sous l'égide du STET, et au cours d'essais en vol en 1977 et 1978 sur Alouette III ont montré tout l'intérêt de cette formule.

La stabilisation fine des différentes voies est assurée par des miroirs asservis. Les performances obtenues au cours des essais sont compatibles avec la précision demandée. Les solutions électroniques de compensation des vibrations permettent de replacer les photons reçus à leur juste position dans l'image, mais n'assurent pas que tous les points du champ soient explorés pendant des temps égaux.

La SFIM a développé également les moyens de simulation et de test qui permettent de prévoir et de contrôler les performances d'un système dans l'environnement réel prévu pour son utilisation.

AN ADVANCED GUIDANCE AND CONTROL SYSTEM FOR RESCUE HELICOPTERS

by

Kenneth W. McElreath
Flight Control Systems Engineer
Collins Government Avionics Division
Rockwell International
400 Collins Road N.E.
Cedar Rapids, Iowa
52406
United States of America

INTRODUCTION

In spite of many efforts to extend the usefulness of helicopters into night and adverse weather operations, they are still not fully effective in other than visual conditions. As new developments in navigation, guidance, controls, displays, and sensors become available, each one supplies an increment of capability to the helicopter pilot.

However, the full effect of these developments in achieving the helicopter rescue potential will be achieved only by careful integration of avionics functions into a total system. This avionics system must particularly consider the pilot's active role as the mission manager.

Collins Government Avionics Division has formulated an integrated Guidance and Control System for rescue helicopters. It comprises the following primary elements, along with other supporting sensors and displays:

(1) A digital mission computer capable of precise positioning and three-dimensional navigation.

(2) A three-axis flight director to follow a computed navigation profile, execute an approach and transition to hover, and maintain a stabilized hover.

(3) A four-axis fail-passive automatic flight control system for improved manual control as well as coupled flight director operation.

(4) An electronic multifunction flight situation display complemented with an electromechanical attitude and steering command display.

After identifying the tasks in accomplishing a search and rescue mission, Rockwell-Collins established guidelines and priorities for designing and integrating the system elements. This paper relates the development of the avionics system architecture and then describes those features of each element which contribute to the adverse weather capability of the integrated system.

I. GUIDANCE AND CONTROL IN THE RESCUE MISSION

In a typical maritime search and rescue mission, the search phase consists of visual surface scanning and the use of electronic sensors; for example, radio homing, search radar, or FLIR, to locate the rescue target. The guidance and control task in this phase is to conduct a search pattern at constant altitude and moderate speed -- a task which may be accomplished with conventional, even fixed-wing derived, equipment.

Once the target is located the rescue phase commences. If visibility is limited (at night or in adverse weather) then the dependency of the pilot upon the guidance and control system increases, for the following reasons:

(1) An approach to a hover at a rescue point near the surface demands a control precision several times greater than that for the search phase.

(2) Changes in aircraft handling qualities, stability and control cross-coupling, which occur as airspeed decreases, force the pilot and the control system to adapt quickly during the approach to a hover. Moreover, the pilot workload to maintain the desired flight path rises significantly.

(3) The proximity of the aircraft to the surface reduces the safety margin for pilot errors or equipment malfunctions.

(4) Increased crew activity during the rescue itself often results in only one pilot controlling the aircraft.

(5) Inadequate low speed air data or ground referenced velocity data confounds pilot attempts to maintain safe path and speed control near a hover.

These problems are inherent in both the rescue mission and the helicopter vehicles.

In developing concepts and systems to overcome these problems, it is advantageous to divide the rescue phase into the guidance and control tasks which the pilot must accomplish. For a typical mission flown at night or in instrument conditions, the pilot must accomplish the following:

(1) Plan the descent profile to a point near the surface downwind of and facing the target.

(2) Follow this descent profile, which may involve a steep path angle at low speeds, and decelerate to a hover.

(3) Transition from instrument to visual hover reference near the target, although at night minimal visual position cues, and no visual velocity, altitude, or attitude cues may be available.

(4) Maintain a precise hover referenced to a fixed or moving target or vessel.

(5) Execute a safe transition and departure from the rescue site.

II. AVIONICS SYSTEM OPERATIONAL REQUIREMENTS

Performing these rescue mission tasks in adverse environments imposes a set of severe operational requirements upon the helicopter avionics system design. Properly identifying these requirements and formulating a system to satisfy them are the tasks of the Avionics System Integrator. In establishing requirements and priorities for the helicopter rescue avionics, Rockwell-Collins considers the interaction with and support to the pilot (the mission manager) to be the most important element.

First, the avionics must be manageable by only one pilot, since the other pilot is often involved in other rescue duties. Single pilot operation thus becomes a reality. This fact dictates that all system functions must be accessible to either pilot in a two-pilot aircraft. Also, the pilot workload must be low so that he can monitor the critical automatic functions as well as make appropriate manual inputs. Further, the system must respond gracefully to failures, allowing the pilot to manually assume partial control at any time during a maneuver. This requirement emphasizes the importance of proper flight displays to enhance the pilot's control and confidence during rescue maneuvers.

Second, the system must be safe, so that no single failure can have catastrophic consequences. This requirement becomes most demanding in a low altitude hover. Further, any failures or limitations must be made obvious to the pilot so that he can reconfigure the system and minimize adverse effects.

Third, the system must be capable of continuing the mission despite failures, with high probability of mission success. The urgent nature of rescue operations dictates this requirement, and although it could be satisfied with sufficient avionics duplication, careful consideration of complementary redundancy affords a more satisfactory solution.

Fourth, the system must be constrained to stringent size and weight specifications. There are several reasons for these restrictions, stemming directly from the rescue aircraft and its mission application. Cockpit visibility forward and downward is essential during a rescue and it directly impacts the allocation of instrument panel and side panel space to avionics equipment. In addition, the mission requirements to hover for extended periods, to pick up as many survivors as possible, and to return to a shore base or vessel cause the avionics system weight allocation to be slashed to a minimum, in direct conflict with the mission's required avionics capabilities. Thus, the job of the avionics integrator/implementor is a continual balance of functional integration; for example, multifunction displays, versus survivability and mission success, versus minimal size and weight allocation.

III. PHILOSOPHY OF THE SYSTEM SOLUTION

Based upon the mission and the operational requirements for the rescue avionics, Rockwell-Collins formulated a system concept which would satisfy these requirements, providing long-sought adverse weather search and rescue capability. In order to accomplish this goal all of the areas of current deficiencies had to be addressed. Such issues as adequate sensors and displays, functional hardware/software integration, graceful degradation with failures, proper pilot awareness and interface, workload, aircraft stability, safety, and system performance had to be resolved before a successful integrated avionics system could result.

Rockwell-Collins established guidelines to control the relative priorities of the overall system design and to ensure that the mission requirements would be met. These guidelines grew out of Rockwell-Collins extensive development work with the US Air Force and Army, the experience gained on the US Coast Guard HU-25A (Medium Range Surveillance) aircraft, direct in-house research, and communication with the operators, particularly in the Navy and Air Force, regarding rescue mission requirements and difficulties. Later discussion will show how these guidelines translated into specific implementations, using state-of-the-art techniques.

First, multifunction displays were used to the maximum extent practicable to reduce and simplify the instrument panel. The CRT was one choice for such implementations.

Second, the system was integrated for consistency and ease of management of navigation, guidance, control, and display functions. Sensors, radio aids, and the pilot selections of navigation displays and guidance modes should also be complementary and appropriate to the piloting task.

Third, guidelines were established for independence of functions, required types and levels of redundancy, and pilot awareness of system status to achieve safety and mission reliability by graceful degradation of functions following failures. Critical flight functions were made directly redundant through duality. Other areas were protected by complementary functions which gave the same reliability as direct duality but provided expanded avionics capability. For graceful degradation, certain functions were kept independent to reduce the impact of failures. Thus, raw navigation data displays were maintained in addition to the integrated Mission Computer displays, and in the Automatic Flight Control System (AFCS) independent axis engagement and monitoring limited the loss of automatic capability with failures.

The fourth guideline was that the operation and maintenance of the system should be simple and expeditious. On-board fault isolation and ease of replacement of failed units ensure that the helicopter is responsive to immediate rescue demands. In addition, sensors and other system elements were specifically chosen to minimize start-up time.

The Flight Guidance and Control System itself was divided into the following subsystems: The Flight Director System, the Automatic Flight Control System (AFCS), and the Flight Display System, as well as those elements in common with other systems, as mentioned previously.

