

P-187

**NASA Contractor Report 181954**

**FLY-BY-LIGHT TECHNOLOGY DEVELOPMENT PLAN**

(NASA-CR-181954) FLY-BY-LIGHT  
TECHNOLOGY DEVELOPMENT PLAN Final  
Report (Douglas Aircraft Co.)  
187 p

N92-32778

Unclass

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G3/08 0116902

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**Contract NAS1-18028  
August 1990**

**NASA**

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ACKNOWLEDGEMENTS

Significant technical contributions made to this plan by the Systems and Research Center and Space Systems Group of Honeywell; Electromagnetic Applications (EMA); John Hopkins University are acknowledged.

### EXECUTIVE SUMMARY

This Fly-By-Light (FBL) Augmentation Plan is specifically intended to facilitate the introduction and certification of U.S. built Fly-By-Wire/Fly-By-Light (FBW/FBL) commercial transport aircraft, a significant U.S. aircraft industry issue. While the applications of this program are ostensibly for aircraft, the technologies and approaches resulting from it will benefit a broad cross-section of U.S. industry.

The proliferation of critical digital fly-by-wire aircraft systems has evidenced a number of new potentially catastrophic failure modes not encountered with conventional mechanical and analog control systems. The most dramatic news to hit the EMC (electromagnetic compatibility) community in some time is the recent revelation that a number of flight critical fly-by-wire systems are highly susceptible to radiated electromagnetic energy. Despite this, the performance and weight requirements imposed on military aircraft have necessitated the use of fly-by-wire flight and engine controls. The problem of designing highly reliable, maintainable, and lightweight fly-by-wire flight controls is further complicated by such factors as:

- o The increasing complexity (hardware and software) and number of aircraft digital systems yields new types of failure modes;
- o The increasing use of composites for aircraft skins decreases the first line of shielding provided by their metal counterparts;
- o The worsening EM (electromagnetic) environment due to the proliferation of radar, microwave, television, and radio sources, as well as the introduction of directed energy weapons;
- o New high speed integrated circuits require less power to change state (or be upset).

The extensive application of FBL technology for data transfer and sensing functions can substantially reduce the electromagnetic susceptibility of critical digital flight control systems. Since fiber optics are virtually immune to electromagnetic interference (EMI) the need for special shielding is eliminated and the number of conductive paths into the digital electronics is substantially reduced. Extensively shielding a critical digital flight control system is costly, heavy, and difficult to maintain. The use of fiber optics will reduce system weight and shielding maintenance.

In general, fly-by-light refers to a wide range of complementary technologies, concepts, design approaches, and computer based tools needed for next generation flight critical digital flight control systems (FCS). While fiber optics are a major part of FBL, many other aspects need to be evaluated, integrated, and balanced to optimize these FCSs.

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LIST OF ACRONYMS

|          |  |
|----------|--|
| AC       | Advisory Circular                                  |
| ADAS     | Architecture Design and Assessment Systems         |
| ADC      | Air Data Computer                                  |
| ADOCS    | Advanced Digital Optical Control System            |
| AFS      | Autoflight System                                  |
| AOPT     | Advanced Optical Position Transducers              |
| ASIC     | Application Specific Integrated Circuits           |
| ATOPS    | Advanced Transport Operating System                |
| BaE      | British Aerospace Corporation                      |
| BCAC     | Boeing Commercial Aircraft Company                 |
| BIT      | Built-In-Test                                      |
| BLM      | Behavioral Language Model                          |
| CAD      | Computer-Aided Design                              |
| CAE      | Computer-Aided Engineering                         |
| CARE III | Computer Aided Reliability Estimator               |
| CASE     | Computer-Aided Software Engineering                |
| CFC      | Carbon Fiber Composite                             |
| CFR      | Carbon Fiber Reinforced                            |
| CMOS     | Complementary Metal Oxide Semiconductor            |
| CPU      | Central Processing Unit                            |
| CW       | Continuous Wave                                    |
| DAC/Hi   | Douglas Aircraft Company/Honeywell Incorporated    |
| DBM      | Power in a 50 ohm load with respect to 1 milliwatt |
| DET/EOS  | Deterministic Exponential Order Statistics         |
| DS/EOS   | Doubly-Stochastic Exponential Order Statistics     |
| DoD      | Department of Defense                              |
| EDIF     | Electronic Design Interchange Format               |
| EM       | Electromagnetic                                    |
| EMC      | Electromagnetic Compatibility                      |
| EME      | Electromagnetic Environment                        |
| EMI      | Electromagnetic Interference                       |
| EMP      | Electromagnetic Pulse                              |
| EMS      | Electromagnetic Specifications                     |
| EMT      | Electromagnetic Threats                            |
| ESD      | Electrostatic Discharge                            |
| FAA      | Federal Aviation Administration                    |
| FAR      | Federal Aviation Regulation                        |
| FBL      | Fly-By-Light                                       |
| FBW      | Fly-By-Wire  |
| FBW/FBL  | Fly-By-Wire/Fly-By-Light                           |
| FCS      | Flight Control Systems                             |
| FMEA     | Failure Mode and Effect Analysis                   |
| FMES     | Failure Mode Effect Simulations                    |
| FMFA     | Failure Mode Fault Analysis                        |
| FMS      | Flight Management System                           |
| FOA      | Fiber Optic Architecture                           |
| FOCSI    | Fiber Optic Control System Integration             |
| FOL      | Fiber Optic Link                                   |
| FSK      | Frequency Shift Keyed                              |
| GGLOSS   | Generalized Gate Level Logic System Simulator      |
| GPS      | Global Positioning Satellite                       |
| GTD      | Geometrical Theory of Diffraction                  |
| HARP     | Hybrid Automated Reliability Predictor             |
| HDW/SW   | Hardware/Software                                  |
| HERF     | High Energy Radio Frequency                        |
| HI       | Honeywell Incorporated                             |
| HIRF     | High Intensity Radio Frequency (replaces HERF)     |
| HPM      | High Power Microwave                               |

LIST OF ACRONYMS (Continued)

|       |  |
|-------|--|
| I/O   | Input/Output   |
| IC    | Integrated Circuit                                   |
| IIDOS | Independent Identically Distributed Order Statistics |
| IRS   | Inertial Reference System                            |
| LABL  | Logic Automation Behavioral Language                 |
| LAI   | Logic Automation, Incorporated                       |
| LED   | Light Emitting Diode                                 |
| LRU   | Line Replaceable Unit                                |
| LSAS  | Longitudinal Stability Augmentation System           |
| LSI   | Large Scale Integration                              |
| LVDT  | Linear Variable Differential Transducer              |
| MBPS  | Megabits Per Second                                  |
| MTBF  | Mean-Time-Between-Failure                            |
| MTBUR | Mean Time Between Unscheduled Removals               |
| MTTF  | Mean-Time-To-Failure                                 |
| NAND  | Not - And Logic                                      |
| NASA  | National Aeronautics and Space Administration        |
| NEMP  | Nuclear Electromagnetic Pulse                        |
| NHPP  | Non-Homogeneous Poisson Process                      |
| OPMIS | Optical Propulsion Management Interface System       |
| OPS   | Operating System (Flight OPS - Flight Operations)    |
| PALAS | Modeling System                                      |
| PBF   | Power-By-Fiber                                       |
| PBL   | Power-By-Light                                       |
| PC    | Printed Circuit                                      |
| PDL   | Program Design Language                              |
| PFC   | Primary Flight Control                               |
| PFCB  | Primary Flight Control Bus                           |
| POE   | Points of Entry                                      |
| R&D   | Research and Development                             |
| RF    | Radio Frequency                                      |
| ROCOF | Rate Of Occurrence Of Failure                        |
| ROM   | Read-Only Memory                                     |
| RPT   | Rapid Prototyping Technologies                       |
| RVDT  | Rotary Variable Differential Transducer              |
| SAE   | Society of Automotive Engineers                      |
| SCFSG | Sperry Commercial Flight Systems Group               |
| SE    | Software Engineering                                 |
| SEMP  | Simulated Electromagnetic Pulse                      |
| SNP   | Standard Networks and Protocols                      |
| SSC   | Sidestick Controller                                 |
| SSI   | Small Scale Integration                              |
| STC   | Supplemental Type Certificate                        |
| SURE  | Semi-Markov Unreliability Range Evaluator            |
| TDFD  | Three Dimensional Finite Difference                  |
| TDM   | Time Division Multiplexed                            |
| TLA   | Throttle Lever Angle                                 |
| TMC   | Thrust Management Computer                           |
| TTF   | Time To Failure                                      |
| V&V   | Verification and Validation                          |
| VHDL  | Very High Level Description Language                 |
| VHSC  | Very High Speed Circuit                              |
| VHSIC | Very High Speed Integrated Circuit                   |
| VLHIC | Very Large High Integrated Circuit                   |
| VLSI  | Very Large Scale Integration                         |
| VMI   | Vibro-Mechanical Interference                        |
| VOA   | Voice of America                                     |
| VVC   | Verification, Validation, and Certification          |
| WDM   | Wavelength Division Multiplexed                      |

1.0 INTRODUCTION

1.1 Background

This report discusses the driving factors and developments which make fly-by-light viable. It also provides documentation, analyses, and recommendations on the major issues pertinent to facilitating the U.S. implementation of commercial FBL aircraft before the turn of the century. To accomplish this goal a unified national effort is required. This plan would be coordinated by the National Aeronautics and Space Administration (NASA) and would involve both government and industry participation.

In this report "FBL" is used in a general sense, referring to a wide range of complementary technologies, concepts, design approaches, and computer based tools needed for next generation flight critical digital flight control systems (FCS). While fiber optics are a major part of FBL, many other aspects need to be evaluated, integrated, and balanced to optimize these FCSs.

It should be noted that continuous involvement of Federal Aviation Administration (FAA) engineers is essential in both planning and executing the proposed FBL Program.

The introduction of fly-by-wire/fly-by-light technology into U.S. manufactured commercial transport aircraft places unprecedented demands upon airframe companies, equipment suppliers, and regulatory agencies. These demands necessitate a high degree of cooperation between all parties involved.

Areas of particular concern include:

- o Ultra-reliable computing (hardware/software)
- o EME
- o Verification and Validation
- o Optical techniques
- o Life-cycle maintenance
- o Basis and procedures for certification

1.2 Ultra-reliable computing

Ultra-reliable computing (i.e., extremely fault tolerant systems) is an area that has received significant Research and Development (R&D) emphasis in past years. This work includes concepts such as "zero unscheduled maintenance" and "extended availability".

### 1.3 Electromagnetic Effects

Protection/quantification relative to the effects of the electromagnetic environment (EME) is a design problem coupled to environmental conditions that has received inadequate attention in the past but now has been elevated to prominence. Such a late recognition of the magnitude of this problem has left industry with major near term design and Verification & Validation related problems to solve. Figure 1.3-1 illustrates the levels of electrical/electronic systems for which EME must be addressed.

### 1.4 Optical Techniques

Optical techniques show potential to resolve significant aspects of the EME problem mentioned previously. Size and weight savings are also expected. It is anticipated that optical networking and data links will be applied to first generation U.S. commercial FBW/FBL aircraft. Subsequent generations will likely make extensive use of optical sensing technologies.

### 1.5 Verification and Validation

Verification, Validation, and Certification (VVC) must also come to the forefront. It is generally agreed by equipment suppliers now developing FBW/FBL systems that state-of-the-art VVC methodologies are inadequate. Of particular interest is the role of modeling the system response to faults (functional and gate level) in VVC as commercial FBW/FBL is introduced. The work already done at NASA-LARC (i.e., HARP, CARE III, SURE, GGLOSS) must be expanded and matured. Software verification and aspects of certification are also areas of concern that must continue to be emphasized.

### 1.6 Life-Cycle Maintenance

Life-cycle maintenance issues become safety issues when considering commercial FBW/FBL. It is anticipated that requirements for flight line maintenance must be sharply reduced. This can only be achieved through new design methodologies which facilitate increased system effective mean-time-between-unscheduled-removals (MTBUR) through the application of "secondary redundancy" techniques. Redundant elements are included and can fail without the need for removal of the line replaceable unit (LRU). Accordingly mean-time-between-failure (MTBF) decreases while MTBUR is sharply increased. That is, additional system redundancy is included not for safety but for "extended system availability." This means that a number of simple faults can be accumulated and tolerated in redundant elements of a system for an extended period of time without impacting system safety. The net result is less unscheduled box removals for minor and oftentimes elusive squaks.

## 1.7 Basis and Procedures for Certification

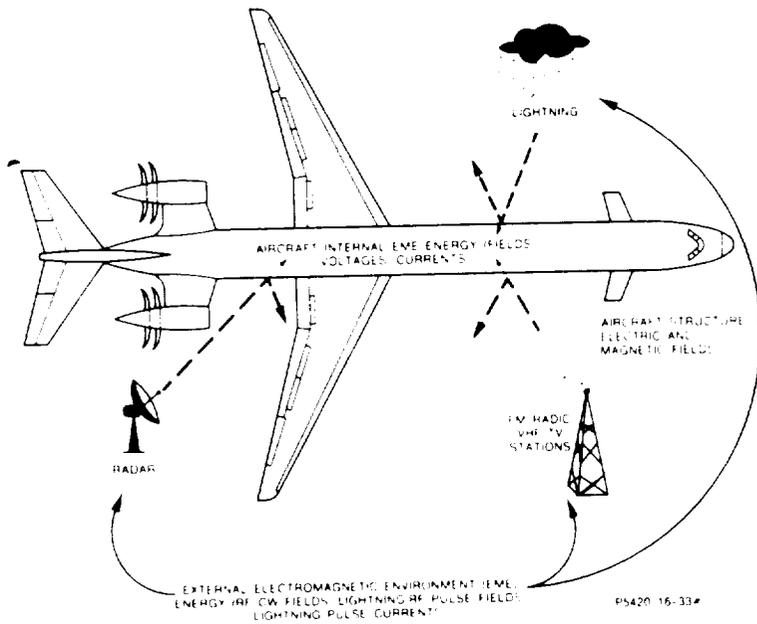
One of the major goals of this program is to provide guidance to the FAA in the development of certification criteria. Generally, noncritical avionic systems for commercial transport aircraft have been designed to various Federal Aviation Regulations (FARs) such as FAR 25.1309. However, rules or guidance materials are not available for designing flight critical systems using FBL technology. Therefore, it is important that the FAA be involved throughout this technology verification program.

In order to accomplish this, FAA certification specialists will be directly involved in the development of advanced technologies. This involvement will assist the FAA in the development of the certification criteria outside of the typical adversarial relationship between the FAA and industry in an actual certification project.

With the FAA establishing the FBL/FBW certification criteria early, the typical uncertainty of certification for the airframe manufacturers will be eliminated. This uncertainty comes from advanced designs not meeting the exact requirements that would be contained in a special condition that does not exist at the beginning of the design process. Since these systems are normally developed for new airplanes, this uncertainty has in the past forced the airframe manufacturer into parallel development of a conventional system for the new aircraft model, thus protecting themselves against the possibility that the new system would not be developed. This practice has proved to be costly in time and money to the manufacturer, the FAA, and the customer.



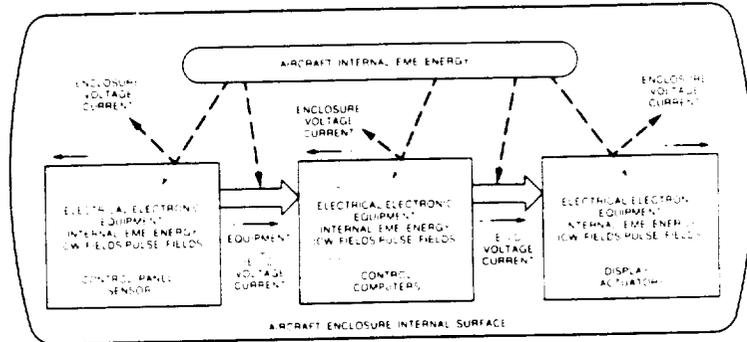
FOLDOUT FRAME



### Aircraft Level

- Environment induces electric and magnetic fields (charge and currents) and injects lightning currents on aircraft exterior
- Wide bandwidth: DC-40GHz
- Transient and CW
- Understanding of coupling mechanisms and the corresponding analytic models/algorithms needs further development
- Test methodologies need further development and validation
- Aircraft level testing needs justification

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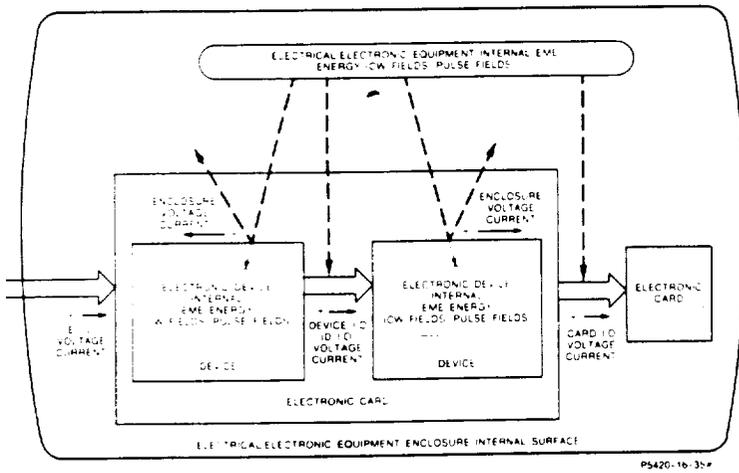


### Control System Level

- External energy penetrates to interior via apertures, composites, seams, joints, and antennas
- Voltages and currents induced on flight control system components and cables
- Coupling mechanisms relatively well understood
- Analytic models/algorithms need further development
- Test methodologies need further development and validation

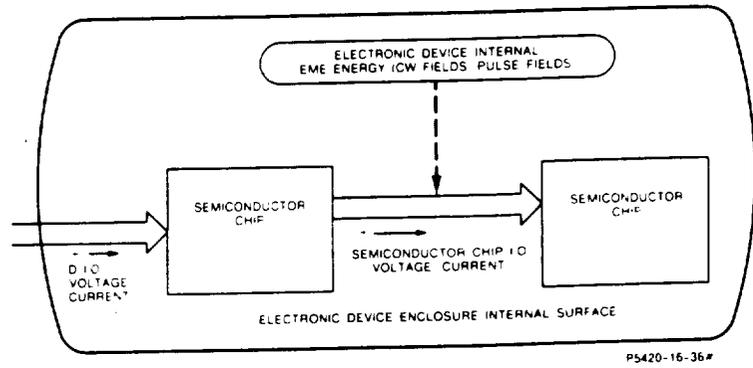
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1-3 A.



### Electrical/Electronic Equipment and Card Level

- Voltage, fields, currents and charge on components penetrate into box interiors via holes, seams, and cables
- Energy (voltage and current) picked up by wires and printed conductors on cards and carried to electronic devices
- Coupling mechanisms relatively well understood
- Analytic models/algorithms need further development
- When applicable, test methodologies need development and validation



### Device Level

- Card and device conductors carry energy to the semiconductor chips
- Possible effects
  - Damage
  - Upset
- Coupling mechanisms relatively well understood
- Analytic models/algorithms need further development
- Device response characteristics and models/algorithms need further development
- When applicable, test methodologies need development and validation

FIGURE 1.3-1

System Topology EME Overview

## 2.0 REQUIREMENTS FOR FLY-BY-LIGHT FLIGHT CONTROL SYSTEMS

### 2.1 Background

Douglas Aircraft Company (DAC) studies conducted in early 1984 showed that a transport aircraft fly-by-wire (FBW) flight control system (FCS) would provide significant benefits in terms of increased performance, weight reduction, and cost effective manufacture, and operation. These studies also indicated that a transport FBW system could be designed for high reliability, maintainability, and fault survivability. Furthermore, if properly designed, no mechanical backup would be required and dispatchability would not be jeopardized.

Military helicopters and fighter aircraft have employed FBW flight and engine controls for years, but the higher reliability, maintainability, and safety requirements of transport aircraft have precluded the use of FBW technologies until recently. Recent demands for increased performance and operational capabilities of transport aircraft have encouraged the development of FBW flight controls.

The Airbus A320, introduced in 1987, has full authority digital FBW in the pitch and roll axis, partial FBW in the yaw axis, partial mechanical in the yaw axis, and a mechanical horizontal stabilizer. The rudder and the mechanical trim stabilizer controls provide minimal mechanical backup for flight control following a complete failure of the primary FBW system, with the goal of maintaining flight until the primary system can be reactivated or until landing.

### 2.2 Safety Requirements

Safety Requirements for a full time, flight critical commercial aircraft fly-by-wire/fly-by-light (FBW/FBL) system with no mechanical backup exceed the requirements asserted for any previous commercial system in the area of probability of loss of function. The probability of loss-of-aircraft demanded by the Federal Aviation Administration (FAA) is  $10^{-9}$  flight/hour.

Loss-of-aircraft can result for a variety of reasons; including FBW/FBL system failure, various structural failure modes, and certain propulsion related failures. The issue of concern here is the FBW/FBL system.

(Note that the A320 has a rudimentary mechanical backup system but it is not given any credit in probability of loss of function calculations.)

## 2.3 System Electromagnetic Environment (EME) Effects Protection

System EME effects protection is required and must receive elevated emphasis in this program. Recent alleged EME-related aircraft losses have revealed a major problem that will likely get worse. Accordingly, the EME effects problem has gone from relative obscurity to perhaps the pacing problem relative to the introduction of U.S. built FBW/FBL commercial aircraft. Both design and verification and validation (V&V) requirements will require emphasis. Recent trends in the EME threat are depicted in Figure 2.3-1.

## 2.4 Fiber Optics

### 2.4.1 Optical Data Bus

Optical busses, particularly for avionics and primary flight controls, are of great interest for first generation FBL aircraft. The avionics bus, which links the aircraft's main flight boxes, must provide bidirectional communication between all (32 to 50) terminals. The primary flight control bus must provide bidirectional communication between one flight control computer in each of 9 to 24 smart actuator terminals; communication between actuators is not required.

In order to achieve fiber optic data bus flight readiness, protocol and architecture requirements must be established for each bus application, component technologies (e.g., connector type, fiber composition, size, etc.) must be evaluated and certified, and installation and maintenance procedures must be developed.

### 2.4.2 Optical Sensors

Optical sensors will be required to be as accurate, reliable and maintainable as the devices which they replace. An evaluation of various sensor technologies is needed to determine their applicability to flight critical avionics.

## 2.5 System Functionality

The system functionality requirements for a first generation U.S. built FBW/FBL system are anticipated to resemble those of the A320. That is, in normal operation, both inertial data and air data would be fully utilized and full envelope protection would be provided. Autoland capability would not be considered part of the FBW/FBL computer's function. However, the impact of its inclusion should be studied. A sidestick interface should be assumed.

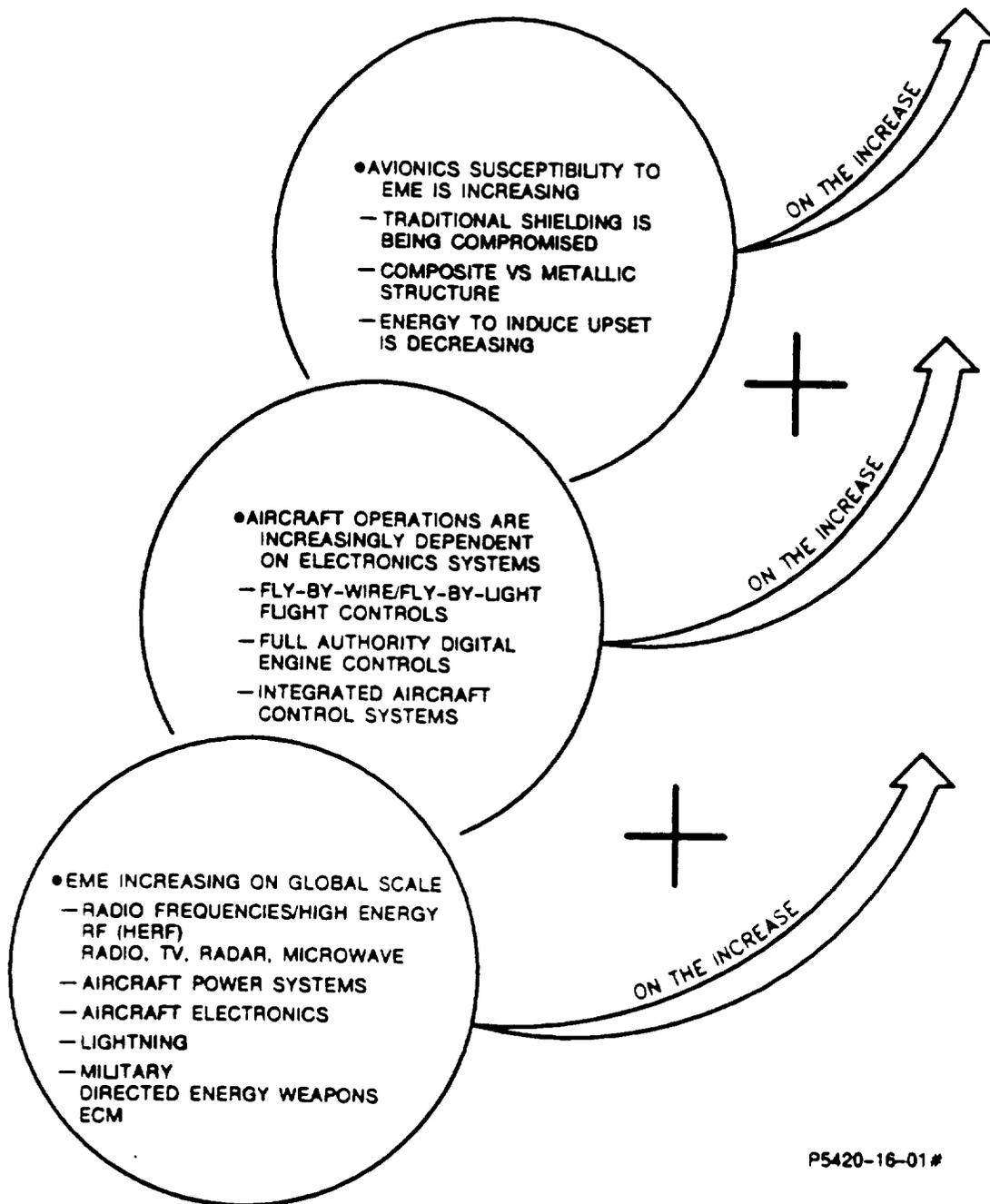


FIGURE 2.3-1  
EME Threat Trends

### 2.5.1 FBW/FBL Computer to Sidestick Interface

The sidestick to flight control computer interface requires study and requirement development. These requirements would address Captain/First Officer sidestick coupling, reliability requirements for sidestick design, sidestick to FBW/FBL interface, and sidestick ergonomics. Other areas of study would include alternatives to sidesticks and the pilot/sidestick interface.

### 2.5.2 FBW/FBL Computer to Actuator Interface

The FBW/FBL computer to actuator interface is an area of particular importance if a "smart" (i.e., computer controlled) actuator is assumed. A smart actuator allows a digital bus interface media and local monitoring of actuator operation with attendant advantages. However, this could open up the possibility of simultaneous loss of all actuators of a given type due to a "generic" fault. To help enhance safety, a concept under study by Douglas Aircraft Company and Honeywell, Inc. (DAC/Hi) is referred to as a smart "actuator override" in which the FBW/FBL computer would be capable of bypassing all computerized internal actuator monitoring functions to either force the actuator to engage or force the actuator to disengage. It is believed that such a capability should be required. The flight control computer has access to aircraft sensor data, comprehensive monitoring, and degradation strategies.

## 2.6 System Availability

System availability for a FBW/FBL system without mechanical backup becomes a safety issue. It has been argued that routine flight line maintenance of FBW/FBL computer systems should not be permitted. Rather, repairs should be restricted to dedicated repair facilities. It can be argued that the mean-time-between-failure (MTBF) of the FBW/FBL computers should be at least as good as the equipment they replace. The above issues lead to requirements such as "no unscheduled maintenance" or possibly "deferred maintenance". To preclude to a high probability the necessity of unscheduled maintenance requires redundant elements within the computers with an associated reconfiguration capability.

Such redundancy is therefore not directly safety related and accordingly has come to be called "secondary redundancy". This secondary redundancy may be required because it is anticipated that a FBW/FBL computer assembly might require a mean-time-between-unscheduled-removal (MTBUR) approaching 100,000 hours; thus achieving zero unscheduled maintenance.

## 2.7 Graceful Degradation

Graceful degradation in the context of FBW/FBL systems refers to control law reconfiguration in flight after the loss of a sensor set being used to control flight. The system must be capable of gracefully and automatically transitioning from fully augmented flight to levels of lesser augmentation and to stick-to-surface operation or even activation of a backup system. Pilot override of this feature should be provided.