In the following sections, each of the subsystems which comprise the Flight Guidance and Control System will be examined in more detail. The prominent features of each system which directly benefit the rescue mission will be emphasized along with the complementary interaction among systems. The Guidance and Control System provides the pilot a complete hierarchy of information and input capability, including complete raw navigation data displays, computed steering commands, and stability augmentation or full automatic coupled operation throughout the rescue operation. Thus he can choose to monitor or be actively involved at any point in the system.

Four primary points of interface allow the pilot to make inputs to the operation of the system. These are identified in figure 1. The first input is one of directing the mission flight plan through the CDU's. Certain features of the flight planning and profile capability of the Mission Computer will be described later. Basically, it relieves the pilot of the burden of coordinating navigation radios, courses, and procedures, and allows him to simply tell the system where he wants to go and what he wants to do at any given point.

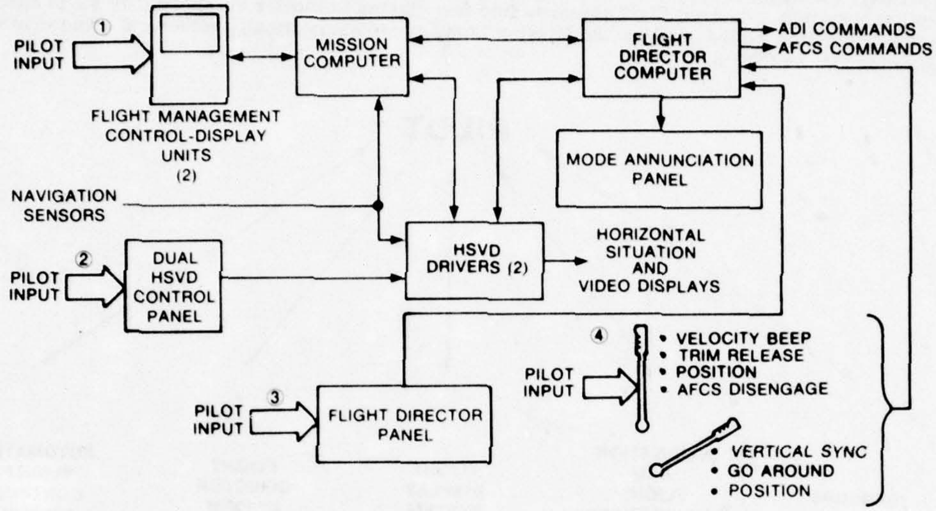

Figure 1. Pilot Interfaces With Flight Guidance and Control System.

The second pilot input point is the Horizontal Situation and Video Display System, or HSVD (an element of the Flight Display System). Here the pilot controls the tactical situation display and assesses his position in relation to the flight plan he has established.

The third point of pilot inputs is the Flight Director Mode Panel, where he may select steering modes to follow a given plan or profile. These modes are linked to the HSVD so that consistency of operation is assured; that is, the flight director can follow only navigation data which is also displayed on the pilot's HSVD.

The fourth area of pilot inputs is through his flight controls. The control of certain flight director data, for example, airspeed or altitude command, and critical modes (go-around) is exercised without the pilot removing his hands from the cyclic and collective sticks. This has been done because it increases the pilot's effectiveness and confidence by making his inputs more natural under automatic operation and because it facilitates manual reversionary operation in the case of AFCS failures.

IV. FLIGHT GUIDANCE AND CONTROL SYSTEM IMPLEMENTATION

A. Overall Description

For improved manageability and pilot flexibility, the functions of the Flight Guidance and Control System have been divided as shown in figure 2. Although the system may be operated automatically, the independence of functions allows the pilot to manually assume control of any function or to supply the manual reversion to a failed element in any of the links.

The Flight Guidance and Control System comprises only a portion of the total integrated Avionics System.

Other major groups within the Avionics System are the Navigation System, the Communication System, the Radar System, and the Flight Management System (a central integrated avionics control system). The Flight Guidance and Control System shares many common elements with these systems, resulting in improved manageability and less hardware. In particular, the Flight Management System's Control Display Units (CDU's), the Mission Computer, and various sensor systems serve several purposes.

Many of the interfaces among systems, including the central integrated avionics control, are implemented by means of dual MIL-STD-1553A multiplex digital buses, as shown in figure 3. This architecture reduces the numbers of analog signal conversions and allows the pilot to manage almost the entire system from the multi-function CDU's. Two identical CDU's are in the cockpit so that the pilot and copilot can split their avionics control functions (for example, navigation and communication) and so that complete redundancy is afforded.

This organized approach to pilot interface permits a graceful degradation of mission capability with failures of any of the systems. The pilot experiences only a minor increase in workload to assume control of the failed function and can continue the mission rather than aborting it.

In all of the above interfaces, pilot manageability is paramount. Extensive human engineering efforts and analysis of rescue operations have been employed in the development of display formats, control algorithms, status feedback and pilot input implementations to ensure that the pilot would be supported and not subdued by the system.

B. Navigation and Flight Management Systems

The Navigation System and Flight Management System complement the Flight Guidance and Control System by providing the navigation radios, pilot interfaces (control-display) and navigation computations to generate the aircraft path and performance data for the search and rescue mission. The interfaces with the Flight Guidance and Control System are shown in figure 3.

The primary elements are the Mission Computer and the two multifunction CRT Control Display Units (CDU's). The pilots may insert desired flight plan data into the Mission Computer via these CDU's. In addition, they may tune navigation radios or allow the Mission Computer to automatically select and tune the navigation frequencies via the CDU's.

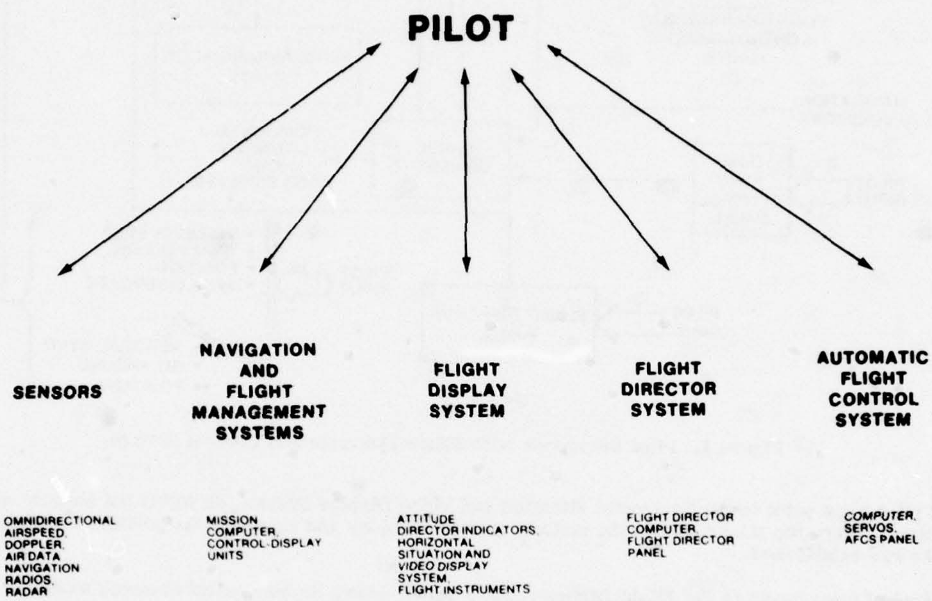

Figure 2. The Division of Flight Guidance and Control Systems.

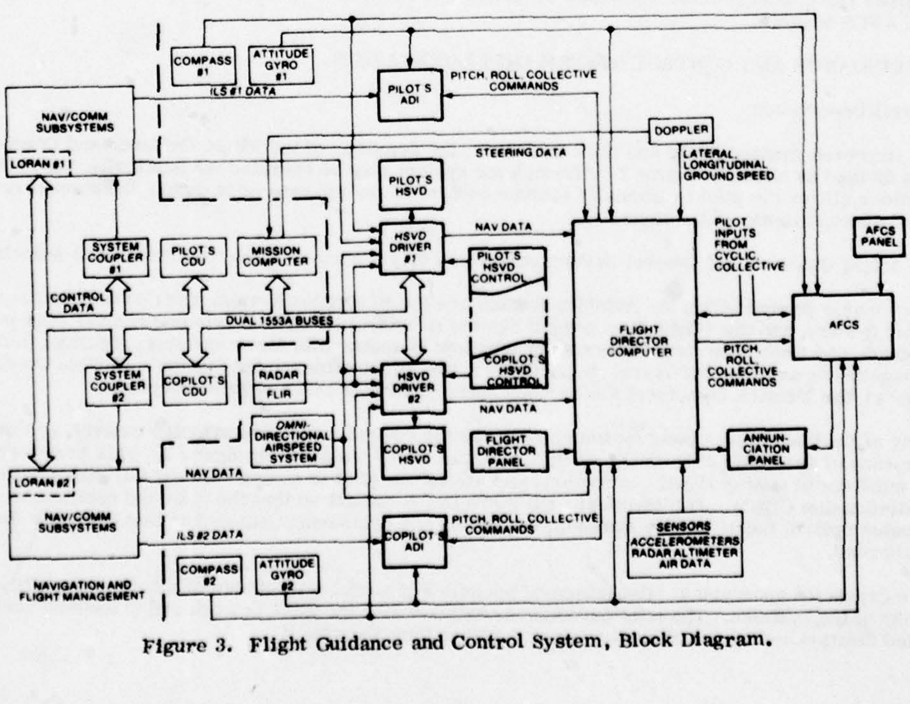

Figure 3. Flight Guidance and Control System, Block Diagram.

The Mission Computer is a 16-bit miniprocessor. In addition to navigation-related functions, it provides capability for minimum fuel cruise optimization, engine condition monitoring, and data link messages.

The Mission Computer has navigation data, including radio station sites and identifiers, waypoints, prestored patrol routes and various search patterns and other navigation algorithms stored in its memory data base. It uses Kalman filtering of multiple source data (loran, doppler, air data, VOR/TACAN) to determine a best estimate of the aircraft present position and velocity, generating path deviation and steering data for the flight director system from those estimates.

Several capabilities of the Mission Computer directly enhance performance of the search and rescue mission. The first of these is the control and tuning of all available navigation radios, including the loran receivers, automatically via the 1553A buses. This feature permits the continual computer selection of stations for the best possible position estimating, based on geometry and signal quality. A second feature is that of having prestored search patterns which the pilot can call up with the push of a button. Ladder, sector, and expanding square patterns are available, as well as conventional holding patterns. An example of how the pilot might call up a sector search pattern is illustrated in figure 4. After selecting the sector search pattern on the CDU, the pilot may choose the leg lengths and sector angles, typing them into the computer on the CDU keyboard. Upon inserting this pattern in his flight plan, the Mission Computer generates situation display and command steering signals and sends them to the flight director system for manual following or automatic AFCS coupling. The third operational feature is the ability to automatically compute a minimum time rendezvous with a moving object, based upon a position and velocity report. The fourth and most significant feature for the rescue operation is the capability to generate a three-dimensional descent path to a point just above a surface target. This position of the target is fixed by either visual sighting, radar detection, or radio homing. One button push by the pilot as he overflies the target marks its position in the Mission Computer. The computer then programs a three-dimensional approach to hover path, illustrated in figure 5, based on estimated geographical position and radio altitude. The path consists of an outbound (downwind) course, followed by a procedure turn and a 5-degree synthetic glidepath approach to the hover point, slightly downwind of the marked target point. This path, in conjunction with the flight director approach and transition steering modes, allows an automatic or manual rescue approach to be flown without visual reference.

An auxiliary capability of the Mission Computer is that of performing a hover power assurance check based upon continuous engine data. This computation allows the pilot to verify that sufficient power is available for the hover conditions to be encountered.

C. Flight Director System

The Flight Director System generates steering commands for the entire search and rescue mission in three dimensions: cyclic pitch, cyclic roll, and collective. It consists of the flight director computer and the flight director panel (figure 6). The flight director computer was made analog; digital solutions are available and would have been used had the necessary sensors and transducers been available for ease of interfacing.

In developing the guidance algorithms, extensive use was made of exact nonlinear equations as well as linearized small perturbation simulations. Real time simulations in both instances encouraged consideration of actual pilot interaction and permitted engineering insight into the resulting effects and sensitivities of the guidance algorithms implementation. Rockwell-Collins has found that helicopter guidance algorithms especially are enhanced by proper use of high quality accelerometers in all three axes to sense path motions. They not only provide true feedback of the path response, allowing high system gains without ill effects, but they are responsive to air mass variations (wind shears), permitting greater immunity than more conventional guidance algorithms. Rockwell-Collins is applying these principles also to path tracking for fixed-wing aircraft to improve their windshear immunity.

In addition to being displayed to the pilot on the Attitude Director Indicator the flight director commands may be coupled to the Automatic Flight Control System (AFCS) for automatic execution if the pilot desires. The flight director steering commands are complemented by yaw computations for turn coordination or heading hold in the AFCS. The flight director system modes are designed to adapt to the pilot's mission task and maneuvering requirements. A simplified mode chart is shown in figures 7 and 8. Most of the modes are conventional and self-explanatory. However, three of the modes afford guidance for the difficult rescue operation, particularly to transition the helicopter from the higher speed cruise search flight regime down to a hover near the rescue target. These three modes are called the approach mode (APPR), the airspeed-vertical speed mode (IAS-VS), and the transition-hover mode (T-HOV).

The APPR mode is the one which the pilot selects to provide steering guidance to follow the 3-D approach to hover Mission Computer profile described previously. This mode computes lateral guidance to maintain the lateral path throughout the maneuver. In addition, the collective steering computations cause the aircraft to capture and track the approach to hover descent path. Airspeed hold is automatically selected as the cyclic pitch mode at vertical path capture. In the airspeed hold mode, the pilot may readily modify the airspeed reference, using a beep slew switch on his cyclic stick. The airspeed reference value is displayed on his Horizontal Situation and Video Display.

The APPR mode is intended to be selected in conjunction with the transition-hover (T-HOV) mode. As shown in figure 9, the T-HOV mode remains in an armed condition until the aircraft passes 100 feet radio altitude. At that point, it causes the APPR mode to drop. Then the aircraft is guided in three dimensions through a maneuver which captures 50 feet radio altitude and decelerates to zero lateral and longitudinal doppler groundspeed. For profile safety, the aircraft is commanded to maintain 50 knots until it descends below 60 feet radio altitude.

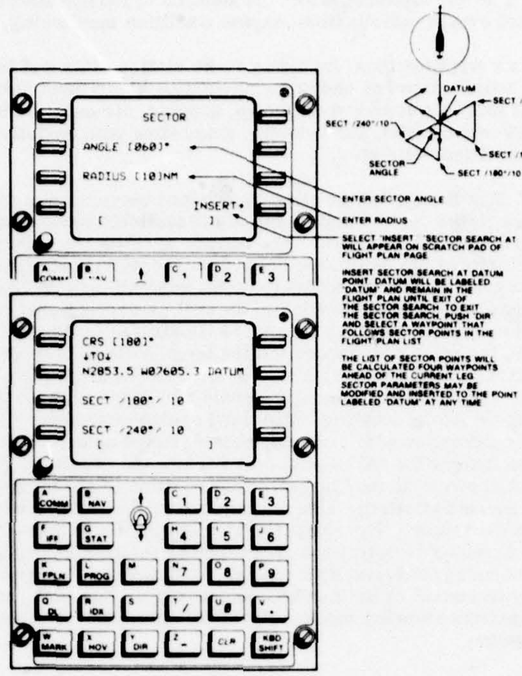

Figure 4. Sector Search Entry in Flight Plan.

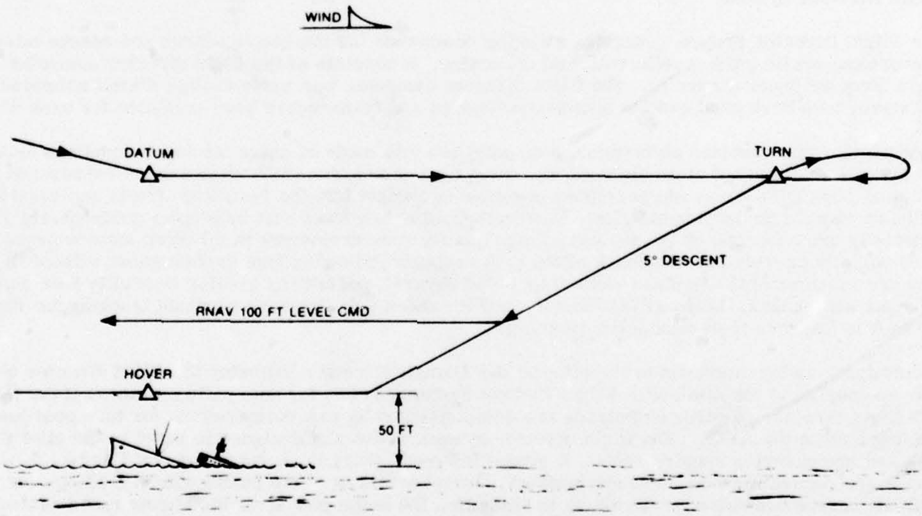

Figure 5. RNAV Approach to Hover Flight Profile.