## 2.8 Generic Fault Tolerance

Design provisions must permit the system to remain operational in the presence of generic faults. Areas of concern include hardware, software generation, and system specification. Relative to these fault classes, N-version techniques are now widely used in the commercial aircraft industry (i.e., DAC, BCAC, British Aerospace [BAe], Airbus). It should be noted that although generally regarded as effective in relation to certain specific fault classes, N-version techniques lack both an established set of design rules and a clear quantification of benefits.

It is widely agreed that a major area of problems and potential safety hazards relates to the specification itself. Specific requirements for V&V of system specifications must be developed.

## 2.9 System Recovery from Upset (Soft Fault Tolerance)

Automatic system recovery from upset must be addressed. It is now becoming increasingly clear that most system faults are intermittent, rather than permanent hardware failures. However, little attention has been given to this area. Upsets can result from many causes, not the least of which is EME. DAC/HI are actively working on techniques which will permit the system to operate through a disturbance via a methodology referred to as "Transparent Recovery". Note that we are not talking about eventual reinstatement of a failed FBW/FBL computer but rather designing FBW/FBL computers which can tolerate the disturbance with no external effect (i.e., "transparent").

## 2.10 Backup Systems

Backup flight control will likely be included in a first generation FBW/FBL systems. An electrical implementation (as opposed to mechanical) is assumed. No safety credit should be taken for this backup during certification. The backup can be single-thread (i.e., no primary redundancy). It would take control automatically in the event of complete loss of the primary system. Note that it can be argued that no means need to be provided to permit the airline pilot to activate the backup. This entire issue warrants further study.



### 3.0 TECHNOLOGY ASSESSMENT

#### 3.1 Background

The proliferation of flight critical digital flight control systems (FCSs) has evidenced a number of new potentially catastrophic failure modes not encountered with conventional mechanical and analog control systems. Recent evidence has indicated that flight critical fly-by-wire (FBW) systems are highly susceptible to radiated electromagnetic (EM) energy.

The extensive application of fiber optic technology for data transfer and sensing functions will substantially reduce the EM susceptibility of flight critical digital FCS. Since fiber optics are virtually immune to electromagnetic interference (EMI) the need for special shielding is eliminated and the number of conducted paths into the digital electronics is substantially reduced. Extensively shielding a critical digital FCS is costly, heavy, and difficult to maintain. The use of fiber optics will reduce system weight and shielding maintenance.

#### 3.2 Safety Requirements

##### 3.2.1 General Issues

Relative to the internal workings of commercial transport fly-by-wire/fly-by-light (FBW/FBL) systems, extensive investigations of ways to guarantee availability of  $10^{-9}$ /hour or better have been pursued. Still there remain unanswered questions as noted in Section 2. Specific to the FBW/FBL flight control computer system internal operation questions not fully resolved include the following:

- A. Quantification of benefits gained from software development methodologies (e.g., N-version).
- B. The effects of long term field maintenance on safety level degradations.
- C. The benefits and risks associated with design for "extended availability" through secondary redundancy.
- D. Validation of system fault tolerance/redundancy management methodologies with new and proposed system fault effects modeling methodologies.

##### 3.2.2 Generic Fault Concerns

Several approaches to protection from "generic" faults have been suggested. These include:

- o N-version software and/or hardware
- o Performance assessment monitor

### 3.2.2 Generic Fault Concerns (Continued)

#### o Proof of correctness

A review of articles documenting numerous N-version software experiments, results in no clear quantification of benefits. Further, it appears likely, even to the casual observer, that the specific type of implementation methodology adopted will heavily impact the effectiveness of this technique. The following two needs relate to our understanding, first of N-version effectiveness experienced in industry, and secondly of the implementation methodologies that have been adopted.

- A. Survey and document industry experience with the application of N-version techniques.
- B. Based upon experience gained in both industrial applications and research experiments, one or more development methodologies should be documented. It is essential that formal, mature methodologies be established.

#### 3.2.2.1 N-version Techniques

Currently N-version techniques are being applied to numerous critical commercial aircraft FCS. Examples include:

- o MD-11
- o 737-300
- o B777 (currently in development)
- o BAe-146-300 (currently in development)
- o A310
- o A320

#### 3.2.2.2 Performance Assessment Monitors

This technique defines specific monitor functions to detect event system behavior. This technique has been applied by Honeywell and others in both commercial and military avionics. To date the benefits and limitations of this technique have not been thoroughly explored.

#### 3.2.2.3 Proof of Correctness

Methodologies to rigorously provide the validity of a specification have been suggested in the literature. The applicability of such techniques has, in general, been viewed negatively. However, no documented study has been conducted to date.

### 3.3 System Electromagnetic Environment Protection

#### 3.3.1 Introduction

Electromagnetic environments produce electrical energy of the same type that is used by electrical/electronic equipment to process and transfer information. As such, this environment represents a fundamental threat to the proper operation of systems that depend on such equipment. For systems providing flight critical functions that depend upon information processed by electronic equipment, the electromagnetic threat to systems translates to a threat to the airplane itself. Specifically, the EM environment is a top level system issue that must be conscientiously accounted for in any airplane design where control is provided by a FBW system architecture. Although the substantial coupling paths resulting from wiring running throughout the aircraft fuselage is eliminated by a FBL data link architecture, the primary means of processing information is still electronic, and the electromagnetic environment (EME) is still a significant threat.

In general the EM environment elements of interest for aircraft include the following:

- A. Lightning
- B. High Intensity Radio Frequency (HIRF)
- C. Electrostatic Discharge (ESD)
- D. Intrasystem EMI
- E. P-static
- F. Special Military Requirements
  - 1. Electromagnetic Pulse (EMP)
  - 2. High Power Microwave (HPM)

For commercial aircraft, the emphasis should be on lightning and HERF, because most of the energy and system hazards arise from these threats. Their interaction with aircraft systems is global and also the most complex, requiring more effort to understand. ESD is of interest from an operational viewpoint, but is especially of concern in the development of latent defects, which are usually controlled during manufacture and assembly stages. EMI is an intrasystem problem which is locally controlled, but still requires sound knowledge and practice. P-static is also usually controlled by rather well known design practices.

### 3.3.1 Introduction (Continued)

Therefore, for this effort it is assumed that the lightning and HIRF environments are those of concern. The other environments may be of occasional interest, but do not drive this technology development.

In addition to the fundamental trend of relying on electronic equipment for flight critical functions, another trend that has increased the concern about EME is the increased percentage of composite materials in aircraft construction. Because of their decreased conductivity, composite materials result in less inherent shielding by the aircraft structure.

### 3.3.2 EME Threat

Lightning and HIRF are two extraordinarily severe EME threats to the overall aircraft. Of the two, lightning produces the most intense EME. The lightning-produced environment is relatively instantaneous (exists for less than 0.5 msec for each lightning stroke occurrence). During a lightning strike (one lightning strike or flash can contain several individual strokes), relatively large currents flow in the wing structure, fuselage, or empennage, which in-turn induce relatively large voltages and currents in the aircraft wiring. Such voltages and currents appear at equipment interface circuits. In an all metal aircraft, induced voltages are usually less than 200 V although 1000 V or more have been projected for some aircraft wiring when excited by the EM environment of a severe stroke. For aircraft employing the extensive use of composite materials, the lightning-produced voltages and currents could increase substantially. Whether they do or not depends upon the shielding measures provided in the aircraft.

In addition to the large currents associated with return strokes, high rates of rise short duration pulses (noise pulse) have been found to occur randomly throughout a lightning flash, interspersed with the other current components. The current amplitude of such pulses are much less than those of a return stroke (first or subsequent) and while not likely to cause physical damage to the aircraft or electronic components, the random and repetitive nature of these pulses may cause interference or upset to certain systems. As previously noted, a typical cloud-to-ground lightning flash contains more than one restrike. In fact, flashes containing up to 24 strokes randomly spaced have been recorded. For evaluation of indirect effects (damage and upset) it is necessary to consider the multiple-stroke nature of an actual lightning flash, because the succession of strokes may induce corresponding pulses in data transfer circuits (for example) causing cumulative damage (in some special interface circuit

### 3.3.2 EME Threat (Continued)

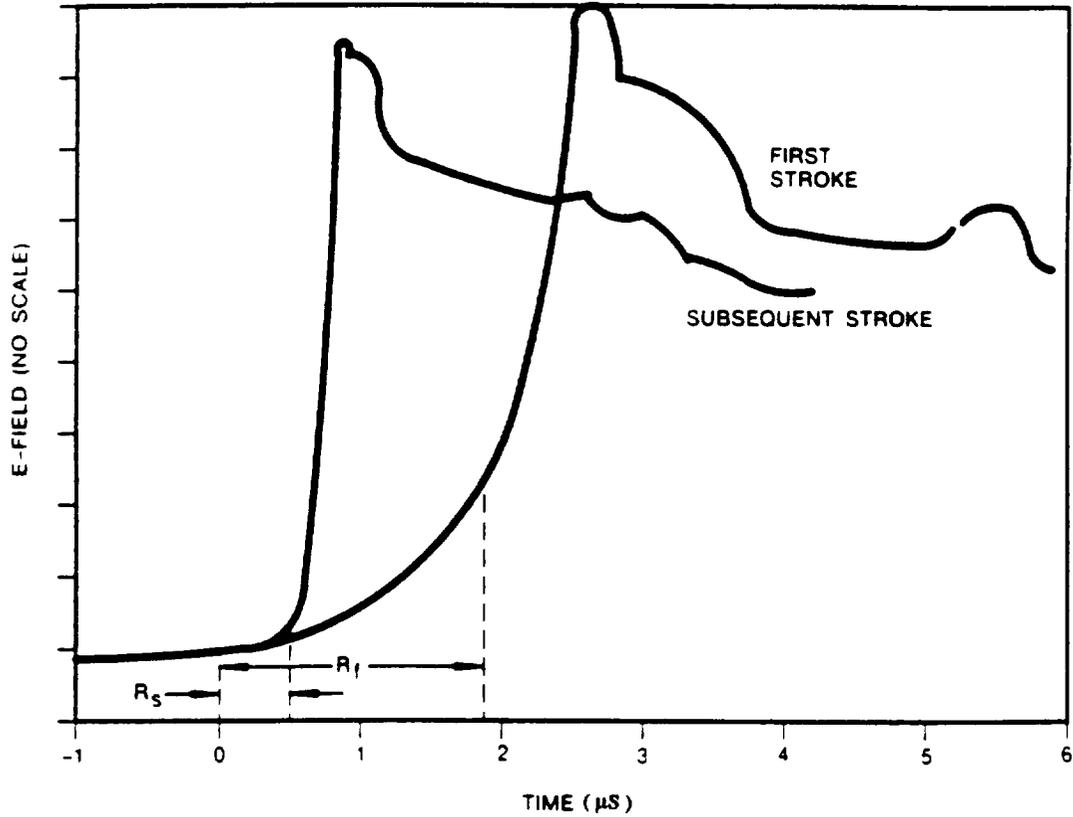
configurations) or upset to sensitive systems or devices. Figure 3.3-1 shows the wave shape of two natural lightning events and Figure 3.3-2 shows the waveshape of engineering waveforms that reproduce important lightning waveform parameters (amplitude, rise time, action integral). Figure 3.3-3 shows severe engineering representations of the environments associated with a multiple stroke and bursts of noise pulses.

Even though lightning produces the most intense EM levels, the radio frequency (RF) environment dramatically exposes electrical/electronic system susceptibility to the effects (induced voltages and currents) of EME. This discovery of system susceptibility was probably made possible because the RF environment is deterministic (not instantaneous, random, and capricious as is the case for lightning), and therefore relatively traceable to the catastrophic events it caused.

The EM spectrum associated with radio frequencies is vast. The radio frequency/electromagnetic (RF/EM) field strength from numerous sources (measured in volts per meter) varies widely. Figure 3.3-4 shows various sources associated with the RF spectrum. Figure 3.3-5 shows the envelope of peak field strengths that could be encountered within the U.S, U.K. and France during a flight scenario. While all aspects of a flight scenario are vital, the takeoff and landing portions exhibit the most potential for catastrophic occurrence due to the short reaction times. Therefore absolute safety in takeoff and landing environment shown in Figure 3.3.5 must be assured.

The wiring lengths (of interest from an RF coupling perspective) anticipated to be encountered within aircraft of the MD-91 size class will be roughly in the 0.5 m to 50 m range. This translates to resonant frequencies of 300 MHz to 3 MHz, respectively. Aircraft wiring resonances could increase the induced voltages produced by external RF to as much as 30 volts (as compared to the less than 1 volt that would occur at nonresonant frequencies). Induced voltages will be present as long as the interfering RF persists. The spectrum of RF energy that penetrates aircraft wiring and electrical/electronic systems can be summarized in three basic areas:

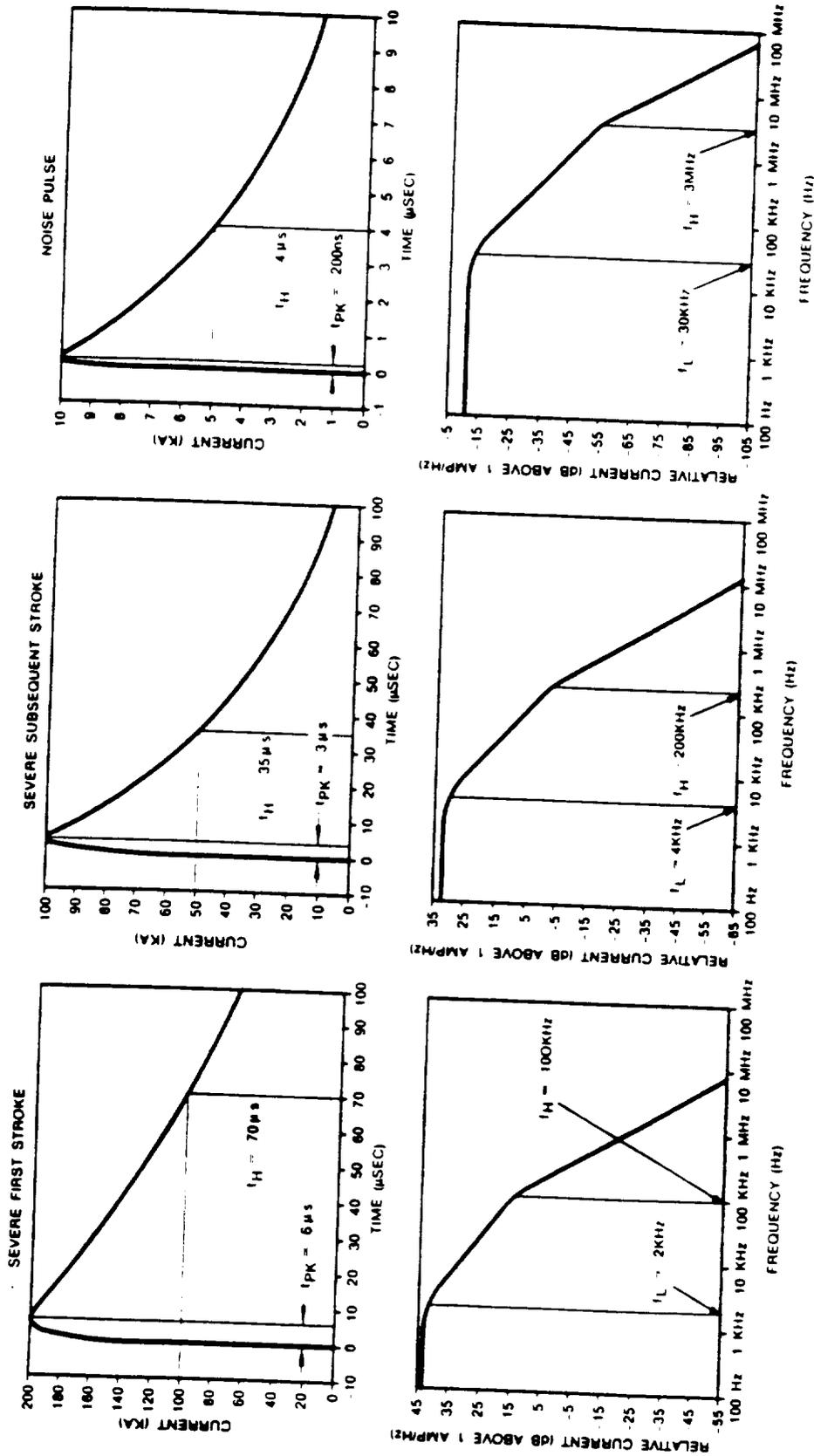
- A. RF energy below 1 MHz - induced coupling at these frequencies is inefficient and thus will probably be of lesser concern.
- B. RF energy between 1 and 300 MHz is of major concern as aircraft wiring at these frequencies acts as a highly efficient antenna.
- C. RF energy coupling to aircraft wiring drops off at frequencies above 300 MHz. At these higher frequencies the EM energy tends to couple through box apertures rather than through aircraft wiring.



CHARACTERISTIC OF FIRST RETURN STROKE FRONT RAMP ( $R_f$ ) AND SUBSEQUENT STROKE FRONT RAMP ( $R_s$ ) AS MEASURED BY WEIDMAN AND KRIDER (1978) E-FIELD AMPLITUDES ARE NOT TO SCALE. WAVEFORMS ARE BASED ON MEAN VALUES OF HISTOGRAMS.

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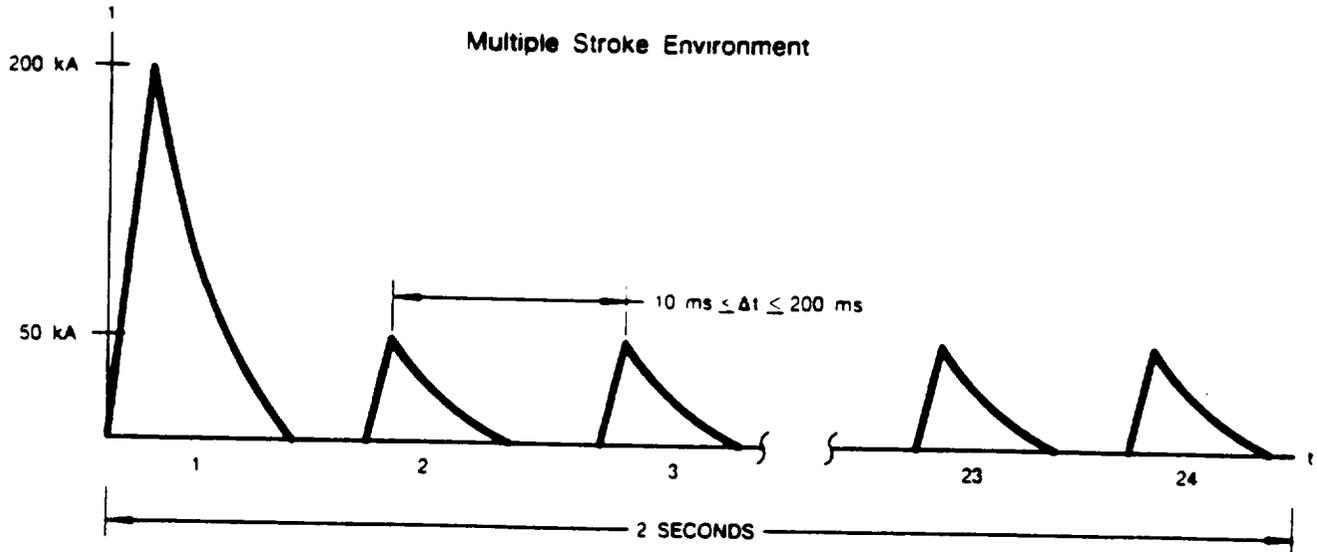
FIGURE 3.3-1  
Waveshape of Two Natural Lightning Events



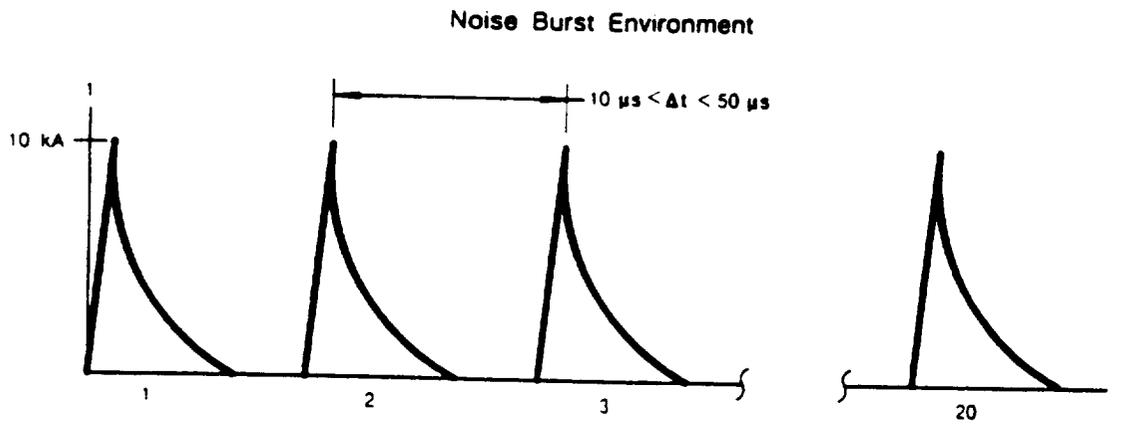
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FIGURE 3.3-2

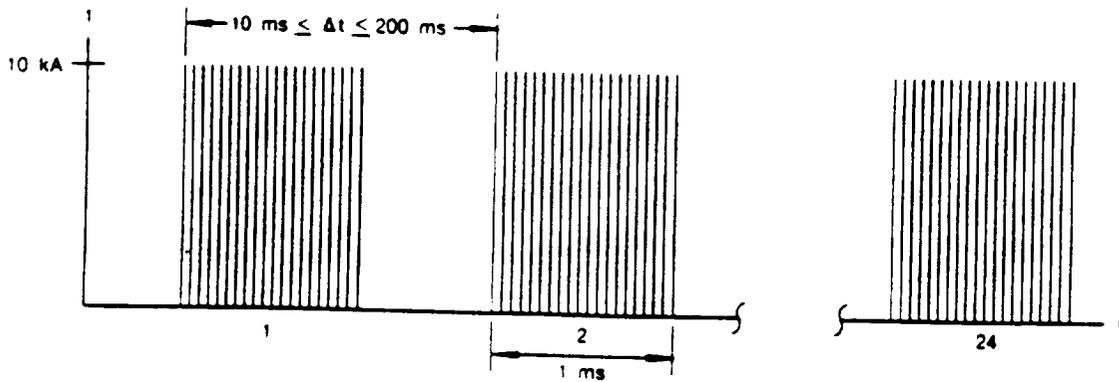
Waveshape of Engineering Waveforms



ONE SEVERE FIRST STROKE FOLLOWED BY TWENTY-THREE MODERATE SUBSEQUENT STROKES, DISTRIBUTED OVER A PERIOD OF UP TO TWO SECONDS.



ONE NOISE BURST IS 20 PULSES IN 1 MILLISECOND.

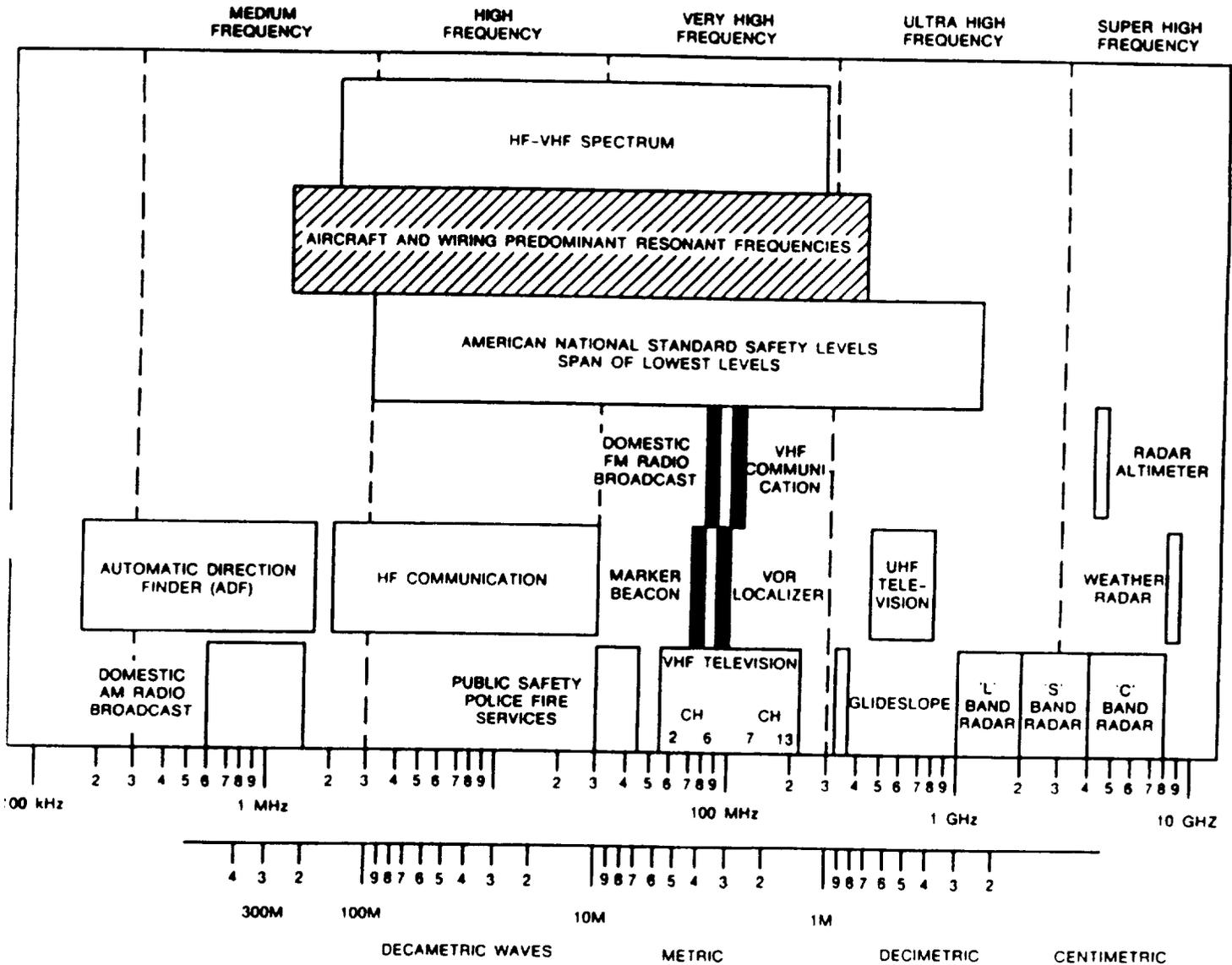


TWENTY-FOUR NOISE BURSTS DISTRIBUTED OVER A PERIOD OF UP TO 2 SECONDS.