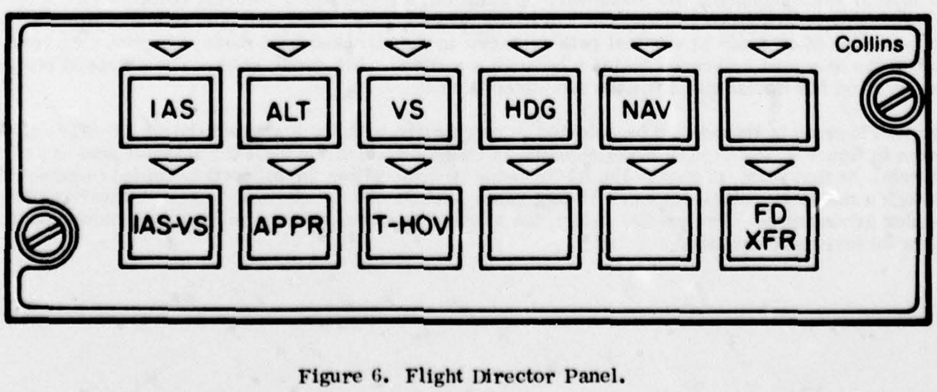

Figure 6. Flight Director Panel.

SELECTED MODE	NO MODE (FLIGHT DIRECTOR OFF)	HEADING	NAVIGATION (VOR, LOC, TACAN, OR RNAV)		APPROACH (VOR, ILS, TACAN, OR RNAV/VNAV)				
					LATERAL			VERTICAL	
SUBMODE CONDITION	AFCS IN BASIC MODE		BEFORE CAPTURE	AFTER CAPTURE	BEFORE CAPTURE	AFTER CAPTURE	BACK COURSE LOCALIZER (AFTER CAPT)	BEFORE VERTICAL CAPTURE	AFTER VERTICAL CAPTURE
ANNUNCIA-TIONS		HDG	HDG, NAV (ARM)	NAV (CAPT)	HDG, APPR (ARM)	APPR (CAPT)	BK CRS, APPR (CAPT)	GS (ARM) OR VNAV (ARM)	GS (CAPT) OR VNAV (CAPT)
CYCLIC ROLL FUNCTION	STEERING BAR OUT OF VIEW	SELECTED HEADING (FROM HSVD)		SELECTED V/L, TCN, OR RNAV COURSE	SELECTED HEADING	V/L, TCN, OR RNAV COURSE	LOCALIZER BACK COURSE	LATERAL CAPTURE INDEPENDENT OF VERTICAL CAPTURE	
CYCLIC PITCH FUNCTION	STEERING BAR OUT OF VIEW	NO MODE OR ALT HOLD OR IAS HOLD OR VS HOLD							IAS HOLD
COLLECTIVE FUNCTION	STEERING POINTER OUT OF VIEW	NO MODE OR VS HOLD (IAS/VS MODE)							GS OR VNAV
YAW FUNCTION	TURN COORDINATION (>20 KTS) HEADING HOLD (<20 KTS)								

Figure 7. Flight Director Mode Chart.

SELECTED MODE	TRANSITION TO HOVER				GO AROUND	ALTITUDE HOLD	VERTICAL SPEED HOLD	AIRSPEED HOLD	IAS/VS HOLD MODE	
		CAPTURE								
SUBMODE CONDITION	ARM	AFTER APPR (ILS OR RNAV)	AFTER IAS/VS	WITHOUT ARM OR AFTER PILOT SYNC/BEEP					SYNC VERT SPD <50 FPM	SYNC VERT SPD >50 FPM
ANNUNCIA-TIONS	IAS APPR (AND) VNAV OR GS OR IAS/VS T-HOV (ARM)	APPR, T-HOV HOV AUG (ARM)	T-HOV, HOV AUG (ARM)	T-HOV	GA	ALT	VS	IAS	IAS, VS	
CYCLIC ROLL FUNCTION	LOCALIZER OR RNAV COURSE		ZERO LATERAL SPEED	BEEPED LATERAL SPEED	WINGS LEVEL	NO MODE OR SELECTED LATERAL MODE		NO MODE OR SELECTED LATERAL MODE		
CYCLIC PITCH FUNCTION	IAS HOLD (OAS)	DECELERATE TO ZERO FORWARD SPEED		BEEPED FORWARD SPEED	FIXED PITCH	MAINTAIN SYNC BARO ALT	MAINTAIN SYNC VERTICAL SPEED	MAINTAIN SYNC OAS FORWARD SPEED		
COLLECTIVE FUNCTION	GS OR VNAV	CAPTURE 50 FEET RADIO ALT		MAINTAIN SYNC RADIO ALT	200 FPM CLIMB	NO MODE		NO MODE OR APPR (GS OR VNAV)	MAINTAIN SYNC BARO ALTITUDE	MAINTAIN SYNC VERTICAL SPEED
YAW FUNCTION	TURN COORD (>20 KTS) HEADING HOLD (<20 KTS)	HEADING HOLD (ALL SPEEDS)			TURN COORDINATION (>20 KTS) HEADING HOLD (<20 KTS)			TURN COORDINATION (>20 KTS) HEADING HOLD (<20 KTS)		

Figure 8. Flight Director Mode Chart.

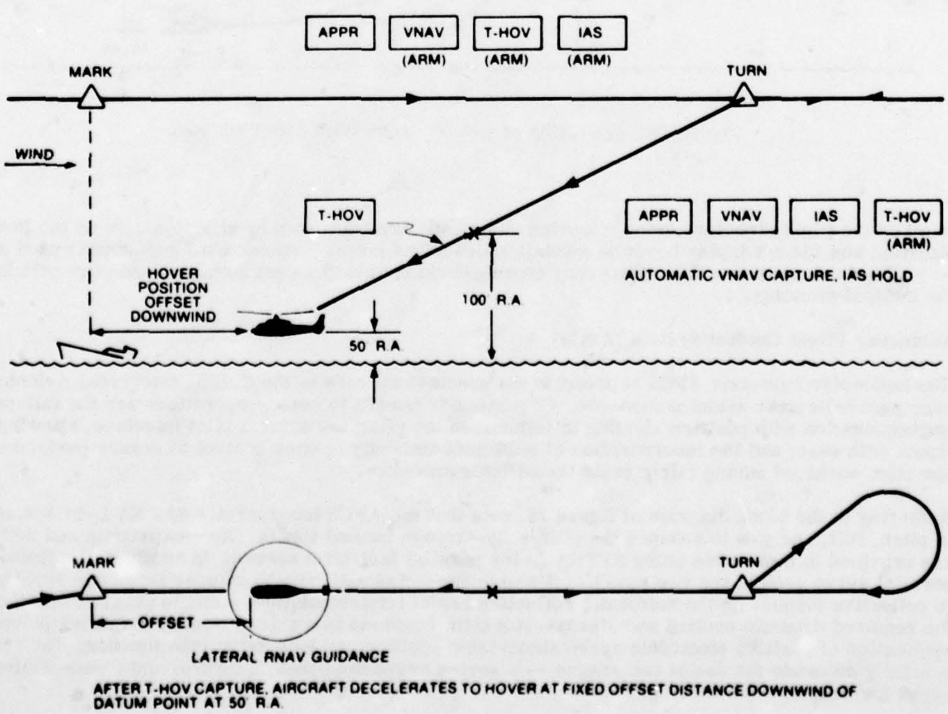

Figure 9. Operation of T-HOV Mode With RNAV Approach to Hover.

During or following the transition maneuver, the pilot may increment the final lateral or longitudinal speed by moving the cyclic beep slew switch. (The same one used for airspeed slewing during the approach.) The resultant hover velocity is then maintained as long as the pilot desires. If he wishes to modify the hover altitude, he simply raises or lowers the aircraft using his collective control and presses a vertical synchronize button on the grip. This action synchronizes the collective steering commands to the new altitude reference. If these flight director commands have been coupled into the AFCS, the pilot has little more to do than to monitor the system's performance and make occasional inputs to maintain his position above the rescue target, which may be at rest or moving in the water.

The T-HOV mode may also be selected without prior selection of another mode. In this case the flight director immediately commands a deceleration to zero longitudinal and lateral doppler ground speed while maintaining the existing radio altitude. As before, the pilot may make inputs into any of the axes of control by his cyclic beep switch or collective vertical (altitude) sync button. Heading control is exercised through the AFCS by inputs to the yaw pedals and is described in detail later.

If a ground speed sensor (doppler) failure occurs, hover augmentation through accelerometers is still available to provide a reversionary short-term inertial stabilization in the T-HOV mode.