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FIGURE 3.3-3

Multiple Stroke and Noise Burst Environments



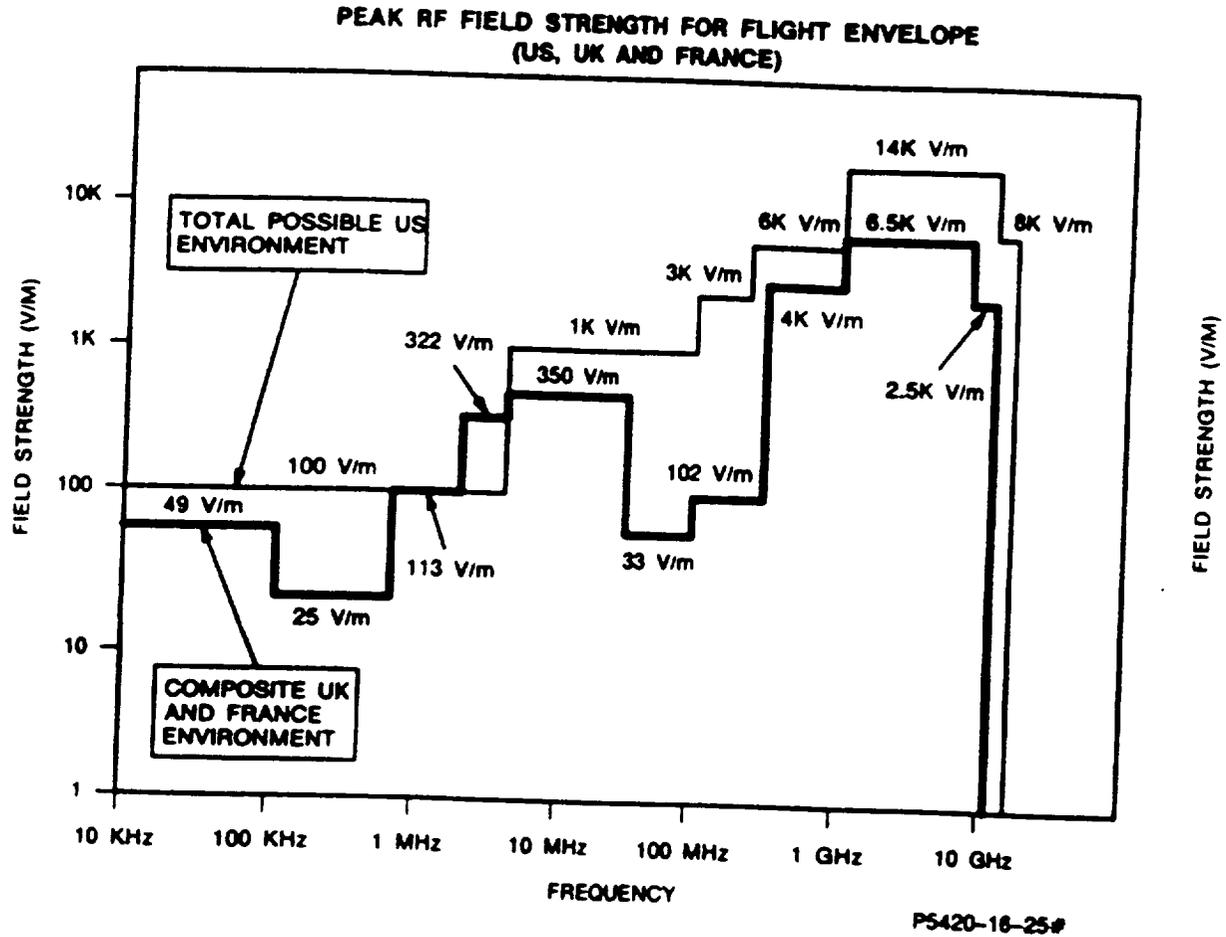
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FIGURE 3.3-4

Sources Associated with Radio Frequency Fields

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FOLDOUT FRAME

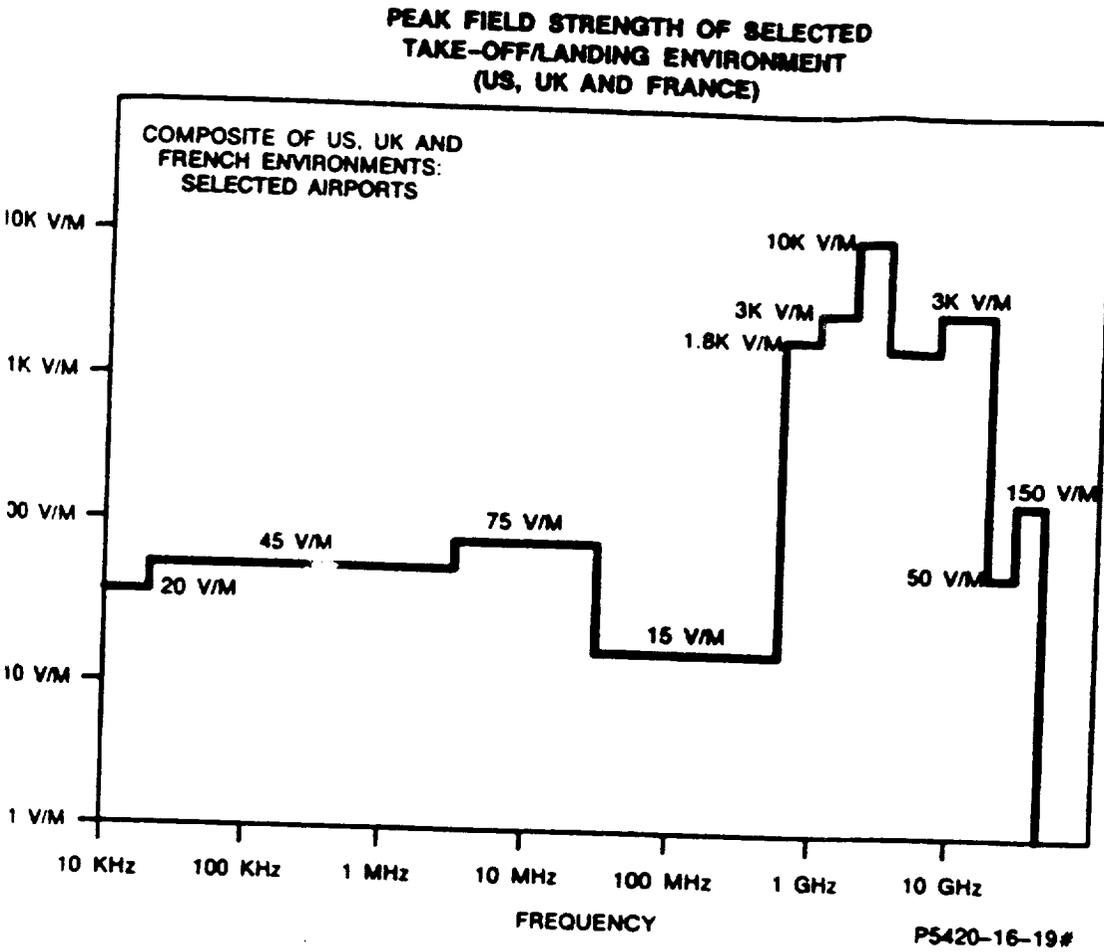


FIGURE 3.3-5

Peak RF Field Strength Envelope

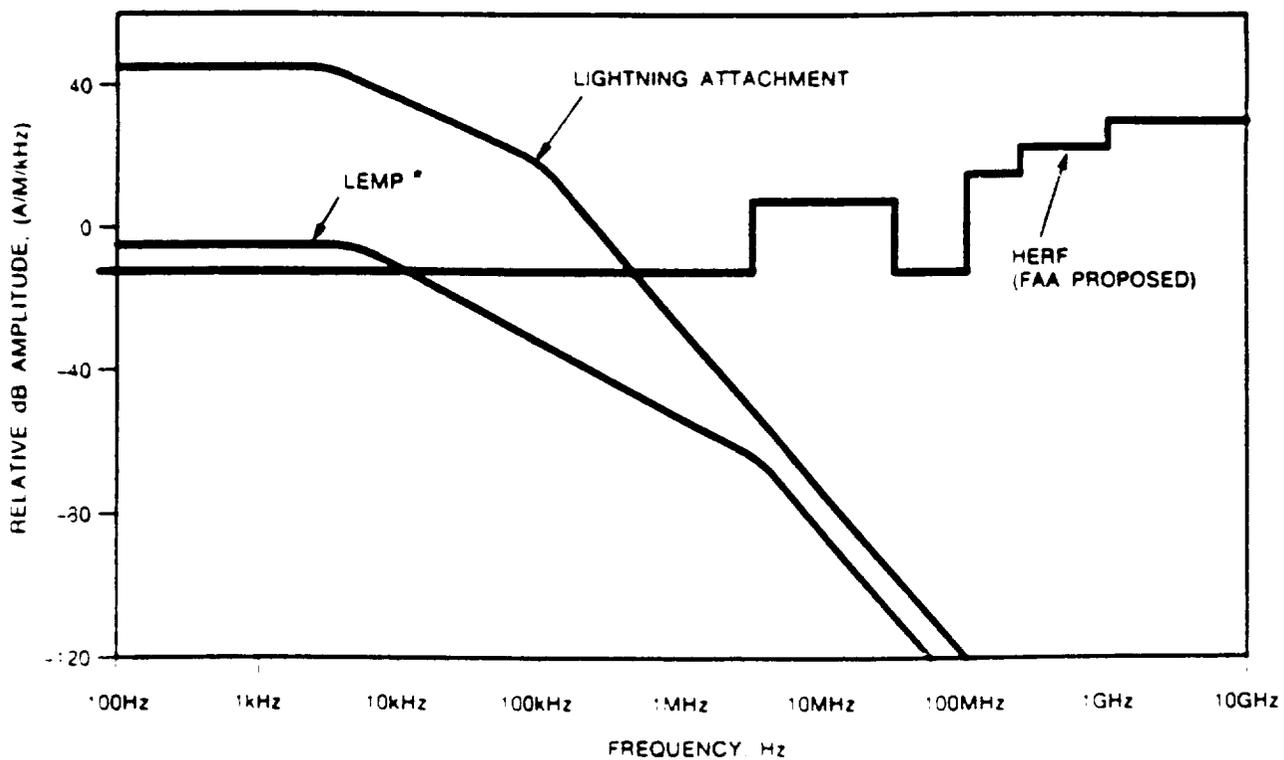
### 3.3.2 EME Threat (Continued)

Figure 3.3-6 shows (in the frequency domain) the lightning and HERF components of the EME threat. The points that make up each curve were derived by normalizing the threats to their equivalent magnetic field spectral density at a fixed bandwidth of 1 KHz (amps/meter/kilohertz). The HERF spectral envelope was derived by assuming a far-field source modulated 100% by a 800 Hz square wave with a carrier frequency equal to the frequency at which the evaluation is made.

### 3.3.3 Response to EME Threat

It is clear that the of EME threats to avionic systems, either digital or analog, are numerous. Although both types of avionics system respond to the same threats, there are factors that make the threat response to a momentary transient far more serious in digital systems than in analog. For example, the information bandwidth and, hence, the upper noise response cutoff frequency in analog devices is limited to, at most, 5 MHz, whereas in digital systems it is often in excess of 100 MHz. This bandwidth difference, which is at least ten times more severe in digital systems, allows substantially more energy and types of energy to be coupled into the digital system. Moreover, the bandwidths of analog circuits associated with autopilot and Flight Management Systems (FMS) are on the order of 50 Hz for servo loops and much less for other control loops (less than 1 Hz for outer loops). Thus, if the disturbance is short relative to significant system time constants, even though an analog circuit device possessing a broad bandwidth is upset by an EM transient, the circuit will recover to the proper state. Unlike analog circuits, digital circuits and corresponding computational units, once upset, probably will not recover to the proper state and will require external intervention to resume normal operation.

It should be noted that in older digital systems using discrete transistor devices the transient energy necessary to cause an upset is on the order of  $10^{-5}$  J. With the advent of VHSIC technology digital systems, upset occurs at only  $10^{-9}$  J or less. This means that advanced technology systems will be four orders of magnitude more sensitive to upset. In addition, these digital devices can now be upset by transient disturbances that last less than 2 nsec; previously, the disturbance had to last a few microseconds to cause upset. Such upsets refer to digital devices, that have been set to an illegitimate state which is correctable (the device is still operational, as opposed to, component damage which is a permanent failure state and the device is no longer operational), are, in turn, referred to as soft faults.



\* LIGHTNING ELECTROMAGNETIC PULSE (LEMP)

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FIGURE 3.3-6  
EME Threat Comparison

### 3.3.3 Response to EME Threat (Continued)

In a normal operating environment, the occurrence of soft faults within digital processing systems is relatively infrequent and random. The condition should be treated as probabilistic in nature. From this perspective, the projected effect of a substantial increase in the severity of the EME would be an increased probability of a soft fault occurrence. That is, in reality a soft fault may or may not occur at any particular point in time, but that on the average soft faults would occur more frequently with the new environmental level.

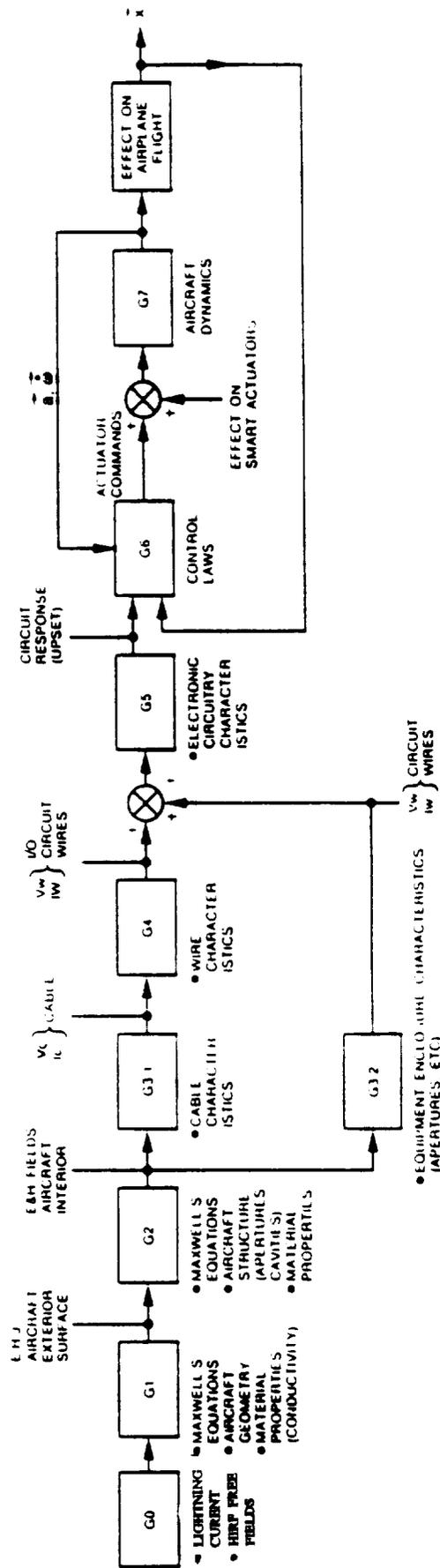
The criticality of functions would be the main factor that determines the degree of EME protection for the design associated with that function. To achieve a satisfactory degree of safety, it appears that all EME protection options available will be necessary for flight critical systems such as full authority digital engine control or FBW primary FCS.

### 3.3.4 System/Topology Approach to EME Immunity

A balanced approach, that would be conducive to an optimum protection design, is to view the protection objective from a system topology perspective. From this perspective, the aircraft is mapped into topology zones that contain a decreasing fraction of the energy and corresponding effect associated with the EME threat. These zones or regions are enclosed by boundary layers that provide the energy attenuation. Such boundary layers could even be extended to the regions within avionic equipment. Within the aircraft the layers would be provided through a variety of aircraft design options (e.g., line filters, isolation transformers, fiber optics, cable shield, equipment enclosures, and the airframe materials). Within equipment similar layers would be provided by equipment design options (e.g., interface circuit configurations, compartmentalization, shielded modules, optical couplers, circuit board layout, wire routing, hardware/software architectural provisions).

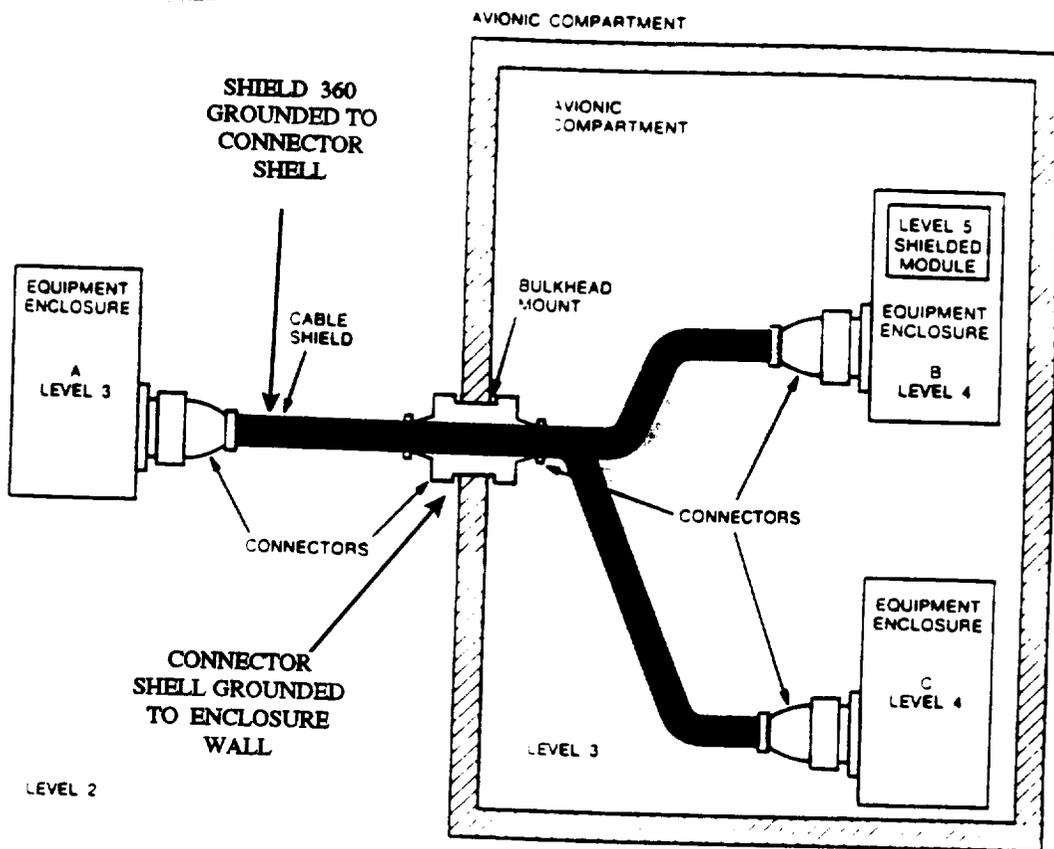
Figure 3.3-7 is a system level (block diagram) overview representation of the transfer processes associated with the interaction between EME and a flight critical system. Both the transfer process and the topology perspective emphasize the system level nature of the EME threat issue and they are complementary.

Figure 3.3-8 is a representation of an aircraft electrical/electronic system from a topological perspective. The figure shows typical items corresponding to electromagnetic considerations. Isolation of zones or topology levels must be maintained by isolating power ground from signal grounds, primary and secondary power isolation, single point circuit ground and cable shields peripherally terminated (not necessarily grounded), at a minimum, at both ends.

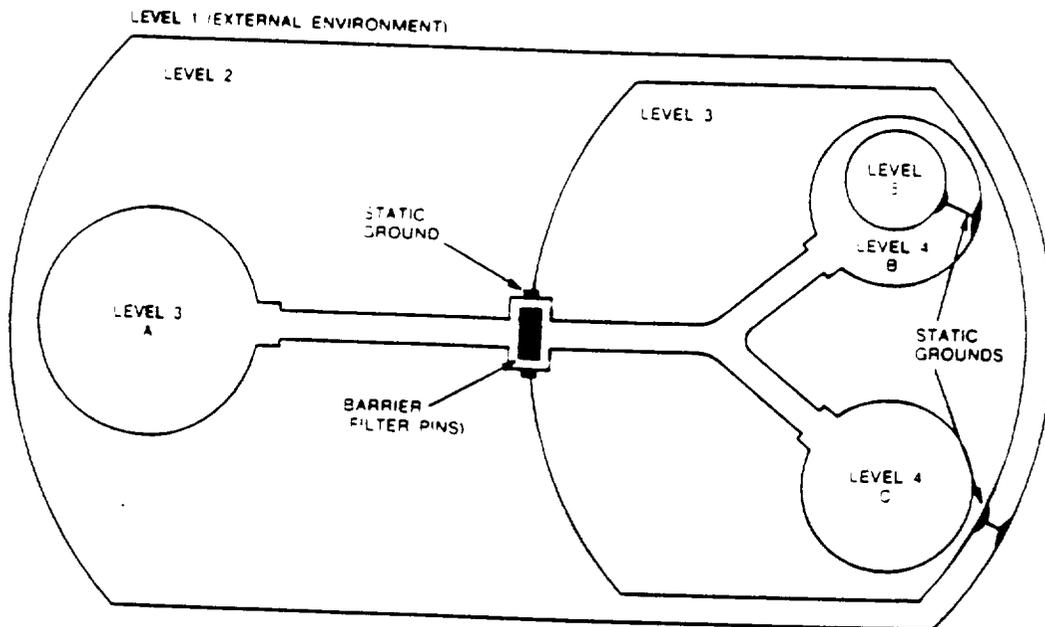


Electromagnetic Environment Transfer Processes  
Flight Critical Systems

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(a) INTERCONNECTED SUBSYSTEMS



(b) SHIELD TOPOLOGY

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FIGURE 3.3-8

System Topology Representation

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### 3.3.4 System/Topology Approach to EME Immunity (Continued)

Within the topology framework the effectiveness and necessity of various options could be assessed through appropriate trade studies.

Immunity of electronic components to damage is a consideration that occurs as part of the circuit design process. This circuit characteristic (immunity to damage) is influenced by a variety of factors: circuit impedances (resistance, inductance, capacitance) which may be distributed as well as lumped; the characteristic (surge) impedance of wiring interfacing with circuit components; properties of the materials used in the construction of a component (e.g., thick film/thin film resistors); threat level (open circuit voltage/short circuit current) resulting in a corresponding stress on insulation, integrated circuit (IC) leads, printed circuit (PC) board trace spacing, etc.; and semiconductor device nonlinearities (e.g., forward biased junctions, channel impedance, junction/gate breakdown). Immunity to upset for analog processors is achieved through circuit design measures, and for digital processors it is achieved through architectural as well as circuit design measures.

The use of composite materials in aircraft construction, electronic computers that use VLSI-type electronic devices, and reliance upon electronic computers for flight critical functions are recent technology trends that have greatly magnified the threat of EME relative to aircraft operations. As previously noted, although a momentary (less than 1 sec) threat may cause disruption in both analog and digital systems, analog systems generally resume normal operation when the threat is removed. Digital systems, on the other hand, when once perturbed, require external intervention to resume normal operation. Unless substantial fault tolerance is built into it, the digital system is far more susceptible to system upset by momentary threats than its predecessor analog device.

When the threat is not momentary, analog systems that contain VLSI devices (e.g., operational amplifiers, comparators) will, depending upon the amplitude of voltages/currents produced by the threat, be disrupted and may take a substantial length of time to recover to normal operation after the threat subsides. Digital system disruption is a probabilistic matter and may or may not occur. Thus, for nonmomentary EME threats, digital systems may offer more promise of achieving system immunity than their analog counterparts.

As a general rule, optimum EME protection (hardening) occurs when the protection burden is partitioned among the various options available. An approach to optimum protection design is to view protection from a system topology perspective. It should be noted once again that it appears that use of all possible options may be necessary to achieve an acceptable confidence level for

### 3.3.4 System/Topology Approach to EME Immunity (Continued)

critical systems (e.g., full authority engine control, FBW primary flight controls). If protection requirements are allocated in an optimum manner, the impact of EME protection on airplane costs (weight, power, performance, monetary) will be minimized and the degree of immunity to lightning and RF effects is maximized.

Ideally, system immunity to EME threat should be achieved through system design measures that provide sufficient inherent immunity so that reliance on dedicated protection devices can be minimized. Such protection measures tend to be self-monitoring through noticeable degradation or actual loss of system function when a protection measure has been degraded or lost.

When balanced protection is distributed throughout the various levels of a FBL system methodologies and capabilities, corresponding to the nature of such a protection strategy, must be available to verify/validate that the desired degree of immunity to the EME threat has been achieved. Such capability would need to be broad in scope and based upon a top down approach. It is perceived that assessments from the device/circuit level would need to be integrated into subsystem and system level assessments.

### 3.3.5 State-of-the-Art of EME Analysis for Aircraft

#### 3.3.5.1 The Role of Analysis

A fundamental understanding of the propagation of EM energy, is the contribution of an analytic resource to the EME effects assessment process. Analysis involves the systematic application of electromagnetic physics and associated mathematical models. Empiric (testing) activity contributes the data from electromagnetic measurement based upon experimentation under controlled laboratory conditions. There should be a complementary interplay between analysis and test throughout the assessment process.

Although analysis can now play a major role in aircraft validation, its potential has not been realized. However, the role of validation is expected to increase in the future. For several reasons:

- A. High fidelity mock-ups suitable for testing are frequently not available. It is expensive to develop mock-ups only for EM evaluation purposes. Also, testing of airworthy prototypes can usually only be done at reduced levels because of concern that the testing may damage the prototypes.
- B. Analysis is generally less expensive than testing.
- C. Analysis can be done using design drawings as input; the hardware need not exist, and parametric variations can be easily done.

### 3.3.5.1 The Role of Analysis (Continued)

- D. Because of the advanced state of numerical/analytical techniques relative to the limitations in test techniques and instrumentation, confidence in analysis results can be equivalent to that of test results.
- E. Analysis tools are expected to become more accurate in the future since the accuracy of the numerical tools is presently limited by available computer capability. Computer capability is rapidly increasing, which continues to make analysis more accurate and cost effective.

Because of the above reasons, analysis will play the following role:

1. Design: Analysis can be used to evaluate designs and perform trade studies. Analysis can be used to identify the critical areas which require more design attention and provide a high payback in EM hardness.
2. Definition of Line Replaceable Unit (LRU) Electromagnetic Specifications (EMS): Analysis can be useful in the development of voltage, current, and EM field specifications for LRUs.
3. Evaluation of EM Environment Mitigation Practices: EM environment mitigation practices, such as volume and cable shielding, and terminal protection designs can be evaluated by analysis.
4. Test Planning: It is not recommended that analysis completely replace testing. Both testing and analysis of systems or subsystems should be done, although the role of analysis will be increasing. The first role of analysis in this regard is to help with test planning. The analysis can identify potentially weak areas of the system design which should be further investigated by tests. The test setup can be analyzed and meaningful test points and expected response levels can be identified.
5. Data Interpretation: Because modern aircraft are such complex systems, test data is often difficult to understand, especially with CFC/metal structures. Analysis can help in understanding the data and the related physical EM interaction processes.
6. Aircraft Validation and Certification: Although partly alluded to in the previous discussions, it should be emphasized that analysis can be used to help validate and certify aircraft. It is expected that analysis will assume a bigger role in the future.

### 3.3.5.2 Analysis State-of-the-Art

The issues of interest for EM analysis include the coupling to the aircraft surface and structure, the penetration through seams and apertures, and propagation along and through cable bundles. Techniques exist for analysis of all these issues. The EM analysis approaches must also have the ability to study the peculiarities of lightning, including channel attachment geometries, as well as the RF illumination of an aircraft in free space or on the ground. Both transient and CW results are required.

It is not the intent here to provide a detailed technical review of each available analysis technique. Instead the purpose is to review the general capabilities and deficiencies.

First of all, in order to cover the lightning and HERF environments, one must note the large frequency range. The HERF environment is basically a CW or modulated CW having carrier frequencies between 15 KHz and 40 GHz. The lightning environment is a transient current waveform having significant spectral content from DC to about 30 MHz. There presently is not a single technique which can be used to completely solve the EM interaction problem for an entire aircraft all at once. However, certain related significant statements can be made:

- A. Numerical techniques exist which could completely solve the entire interaction problem all at once over the desired frequency range if there were unlimited computational power. That is, the present limit of the application of numerical solutions to Maxwell's equations for entire aircraft is caused by limits in computer memory and speed. For example, with available supercomputers (e.g., Cray II), the Three Dimensional Finite Difference (TDFD) approach can accurately determine the response from DC to about 200 MHz for commercial size aircraft. Results up to 100 MHz can now also be obtained with minicomputer technology. Accuracy of this approach has been validated in numerous instances. Significant amounts of detail can also be put in this model, including apertures such as windows, the structure of the interior, and internal cables. The inside, outside and cable propagation, are solved self consistently and simultaneously, thus eliminating two or three extra calculations. Both time and frequency domain responses are used to analyze aircraft sub-elements, such as LRUs, modules, PC cards, and localized Points of Entry (POE) on aircraft surfaces. Thus, an important part of the spectrum is already covered by existing technologies, and the upper frequency limit will increase with increased computer capability.

3.3.5.2 Analysis State-of-the-Art (Continued)

The required increase in computational power to completely solve a commercial aircraft up to 40 GHz can be estimated. For the TDFD approach, for example, the memory requirements go up as the cube of the upper frequency limit (for the same size object), and the computation time requirements increase as the fourth power of the upper frequency limit. Therefore, if present technology limits the upper frequency to 200 MHz, and the goal is 40 GHz (factor of 200) memory requirements will have to increase by a factor of  $8 \times 10^6$ , and computer speeds will have to increase by a factor of  $1.6 \times 10^9$ . These are huge numbers; however one may reasonably expect that within a few years, the upper bandwidth may be extended to 1 GHz (factor of 5), because of advances in supercomputer technology.

- B. It is not necessary (although it would be nice) that the entire aircraft interaction problem be solved all at once for all frequencies. As mentioned above, present techniques are band limited by computer capability to about 200 MHz, but fortunately, however, all of the major structural resonances of the aircraft are below this frequency. This means that for the higher frequencies it is not necessary to include the entire aircraft in the coupling problem. For example, coupling through a cockpit window with 10 GHz RF illumination is fairly independent of aircraft length and geometry, whereas this would not be the case at 3 MHz, especially if this were the first fuselage resonance frequency.
- C. Techniques are available which can be used to solve the upper frequency parts of the interaction. As mentioned, it is not necessary to have the entire aircraft in the problem space. In the example given in B. above, for example, the TDFD approach can be now used to solve the coupling at a more local level. Other techniques also exist such as Geometrical Theory of Diffraction (GTD), physical optics, method of moments, and finite elements.