The third unique flight director mode, the IAS-VS mode, operates much like the APPR mode, except that it is referenced to air mass parameters instead of fixed approach path. This mode is useful for rescue approaches at night when the target may be visible (illuminated) but no visual altitude, attitude, or velocity cues are available to the pilot. By selecting the IAS-VS mode, the pilot may stabilize the descent path of the helicopter, since both forward speed and descent rate are held constant. With his workload thus relieved, the pilot can devote his attention to the target and its apparent motion in the windshield. Changes in descent rate are effected by raising or lowering the collective lever to achieve a new rate of descent and pressing the vertical sync button on the grip. The collective commands will then maintain the new reference. Likewise, his forward speed may be changed by beep inputs on the cyclic slew switch. As with the APPR mode, the IAS-VS mode allows the T-HOV mode to be armed above 100 feet radio altitude as shown in figure 10. When the aircraft reaches 100 feet, the T-HOV mode causes the IAS-VS mode to drop and transitions the aircraft to a hover as described in its operation with the APPR mode.

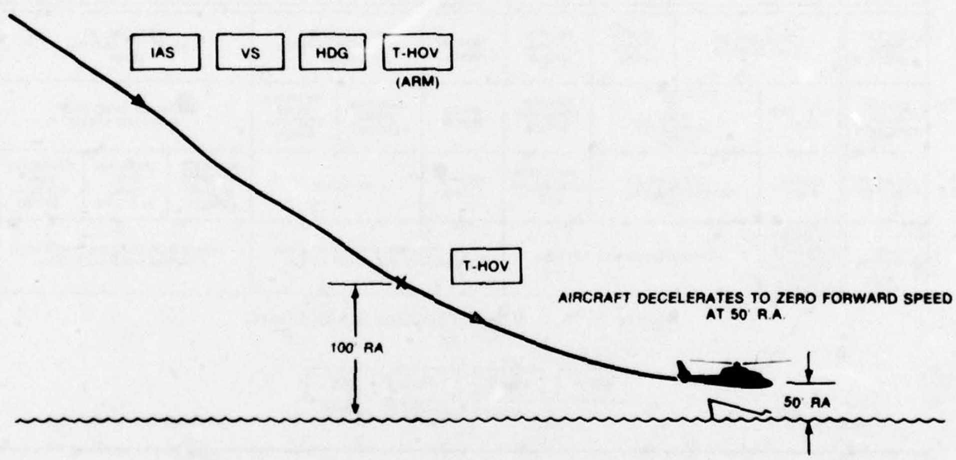

Figure 10. Operation of T-HOV Mode With IAS-VS Mode.

In all of the flight director modes, steering commands are supported by situation data on the Horizontal Situation and Video Display hover presentation described later. Lateral and longitudinal speed errors as well as path deviation data allow easy cross-checking of system performance and pilot orientation with the tactical situation.

D. Automatic Flight Control System (AFCS)

The helicopter four-axis AFCS represents the greatest advance in the Collins Integrated Avionics System over past helicopter avionics systems. Of particular benefit to rescue operations are the fail-passive implementation with positive alerting of failures to the pilot; the natural pilot interface, allowing manual inputs with ease; and the incorporation of sufficient authority in each control to ensure performance with low pilot workload during fairly rapid transition maneuvers.

Referring to the block diagram of figure 11, note that the AFCS incorporates SFENA L-24 series servos in pitch, roll, and yaw to enhance the pilot's fly-through manual inputs. Automatic trim and artificial feel are provided in these axes using SFENA L-109 parallel feel/trim servos. In addition, the Rockwell-Collins parallel servo used in the yaw axis has dynamic range and authority to counter large and rapid variations in collective torque. In the automatic collective assist implementation, a single parallel servo combines the required dynamic control and steady-state trim functions in a unique Rockwell-Collins patented application of positive electronic servo short-term position and long-term rate limiting. Such capability normally demands the use of two servos -- a series servo for dynamic control and a rate-limited parallel servo for trim.

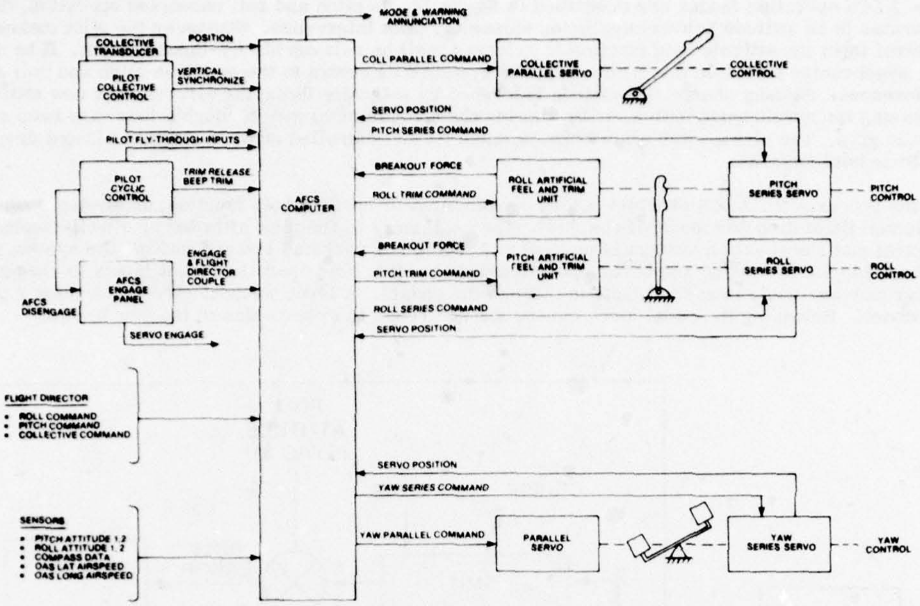

Figure 11. Automatic Flight Control System.

All of the primary servos (series and parallel) and trim functions are driven from dual computations, voted and compared within the AFCS computer. Thus the entire system is fail passive; that is, no failure can result in a servo hardover, to an FAA certified probability of 1 x 10^{-9} (extremely remote). Furthermore, even "soft" failures to zero are detected with the same probability and annunciated to the pilot. If not detected, these are as hazardous to a helicopter hovering 50 feet off the water as hardovers are. As an example of the fail-passive AFCS implementation, figure 12 shows the dual channel computations for the yaw series servos with the associated voters and comparators. The voters nullify any immediate effect of a disagreement between channels and allow the comparator time constants to be set up to 3 seconds. The comparators then activate the automatic warning and disconnect function. Thus, nuisance disconnects and warnings are minimized. Note that even a gyro failure cannot cause a system hardover, that failures are detected and voted out even before any control movement results, and that hardover commands coming from the flight director system, when coupled, cannot exceed the attitude limit for that mode (cruise or hover) (figure 13).

Careful consideration was given to the pilot interactions and operation of the AFCS. Ideally the AFCS should be a full time pilot assist system. Control of the aircraft with the AFCS should be more natural and precise with less workload than without it. Pilot inputs should be facilitated, not discouraged.

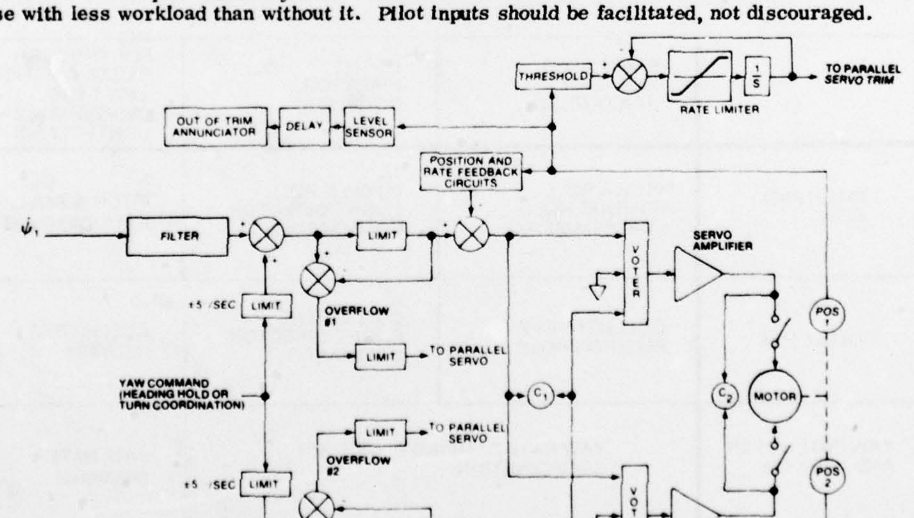

Figure 12. Fail Passive Yaw Computation and Series Servo.

The AFCS operation modes are described in figure 14. In pitch and roll uncoupled operation, the AFCS operates in an attitude hold configuration unless the pilot intervenes. Whenever the pilot makes a manual control input the attitude hold function is released until he relaxes his fly-through input. If he does not press the synchronize button on the cyclic grip, the system will return to the previous pitch and roll attitude references. He may change the attitude reference by manually flying the aircraft to a new attitude and pressing the synchronize button, or by slewing the attitude reference through a four-way beep switch on the cyclic grip. The slew switch allows him to make small controlled attitude changes without dropping the attitude hold function.

In the yaw axis the AFCS provides a turn coordination or heading hold function, depending upon airspeed and selected flight director mode (if coupled). The assistance to the pilot afforded by a well-mechanized yaw control and stabilization system is critical to keeping his workload low and making the system performance acceptable for low speed and hover operations. In heading hold operation, pilot inputs to change the heading reference are made in normal fashion through the pedals. A force washout circuit provides a pedal force feedback. Releasing the pedal force causes the reference to synchronize to the new heading.

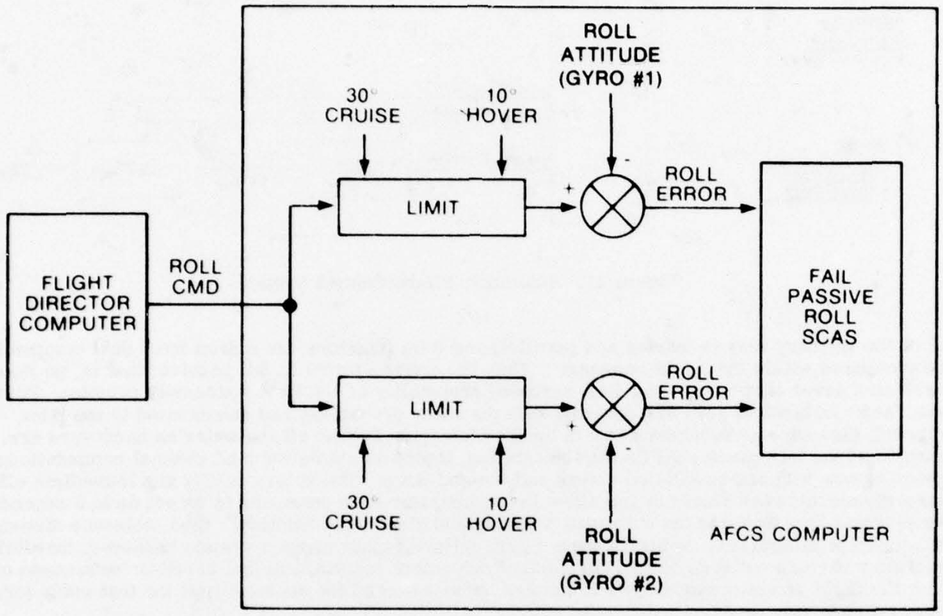

Figure 13. Roll Flight Director Command Interface With AFCS.

	BASIC AFCS ENGAGED (UNCOUPLED)	FLIGHT DIRECTOR COUPLED	FLY THROUGH CYCLIC/VERTICAL/ YAW SYNC (INDIVIDUALLY CONTROLLED)
PITCH AND ROLL	PITCH & ROLL ATTITUDE HOLD + RATE DAMPING	PITCH & ROLL FLIGHT DIRECTOR COMMANDS	PITCH & ROLL RATE DAMPING
COLLECTIVE	COLLECTIVE LEVER POSITION HOLD	COLLECTIVE FLIGHT DIRECTOR COMMANDS	COLLECTIVE RELEASE
YAW (NOT HOVER AND >20 KTS)	YAW RATE DAMPING PLUS TURN COORDINATION		YAW RATE DAMPING
YAW (HOVER OR <20 KTS)	YAW RATE DAMPING PLUS HEADING HOLD		YAW RATE DAMPING

Figure 14. AFCS Mode Chart.

The automatic collective assist, when the AFCS is not coupled, holds the collective lever in a fixed position without creeping. To move it the pilot presses the vertical synchronize button and moves the lever with low frictional forces. Releasing the button causes the lever to lock in the new position.

Coupled AFCS operation consists of the pitch, roll, and collective following the flight director commands. In each case, pilot fly-through inputs cause the AFCS to temporarily cease following the flight director steering commands, until the pilot releases his input.

For the transition and hover tasks of the rescue mission, it is imperative that sufficient automatic control authority be provided. Otherwise, performance will suffer. The fail-passive AFCS allows "inner loop" servo authority to be incorporated which exceeds that allowable in a single computation, fail-safe system. Furthermore, the use of a dynamic parallel servo in yaw to complement the series servo overcomes the greatest weakness in helicopter low speed stability. Large and rapid changes in collective torque or winds near the surface and other vessels result in poor heading performance in conventional systems. The full-authority Rockwell-Collins yaw system overcomes this difficulty and relieves all cross-coupling pilot workload in maintaining the heading in or near a hover. However, most of the high-frequency inputs are accomplished through the series servo to minimize the pedal motion feedback to the pilot.

E. Flight Display System

Integration of raw situation and computed steering data into meaningful display formats is a key element in achieving the maximum pilot confidence and capability while managing the guidance and control of the rescue operation. The two distinctive elements of the Flight Display System are the Horizontal Situation and Video Display (HSVD) and the Attitude Director Indicator (ADI). Together they enable the pilot to make assessments of task performance, safety margins, and required inputs for mission success.

The HSVD serves as the focal point for three functions in the guidance and control system.

(1) Pilot selection of navigation signals for direct situation display as well as inputs to the flight director system. Pilot selections of navigation data source, course, and heading are accomplished on the HSVD Control Panel, shown in figure 15. These selections are displayed and transmitted to the flight director system. The flight director generates commands to track only the navigation data displayed on the HSVD. As an example, if the pilot selects the TACAN navigation source for his HSVD display then the flight director NAV mode will capture and track only that TACAN course and not a VOR or RNAV course.

(2) Signal conversion and processing of data used by the HSVD and also by the Mission Computer. Analog signals representing airspeed, radio altitude, heading, and radio deviations and bearings are managed by the HSVD driver and converted into MIL-STD-1553A format for use in the Mission Computer, as shown in figure 3. The number of conversions is thus kept to a minimum and the status of the signals is obvious to the pilot on his display.

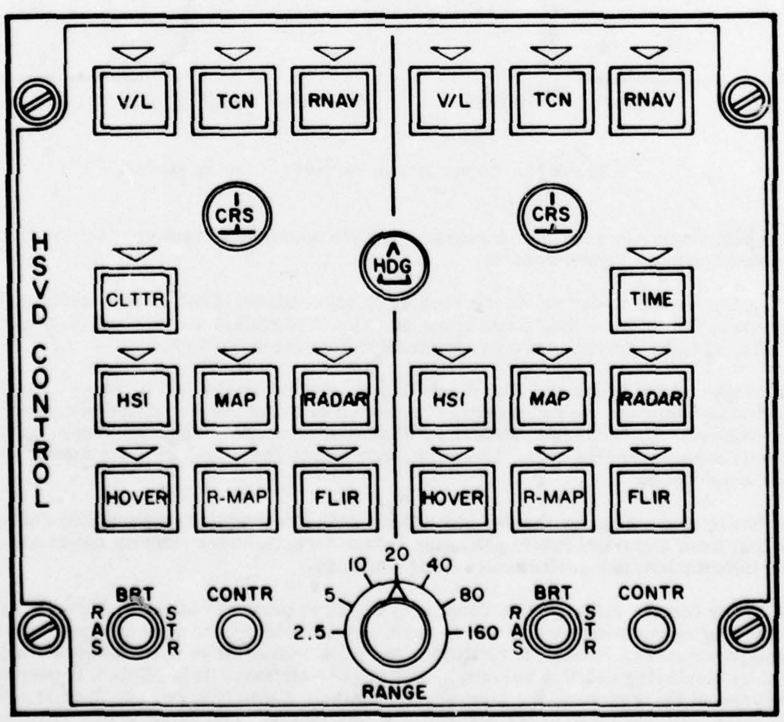

Figure 15. HSVD Control Panel.

(3) Centralized management of the tactical mission display according to the pilot's requirements. From the control panel shown in figure 15, the following modes may be selected for viewing:

Horizontal Situation Indicator (HSI)

Tactical Map

Weather/Search Radar

Radar + Map (Radarmap)

Hover

Forward Looking Infrared (FLIR)

These modes display data pertinent to the situation at hand and give the pilot display flexibility in a single instrument. Representative display formats of the modes are shown in figure 16.