In addition to the computational capability limitations, there are several other limitations as well:

- D. Present algorithms are not user friendly. Generally speaking, the use of sophisticated numerical analysis techniques is done by skilled EM theoreticians with extensive computer skills. Although there are on-going efforts to make these tools friendly, much work needs to be done in the following areas:
  - 1. Input of Geometry and Materials: Work needs to be done in this area in which the user enters the aircraft shape and materials. A CAD/CAM type of front end needs to be developed which would make this a relatively painless process. This would also include internal cables, and details of apertures, etc.

3.3.5.2 Analysis State-of-the-Art (Continued)

2. Debugging Capability: Capability needs to be built into the software to tell the user of faults within his problem. This can include help menus and information such as the viability of his input data, possible locations of sources of instabilities or inaccuracies, possible problems with boundary conditions, and others. The idea is to relax the requirement that the user be an expert EM technologist.
  3. Interactive Graphics Output: High quality interactive graphics output is required. 3D plotting routines, videos showing temporal response development, and similar capabilities are required to help the user easily visualize his data. Output points should be mouse or menu selected, instead of by inputting x, y, z coordinates.
- E. High frequency models have not been fully developed. As mentioned earlier, present techniques may now be accurately applied to jumbo jet size aircraft up to 200 MHz. This is primarily because much effort has been spent on developing models for lightning and Nuclear electromagnetic Pulse (NEMP), while the HERF environments have not been as thoroughly considered. Therefore, development and applications of models in these frequency regions to the aircraft hazard evaluation problem is rather new territory.
- F. The EM description of an aircraft is often not known in great detail. This problem is not associated with the techniques themselves, but is associated with the input data. For example, the layout of cables and branches within an airframe is not well controlled and is difficult to define. The termination impedances are also difficult to determine. The same is true with regards to bulkhead locations, box (LRU) locations, and the like. Even though the numerical methods may be capable of analyzing the configuration, the results cannot be more accurate than the input data.

Two aspects of this problem need some discussion. First, one may question the need to know this detail. Historically, the approach has been to make worst case or typical models, and this is generally thought to be adequate. However, if the results are too worst case, unacceptable hardening penalties may result.

Second, the approach has not been adequately verified on a full scale aircraft. Therefore the possible hardening penalties mentioned above have never been quantified.

### 3.3.5.2 Analysis State-of-the-Art (Continued)

- G. The models need extensive validation to be accepted by the community. Validation of the TDFD approach under a variety of circumstances has been accomplished. While the results from this validation effort have been encouraging, none of the cases examined included coupling to the cables and LRU pins within commercial aircraft. Also, validation was achieved at relatively low frequencies (appropriate for lightning and NEMP), and there is none at the higher frequencies. Therefore, an extensive effort must be made to validate the models and quantify the errors.

### 3.3.6 Approach for the Development of an EME National Resource

#### 3.3.6.1 Introduction

The proposed technical approach for this program is summarized in Table 3.3-1. The essential features of this approach are as follows:

- A. To provide a credible solid technical scientific basis for the EME national resource, the program emphasizes the development of first principle EME models and rigorous testing for validation.
- B. The development of a 1/10 scale model facility is recommended. This requires a scaling of the frequency along with the physical scaling of the model. A 1/10 scale model requires illumination frequencies that are ten times larger than actual frequencies for full scale aircraft. Thus, with a 40 GHz facility, scale model aircraft interaction studies can economically be accomplished for a variety of sources and aircraft geometries and orientations for frequencies below 4 GHz (1/10 scale illumination frequencies below 40 GHz).
- C. The scale model facility will also be used to evaluate interactions with full scale LRUs, cable assemblies, and printed circuit cards up to 40 GHz.
- D. A full scale test bed will also be used for validation at the low frequencies and also up to 40 GHz.
- E. In addition to the EME models, another valuable product will be produced, which is a definition of threat levels inside aircraft for lightning and HERF environments.
- F. The program's main output will be a set of scientifically validated user friendly EME interaction codes.
- G. LRU/aircraft test and verification methods will be developed.
- H. The project should be validated with the flight test aircraft in a ground test demonstration.

**TASK DESCRIPTION**

**OBJECTIVE**

|   |   |
|---|---|
| <p>1. <b>Development of Interaction Technology</b><br/>           1.1 Literature Review<br/>           1.2 Selection of Models<br/>           1.3 Numerical Model Development</p>   | <p>To determine and develop the best numerical models to be used in the National Resource</p>   |
| <p>2. <b>Scale Model and Sub-Element Facility Development</b><br/>           2.1 Requirements Definition<br/>           2.2 Hardware Design<br/>           2.3 Software Design<br/>           2.4 Measurement Tool Development up to 40 GHz<br/>           2.5 Procurement/Assembly<br/>           2.6 Calibration/Checkout</p>   | <p>To define and build a scale model facility to cover approximately 0 to 4 GHz for full size aircraft, and 0 to 40 GHz for sub-elements (LRUs, cables, assemblies, printed circuit boards, etc.)</p> |
| <p>3. <b>Scale Model/Sub-Element Validation</b><br/>           3.1 Design of Three (3) Physical Aircraft Models<br/>           3.2 Procurement/Construction of the Three (3) Models/Sub-Elements<br/>           3.3 Pretest Analysis<br/>           3.4 Test Planning<br/>           3.5 Test Conduct<br/>           3.6 Data Analysis<br/>           3.7 Reporting<br/>           3.8 Interaction Model Modification</p> | <p>To validate and improve EME interaction models and mitigation practices up to approximately 4 GHz for full size aircraft and 0 to 40 GHz for sub-elements</p>                                      |
| <p>4. <b>Full Scale Test Bed Development</b><br/>           4.1 Test Bed Aircraft Selection<br/>           4.2 Measurement Tool Development up to 40 GHz<br/>           4.3 Facility Design (Hardware/Software)<br/>           4.4 Assembly<br/>           4.5 Calibration/Checkout</p>   | <p>To Define and implement a full scale test bed to cover frequencies up to 40 GHz for full size aircraft</p>   |
| <p>5. <b>Full Scale Validation</b><br/>           5.1 Pretest Analysis<br/>           5.2 Test Planning<br/>           5.3 Test Conduct<br/>           5.4 Data Analysis<br/>           5.5 Reporting<br/>           5.6 Interaction Model Modification</p>   | <p>To validate and improve interaction models and mitigation models and mitigation practices up to 40 GHz for full size aircraft</p>  |
| <p>6. <b>Threat Level Definition</b><br/>           6.1 Review and Correlate Previous Test and Analysis Data<br/>           6.2 Develop a Set of Threat Levels for Cable Coupling, Internal Fields, Internal LRU Levels, Suitable as a Basis for a Test Standard</p>  | <p>To define a set of standard threat levels for cable coupling, internal fields, etc. to form the basis of an integrated EME Standard</p>  |
| <p>7. <b>Code Enhancement</b><br/>           7.1 Add CAD/CAM Front End<br/>           7.2 Add debugging/Help Features<br/>           7.3 Interactive Graphics Output</p>  | <p>To make the numerical analysis codes user friendly and well documented</p>   |
| <p>8. <b>Test Method Development</b><br/>           8.1 Define Methods Based on Threats<br/>           8.2 Bench Test Validation</p>  | <p>To develop test methods for LRU/aircraft EME testing</p>   |
| <p>9. <b>Flight Test Demonstration</b> TABLE 3.3-1</p>  | <p>To demonstrate that results and methodology apply to airworthy aircraft</p>  |
| <p>EME National Resource Development: Program Plan Overview</p>   |   |

### 3.3.6.1 Introduction (Continued)

Many of the items in Table 3.3-1 are self-explanatory. However, expansion of some of the key features is presented.

### 3.3.6.2 Interaction Technology Development

In Task 1, and in many of the following tasks, attention will be given to the development of numerical EME interaction models. At first a literature search will be done, with particular emphasis at the higher frequencies. A set of numerical models will be determined and developed. They will be modified as appropriate by the test results.

### 3.3.6.3 The Scale Model and Sub-Element Test Facility

#### 3.3.6.3.1 Purpose

This facility will conduct EME tests up to 4 GHz for 1/10 scale aircraft models in support of numerical models and to obtain data for establishing threat levels. The tests include CW illumination and direct current injection. In addition, full scale testing up to 40 GHz will be conducted for LRUs, cable assemblies, and printed circuit cards.

The facility description will address stimulus equipment, the physical scale models, measurement probes and sensors and data acquisition equipment.

#### 3.3.6.3.2 Stimulus

The stimulus equipment in the facility will generate and apply swept CW fields to reduced scale aircraft models. This equipment will be reconfigured for current injection to simulate lightning. The stimulus generation equipment consists of network analyzers and RF amplifiers.

The stimulus fields are applied to the models by means of a parallel plate line and a log periodic antenna. Lightning currents are simulated by direct injection of CW test currents into the model.

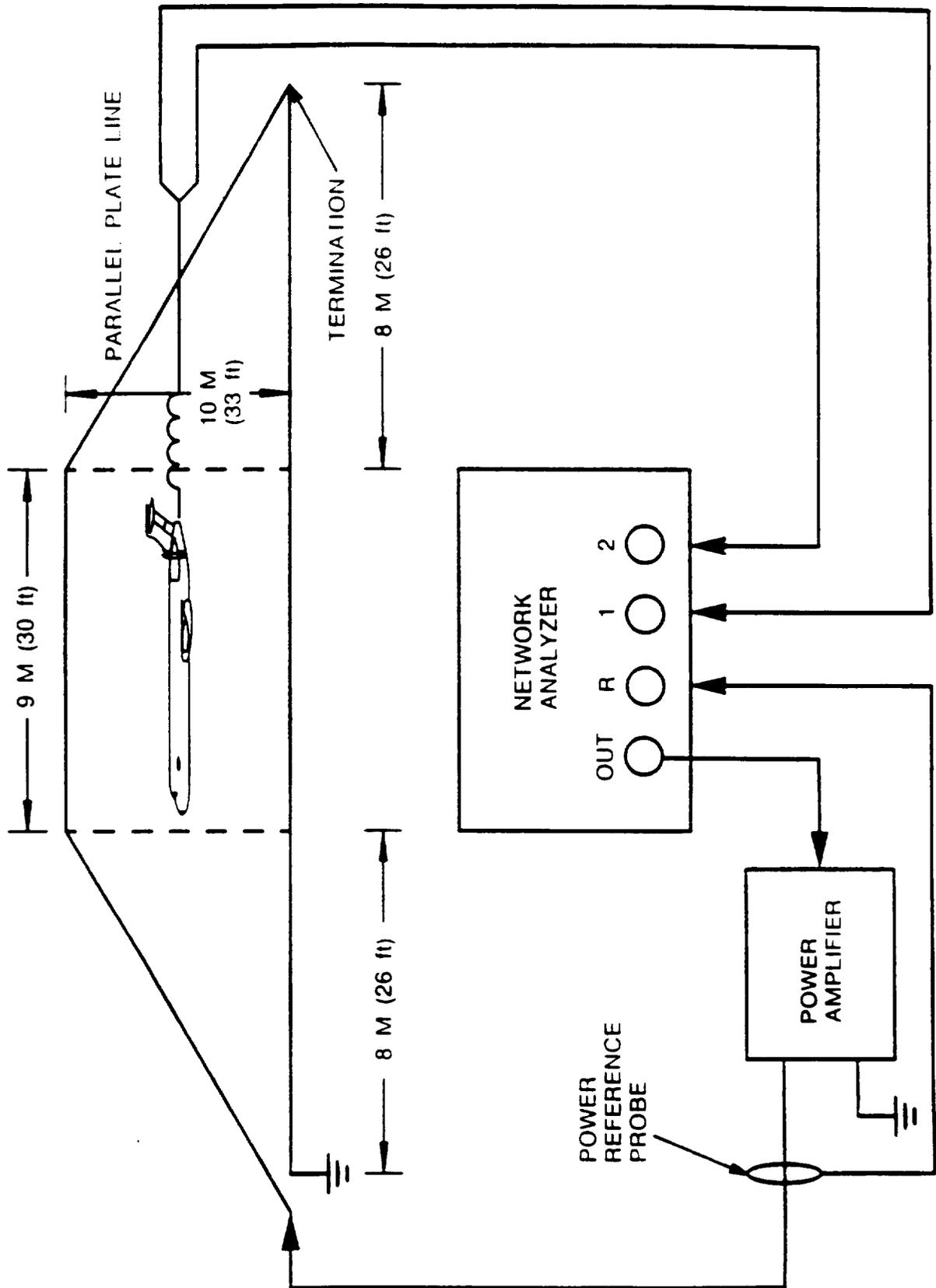
- A. Parallel Plate Line: The parallel plate transmission line shown in Figure 3.3-9 will be used for frequencies up to 100 MHz. The aircraft will be initially replaced by a reference probe to calibrate the illumination signal with the power reference. The reference probe will be inserted in a similar manner to the aircraft. Once the calibration has been obtained, the reference probe will be removed and replaced by the aircraft model. The dimensions of the line will be determined by pretest analysis to optimize the useful frequency range. Reconfiguration of the termination end by absorber materials and/or opening the end will extend the upper frequency limit.

### 3.3.6.3.2 Stimulus (Continued)

The response data from the aircraft can be taken out either by Fiber Optics Link (FOL) or by hard wire. The fiber optics approach is the best from a purely EM point of view. There are two disadvantages, however. The first is the cost associated with a FOL having several GHz bandwidths. The second is that the FOL has a rather high noise floor, thus limiting measurement dynamic range.

The data acquisition cables can be isolated from the test object by the use of ferrite beads and RF chokes. The choke can be made by coiling the cable around a ferrite material. Ferrite beads can be used directly on the cable for attenuation and isolation. The cable can be geometrically attached to the test object in such a way that coupling to the cable is very small. These attachment locations are at E-field nulls on the test object, and are determined by analysis. The cable will also be oriented normal to the incident field so that there is very little coupling to it. The current on the cable will be measured during test setup to verify minimum interaction.

- B. Antennas: A log periodic antenna will be used to illuminate the model, as shown in Figure 3.3-10, for frequencies above the capabilities of the parallel plate line up to 40 GHz. The model pictured in this figure will be initially replaced by a reference probe to calibrate the illumination signal with the power reference. The reference probe will be inserted in a similar manner to the aircraft. Once the calibration has been obtained the reference probe will be removed and replaced by the model. Data acquisition can be hard wired, with the same considerations as described for the parallel plate line, above.
- C. Simulated Lightning Current Injection (up to 300 MHz): Coaxial cables will connect the output of the RF amplifier to the aircraft models as shown in Figure 3.3-11. The point of connection will be at a probable lightning attachment location. Again, data acquisition will be hard wired. This is easier in this case than it is for the illumination cases. Both the reference injected current and the test point responses can be carried on shielded coaxes contained within an overall shielded cable which forms the lightning current detachment point, as shown.



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FIGURE 3.3-9

Parallel Plate Transmission for Frequencies Up to 100 MHz

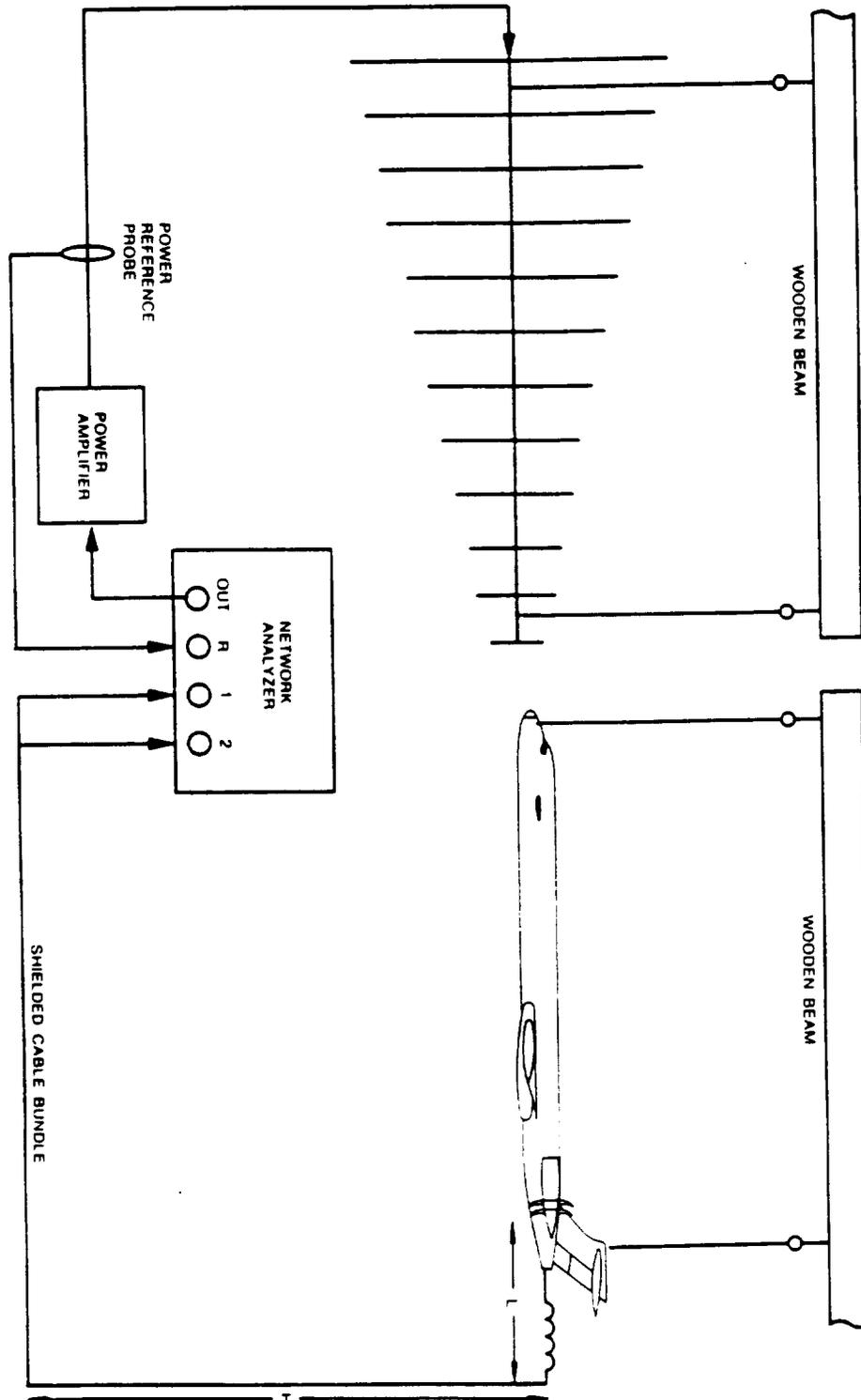


FIGURE 3.3-10

Log Periodic Antenna for Frequencies from 100 MHz to 40 GHz

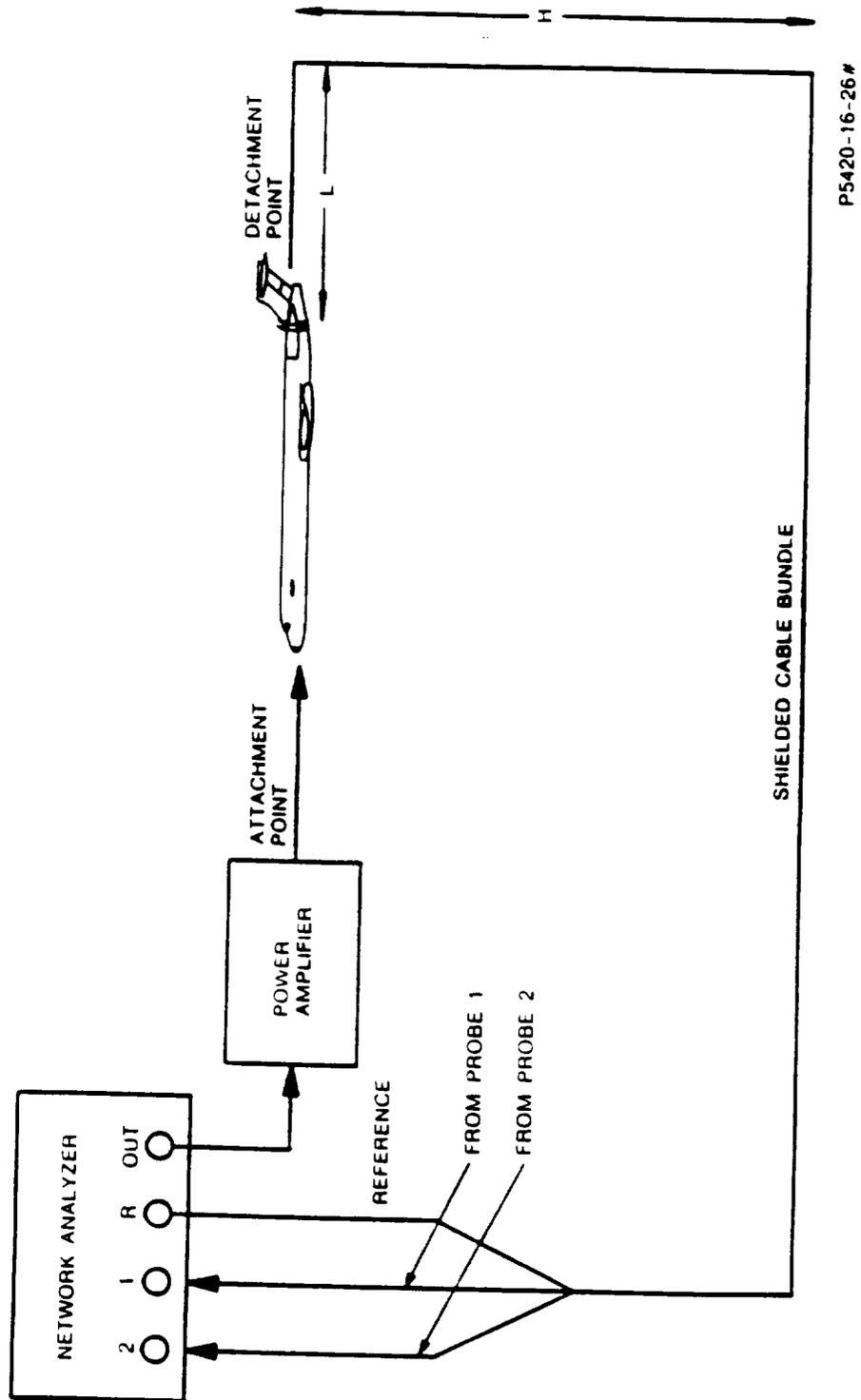


FIGURE 3.3-11

Simulated Lightning Current Injection Set-up

### 3.3.6.3.3 Aircraft Models

Three aircraft models will be used to determine current and voltage amplitudes and electric and magnetic fields at the avionics. The aircraft to be modeled will be a 747 or a DC-10 size, an MD-80 size, and a business size. It is possible that the coupling can be studied for the smaller aircraft at higher frequencies than for the large aircraft due to a larger scale size for small aircraft. A B-747 has a length of about 70 M, which results in a 7 m fuselage for a 1/10 scale model, thus creating the need for a large parallel plate line to study all significant angles of illumination. Scale size issues will be resolved during the planning activity. The models will have scaled apertures, cables, and internal LRUs and will be as high fidelity as possible. Skin thicknesses will not be scaled, and the models will not be used to study diffusion or current redistribution effects. These effects are especially important for lightning and will be studied with full scale test articles.

The reduced scale will allow the model to be oriented in each of three orthogonal directions in the transmission line and antenna illumination cases, thereby assuring that all directions and polarizations are provided. The longest dimension on the model will be no larger than half the shortest dimension in the active area in the cell to minimize the field perturbations. The model will be large to provide adequate internal volume for placement of the measurement probes and sensors.

### 3.3.6.3.4 Measurement

The use of scale models requires that the stimulus frequency be extended by the corresponding amount (i.e., 1/scale factor). Therefore, measurement instrumentation must have adequate frequency response. In order to obtain currents, voltages and fields, the measurement probes and sensors will be made.

- A. Current and Voltage Probes: Because the avionics system is the object to which the internal fields inside couple, the short circuit current and open circuit voltage probes will be built as simulated scaled avionics as shown in Figures 3.3-12 and 3.3-13. Other avionics boxes are simulated by the load on the other end of the simulated cables. The current probe utilizes a 1 ohm resistor shunted to ground. The current measured by a 50 ohm input network analyzer will thus be 1/50th of the actual current value. The voltage probe uses a 1 k ohm resistor in series with the simulated cable to reduce the current and increase the voltage thereby approaching the open circuit voltage value.

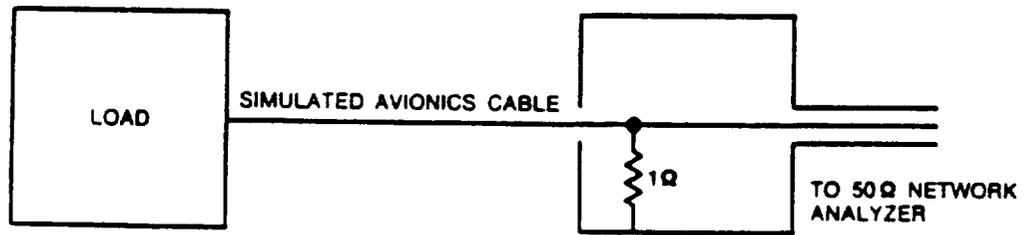


FIGURE 3.3-12  
Short Circuit Current Probe

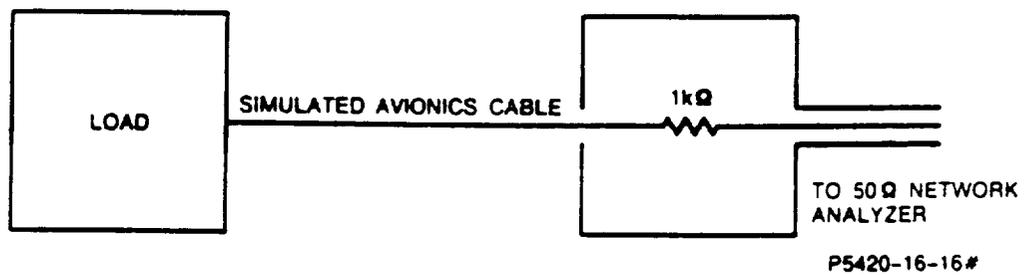


FIGURE 3.3-13  
Open Circuit Voltage Probe

#### 3.3.6.3.4 Measurement (Continued)

- B. Field Sensors: Field sensors will measure the fields at the avionics box and act as a reference for the network analyzer. The electric and magnetic field sensors are illustrated in Figures 3.3-14 and 3.3-15. The electric field probe is monopole above a ground plane where the center conductor of the cable lead is the antenna and the avionics box makes up the ground plane. Two such sensors will be needed. The first, as shown in Figure 3.3-14, will be rotated by 90 degrees for two orthogonal electric field components. The second probe will be similar to the first except that the coax conductor will pass straight through the box and penetrate the vertical wall on the opposite side. This sensor will measure the third orthogonal electric field component. The magnetic field sensor is a semicircular current loop. Three such sensors will be needed, one for each of the three orthogonal field orientations.
- C. Probe and Sensor Cables: The cables which conduct the probe and sensor signals to the network analyzer will be made of SMA semi-rigid coax. The points on the scale model where these cables penetrate the skin will be chosen to minimize the perturbations of the surface current density and where the minimum electric fields have been identified by numerical modeling results. The points chosen will be different for different scale model orientations in the illuminating fields and for the injected current configurations. The shields on the cables will be shorted to the inside surfaces of the scale model metal struts. The cable exit from the scale model will be through a connector jack placed on the outside surface of the skin. When a given connector is not in use a metal cap will cover the connector thereby shielding the center conductor of the probe/sensor from stray fields. For the cable shield outside the scale model, an RF choke will be incorporated into the cable. Within 3 inches ( $G/4$  for  $f = 1$  GHz) of the connector the semi-rigid coax cable will be coiled into several turns to block the RF signal on the shield of the coax cable. The optimum inductive reactance will be determined, but as an example, 6 turns with radius of 25 cm will provide 50 ohms at 10 MHz. In order to further minimize the RF signals coupled to the cable shields, the cables will be laid out for the parallel plate line and the log periodic antenna illumination areas with orientations perpendicular to the electric fields produced by the test equipment.

#### 3.3.6.3.5 Data Acquisition

The data acquisition system will consist of the receive circuits of the network analyzers, attenuators, a personal computer based system controller/VAX terminal and interface and local mass data storage. An 80 dB screen room will contain this system as discussed in Subsection A. below. The network analyzers will be selected during procurement of the data acquisition equipment.

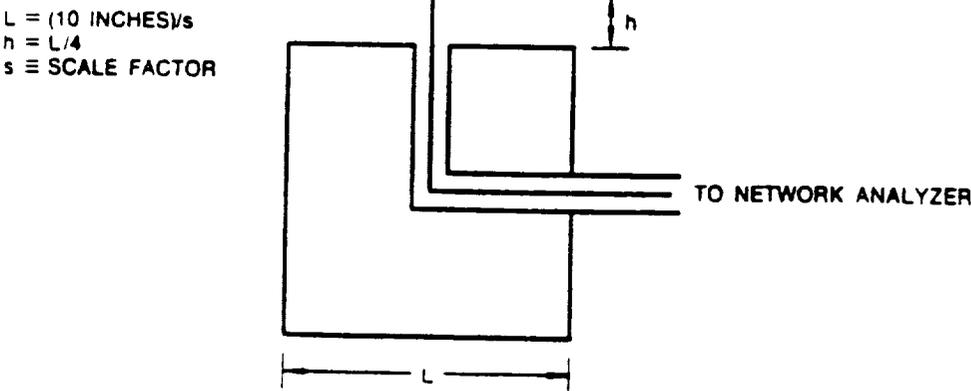


FIGURE 3.3-14  
 Electric Field Probe

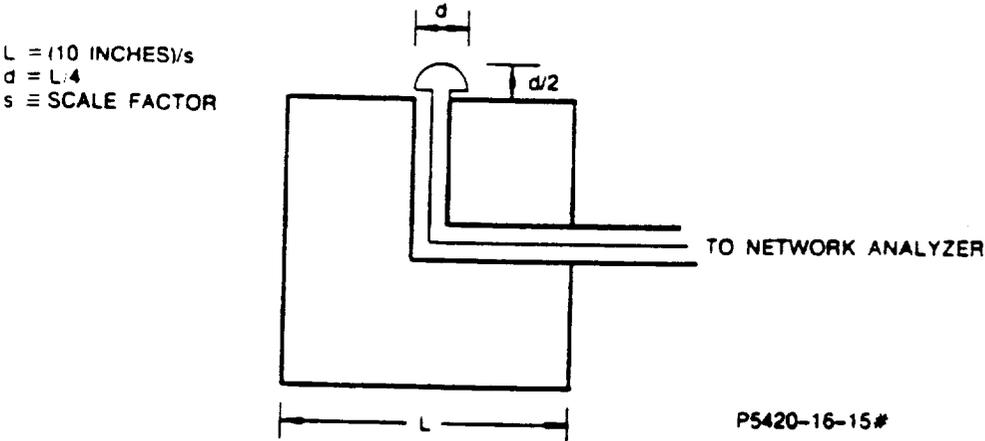


FIGURE 3.3-15  
 Magnetic Field Probe

#### 3.3.6.3.5 Data Acquisition (Continued)

Figure 3.3-16 shows a functional block diagram of the data acquisition system.

- A. The data acquisition system will be contained in an 80 dB shielded room or enclosure. The shields of the cables that pass through the enclosure wall will be shorted to the inside or outside of the wall for cables on the inside or outside, respectively. The enclosure will be adequately grounded.
- B. Attenuators: 60 dB of switchable attenuation will be provided for the inputs to the network analyzer. This will assure a safety margin for the network analyzer.
- C. Data Acquisition Controller: The system controller will be personal computer/microcomputer based. The network analyzer and attenuator control interfaces will be IEEE-488 implementations. The controller will be programmed to control the test initiation, the frequency step size and sweep speed. Signal to noise ratios will be monitored and the attenuators adjusted for optimum S/N if adequate signal levels can be maintained. At the completion of the sweep the data will be stored on local mass storage and then uploaded to the VAX computer by means of RS 232 serial interface in the personal computer. Custom software will be written to implement the control, monitoring and data manipulation.

#### 3.3.6.3.6 Full Scale Sub-Element Testing

The facility would also have the capability of testing aircraft sub-elements at full scale up to 40 GHz. One can evaluate cable coupling, mitigation techniques, shielding of LRU enclosures, and coupling to printed circuit (pc) boards.

#### 3.3.6.4 Full Scale Validation

A full scale test bed will need to be developed to extend the coupling validation to aircraft at the higher frequencies. Much of the same instrumentation previously obtained can be used here. The full scale test bed will also be used to spot check some of the lower frequency scale model results at lower frequencies.

A full scale test bed is envisioned to consist of a full size aircraft (not necessarily airworthy), plus perhaps some full scale major aircraft structures (such as a wing). Here is where the impact of composite structures will also need to be examined.

#### 3.3.6.5 Threat Level Definition

One of the outputs of this effort will be a detailed knowledge of internal EM environments, such as the current, voltages, and field levels. These levels can then be used to define waveforms suitable for LRU standards and testing specifications.

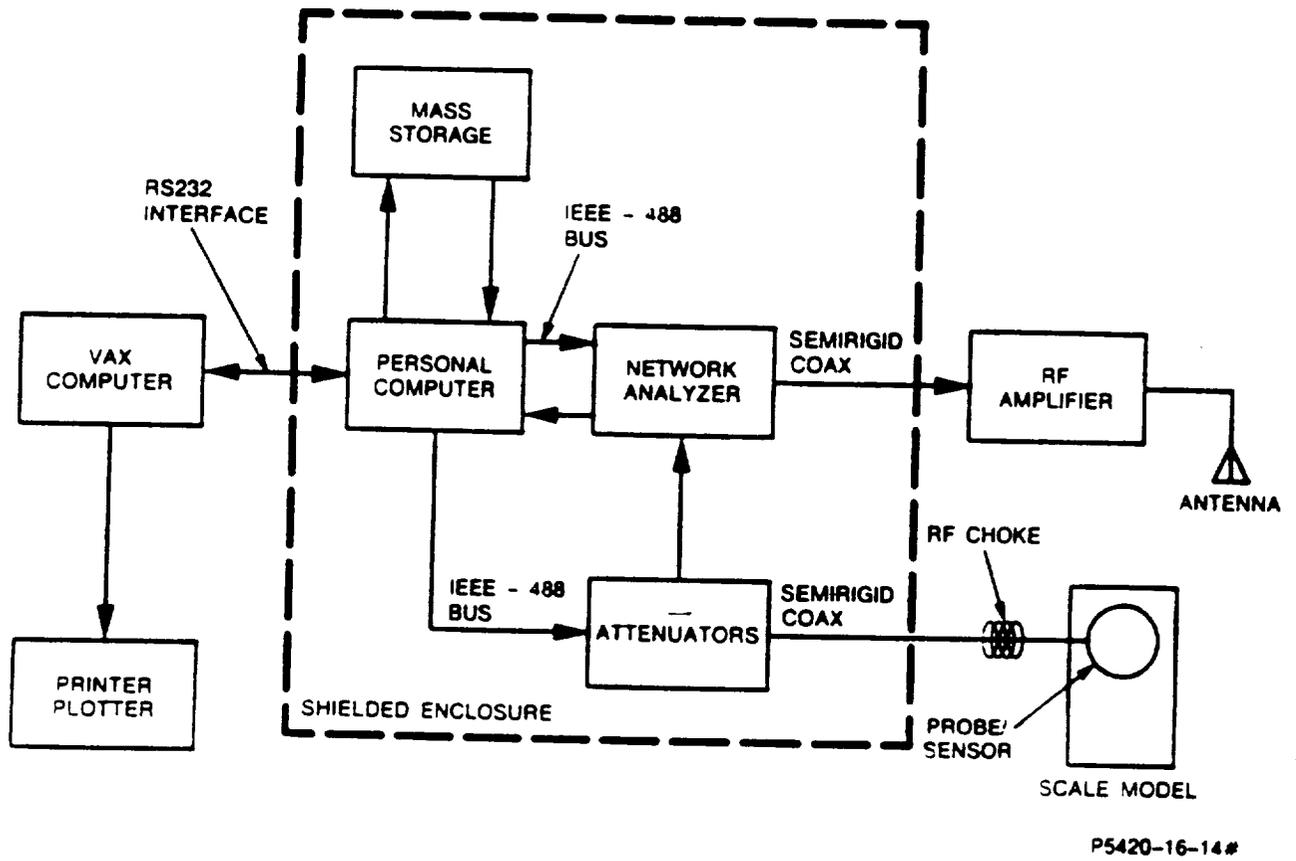


FIGURE 3.3-16  
Functional Block Diagram of the Data Acquisition System

3.3.6.6 Code Enhancements

Once the technical features of the code are in place, the next step is to make them user friendly as described in Task 7. The intent here is to make the codes easily usable to someone who is not an EM theoretical expert.

3.3.6.7 Test Method Development

Another major output of this effort is the development of aircraft EME test methods. The intent here is to define LRU specifications and test methods so that entire aircraft testing will not be required for certification.

3.3.6.8 Aircraft Demonstration

The methodology will be applied to the EME hardening of a demonstration aircraft. Its EME hardness will be demonstrated by the analysis and test methods developed under this effort. The approach here is to perform the EME demonstration with a ground test that covers HERF and lightning. A full up airworthy vehicle with all systems powered and functional will be the test object. Test planning and analysis activities will be done to ensure proper test configurations and procedures so that the test results are applicable to an in-flight aircraft.

3.3.7 EME Protection Technologies/Approaches

Listed below are some EME protection approaches.

CONVENTIONAL

Shielding  
Filtering  
Redundancy  
Mechanical Controls  
(including Hydraulic)  
Diversion Paths  
Grounding  
Balanced Circuitry  
Electrical Bonding  
Location

EMERGING TECHNOLOGIES

Fiber Optics  
Transparent Recovery  
Software Tolerance  
Reconfiguration  
Hardened ICs  
Conductive Composites  
Immune Electronic Backup  
Optical Computing and Storage

3.4 Fiber Optics/Optics

Over the past decade the effort to develop aircraft suitable fiber optics has grown exponentially. These efforts have provided aircraft systems designers with a host of new fiber optic components, cables, sensors, switches, and data links/buses needed for viable FBL systems. Table 3.4-1 provides a very partial list of fiber optic developments.

FIBER

High Temperature Glass and Plastic  
Improved Bend Radius  
Radiation Hardened  
Polarization Preserving  
Ribbon

CONNECTORS

Vibration and Shock Resistant  
Repeatable Terminations  
Crimp and Cleave  
Environmentally Sealed  
Temperature Insensitive  
Ribbon  
Multipin  
MIL-STD  
Automated Termination  
Reduced Loss

COUPLERS

Temperature and Shock Resistant  
Etched Glass  
Temperature Insensitive  
MultiChannel Wavelength Division Multiplexers  
Improved Packaging  
Reduced Loss

EMITTERS/DETECTORS

Improved Launch Power/Sensitivity  
Reduced Electrical Power Consumption  
Reduced Temperature Drift

SENSORS AND SWITCHES

Viable Optical Powering  
Digital and Analog Techniques  
Improved Performance and Reliability  
Parameter Sensitive

OTHER

Integrated Optics  
Smart Skins  
Optical Computing

TABLE 3.4-1

Partial List of Fiber Optic Developments for Aircraft

### 3.4 Fiber Optics/Optics (Continued)

While current fiber optics for aircraft have ample room for improvement in terms of manufacture, installation, reliability, and maintainability, they are mature enough for implementation on a limited basis to address some particularly troublesome problems.

A wide variety of potential benefits are commonly attributed to the application of fiber optics as noted in Table 3.4-2; but EMI protection, potential weight savings, performance requirements, and higher data rates are driving FBL.

#### 3.4.1 Optical Data Bus

The following fiber optics data bus issues have received a great deal of attention, but have not been resolved:

- A. Splitters/Coupler: Fused couplers are sensitive to humidity. Grating types are sensitive to vibration and shock.
- B. Light Sources/Detectors: Temperature range is a significant problem. LEDs lack reliability at high temperatures, while lasers and photo detectors exhibit wavelength drift with temperature variations. Peltier elements are sometimes required for cooling or temperature stabilization. They are unreliable and introduce added complexity to a system. For wavelength discriminating sensor methods, a reliable, stable, solid state broadband source is needed.
- C. Connectors/Terminations: Further progress must be made before fiber optics can be generally considered for commercial aircraft installation. Generally the available systems lack reliability; some specific problems are listed below:
  - 1. High insertion losses
  - 2. Loss variability from connector to connector
  - 3. Unrepeatable losses for reconnect
  - 4. High aging losses
  - 5. Connector back reflection
  - 6. Sensitivity to small amounts of contamination

A major concern is that current technology, particularly with regard to connectors, uses adaptations of electrical technology which do not address the unique properties of optical fibers. In addition, there is a lack of standardization among airframe manufacturers and fiber optic technology developers.

| <u>SIMPLIFICATION BENEFITS</u> | <u>PERFORMANCE BENEFITS</u>         |
|--------------------------------|-------------------------------------|
| WEIGHT                         | ELECTRICAL ISOLATION                |
| LESS CABLE                     | NO SPARK OR FIRE HAZARDS            |
| LIGHTER CABLE                  | NO SHORT CIRCUITS                   |
| NO CABLE SHIELDING             | NO GROUND LOOPS                     |
| FEWER CONNECTORS               | NO CROSS-TALK BETWEEN CABLES        |
| RELIABILITY                    | IMMUNITY TO EMI                     |
| FEWER TERMINATIONS             | IMMUNITY TO LIGHTNING SURGE CURRENT |
| CORROSION-RESISTANT CABLES     | WIDE BANDWIDTH                      |
| INSTALLATION COSTS             | GREATER TRANSMISSION SECURITY       |
| FEWER RUNS                     |                                     |
| FEWER CLAMPS                   |                                     |
| MATERIAL COSTS                 |                                     |
| LESS CABLE                     |                                     |
| FEWER CONTACTS                 |                                     |
| DESIGN SIMPLIFICATION          |                                     |
| FEWER SIGNAL PATHS             |                                     |
| SIMPLER WIRING DIAGRAMS        |                                     |

TABLE 3.4-2

Variety of Benefits Attributed to the Application of Fiber Optics

### 3.4.1 Optical Data Bus (Continued)

Glass fiber is inherently fragile as compared to copper wire. This is a factor which must be examined in attempting to adapt fiber optics to aircraft. Currently copper wires are assembled into bundles using tie-wraps every few inches. The bundles are installed into the aircraft sections after which the sections are joined and the wires mated via connectors. Repair can be performed by slicing or in some cases by pulling a new length of wire.

These installation and repair methods may not be acceptable for fiber optics. The present system requires connectors at every aircraft section and optical systems are sensitive to connector losses. Tie-wraps may induce microbending losses. Efficient, reliable splicing techniques for optical fibers have not yet been demonstrated, and pulling a fiber through a tightly wrapped bundle may cause breakage.

Some work is being advanced in the area of fiber optic connectors. DAC is currently reviewing fiber optic ribbon cables (12 optical fibers in a 1/4 inch wide ribbon). All 12 fibers are terminated as a unit with etched silicon wafers and then polished. A 1/3 inch diameter cable with 144 fibers weighs 1.4 pounds per 100 feet. This order of density is not possible with copper wire due to cross talk. The application of integrated optics to connector technology is a positive step in applying optics to aircraft use.

Some of the other shortfalls of fiber optics in aircraft applications are:

- A. Large differences in path losses of linear and multistar networks.
- B. Optical receiver limitations with regard to these large and varied path losses.
- C. Optical transmitter limitations of coupling sufficient power into lossy networks.
- D. Difficult network installation and uncertain reliability in avionic environments.

First, the limitations of linear networks will be discussed. There are four basic linear topologies in consideration for a FBL System (see Figures 3.4-1 through 3.4-4). The shortfalls of these configurations are high path losses and large differences in path loss between adjacent terminals and widely separated terminals. The number of terminals which can be implemented in a linear configuration is limited due to receiver sensitivity and dynamic range, low coupled power from optical transmitters, connector loss and a system margin for aging.

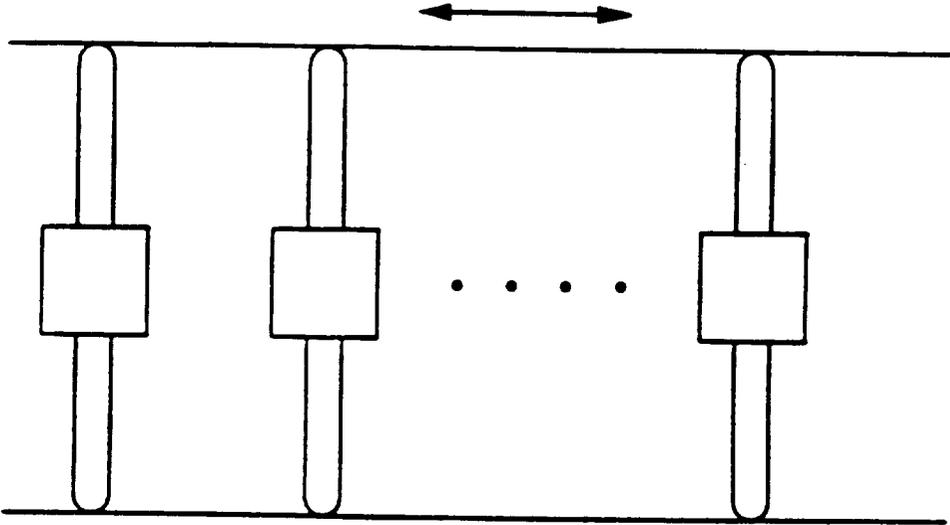


FIGURE 3.4-1  
Dual K Configuration (Two Bus Fibers)

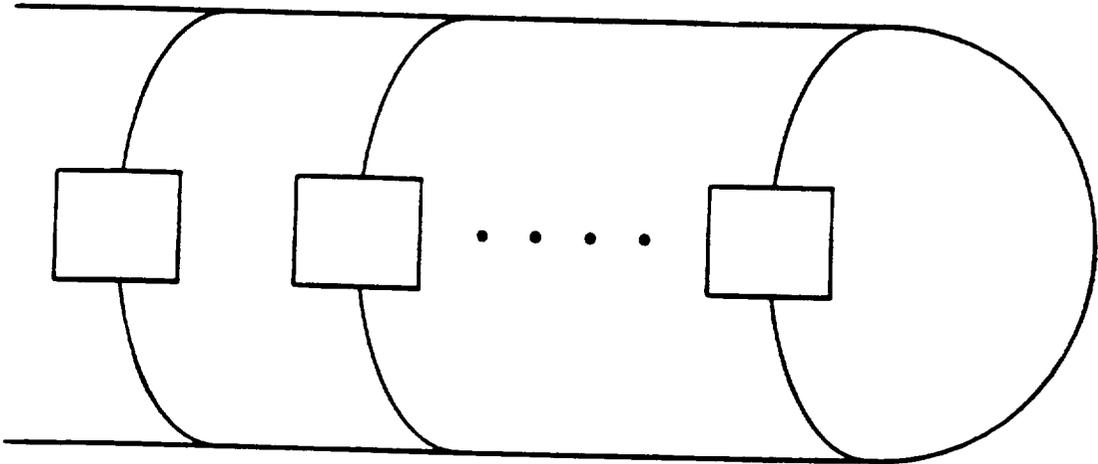


FIGURE 3.4-2  
Linear Loop Configuration

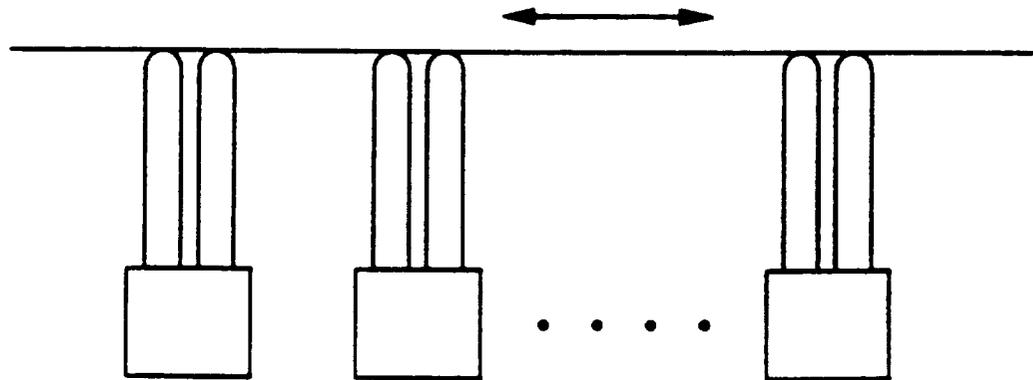
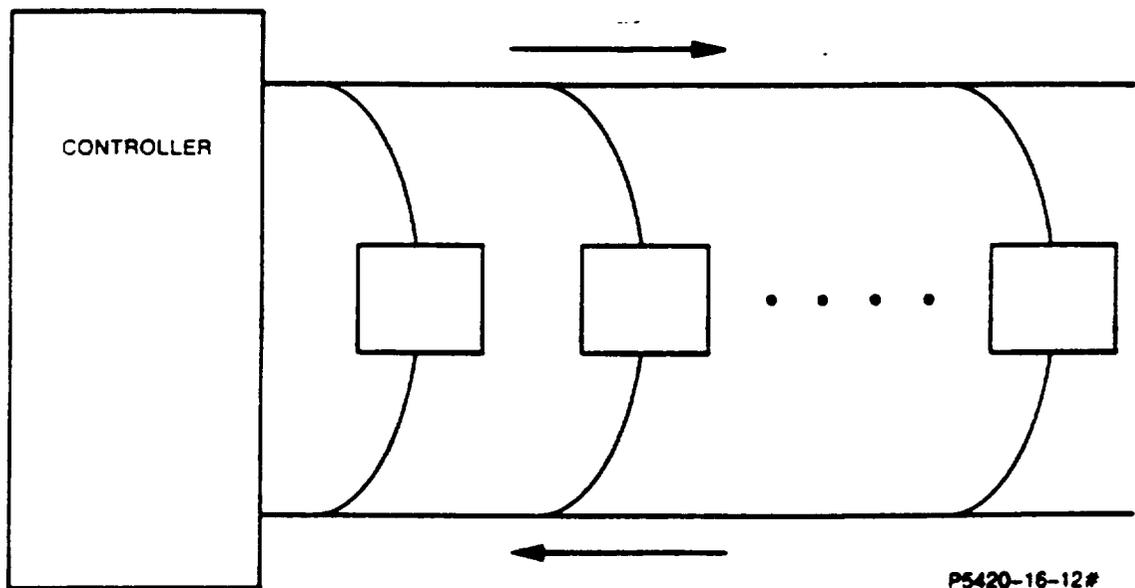


FIGURE 3.4-3

Dual K Configuration (One Bus Fibers)



P5420-16-12#

FIGURE 3.4-4

Broken Linear Loop Configuration

### 3.4.1 Optical Data Bus (Continued)

To analyze this problem quantitatively, a computer program for which the data is presented in Figure 3.4-5, has been developed by DAC. This program makes three assumptions about the couplers in the network:

1. All couplers in a given network have identical insertion loss.
2. The asymmetry of the couplers is constant, regardless of the tap ratio.
3. The efficiency of the couplers is constant, regardless of the tap ratio.

Our discussions with suppliers and lab tests to date indicate that these assumptions are roughly true (+-15%) and adequately conservative for the typical tap ratios which will be used (10-20 dB).

State-of-the-art multimode fiber optic couplers can be produced at any tap ratio between 3 and about 40 dB. In a multiterminal data bus there will be some optimum tap ratio such that sufficient energy is tapped from the bus but not so much that energy is depleted for terminals further down the bus. The optimum ratio is a function of the number and efficiency of the couplers on a bus and the configuration utilization.

Applied to a typical FBL system (25 terminals), Figure 3.4-5 shows the relationship between the maximum loss budget (loss between Terminal 1 and 25) and the tap ratio for a dual K configuration (Figure 3.4-6) with the characteristics noted in the figure. The graph indicates the minimum loss is 48 dB and will occur when a 14 dB tap ratio is specified. Figure 3.4-7 shows the loss profile of the dual K linear data bus which has the same characteristics as the bus of Figure 3.4-5 and utilizes the 14 dB optimum tap ratio. The discontinuities indicate the five bus connector locations (each tap fiber has one connector also). Also indicated in this figure is a typical wrap around loss (insertion loss between the transmitter and receiver of a given terminal) for this configuration. Two connectors are included in the wrap around path. The importance of wrap around will become apparent in a subsequent discussion of optical receiver limitations.

Closely related to the previous network loss budget considerations are optical transmitter and receiver limitations. The transmitter and receiver must operate properly with both the minimum and maximum path losses of a given network. The critical requirements are the amount of optical power which can be launched into the network from the transmitters, and the capability of the receivers to detect a wide range of optical power levels as a result of the varied path losses of the linear networks indicated.