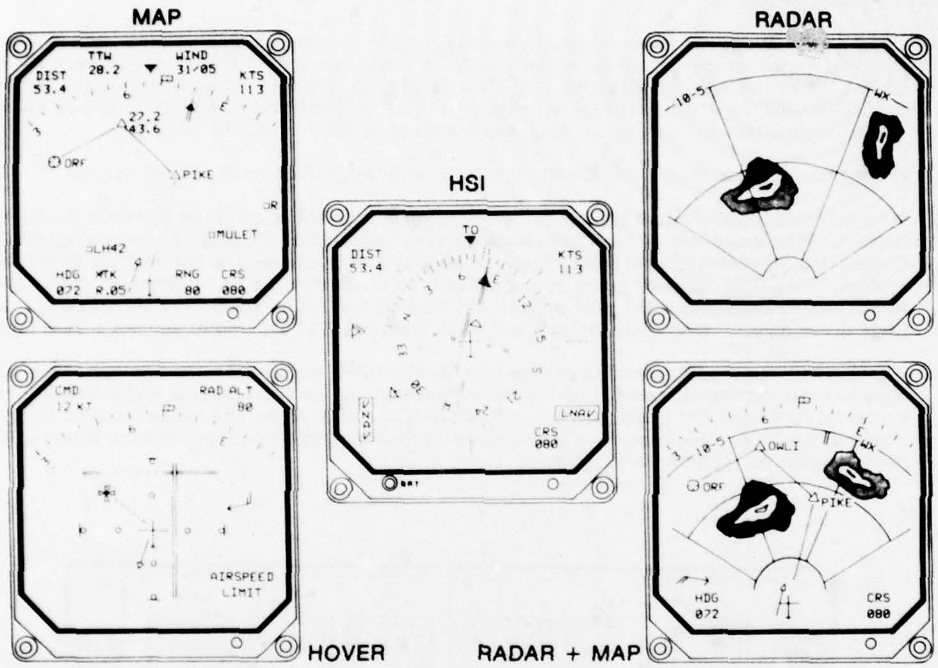

Figure 16. Composite of the HSVD Display Modes.

The FLIR display format (not shown) is a growth option dependent upon the particular equipment interface and customer requirements.

The HSI mode and Radar mode are similar to conventional displays with additional numeric data. The Map mode and Radar + Map mode allow the pilot to visualize his planned flight path in relation to radar targets, weather activity, and other geographic fixes or waypoints.

The ranges of the Radar and Map displays are selected on the HSVD Control Panel (figure 15). Consistency between the radar generated raster data and the navigation computer stroke map data is thus assured. The combined Radarmap display mode results from the stroke writing being done between scans of raster data. The pilots may select these and all other modes without regard to each other's selections.

The Hover mode displays the omnidirectional low airspeed vector (magnitude and direction), relative position from a marked datum point (the rescue target), hover velocity commands, computed real time wind information, and performance limit warnings.

The Hover display mode was mechanized with three purposes in mind: First, it assists the pilot in monitoring an automatic transition to hover and in relating the system's performance to his desired hover parameters. Second, it facilitates the pilot's transfer to outside visual references at the rescue point by displaying relative velocity, position, and air mass data. Third, it permits him to manually intervene in the control of the aircraft and continue a specific hover task or to execute a safe departure, well apprised of his aircraft's safety margins and performance envelope.

Note in figure 16 that the velocity error and airspeed vector displays are predominant in the Hover mode. In hover operations, rate information is more valuable to the pilot than position data. It assists him in appraising his immediate situation and in making effective short-term control inputs. Because he can assess his position data more slowly and integrate it over time, the quality of position data does not need to be as high as that of the displayed rate information. In addition to the HSVD Hover mode data, the Mission Computer compares the omnidirectional airspeed vector with the aircraft safe flight envelope. Upon detecting an out-of-limit condition; i.e., excessive rearward/sideward airspeed, it flashes an "AIRSPEED LIMIT" Annunciation. On the ground, the Mission Computer compares the relative wind to the aircraft rotor engagement/disengagement envelope and flashes a "NO ROTOR ENGAGE" message when it is exceeded.

The second important element in the flight display system, the ADI, incorporates flight director steering data for pitch, roll, and collective, along with raw attitude, deviation, and radio altitude data (figure 17). Emphasis has been placed upon the display presentations to assure that the HSVD and the ADI present a consistent picture of complementary steering and situation data. Since their purpose is one of enhancing manual path control as well as pilot monitoring, the three cue (pitch, roll, collective) commands were implemented to be consistent with good piloting techniques. Thus, the control axes were decoupled as much as possible and optimized for the transition to powered lift flight at low speeds.

F. Sensors

The primary sensor systems which support the navigation, guidance and control systems are the air data system, heading system, vertical reference system, radar altimeter system, doppler, loran, and other navigation radios and homing equipment. The most interesting of these sensors is the omnidirectional airspeed system for airspeed data. The lack of omnidirectional airspeed data at low speeds has been a large single factor in preventing the use of the helicopter's tremendous flight capabilities to their full potential. Not only does the lack of data restrict the maneuvers which can be flown with confidence, but it is the cause of many accidents and aborted rescue attempts. Figure 18 shows a typical helicopter airspeed flight envelope and the poor coverage of data offered by pitot-static airspeed sensors. This figure illustrates one reason why helicopters have been restricted from low speed instrument flight for normal operations. Even automatic approach couplers do not permit the pilot to adequately monitor his safety envelope and thus are restricted in their usefulness.

Rockwell-Collins has selected the PACER Systems, Inc., LORAS (Low Range Airspeed) System for this application because of its proven performance, its advanced development status, and its ability to operate without rotor downwash, useful for shipboard launch and recovery.

The sensor element shown in figure 19 operates above the rotor mast. The sensor rotates, sensing differential pressure at the tips. This pressure difference is resolved into orthogonal components to produce lateral and longitudinal airspeed. The system's accuracy at low speeds is ±2 knots.

The airspeed data from this Omnidirectional Airspeed System (OAS) is displayed to the pilot as a velocity vector in the HSVD hover display mode, shown in figure 16. Further, it permits the navigation computer to calculate wind velocity information in real time and also display it in the HSVD. With these data the pilot

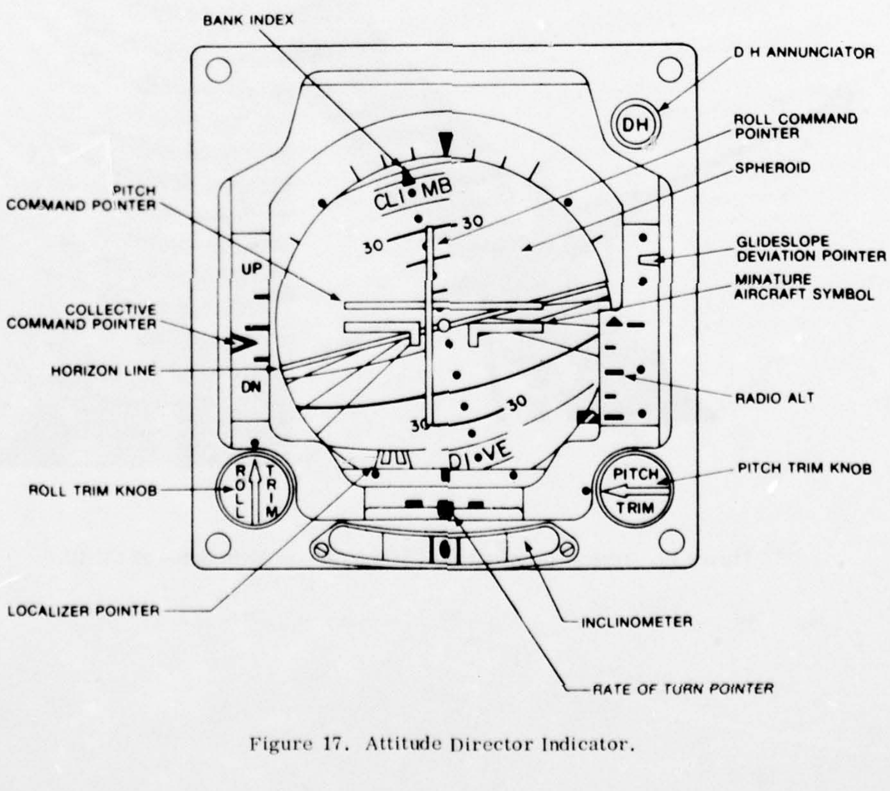

Figure 17. Attitude Director Indicator.