25 NODE LINEAR NETWORK LOSS CHARACTERISTICS

|   | OPTIMUM<br>TAP RATIO (dB) | MAXIMUM<br>LOSS (dB) | MINIMUM<br>LOSS (dB) | COMMENTS  |
|---|---------------------------|----------------------|----------------------|---|
| DUAL K WITH<br>TWO BUS FIBERS<br>(FIGURE 1) | 14                        | 48                   | 24                   | COMPLETE INSTALLATION<br>REQUIRES TWO NETWORKS  |
| DUAL K WITH<br>ONE BUS FIBER<br>(FIGURE 3)  | 18                        | 57.5                 | 24                   | MAX LOSS INCLUDES<br>50 0.1 dB SPLICES AND<br>50 COUPLERS<br>(TWO OF EACH PER TERMINAL)           |
| LINEAR LOOP<br>(FIGURE 2)                   | 17                        | 67.5                 | 34                   | MAX LOSS INCLUDES<br>10 BUS CONNECTORS<br>(5 ON EACH SIDE)<br>AND 50 SPLICES<br>(25 ON EACH SIDE) |
| BROKEN<br>LINEAR LOOP<br>(FIGURE 4)         | 17                        | 37                   | 12                   |   |

FIGURE 3.4-5

Results of Calculations for Each Linear Configuration

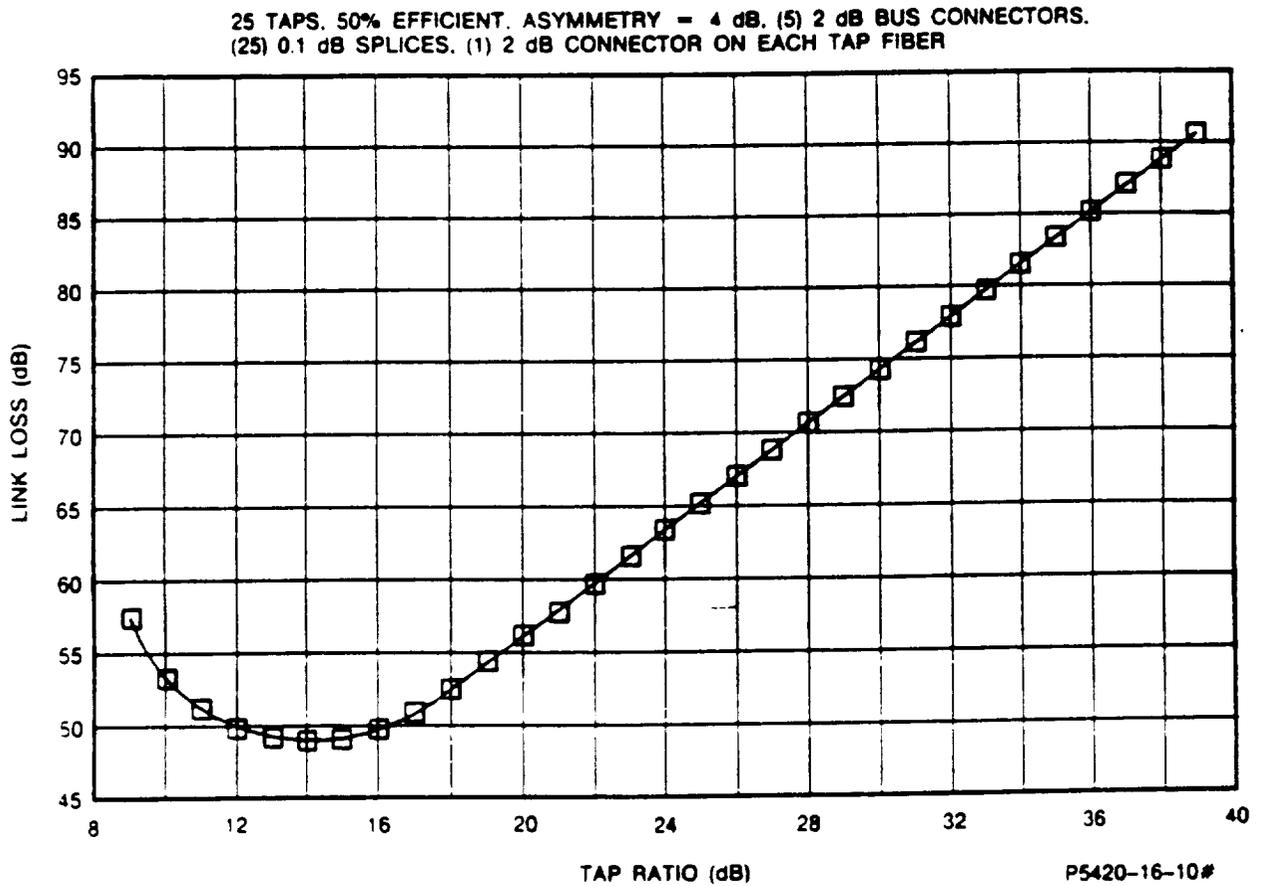


FIGURE 3.4-6  
Dual K Fiber Link Loss Vs. Tap Ratio

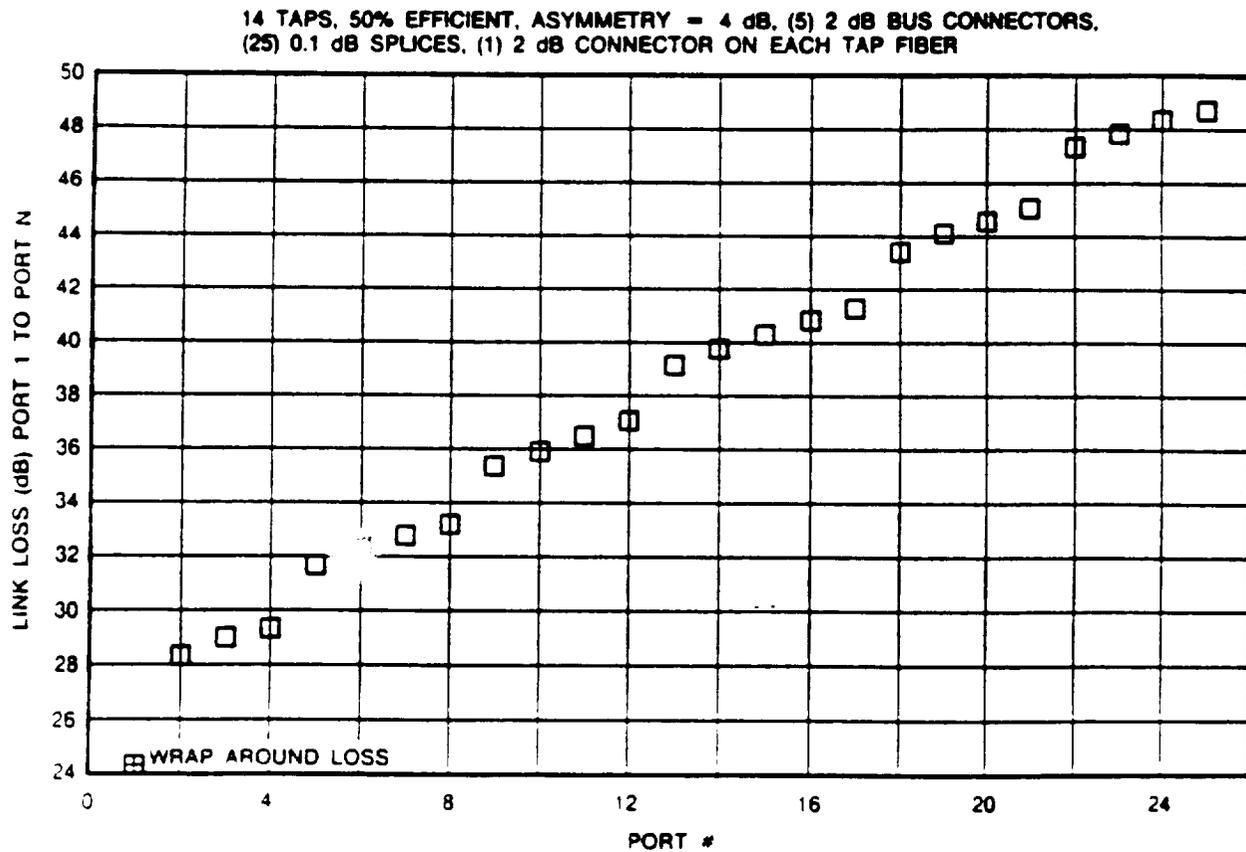


FIGURE 3.4-7  
Dual K Fiber Optic Link Loss Profile

### 3.4.1 Optical Data Bus (Continued)

The launched power is limited by the fiber core size and the type of optical source used. Candidate optical sources include laser diodes, Burrus LEDs, edge emitting LEDs, and superluminescent LEDs. Although the laser diode has the greatest launched power capability, the drive circuitry is complex and must be temperature stabilized. Typically thermoelectric coolers, which have large power consumption, are used for temperature stabilization. The laser diode is generally less reliable ( $10^6$  hours compared to  $10^7$  hours) and more costly (\$500-\$2,000 compared to \$10-\$300) than a Burrus or edge emitting LEDs and presents a safety hazard to the user. The superluminescent LED is also less reliable and more costly, leaving the Burrus or edge emitter as the only practical alternatives for avionic data bus applications. The edge emitter, although more expensive than the Burrus LED (\$200-\$300 compared to \$10-\$100) generally couples more power into a given fiber. The launched power improvement is generally more pronounced with smaller fiber core sizes (100 micrometers or less).

The trade offs between edge emitting and Burrus LEDs are currently being considered at DAC in the development of an optical transceiver pair suitable for the ARINC 629 terminal avionics data bus. The best surface emitting LED found to date will couple 0 dBm peak into a 100 micrometer core fiber but costs \$600. The receiver alternatives are more complex as its sensitivity and dynamic range specifications are not only a function of the optical detector utilized (pin diode or avalanche photo diode) but also a function of the associated electronics, signal format and data rate. These design trade offs and alternatives will not be discussed here as there are various papers available which discuss these issues in detail.

A state-of-the-art receiver developed for a 2 megabits per second (Mbps) Manchester II encoded multitransmitter data bus (ARINC 629) has a sensitivity of -42 dBm minimum and a dynamic range of 20 dB. Assuming 0 dBm launched power these specifications fall short of that required for each of the linear networks discussed previously. One of the factors which limits the sensitivity of this particular receiver is the fact that it must be DC coupled. This requirement is a result of the fact that the ARINC 629 data stream has no preamble and operates in burst mode (i.e., there are relatively long quiet periods between data transmissions). The DC coupled design extends the bandwidth and therefore the noise in the receiver limiting sensitivity.

A general consideration for a FBL System is a given terminal's capability to check itself during transmission and also to detect the presence of a collision on the bus and respond accordingly. In order for a terminal to check itself during transmission, it must be able to receive its own data from the bus. Thus the wrap around loss as discussed previously must be accounted for when determining a receiver's dynamic range requirements. This loss must be large enough such that a given terminal does not saturate its own receiver when it is transmitting.

### 3.4.1 Optical Data Bus (Continued)

This is particularly important in the dual K configuration where the wrap-around loss is considerably less than the loss between two adjacent terminals. It seems tempting on the first look at this problem to temporarily "desensitize" a given terminal's receiver when that terminal is transmitting or simply disable the receiver during that terminal's transmission and provide the wrap around signal electronically within the terminal itself. Although either approach is feasible, each would defeat that terminal's ability to detect a collision during its own transmission. This would not be acceptable for a FBL application as fault tolerance would be compromised. Thus wrap around loss and the associated limitations are an additional shortfall of fiber optics, at least in the dual K network configuration.

A third shortfall of fiber optic technology in aircraft applications is a producibility issue. The installation of fiber networks, particularly in linear configurations will involve several challenges. To install fiber optic network harnesses with acceptable losses will require they be installed with a minimum of connectors and splices implying harnesses be produced in as few pieces as possible. This will require the capability to manufacture sections of linear harnesses with couplers in single pieces with accurate dimensions quickly and easily on the aircraft assembly line. Current technology requires an extremely labor intensive process to produce couplers, terminations and splices and cannot be readily adapted to aircraft manufacturing environments. It is our experience that chipping and cracking during these processes are not at all uncommon, even for the experienced technician. Any fiber damage during the harness development process will require rework and possibly additional splices to avoid compromising the harness dimensions.

To address the link loss and some of the installation issues of fiber optic networks, DAC can utilize the "Fiber Optic Integration Interconnection System" which has been developed at DAC to offer a new way to cost effectively incorporate fiber optics into a production aircraft and minimize connections. This concept involves new installation handling and protection techniques which are needed to ensure first-time quality and to minimize production and in-service costs of fiber optics. This approach is based upon treating the entire aircraft as a single system. It has three major components:

- A. Trunk Line: This is a fiber optic cable assembly that runs the entire length of the aircraft and has breakouts at designated locations found within a given "zone". The trunk consists primarily of AT&T's 12 fiber ribbon cable and utilizes their multiple array connectors which connects all 12 fibers of the ribbon simultaneously.

### 3.4.1 Optical Data Bus (Continued)

- B. Zone: Divides the aircraft into interconnection areas. These "zones" are determined from the production procedures which manufacturing utilizes to build and assemble the aircraft.
- C. Fiber Optic Integration Unit is a junction box used to branch the fiber optic cable from the "trunk line" to the end destination point (data bus terminal in this case) within a designated "zone".

This installation approach offers many advantages such as the optimization and standardization of engineering design, maintainability, consistent quality installations, reduced parts damage, reduced rework and increased production efficiency. A prototype has been successfully installed in a MD-80 mockup. It was a simple installation task to route the ribbon fiber through the entire length of the fuselage with no connectors.

In order to take full advantage of this installation concept, a local star configuration can be utilized. A FBL System would use three transmissive stars as shown in Figure 3.4-8, one in the avionics bay, one in wing root areas and one in the tail area. As shown in Figure 3.4-9, using 16 X 16 star couplers with an insertion loss of  $14 \pm 1.5$  dB with three receive fibers on each star connected to three transmit fibers on each of the other two stars a thirty terminal network is realized with a maximum and minimum loss of 39 and 22 dB, respectively, assuming each transmit and receive fiber has two connectors at  $1.5 \pm 0.5$  dB each. This only leaves a 3 dB system margin, leaving little room for component variations or additional connectors.

In addition to improving the network loss profile previously described, the transceivers can be improved to increase the terminal count and system margin limitations. DAC is currently investigating a Frequency Shift Keyed (FSK) signal format approach instead of the standard Manchester II signal. A transceiver manufacturer has realized an additional 10 db of dynamic range with this signal format. This is reflected in specifications of an optical transceiver recently developed. To address the limitations of receiver sensitivity a preamble can be added to the data bus messages. This would require a modification to the ARINC 629 and MIL-STD-1773. Preliminary investigations indicate a 5 to 10 dB increase in receiver sensitivity can be realized.

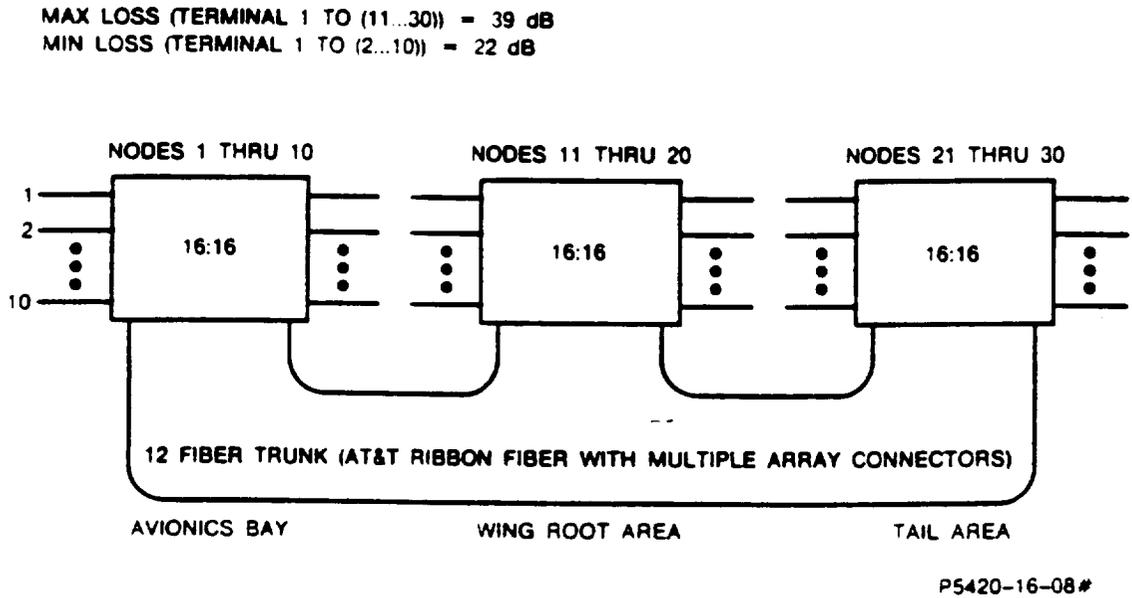


FIGURE 3.4-8

Terminal Local Transmissive Star Fly-By-Light Network  
Compatible With Fiber Optic Integration Interconnection System

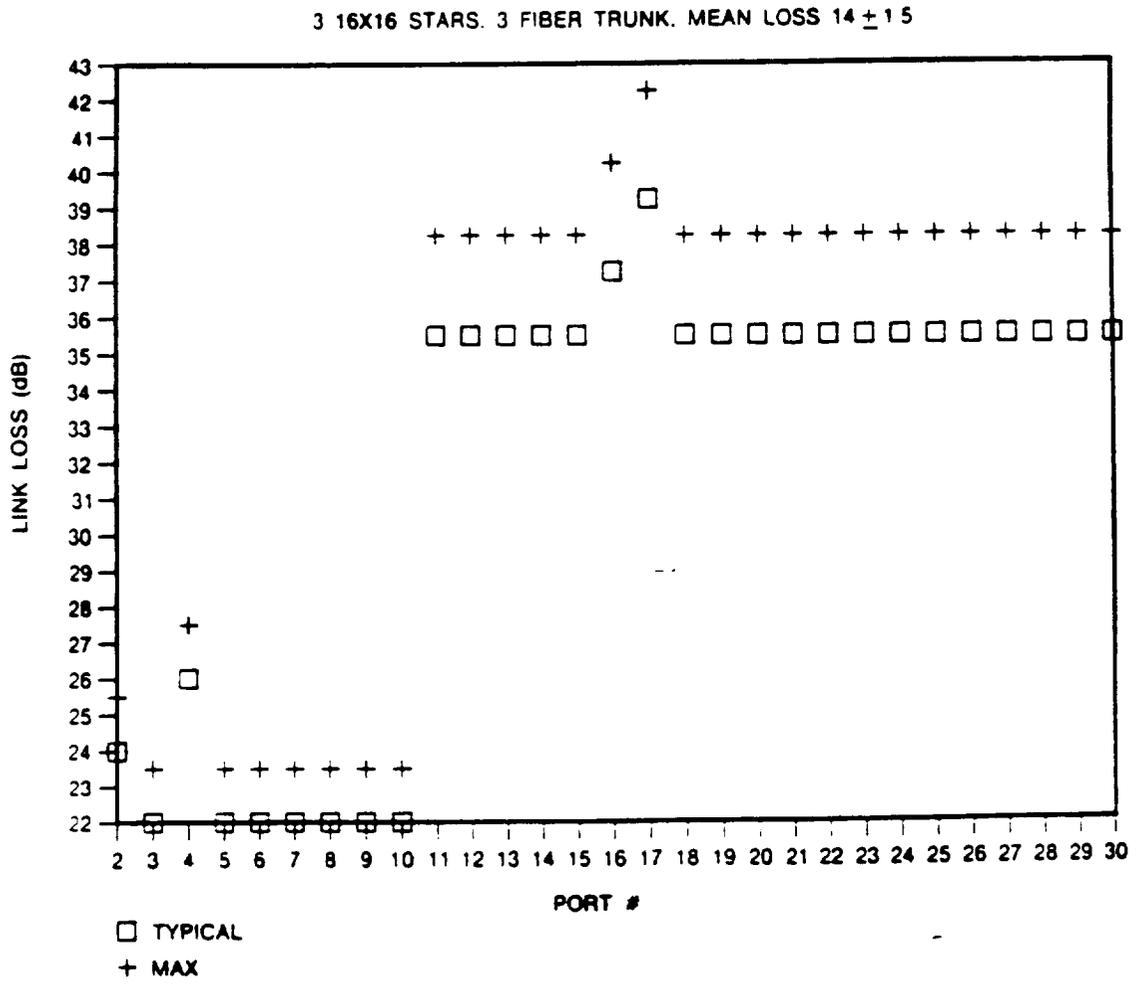


FIGURE 3.4-9  
Link Loss Terminal 1 to N for Local Star

### 3.4.2 Optical Sensors

Voluminous amounts of research and developmental work have been done in investigating fiber optic sensors. Except for an on/off switch and a temperature sensor, however, not very many of these sensors are available as products. The reason for this is that most of the effort has been concentrated on demonstrating new ideas and configurations of fiber sensors rather than on their reliability over environmental conditions. As generally happens with a new technology, product availability depends largely on the economics of replacing the existing technology. Limitations of fiber optic sensors include the dynamic range and the sensitivity to physical parameters other than the one being measured.

There are a variety of physical mechanisms that have been adapted for use in sensors employing fiber optics, comprising a set of optical modulation principles appropriate for optic sensors. In each of these principles a sensing element mechanism modifies power generated by some optical source to convey information about the sensed physical parameter. These optical modulation principles can be grouped into five major categories: Interferometry, Time Division Multiplexing (TDM), Wavelength Multiplexing (WDM), Intensity Modulation (IM), and Optically Powered Sensors (OPS).

Interferometric type fiber optic sensors are more susceptible to changes in physical conditions, such as temperature variation and mechanical shock, than the intensity modulated type. However, these problems can be overcome through proper instrumentation. For example, the fiber optic gyro is a Sagnac interferometer, and it is a highly sensitive rotation sensor. Honeywell has demonstrated building an interferometric fiber optic gyro as a product for the Attitude Heading Reference Systems (AHRS). The intrinsic or extrinsic IM type fiber optic sensors are based on simple optical principles and have some limitations. These include optical component contamination problems for sensors, output intensity normalization requirements to compensate for connector and other losses, and limited capability to multiplex the output signals from several sensors on to a standard data bus. TDM and WDM approaches have the disadvantage of complex encoding schemes, high insertion losses, and non-uniform bit response. Since both of them are based on extrinsic IM, they have the problems of encoder and fiber ends' contamination.

Fiber optic sensors to measure switch state, position, pressure, and other parameters have been demonstrated in the laboratory, but their use in harsh environments such as an aircraft is very limited. The present status and what might be needed to put some of these fiber optic sensors in FBL systems are described in the following subsections.

#### 3.4.2.1 Proximity and Other On/Off Switches

The push-button switch relies on reflections of optical power from a mirror which is switched either into or out of the reflecting position. Though good on/off ratios may be achievable, failure modes include susceptibility to contamination of the optical components and reliability considerations associated with the mechanical switching mechanism. Hermetic sealing of these switches is difficult. This problem can be solved by using some sort of magnetic coupling to the switching element. Fiber optic switches which would pass the aircraft specifications are seen as plausible for the near term.

#### 3.4.2.2 Pressure Sensors

Several fiber optic pressure sensors including interferometric and IM types have been proposed. Detection capability of these sensors are often limited by their sensitivity to temperature changes. In one IM type the pressure induced displaces a small diaphragm which acts as a Fabry-perot cavity attached to an optical fiber. Pressure is then measured as a function of the resultant modified optical spectrum. Temperature sensitivity is minimized by using a small cavity and proper selection of materials. Polarization cross coupling in a properly aligned polarization maintaining fiber can be used as a pressure gauge. Realization of pressure sensors for the FBL system may occur in the range of 10 years or less.

#### 3.4.2.3 Position Sensors

Linear variable differential transformers (LVDTs) are the most commonly used electrical position sensors used in aircraft and have proven to be very reliable. Most of the fiber optic position sensors are based on the principle of intensity modulation. The position on a "n" bit gray-coded encoder is read with a read head, which provides digital pulses that are wavelength or time division multiplexed. Since the multiplexing needed for a 10 bit resolution is complex, non-uniform bit response is seen as a common occurrence. In a delay-line TDM sensor, the non-uniformity is caused by an imbalance of delay-line taps or by imperfections in the optical apertures of the read head. In a WDM system, it can be caused by intensity modulations in the spectrum source, or by the optical properties of the fibers, connectors, filters, etc. The other most common problem with fiber optic position sensors proposed in the past is that since light leaves the fiber, contamination of optics can degrade their performance.

The digital-optical time division multiplexed (TDM) method of position measurement has been demonstrated on the Advanced Digital Optical Control System (ADOCS) program and a second generation TDM sensor is being developed under the Advanced Optical Position Transducers (AOPT) program. In this concept a 20 nanosecond optical interrogation pulse is split up into 12 delay lines. A reflective grey code plate reflects bits of position information back down the respective delay lines.

### 3.4.2.3 Position Sensors (Continued)

Thus the interrogation pulse is split into 12 on/off serial position bits at the receivers. Although simple in theory, difficulties arise in connector losses and reflections. High speed automatic gain control circuitry is required to accurately read the serial data in the presence of bit intermodulation from connector reflections and delays.

Eldec Corporation is in the process of developing a hybrid electrical/optical sensor concept termed "El-Optic". This family of sensors uses existing electrical sensors powered by either a battery or an optical power source supplied via optical fiber. Sensor data is communicated by optical fiber. Two simple battery powered sensors are now in flight testing in scheduled commercial service. Optically powered sensors are in a research stage at Eldec. Issues involved in the design of these sensors include a lack of reliability data for batteries, micropowered circuit design, and reliability at elevated temperatures. Advantages include EMI immunity due to purely optical external connections and a wide choice of sensing functions.

A passive fiber sensor concept in development at Eldec is called Time Delay Intensity Normalization (TDIN). Based on the intensity modulation sensor concept, an optical pulse is split into a reference delay arm and a sensing arm. Some sort of transduction technique transforms an environmental parameter into a reflected optical power level. The resulting signal pulse and delayed reference pulse are compared at the detector.

After suitable analog pulse averaging and normalization, accuracies approaching 0.25% (9 bits) have been demonstrated over a source power variation of 12 dB. Thus, the principal problem of analog intensity-based sensors has been addressed, namely source intensity and connection loss variation. Concepts for multiplexing have been presented. (Reference 5).

TDIN is essentially a transducer readout concept and is separable from the actual transducer design. The transducers themselves are still in a research/development stage. Accuracies and ripples are on the order of 1%. Optimization is still required. The main concern for position sensors, for example, is sealing against environmental effects. Temperature sensors and pressure sensors have not yet been developed.

Litton is developing fiber sensor concepts based on the WDM approach. Existing designs for rotary and linear position, temperature and pressure sensing, are under development for aircraft propulsion system applications. Areas of technology advancement needed include better fiber jacket temperature performance for microbending losses, high reliability detectors and circuitry for high temperature environments, and broad spectrum LEDs.

### 3.4.2.3 Position Sensors (Continued)

Flightworthy approaches for position sensing should incorporate the following features:

- o Light must not leave the fiber
- o Multiplexing schemes must be kept simple
- o Must have the capability to seal the sensor hermetically
- o The insertion loss of the sensor must be kept as low as possible through careful design

Passive optical sensors have achieved flightworthy status in some cases, but cost, ruggedness and production engineering issues still limit the applicability of these sensors in flight controls systems.

### 3.4.3 Optically Powered Sensors

Electrically passive optical sensors as previously described are ideal for immunity against EMI. However, an engineering penalty is paid for such sensors. Without the benefit of reliable electronic signal processing at the sensor, the types of sensors that can fill this category are severely limited. For example, the cost and maintainability penalties paid on the ADOCS program, which used a passive optical displacement sensor, were significant.

It is conceivable that electrically passive fiber optic position sensors that would satisfy the FBL system specifications can be made available for second generation FBL commercial aircraft. For the near term, Honeywell is investigating optically powered sensors. In this concept, optical fiber provides EMI immunity by electrically isolating the existing electrical sensor completely and overcomes the disadvantages of the passive sensor by providing powerful electronic signal processing capability.

What made this possible was a breakthrough in micropower circuit elements. Honeywell was able to reduce normal power requirements of I/O circuits and signal processing circuits to extremely low levels and achieve high yields in production. Honeywell is thus able to take full advantage of reliable and rugged sensors that have been proven in the field on operational aircraft and simply replace the I/O circuitry with those that could perform the same function utilizing significantly less power. An optically powered throttle lever angle (TLA) RVDT for a full authority digital engine control system is being developed for the Optical Propulsion Management Interface System (OPMIS) program and will be flight tested on the Advanced Transport Operating System (ATOPS) aircraft at NASA Langley. Honeywell is evaluating the application of this technology to FBW aircraft and the extent of EME immunity that is achievable. A proximity switch based on the same technology has also been developed at Honeywell.

### 3.4.3 Optically Powered Sensors (Continued)

Principle advantages of optical powered sensor (OPS) technology include:

- A. Mature electronic technologies
- B. EMI immunity and verifiability
- C. Established common interface - multiplexibility

#### 3.4.3.1 OPS Technology Development Areas

Optically powered sensor development centers around several key areas. These areas include:

- o Temperature stabilization of sensor output.
- o EME-immune sensor module packaging/shielding.
- o Optical source power and reliability.
- o Multiplexing architectures and protocol.
- o High temperature (250°C) circuitry.

These areas will be discussed briefly below.

- A. Temperature stability is completely independent of optical effects in an optically powered sensor since the optics are used only as a digital information and power channel. Electronics effects at the sensor alone determine the temperature stability. Considerable progress has been made in designing ratiometric calibrating micropower circuitry that to a first order cancels temperature drift in components. Circuitry for the TLA sensor provides less than 0.5% drift over the temperature range of -55°C to +90°C.

Beyond this temperature, leakage currents from components such as diodes and CMOS gates become noticeable, requiring more optical power to maintain the electrical power budget.

- B. An isolated electrical circuit can be shielded to an arbitrary level from EME at expected frequencies, i.e. to the GHz range. When a small aperture in the shielding is opened for optical connections, short wavelength radiation in the GHz frequency range may be coupled to the optical photodiode or LED. Low frequency cutoffs of KHz and MHz on these devices will attenuate coupled microwave radiation to a large extent. Another EME coupling aperture may be provided by the electrical transducer itself, such as RVDT, LVDT or temperature sensor. Further study and EME modeling is needed to accurately determine the EME attenuation of the sensor shielding. It is expected that the elimination of long electrical or sensor cabling will virtually eliminate EME coupling into the OPS system.

### 3.4.3.1 OPS Technology Development Areas (Continued)

- C. High power, high reliability LED technology is being actively pursued by Honeywell's Optoelectronic Division (OED) and Corporate Systems Development Center (CSDC). At OED, new lensing techniques that increase fiber-coupled power by up to a factor of six for 100m fiber are in development. At CSDC, a dielectric quarter wave stack acts as a mirror to reflect light towards the fiber, nearly doubling power output. These technology developments will drive the demonstrated high reliability of LED's to a sufficient power margin for most optically powered sensors.

For longer term sensor technology development, sensor multiplexing and architecture issues indicate the use of laser diodes as optical power sources. Reliability and temperature issues are presently hindering the acceptability of these devices. Lifetime enhancement by low duty cycle usage and near threshold operation are being investigated by Honeywell. Thermoelectric cooler technology is not yet reliable enough for application. Further development is needed by the laser and thermoelectric cooler manufacturers.

- D. The development of high power optical sources will drive the multiplexing capabilities of optically powered sensors. Issues to be analyzed include sensor architectures power margins, redundancy, and data communication protocols. It may be possible to interface in this system to an ARINC 629 type of optical network.
- E. Operation over the range of  $-55$  to  $+125^{\circ}\text{C}$  is expected with silicon-based technology. Longer term research will center on the use of silicon-on-insulation (SOI), Gallium Arsenide (GaAs), and Silicon Carbide (SiC) technology. Higher bandgaps in these semiconductors allow higher temperature operation with reduced leakage current. Honeywell is active in both SOI and GaAs technology, and significant progress is being made in SiC technology at companies such as Cray Research.

### 3.4.3.2 OPS Development in Industry

The optically powered sensor concept is also known as power-by-light (PBL) and power-by-fiber (PBF). EG&G and Simmonds Precision also are developing OPS systems. Moog, Inc., with Banks Electronics and Varion, has also developed a prototype OPS.

### 3.4.4 Optical Gyro Technologies

#### 3.4.4.1 Overview

Three types of technologies are currently being pursued for gyros: the interferometric fiber optical gyroscope (IFOG), the resonant fiber optic gyroscope (RFOG), and the ring laser gyroscope (RLG). Currently, RLG's are widely used for many types of military and commercial applications, while the other technologies range from research and development to early product development. All three kinds are based on the Sagnac effect. This phenomena, discovered in 1913, states that two counter-rotating optical beams traveling around the same closed optical path will experience pathlength differences that are proportional to the rotation rate of the closed optical path.

The various optical gyro technologies are distinguished by different techniques for measuring this rotation-induced pathlength difference. The RFOG and RLG pathlength differences are sensed by measuring the difference between two cavity resonant frequencies. In the IFOG, the rate is sensed through direct measurement (open-loop) or the nulling (closed-loop) of the optical phase differences of the two beams.

The means of implementing the measurement of pathlength differences are quite different among the RLG, the IFOG and the RFOG. Both the IFOG and the RFOG utilize an external light source to launch light into a passive fiber loop. The IFOG requires a broadband light source to eliminate many error sources while the RFOG, whose operation most closely resembles the RLG, requires an extremely narrowband light source. For the RLG, a narrowband gas laser with its active gain medium is an integral part of the sensing cavity. Therefore, the RLG is characterized as an "active" optical gyro, as opposed to the IFOG and RFOG which are considered passive.

The fiber optic gyroscope of choice is the IFOG because it can meet the aforementioned objectives while offering additional advantages. These include low weight, low power consumption, rapid start-up time, long shelf life and flexible geometries. The advantages of the IFOG are attributed to its solid-state construction and its limited number of components, which are a light source, a coupler to split light, a length of optical fiber as the rotation sensing loop, a photodetector and a polarizer. IFOG's are potentially low in cost.

As mentioned earlier, IFOG's are essentially classified as either open-loop or closed-loop.

#### 3.4.4.2 Open-Loop IFOG

In the open-loop IFOG, light is directed into the fiber to sense the rotation rate. Upon exit, a photodetector captures the two counter-propagating beams having some relative phase shift. (If the gyro is at rest, the two beams will travel the same distance and will be in phase, and the intensity of the combined beam will be maximum.) Destructive interference due to any phase mismatch, as a result of a rotating gyro, will result in a reduction of the output intensity. The rotation rate can be determined by monitoring signal changes at the output.

Despite its presumed simplicity, the open-loop design is a sophisticated instrument requiring additional refinements. The output photocurrent, for example, is a raised-cosine function of the phase difference between the two counter-propagating beams. Sinusoidal bias modulation is introduced at one end of the sensing loop by means of an optical phase modulator that is used to overcome at-rest minimum sensitivity. Synchronous demodulation of the detected optical signal is then applied to recover the rotation rate and direction of rotation. Such a modulation-demodulation technique transforms the gyro transfer function from that of a raised-cosine to a sine. The latter function exhibits maximum sensitivity at rest. The polarity of the signal indicates the direction of rotation.

Problems arising from the sinusoidal transfer function are twofold. First, the periodicity of the sinusoidal function yields an unambiguous measurement only between  $-90$  and  $90$  degrees. Thus, the dynamic range is limited. Secondly, the scale factor linearity becomes worse as rotation rates increase. Achieving good scale factor performance over a large dynamic range is an on-going research topic for the IFOG.

Both dynamic range and scale factor linearity can be improved with open-loop signal processing techniques and the use of a closed-loop servo. Open loop processing relies on post processing of the gyro output signal to linearize the sinusoidal dependence. The closed-loop servo, on the other hand, uses a feedback loop to force the gyro to operate at the null condition. Both techniques help minimize the dependence of the output on fluctuations of light source intensity, which is a sensitive design concern. Over the past 10 years, various open-loop processing approaches have been proposed. The success of these open-loop techniques have resulted in only modest gains in dynamic range and scale factor accuracy. The open-loop IFOG is limited to low performance applications because of the limited dynamic range and a scale factor accuracy.

#### 3.4.4.3 Closed-Loop IGO (Serrodyning)

To achieve appreciably improved performance, IFOGs are best operated in a closed-loop mode, the present focus of IFOG development. In a closed-loop IFOG, the Sagnac phase shift is compensated by introducing a phase shift of opposite polarity using optical components within the fiber loop. One of the preferred approaches for generating this phase nulling effect is to apply an optical frequency shift at one end of the fiber using a technique known as serrodyne phase modulation. Serrodyne phase modulation was first applied to microwave circuits to perform the function of a frequency shifter. "Serra" in Latin means saw or sawtooth.

By placing a phase modulator at one end of the fiber sensing loop, and applying a sawtooth drive voltage to the modulator, clockwise and counterclockwise waves each experience a saw-tooth phase modulation but delay relative to each other by a time interval equal to the loop's transit time. The effective phase shift at the output of the loop is equal to the net phase difference of the two sawtooth waveforms. Consequently, there is nonreciprocal phase shift that is then used to counterbalance the rotation-induced Sagnac phase shift. With this scheme, the sensor is always operated at its most sensitive null condition. The dynamic range of the serrodyne closed-loop IFOG will only be limited by the bandwidth of the loop-closure transducer and the serrodyning electronics that return the sensor to its null position.

In practical implementations, the output of the synchronous demodulator is used as the error signal of a closed-loop servo. This signal, after integration and amplification, is used to control the frequency of the serrodyne electronics that drive the phase modulator. The amount of phase shift applied to the sensing loop is proportional to the serrodyne frequency which is a measurement of rotation rate.

The null operation of closed-loop IFOGs also removes the problem of source power fluctuation and electronic gain instability. The digital readout electronics of the serrodyne IFOG generate a pulse at the falling edge of each sawtooth cycle. Each pulse corresponds to a small, fixed angular rotation and thus this device integrates the rotation rate.

The two required elements for a high performance closed-loop IFOG are a high speed serrodyne driver, and a high quality serrodyne phase-modulator.

#### 3.4.4.4 IFOG Highlights

##### A. Open-loop IFOG

1. First FOG to be introduced as a product
2. Low-medium scale-factor-performance applications
  - o Aircraft attitude, heading, and reference (AHRs)
  - o Space applications with modest scale factor requirements
3. AHRs flight test scheduled for latter 1989, 1990
4. Production to begin by 1991
5. Technical Data
  - o Sinusoidal output limits dynamic range to  $\pm 90^\circ$  optical phase shift
  - o Scale factor linearities of 1000 ppm over temperature demonstrated
  - o AHRs IFOG bias stability of  $1^\circ/\text{hr}$ , random noise  $0.004^\circ/\text{sq. rt. (hr)}$  demonstrated
  - o Higher performance open-loop gyro have demonstrated  $0.1^\circ/\text{hr}$  and  $0.004^\circ/\text{sq. rt. (hr)}$  bias stability and random noise performance, respectively

##### B. Serrrodyne IFOG

1. Closed-loop IFOG
2. High-performance applications
  - o Commercial airliners
  - o Military aircraft
3. On-going prototype efforts
4. High-speed serrrodyne driver
5. High-quality serrrodyne phase-modulator
6. Technical Data (ten-hour prototype run)
  - o Scale factor linearity is 30 ppm from  $-199 \text{ deg/s}$  to  $100 \text{ deg/s}$
  - o Bias stability  $0.02 \text{ deg/hr}$
  - o Random noise of  $0.0016 \text{ deg/sq. rt. (hr)}$

3.4.4.4 IFOG Highlights (Continued)

- o Scale factor repeatability is better than 5 ppm at room temperature, and better than 15 ppm from -55°C to +45°C
- o Optical phase shift stability of 0.4 microradians

7. Forecasted flight test - two (2) years

3.4.4.5 Necessary Developments to Enable FOG

In order for the fiber optic gyroscope (closed-loop configuration) to become fully operational and a prevalent product in commercial and military aircraft markets, the following systems, technological, and cost issues will have to be resolved. The issues or concerns included in this document do not enumerate all the problems associated with this project; however it does provide details as to the developments required to enable this technology for application in commercial avionics. Background information about optical gyroscopes and specific fiber optics gyroscopes was included to familiarize the reader with current technology and as explanatory data.

A. Technology Issues

1. Wavelength stabilization of the light source to ensure long term scale factor repeatability.
  - o Accurate measurement of the wavelength
  - o Temperature control/measurement compensation over wide environments
2. Testing over environments
  - o Thermal transient gradients across the sensing coil
  - o Radiation hardening
  - o Shock and vibration
3. Light source lifetime
4. Electronic development

B. Cost Issues

1. Fiber - reduce fiber costs
2. Components - current costs expected to go down rapidly with large production volumes
3. Manufacturing and Production - unknowns
4. Factory Capital Costs - simple factory expected to be an advantage for FOGs

### 3.4.5 Some systems that have flown

There have been many fiber optic system flight test programs since the late 1970's. A variety of data links and a few optical sensor systems have been reported. Most of the optical systems generally performed up to expectations, although after extensive production engineering for difficulties, especially with sensors. Problems common to all systems centered on the interconnection hardware. The following are among the notable systems to date:

- A. AV-8B Harrier Digital Data Link
- B. ADOCS JUH-60A Digital Flight control side-arm controllers position transducers
- C. DC-10 Aileron Position - Sensor, Passenger Entertainment System
- D. 737-200 - Nose gear downlock sensor
- E. Boeing 8 X 8 Star Bus Ethernet Command/Control

In general, connection hardware was the main problem on these flight tests. Douglas Aircraft has conducted an extensive connector evaluation program and has specific recommendations and technology plans for solving these problems.

#### 3.4.5.1 OPMIS

As part of its Fiber Optics Readiness program, Douglas Aircraft is currently evaluating optical propulsion system technology in its Optical Propulsion Management Interface System (OPMIS) program. This is a cooperative program between DAC, United Technologies, several fiber optic sensors vendors, and NASA Langley Research Center. NASA is providing flight testing for vendor-supplied fiber optic engine sensors and throttle sensors. This is an opportunity for these vendors to gain valuable flight test data on their optical components starting in 1990.

#### 3.4.5.2 ADOCS

Production and flight testing of the first fiber-optic position transducers revealed many of the problems of production and reliability of electrically passive fiber sensors. Teledyne-Ryan provided a set of TDM fiber sensors for the Advanced Digital Optical Control System (ADOCS) program, based at the Army Applied Technology Laboratory in Fort Huestis, VA.

#### 3.4.5.2 ADOCS (Continued)

Displacement transducers were designed to measure the position of the helicopter swash plate and hence the main rotor angle. Short stroke transducers were designed with 8 bits of precision, and long stroke transducers with 12 bits of precision. Some 30 of these sensors have logged over 500 hours of helicopter flight time on the JUH-60 "Light Hawk". Some of the main technology development problems faced for these transducers were as follows:

- A. Special 50m fiber delay line coil technology, with thermally matched mandrel and fiber.
- B. Transducer sealing against "pistonning" effect in a highly contaminated environment.
- C. High power pulsed laser diodes.
- D. Extensive multiplexing, both on the bit level and on the sensor level.
- E. High frequency automatic gain control electronics required for non-uniform bit response.
- F. Optical code plate thermal expansion over the -65°F to 165°C environment.
- G. Low power budget margin.
- H. Connector technology in a severe vibration environment. Fiber breakage and connector contamination remains a problem.

The tremendous amount of engineering involved to overcome these problems resulted in a price tag of \$25,000 apiece for these transducers. The system included 24 transducers. A second generation of Teledyne-Ryan's TDM sensor is being developed under the Advanced Optical Position Transducer (AOPT) program. It is evident from this program that the simple, elegant theoretical idea of TDM sensors is very difficult to implement in a production engineering sense.

#### 3.4.5.3 FOCSI

The Fiber Optic Control System Integration (FOCSI) program has concentrated on military applications of FBL FCSs.

#### 3.5 FBW/FBL System Functions

The necessary functionality for commercial transport FBW/FBL is generally agreed. In addition to augmented pitch/roll/yaw control, full aircraft flight envelope protection is necessary.

### 3.5 FBW/FBL System Functions (Continued)

Autoland is generally not considered to be a part of FBW/FBL.

Sidestick control is likely to be of the rate-command/attitude hold type, or an equivalent.

### 3.6 FBW/FBL Computer to Sidestick Interface

The technology drivers for sidestick controllers (SSC) indicate that a fly-by-light (FBL) configuration is desirable. Fly-by-wire (FBW) SSCs have used electrical transducers to detect the motion of gimbals having mechanical mechanisms which provide 'feel' characteristics illustrated in Figure 3.6-1. The controller always exhibits a return-to-center characteristic. The force versus displacement characteristics are characterized by:

- o Break-out force
- o Primary force gradient
- o Softstop stop
- o Hardstop magnitude

The breakout force provides the pilot with a definite input force threshold before manual control is applied. The softstop step provides a definite cue to the pilot about the limit of the flight regime. The softstop step can also be used to implement different control laws with the force gradient change, providing a cue to the pilot.

Mechanization of conventional FBW aircraft SSC for roll and pitch incorporates a base pivot in such a way that the grip pivots about a point below the forearm, providing both force and displacement to the pilot. Conventional force gradients are achieved by a spring/scissors mechanism allowing complete adjustability during build. Viscous damping provides the best feel for aircraft SSCs. The mechanisms incorporate a balanced pitch gimbal that prevents inputs due to longitudinal aircraft acceleration.

Left and right hand grip configurations are provided. The switch functions provided on each grip are tailored for each cockpit configuration. Human factors specialists determine proper location and force magnitude, optimized for ease of actuation.

Assemblies under mechanical loading where stress considerations relate to safety hazards are:

- o Column
- o Pitch and roll gimbals
- o Housing
- o Boots
- o Springs
- o Rolls and pitch damper linkages
- o Interconnect cabling flexing.

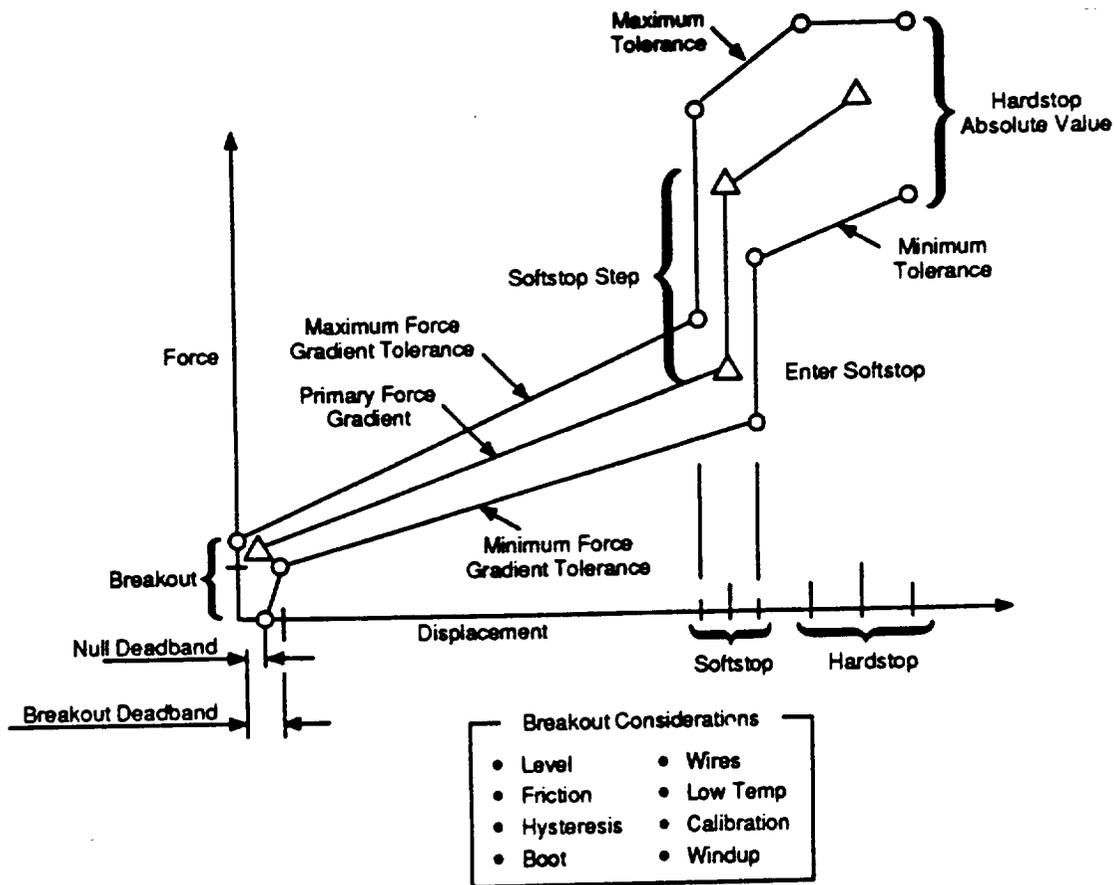


FIGURE 3.6-1

Interactions of Force and Displacement, and Other Elements  
That Enter Into Hand Controller Design Considerations

### 3.6 FBW/FBL Computer to Sidestick Interface (Continued)

Assemblies where mechanical wear-out-mechanisms come into play are:

- o Microswitches
- o Roll and pitch gimbal bearings
- o Damper seal mechanism
- o Roll and pitch RVDTs
- o External connectors
- o Roll and pitch scissors mechanism.

These are typical parts for an SSC.

#### 3.6.1 Active SSC

Lack of cross-cockpit coupling between SSC using conventional FBW mechanical design may be overcome by active hand controller technologies. Active SSCs use gimbals that are motor driven rather than spring-restrained. The motors are actively controlled by the flight control computer; thus, the two SSCs in the cockpit will be electromechanically coupled. Forces exerted in opposite directions will be algebraically added, with the net force resulting in movement of the SSC.

Motor driven hand controllers are operational in our lab and utilize the following components:

- o Brushless dc torque motors
- o Sinusoidal motor commutator
- o Rotational variable differential transformers (RVDT) or digital encoder pick-off
- o Planetary gear reducers
- o Internal torque sensors.

The hand controllers require real-time computer control where position, rate, and force is output by the SSC and force command is received. The flight control computer will provide the processing capacity, performing the control algorithms for the pilot and copilot SSCs.

The technologies for active SSCs are available without research breakthrough. Areas such as high torque/low weight brushless dc torque motors, available because of the new rare earth permanent magnets coupled with relatively easy computation power provided by microcomputers, makes active SSCs feasible. Engineering implementation is the most significant factor to consider the following critical commercial transport items:

- o Safety
- o Reliability
- o Weight

### 3.6.1 Active SSC (Continued)

An FBL version of an active SSC requires optical data to be transmitted between the SSC and flight control computer (Figure 3.6-2). Since an active SSC uses internal electrical signals, an optical-to-electrical converter is required.

### 3.6.2 SSC Sensors

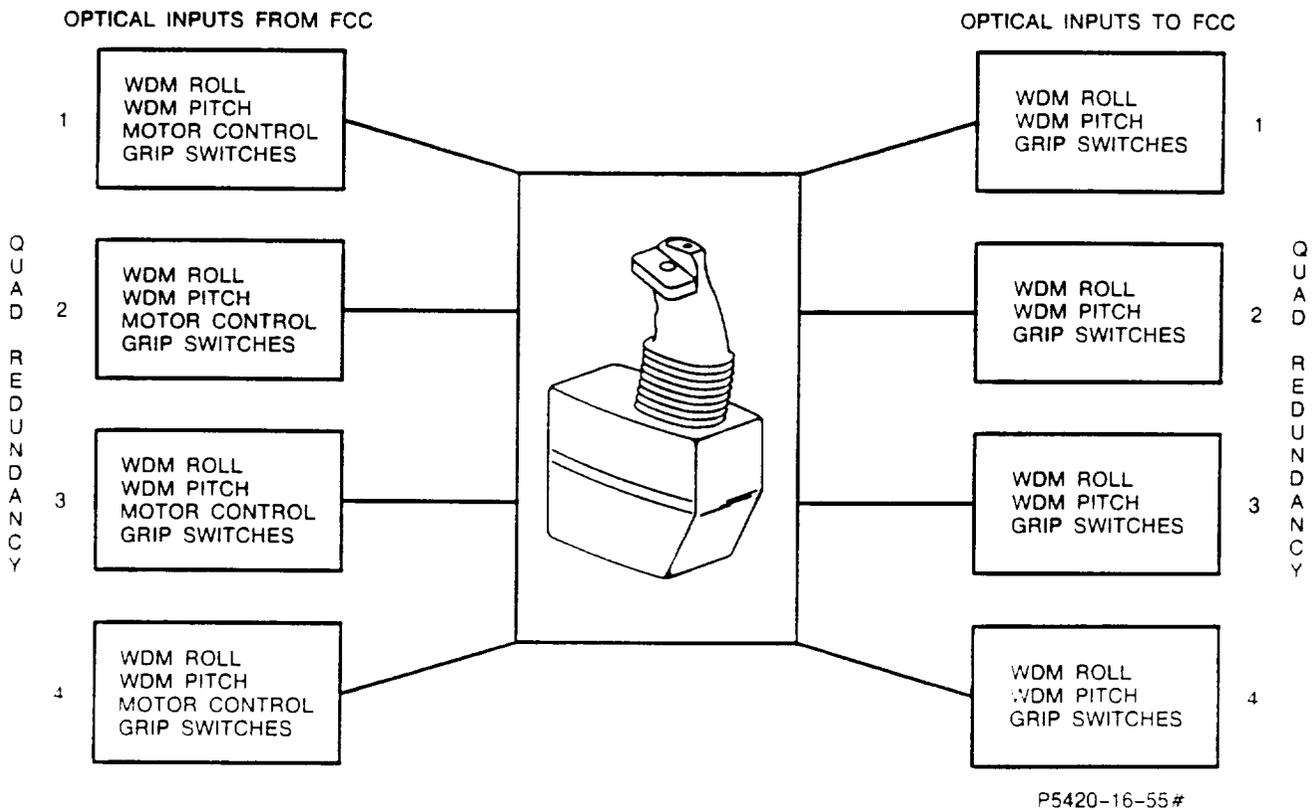
SSCs for use in an FBL flight control system must provide an optical output signal to the flight control computer. For near-term applications, the output from an FBL SSC will be a direct optical link to the flight control computer. As sensor technology and optical data bus architecture mature, SSCs will incorporate multiple axis sensors that interface both input and output to a single optical fiber. The need for more than one fiber will result from safety (redundancy) requirements rather than data transmission needs. In the medium-to-long, SSCs will interface with flight control computers through a standard avionics optical data bus.

#### 3.6.2.1 Preliminary Analysis and Trades

For initial analysis of human factors, the side sticks for air transport FBL systems will employ standard electronic sensors with the output converted to light by the use of a transmitter. Coincident with human factors design, all position sensors used in the FBL system will be reviewed for applicability to side stick controls.

The paragraphs below outline several options which exist for suitable sensors, including:

- o Analog electrical sensors with output converted to optical. These sensors require electric input and provide a voltage output dependent on the magnitude of the stick displacement.
- o Digital electronic sensors with output converted to optical. These sensors require electric input and output a digital position signal to the flight control computer.
- o Digital optical sensors with or without multiplexed light output. These sensors are passive since no electric power (only light) is required for input and output.



Optical Interface of Sidestick with Flight Control Computer

FIGURE 3.6-2

Optical Interface of Sidestick with Flight Control Computer

### 3.6.2.2 Analog Electrical Sensors

Analog sensors such as RVDTs have been used as position sensors for many years in flight control systems. They are essentially transformers with a primary coil that couples two secondary coils through a movable core. As the core is moved, a larger signal is coupled into the secondary. The direction of motion is determined by the phase of the output signal. Triple or quad redundancy is currently available. Other analog sensors are available such as resolvers and synchros.

Demodulation circuitry is available to convert an RVDT output to a digital signal. The digital signal can then be converted to a digital optical output signal. However, the side stick unit will require electric power to perform the conversion.

### 3.6.2.3 Digital Electronic Sensors

A second type of position sensor that has been used for side stick control applications is an absolute shaft angle encoder. This device consists of LEDs and phototransistors separated by a code wheel. The wheel is a metal disk etched with an array of precise holes. As the code wheel turns, the phototransistors will provide a digital output signal that depends on shaft position. Although encoders can be made with 16-bit accuracy, hand controllers require 12-bit accuracy. This type of sensor has several advantages: (1) no moving electrical contacts, (2) the device uses long life electronics, (3) the signal generated by the code wheel is digital and doesn't require A-to-D conversion. The disadvantage of encoders is that they are single-redundant; thus, multiple sensors must be used to accomplish redundancy.

### 3.6.2.4 Digital Optical Sensors

The digital electronic output from any of the above devices may be converted to light and injected into a fiber link using existing transmitter/receiver circuitry. However, these devices require electrical excitation and output digital electrical data. Optical sensors, on the other hand, are passive since no electric power (only light) is required for input and output.

For the short term, a small rugged optical sensor is available from Litton Encoder. This sensor employs encoder technology as described above; however, light originates from LEDs located outboard of the sensor (e.g., in the flight control computer) and is delivered to the sensor via fiber optic (F/O) bundles. The encoded position of the shaft angle is then delivered to the flight control computer via F/O bundles. These sensors offer several advantages: (1) sensors are smaller than electronic encoders, (2) F/O bundles will still function even if some fibers break, and (3) the output of the control unit is available as 0-10 Vdc, as well as digital, thus providing dissimilar

#### 3.6.2.4 Digital Optical Sensors (Continued)

redundancy. This type of sensor may be incorporated into an existing design immediately. The disadvantage of this sensor is that dedicated links are required for optical transmission and that the parallel 'word' output requires several transmission channels.

For the medium term (three years), single fiber encoders may be available. These sensors provide light input and output on a single fiber. In addition, the input/output of both the pitch and roll axis may be placed on a single fiber using a splitter. Two types of single fiber sensors are today available as prototypes. TDM sensors were developed under the ADOCS program by Teledyne-Ryan, and WDM encoders have been developed by Litton-PolyScientific. The output of these sensors will be fed directly to the flight control computer through a fiber optic link.

Long term sensors will be similar to those identified for medium term except that their output will be interfaced to an optical avionics bus.

#### 3.6.3 Human Factors

##### 3.6.3.1 Handling

The force per displacement and force per aircraft response characteristics have been researched in previous studies, but additional work remains to obtain the optimal levels for both these characteristics. Figure 3.6-1 illustrates the relationship of the various sidestick controller force per displacement characteristics that will be determined. The tactile feedback for the pilot to sense the neutral point is achieved with a breakout force. A softstop position will be included in the sidestick controller design and the force and displacement limits will be experimentally determined. The location of the sidestick controller X-Y axes should be rotated inboard and forward to minimize cross-coupling between the pitch and roll axes. If force feedback is to be included in the sidestick controller (forces from either the other pilot or the autopilot), the time delay of pilots input forces to the actuator movement should be 100 ms to maintain pilot-system stability.

These issues will be thoroughly reviewed in the available literature. Especially relevant are those studies conducted by Airbus, Boeing, and DAC. Unresolved issues will be initially evaluated in the laboratory and later moving to a high fidelity simulator and then to flight test for final calibration.

##### 3.6.3.2 Human Interface

The increasing number of female pilots in the airline industry impacts future hand controller configurations. The anthropometric

### 3.6.3.2 Human Interface (Continued)

implications of the grip design and the adjustability of the grip position with respect to the seat reference point will need to be addressed.

If force feedback is to be incorporated in the design, male and female strength capabilities must be considered during the selection of human interface parameters.

Human factor specialists are utilized in the proper selection and location of gyro mounted switches to insure ease of activation without cross coupling to SSC motion.

### 3.7 FBW/FBL Computer to Actuator Interface

The application of smart actuators to commercial transport aircraft is without production precedent. It is anticipated that "smart" actuators which contain processors that perform most or all of their own monitoring would be used for a FBW/FBL System. This simplifies the interface since they need only be connected to a common bidirectional data bus.

However, the aircraft is now potentially vulnerable to a generic CPU or software fault. Relative to this concern, DAC/HI have been actively evaluating a concept referred to as smart-actuator-override. The actuator's processor/software could be bypassed by the flight control computers and forced either to engage unconditionally or disengage. Accordingly generic fault protection could be provided for the smart actuator by the flight control computer.

### 3.8 System Availability

The use of redundancy to extend the average interval of time between required maintenance actions is of particular importance for FBL. The mechanical systems being replaced by electronics are themselves seldom replaced. It can be argued that to avoid uncertainties and risks associated with comparatively frequent equipment replacements, that the unit removal frequency for an FBL System must be in the same range as the mechanical assemblies being replaced.

The redundant elements which act in effect as "spares" have come to be referred to as "secondary redundancy". Redundancy of the more traditional type is by contrast referred to a "primary redundancy". The failure of an element of "secondary redundancy" results in no loss of functionality or reduction in safety level. The mean-time-between-unit-removal (MTBUR) that is believed necessary is in the range of 100,000 hours. It should be noted that such an extension of availability worsens the problem of fault latency. Accordingly faults can be sustained without unit removal despite internal "self-repaired" faults.