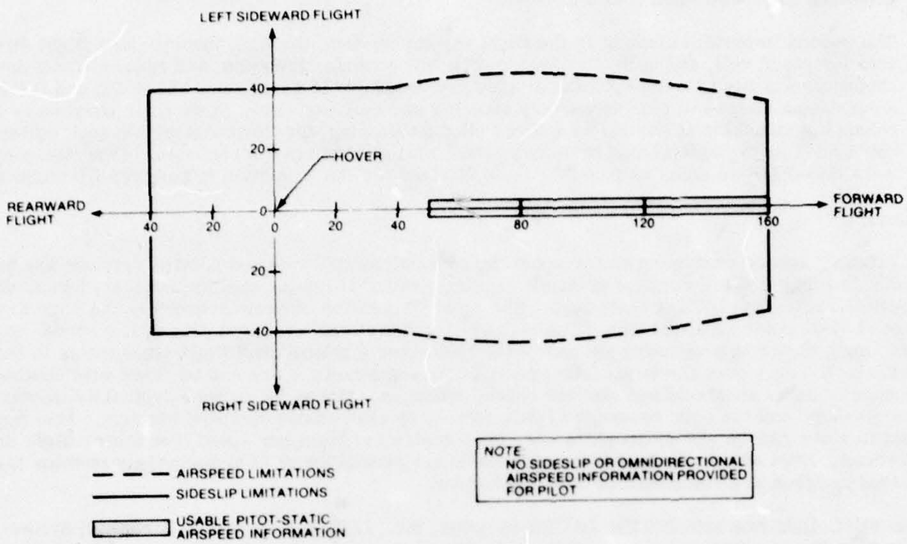

Figure 18. Helicopter Omnidirectional Airspeed Envelope.

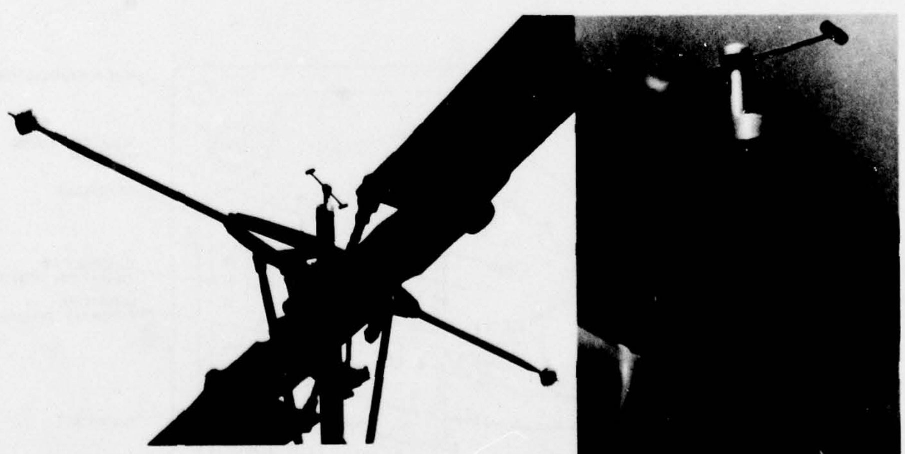

Figure 19. Typical Installation of LORAS Above Main Rotor of UH-IN.

knows his exact location in the helicopter's flight envelope and can control it with safety and confidence. Further, he can plan his hover maneuver to take best advantage of the added lift available from the wind.

V. SYSTEM DEVELOPMENT STATUS

The Collins Helicopter Flight Guidance and Control System is presently in a system engineering and design phase prior to prototype development. The system configuration and control algorithms are being finalized. However, flexibility has been maintained so that certain customer requirements, for example, peculiar sensors, etc, may be incorporated to make the system suit particular mission applications.

The implementation of the Flight Guidance and Control System elements described in this paper comprise an avionics solution to the nagging problem of all-weather helicopter rescue operations. Since the Avionics System Integrator is concerned about the end product - mission effectiveness - and not simply a demonstration of advanced technology equipment, a further responsibility is to verify the operational status, interface and acceptability of the entire avionics suite before delivery.

Before delivering a final equipped aircraft to the customer, several steps are taken to assure complete satisfaction and compliance with the intended use. One such step is the hot mockup test, where the entire system is set up with an actual aircraft wiring harness and an end-to-end system functional test is made. This testing not only ensures compatibility and proper interplay of the avionics, but it also verifies the power and cooling requirements. The Coast Guard HU-25A Falcon 20G hot mockup is shown in figure 20. A second step is called Dynamic Simulation. The Collins Dynamic Simulation Facility, shown in figure 21, ties actual hardware to a computer-simulated aircraft to verify the proper dynamic function of the Guidance and Control System. The third step comprises static ramp and actual flight testing to achieve government certification and customer acceptance of the avionics system as it is installed on the aircraft and put to its intended use.

VI. CONCLUSION

The solution to the challenge of making our helicopters and pilots truly effective rescue systems, capable of performing in adverse weather conditions with little or no visual cues, lies not just in the advancing technologies of guidance, controls, displays, and sensors. To tap the potential of these devices they must complement each other in a total avionics system. The demands of the rescue mission upon the pilot simply do not allow him the luxury of devoting his attention to one element of the system while ignoring others. Therefore manageability is a crucial factor. This means that the system's displays and controls (the points where the pilot receives information and makes inputs) must be consistent and well tuned to the pilot's capabilities. Such matters fall in the category of avionics system integration. The Rockwell-Collins approach to the integrated helicopter avionics system provides the pilot with not just a new set of avionics equipment but a mission potential to do the job more effectively than in the past.

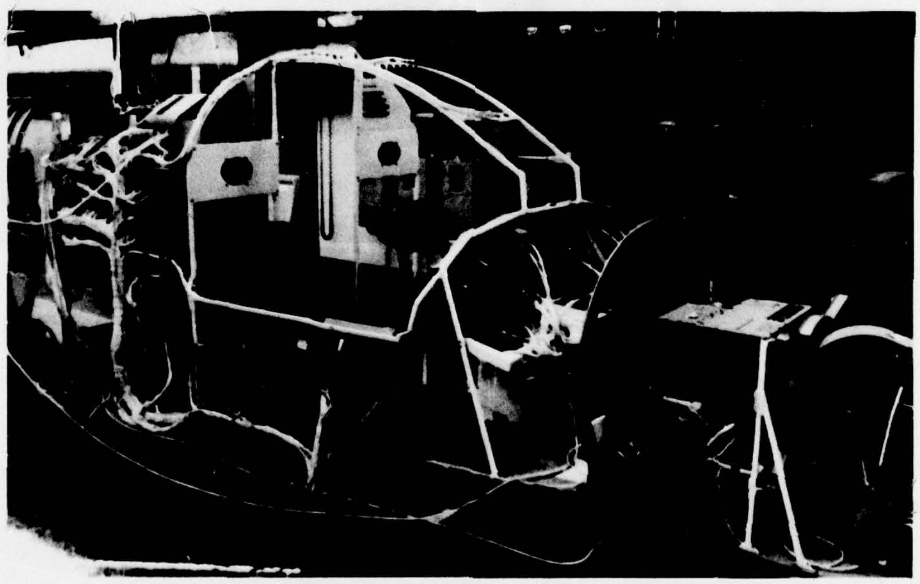

Figure 20. US Coast Guard HU-25A (MRS) Avionics Hot Mockup.

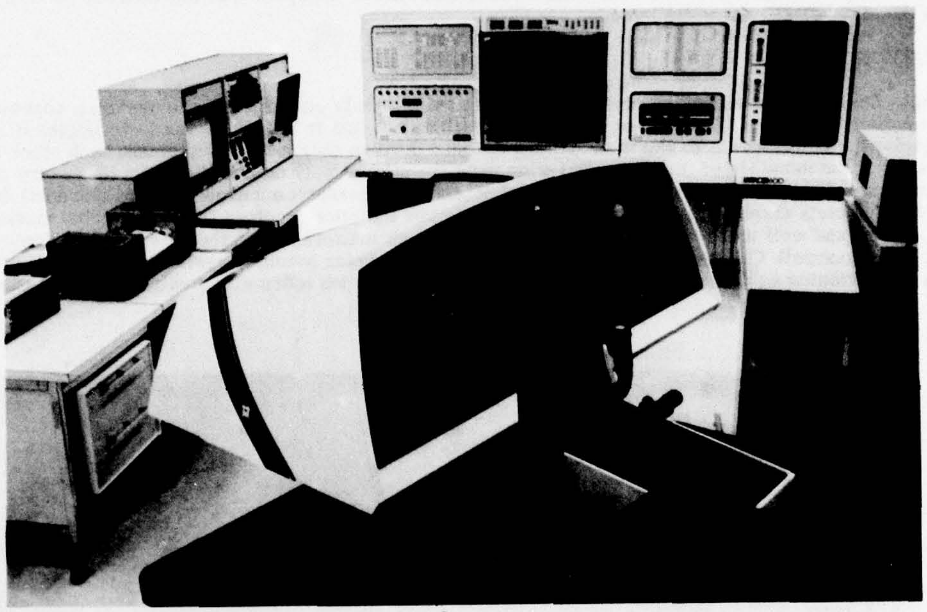

Figure 21. Collins Flight Simulation Facility.

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