### 3.8 System Availability (Continued)

The suggested task(s) to be performed address the benefits, problems and, of course, costs associated with the incorporation of "secondary redundancy" for extended availability.

Candidates must be developed and compared. The impact on maintenance practices and installation mechanisms must be addressed.

Two ways of controlling reconfiguration of secondary redundancy elements should be each investigated. Reconfiguration decisions can be made by hardware voters or via software comparisons and decision making algorithms or a combination of these methods.

### 3.9 Graceful Degradation

In Section 3.3, normal functionality was defined as requiring augmented flight and automatic attitude hold. There is a need to explore methods such that the system performance will incrementally degrade as sensor sets are lost, rather than shutdown, as is the predominant approach in current flight control systems.

### 3.10 Generic Fault Tolerance

At the current time, N-version techniques are being applied by most airframe manufacturers and avionic suppliers to provide limited protection from certain types of generic faults. If this is to continue, we need to specify design guidelines to maximize version independence. A survey of experiences with N-version techniques is needed to aid in providing a quantification of benefit.

### 3.11 System Recovery from Upset (Soft Fault Tolerance)

#### 3.11.1 Introduction

Dependence upon electronic equipment to provide functions that are critical to the safe flight and landing of aircraft, is one of the most profound of the recent advances in avionics. The conversion of analog systems to a digital counter part is a natural extension of this continuing evolution in avionics. Because of the immense emphasis on research and development for the advancement of the electronic devices associated with digital processors and the extraordinary advancements that have been achieved as a result of this emphasis, digital processing provides the opportunity for a vast expansion of the role of electronics in the control and guidance of aircraft.

However, along with their data processing power, digital processors have unique considerations that must be taken into account. Digital processors are extremely complex general purpose

### 3.11.1 Introduction (Continued)

machines. The complex software and hardware defy deterministic assessment methodologies, thus, digital processor behavior should be viewed as a probabilistic issue. Soft faults refer to a probabilistic characteristic of digital processors that is not found in analog processors and is, therefore, a relatively unrecognized, unknown and unfamiliar factor. The more familiar and commonly recognized failure characteristic is referred to as a hard fault where an electronic device has failed in a permanent or irreversible fashion and is noncorrectible (i.e., the device has been destroyed and the needed operational characteristic permanently lost). Hard faults are a characteristic that is common to both digital and analog processors.

Unlike hard faults, soft faults can be corrected by correcting the state of the logic circuits (program counters, registers, etc.) that govern the control of the digital processor. When a soft fault occurs a digital processor may not recover to a proper operational state and may require some form of external intervention to resume normal operation. Quite often this intervention consists of recycling power to the digital processor. When power is recycled, the control circuits are automatically reset to legitimate states. Soft faults have been identified by several aliases (digital computer disruption, circuit upset, transient faults, correctable faults, single event upsets, faults with nonstationary observability, etc.). They are known to occur even when the operating environment is relatively benign and despite the substantial design measures (timing margins, transmission line interconnects, ground and power planes, and clock enables of digital circuits) taken to achieve a relatively high degree of integrity in digital processor operation. However, their occurrence, which is relatively infrequent and random, can be induced by any or all of the following factors:

1. environment (electromagnetic, nuclear, etc.),
2. Hardware (timing margins, circuit layout, etc.),
3. Software.

Soft faults may be the limiting factor in achieving improved reliability in the next generation of avionics. In-service experience with digital systems indicates that the confirmed failure rates are better than the predicted values which, in turn, are significantly better than previous analog equipment. However, the unscheduled removal rate remains about the same. There are several possible causes for this. A leading candidate is soft faults, which cause the system to fail and result in a flight crew write-up, are no longer present when the line or shop tests are performed on the suspected unit. Airline anecdotal data indicates many instances of "BITE check OK and returned to service".

### 3.11.2 System Architecture Measures for Recovery

Until recently, soft fault in digital avionics were manually corrected. More recently, the viability of systems level measures for the automatic correction of soft faults has begun to be investigated. It is perceived that significant benefits can be gained through soft fault protection measures designed into the basic system mechanization. System-level soft fault protection methodologies provide the ability to tolerate disruption of either input/output data or internal computation. Accordingly, there are two distinct classes of disruption:

1. Disruption at the subsystem interface boundary causing corruption of data flowing to or from the affected subsystem.
2. Disruption which reaches within the subsystem to corrupt internal data and computation. As a worst-case scenario, it must be presumed that all memory elements within the computation system are affected at the time of disruption.

The short term disruption of input/output data at a system boundary can be managed via a variety of existing methodologies. Data errors must be detected and the associated data suppressed until the error status is cleared. The data processing algorithm should tolerate data loss without signaling a hard fault. The length of time that can be tolerated between valid refreshes is dependent upon the data item and the associated time constants (response) of the subsystem.

The ability to tolerate disruption which reaches computation and memory elements internal to the subsystem, without propagation of the associated fault effect, is a more difficult problem. For systems with redundant channels, this means toleration of the disruption without loss of any of the redundant channels. Fault clearing must be "transparent" relative to functional operation and cockpit effect. Hence, the term "transparent recovery".

Transparent recovery requires that the disruption be detected and the system to be restored. Safety critical systems are almost always mechanized with redundant channels. Outputs of channels are compared in real time, and an errant channel is blocked from propagating a fault effect. One means available for safety-critical systems to detect disruption is the same cross-channel monitor. If a miscompare between channels occurs, a recovery is attempted. For a hard fault, the miscompare condition will not have been remedied by the recovery attempt.

### 3.11.3 Transparent Recovery

There are two basic approaches to transparent recovery. These are:

1. State variable data are transmitted from valid channels to the channel which has been determined faulted and for which a recovery is to be attempted (Figure 3.11-1).
2. A valid set of state variables is stored away prior to the disruption in memory which is protected from the effects of disruption to a level far greater than the rest of the subsystem (Figure 3.11-2).

The cross-channel mechanization is ineffective against a disruption which has the potential to affect all channels. Accordingly, the in-channel methodology has been emphasized in transparent recovery development activities.

Ideally, the goal of the system designer is to provide a means to tolerate the disruption without propagating the effects outside the subsystem. Stated in a different way, the goal is to provide a means such that disruption can be tolerated without causing a "cockpit effect". It follows that for this to be possible the duration of the disruption must be short relative to the time response of the system, and mechanisms instituted to achieve toleration of disruption must be capable of, in effect, clearing a fault caused by the disruption rapidly relative to the time response of the affected system.

The qualification relative to duration of disruption is reasonable when dealing with a safety-critical system. For such a system, the probability which must be guaranteed relative to total system disruption is typically extremely small. In some cases, less than a  $10^{-9}$ /hr probability of disruption (of a safety-critical system) must be guaranteed. As a result, system level implementations which provide a means to tolerate a disruption without propagation of an induced fault outside the affected subsystem are perceived (for safety-critical systems) to be necessary. Without such mechanisms for commercial aircraft, physical protection (and its maintenance) against EME which has been designed into the subsystem, can become a potential "single-point fault".

Because the occurrence of soft faults as an inherent feature of digital processors should be acknowledged and accounted for, new hardware elements are proposed for processor architectures that would enhance traditional design practices to achieve an extraordinary degree of processing integrity for the digital processor. These new functional elements would provide the detection of processing anomalies (upsets/soft faults), a protected region for the preservation of state data (logic and control), and the management of "transparent recovery" of data processing.

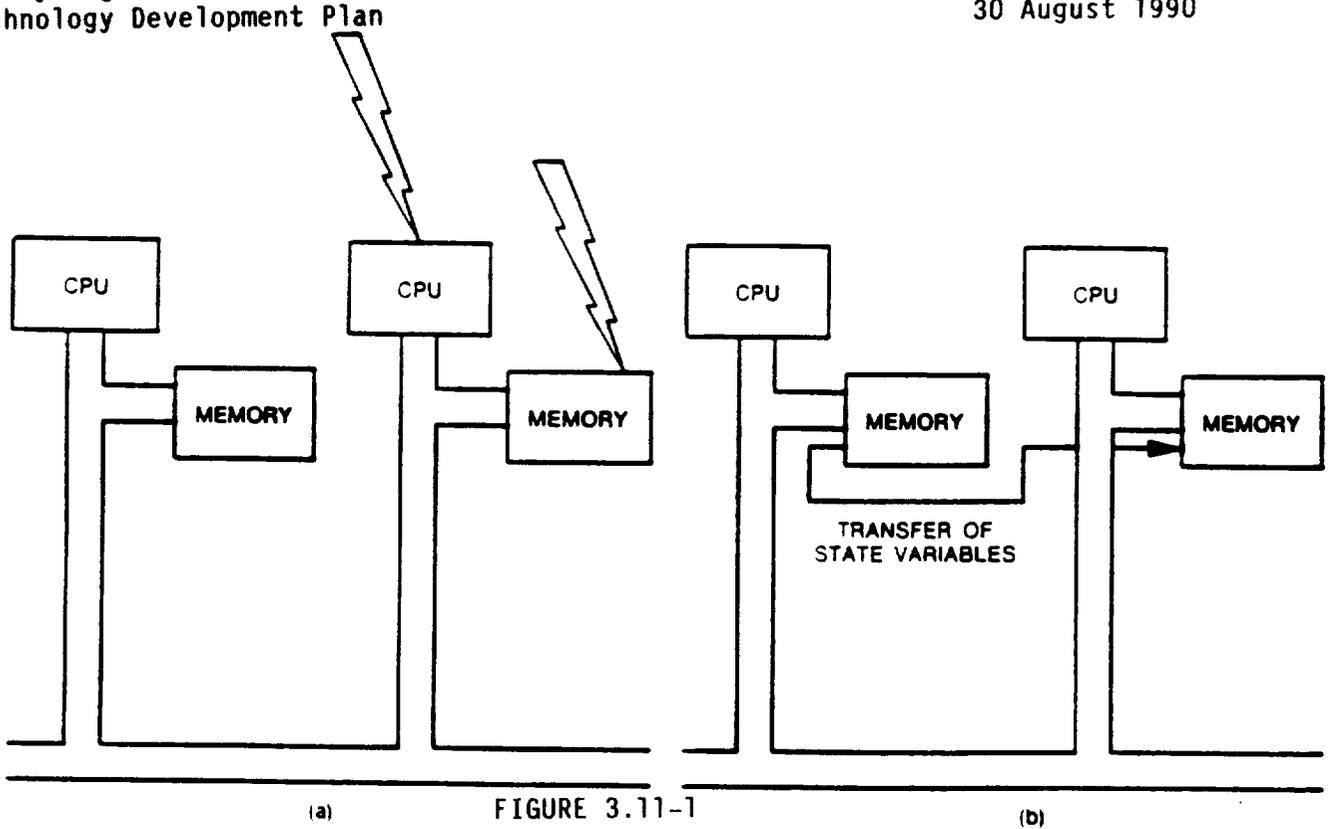


FIGURE 3.11-1

Cross-Lane Recovery

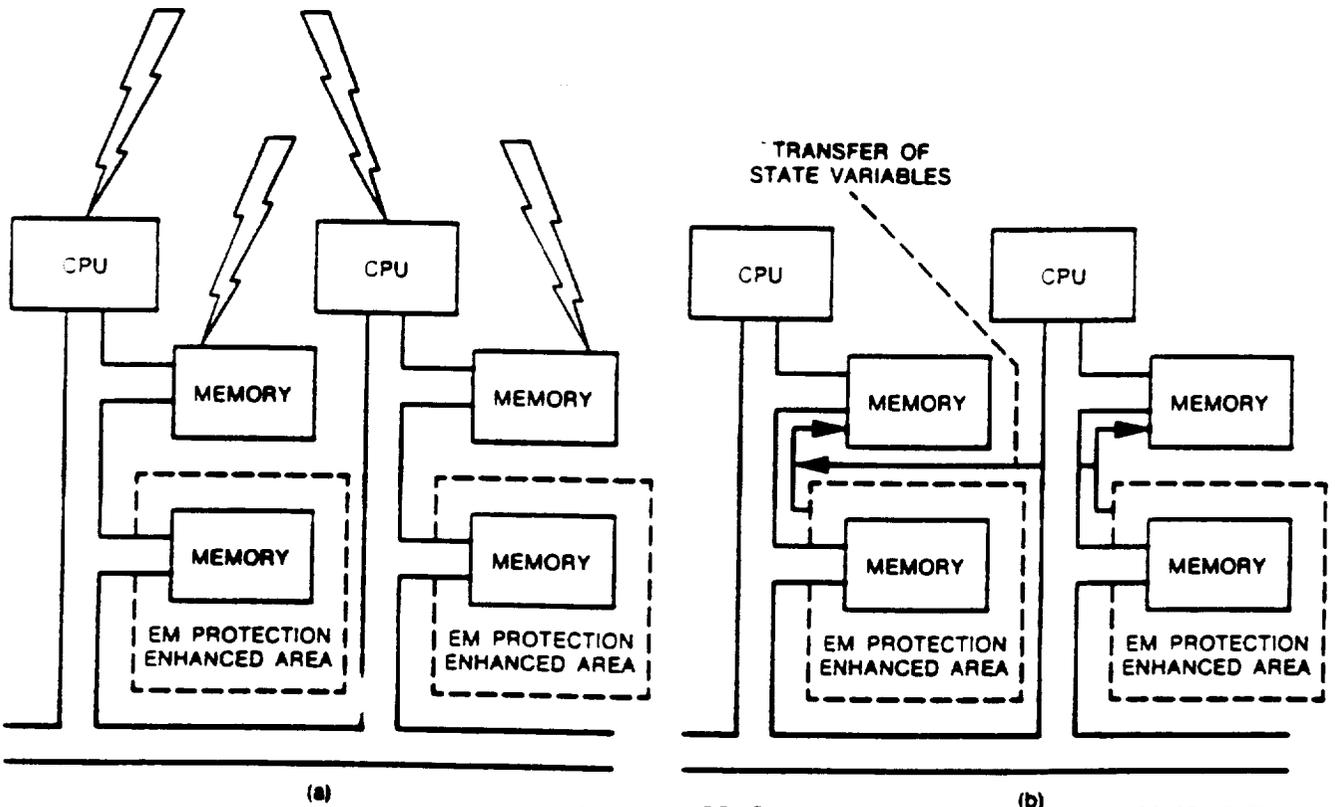


FIGURE 3.11-2

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In-Lane Recovery

### 3.11.3 Transparent Recovery (Continued)

Detection of processing anomalies, protected storage of state variable, and recovery of correct processing are additional capacities needed to achieve the resulting increased processing integrity that is associated with and achieved by transparent recovery. Figures 3.11-3 and 3.11-4 are diagrams of a digital processor architecture with transparent recovery elements.

The detection of processing anomalies might be accomplished through a variety of measures. The most desirable measures would be those that are inherent in the characteristics of processor operation. Essentially, these measures would probably amount to various types of reasonableness tests and could be in-line or cross-processor detectors or a combination of both. In addition to rough checks on data reasonableness, primarily, in-line detectors would check the reasonableness of processor behavior. For example, measures for concurrent in-line monitoring could correspond to checks of operational features of system performance. Concurrent monitoring techniques have been studied which can be implemented in the form of a separate hardware device that operates in parallel with, and transparently to, the system. A key aspect of the monitoring process is that its implementation, whatever form it may take, should be relatively simple when compared with the complex system being monitored. In other words, the process is not meant to duplicate the performance of the system.

The in-line monitoring process would record signals from a set of observation lines in the system to capture a data block and then analyze this block of data to determine whether the system is upset. The process is assumed to be dedicated; hence, its two-phase function of first capturing and then analyzing a block of data will be repeated continuously while the system executes its application program. For specific equipment the effectiveness of an in-line detector could be further enhanced by taking into account salient characteristics of the executing software (e.g., program flow, constraints on data storage) associated with that equipment.

Cross-lane monitoring becomes an option that naturally follows from the redundancy associated with fault tolerant data processing architectures. Cross-lane monitoring is an additional complementary option to in-lane monitoring. Fault tolerant architectures are absolutely necessary when flight critical functions (e.g., FBW flight controls) are provided by electrical/electronic equipment. If the system architecture consists of additional redundant processors (one or more additional processors performing the same tasks), then cross-processor detectors could be used to provide additional coverage of processor upset. Data reasonableness would

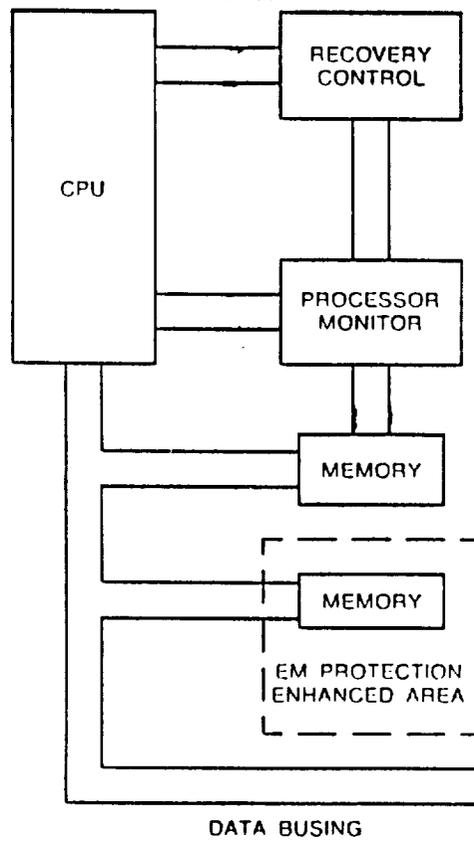


FIGURE 3.11-3

Digital Processor With Elements for Transparent Recovery

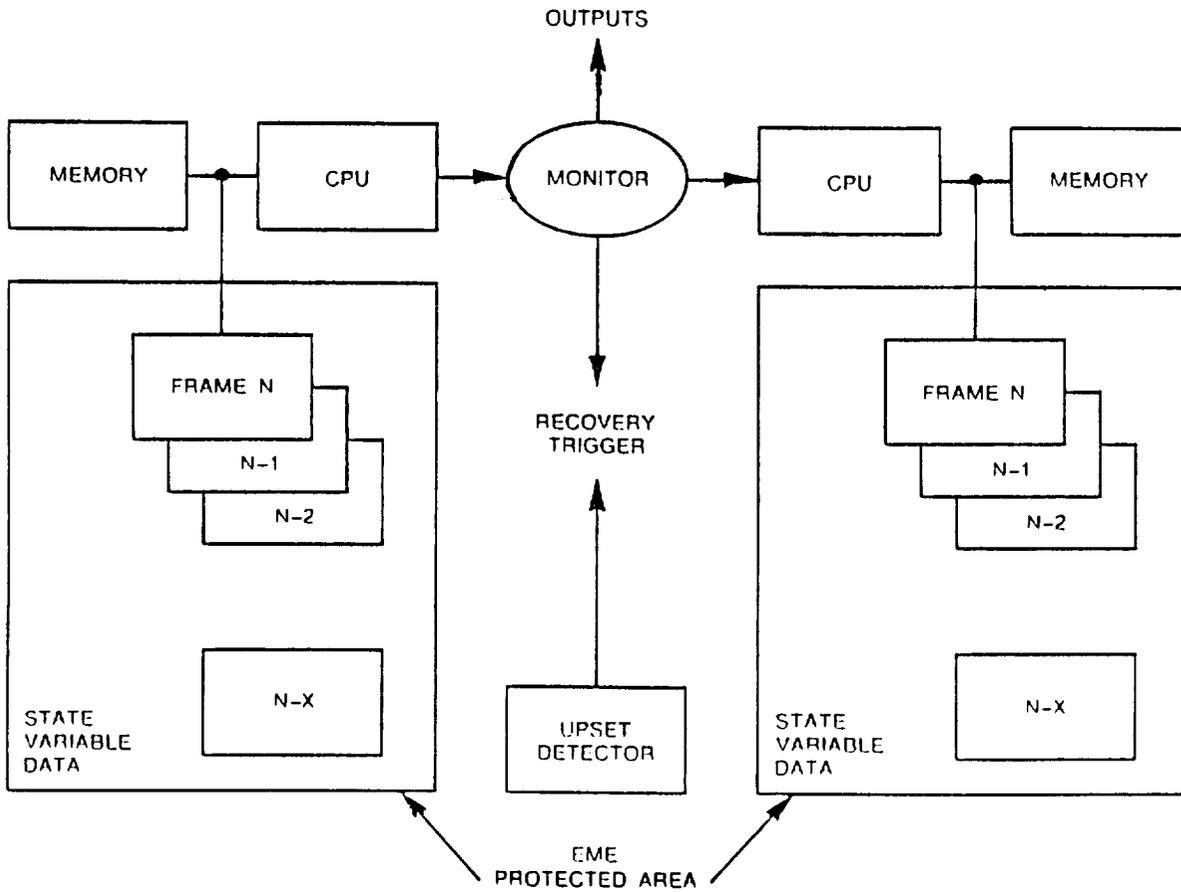


FIGURE 3.11-4

Transparent Recovery Upset Detection and Recovery Flow

### 3.11.3 Transparent Recovery (Continued)

be the primary parameter to be monitored by a cross-processor detector. This detector would complement the in-line detector by applying much finer criteria to corresponding data. As with in-line monitors (relative to the complex processors it is monitoring), a cross-processor monitor should be a simple hardware device.

Dynamic data are continually read from a storage region, updated by the digital processor, and then written back into the storage region. A certain portion of that data contains information that has been developed over time by the processing activity and is not reproducible. This type of data is referred to as state data. The set of state data along with the set of sensor data establishes the system state at each instant of time. In the event of a temporary disruption of the digital processor, the state and sensor data can be used to quickly recover the processing activity in a relatively transparent manner.

Thus, from a macro time perspective, system functions will continue to be performed, even though on a micro time basis a soft fault occurred. The key to transparent recovery is the availability of state data. To guarantee the availability of the needed dynamic data, a protected storage region must be provided. Isolation from the energy of a potentially disruptive electromagnetic event or the use of storage devices that would not be permanently changed by the energy in such an event, are two general approaches to achieving a protected storage region.

When a soft fault has been detected, high-level supervisory override hardware elements would cycle the digital processor through the recovery process. Basically, this process would involve reinitialization of the system state using the data from the protected storage region, resetting the program counter to an appropriate address to restart program execution, and then restarting program execution. Another function of the supervisory hardware would be to keep track of the number of restart attempts within the appropriate period of time. If the number attempts exceeds the appropriate criteria, a noncorrectible condition (e.g., hard fault) has to be presumed to have occurred. In that case, the supervisory electronics would provide a warning indication, disengage the computer, or both.

### 3.11.4 Additional Considerations

It is appropriate to note at this point, that as a result of the generalized nature of digital computers, such general-purpose equipment can fail in a general manner (failure manifestation can range from wholesale and obvious to subtle, unpredictable, and insidious). Soft faults are a classic example of such general unpredictable operation. Transparent recovery is an approach to handling soft faults that is general in nature. As such, it is consistent with the nature of digital processor based equipment

#### 3.11.4 Additional Considerations (Continued)

and will be effective for such general purpose programmable machines. In addition, transparent recovery has received strong endorsement by key technology specialists within the FAA. Generic faults is another example of a fault category that is general in nature. N-version hardware and software (dissimilar redundancy) is a general approach for handling generic faults. To eliminate the threat of destroying or permanently scrambling program instructions stored within digital avionic equipment memories, software resides in Read-Only Memory (ROM) so that even if the logic states of ROM elements are momentarily changed by transients, they will return to normal state after the transients die out. The use of ROM is yet one more example of the application of a general approach to effectively manage digital processor operation. The point being made here is that general purpose machines require approaches that are general and broad enough in scope to provide the coverage needed to eliminate or minimize (to an acceptable level of confidence) undesirable and unpredictable operation.

When such general approaches are implemented, the resulting general purpose machine will require correspondingly general purpose methodologies and associated capabilities to verify/validate its proper operation. Such capability would need to be broad in scope and based upon an integrated top down approach.

#### 3.12 Backup System

##### 3.12.1 Mechanical Backup

A mechanic backup provides the greatest degrees of dissimilarity and physical isolation with regards to a microprocessor based software implemented FBW primary flight control system. The Airbus A320 has incorporated a "minimal flight control" mechanical backup. While a mechanical backup is relatively easy to verify with conventional FMEAs, it cannot be incorporated in the FCCs, requires special interfaces, and defeats some of the weight savings achieved by a FBW system. Essentially a mechanical backup is a step backward for a FBW flight control system.

##### 3.12.2 Electronic Backup

Electronic backups can either be packaged in separate LRUs or incorporated in the FCCs. Electronic backup flight controls packaged in separate LRUs provide a higher degree of physical isolation, but also requires that the new electronics box(es) be validated, shielded, and interfaced separately. Thus electronic backups packaged in separate LRUs tend to increase system complexity in terms of installation and maintenance. Since sufficient physical isolation can be achieved for an electronic backup incorporated in the FCCs this method is more attractive and

### 3.12.2 Electronic Backup (Continued)

cost effective than separately packaged LRUs. Furthermore, if the electronic backup is incorporated in each FCC we would have redundant backups whereas if a separate backup LRU is used there would probably be only a single backup LRU.

The particular type of electronic backup flight control system chosen (e.g. analog vs digital) and whether a dedicated backup channel is needed or desirable will depend somewhat on the overall FBW configuration.

#### 3.12.2.1 Analog Backup

Analog backups have the benefit of being extremely dissimilar with respect to microprocessor based software implemented FBW flight control systems. They are also relatively easy to verify using standard FMEAs. It is presumed a separate analog backup channel would be used.

#### 3.12.2.2 Digital Backup Using Discrete Hardware

This type of backup is very dissimilar with respect to microprocessor based software implemented FBW flight control systems. These systems can be verified using standard FMEAs but analyses are more difficult than in the analog case.

#### 3.12.2.3 Digital Back Using Microprocessor Based Software Implementations

This technique could use a "minimal complexity" unique software or a "recovery block" technique which reverts to a simplified primary software. The unique method provides greater backup software dissimilarity, but would cost more to develop. Both unique backup software and simplified primary software would require extensive Level 1 software verification if included as a production system.



TABLE 4.0-1

FLY-BY-LIGHT Subsystems and Associated Optical Hardware

| <u>SUBSYSTEM</u>                                    | <u>OPTICAL HARDWARE</u>   |
|---|---|
| Avionics Data Bus                                   | Linear and/or Star Data Bus with ARINC 629 or MIL-STD-1553B Protocol<br>IRS Optical Rate Gyro<br>Optical Taps<br>Bidirectional Couplers<br>Electro-Optic Converters |
| Fly-By-Light Data Bus                               | Tree Data Bus with ARINC 629 or MIL-STD-1553B Protocol<br>1x2 Bidirectional Couplers<br>1x3 Bidirectional Couplers<br>Electro-Optic Converters                      |
| Sidestick Command Optical Sensing                   | Position Sensors<br>Pressure Sensors<br>Optical Taps<br>1x2 Bidirectional Couplers<br>Multiplexers<br>Electro-Optic Converters                                      |
| Rudder Command Optical Sensing                      | Position Sensors<br>Pressure Sensors<br>1x2 Bidirectional Couplers<br>Multiplexers<br>Electro-Optic Converters  |
| Speedbrake Lever Command Optical Sensing            | Position Sensors<br>Rotary Switch<br>1x2 Bidirectional Couplers<br>Multiplexers<br>Electro-Optic Converters   |
| Optically Powered Flap and Slat Position Indicating | Optically Powered Modules<br>Position Sensors<br>Star Couplers<br>1x2 Bidirectional Couplers<br>Multiplexers<br>Electro-Optic Converters                            |
| Auto-Throttle Input Optical Sensing                 | Acceleration Sensor<br>Position Sensors<br>Data Links<br>Electro-Optic Converters   |
| Throttle Lever Command Optical Sensing              | Position Sensors<br>1x2 Bidirectional Couplers<br>Multiplexers<br>Electro-Optic Converters  |
| Auto-Throttle Optical Switch Sensing                | Lever-Actuated Switches<br>Pushbutton Switches<br>1x2, 1x3, 1x4 Bidirectional Couplers<br>Multiplexers<br>Electro-Optic Converters                                  |

Fly-By-Light Subsystems and Associated Optical Hardware

#### 4.0 FLY-BY-LIGHT ARCHITECTURES AND TECHNOLOGY SUITES

- A. In this section some possible architectures for near term, medium term, and long term fly-by-wire (FBW) are described. Probable technology suites and some general subsystem designs and approaches will also be provided and discussed. Table 4.0-1 is a summary of the fly-by-light subsystems and Associated Optical Hardware.

#### 4.1 Flight Control Review

##### 4.1.1 Fly-By-Wire/Fly-By-Light Flight Control Computer Internal Architecture

- A. The safety requirements associated with the introduction of FBW/FBL systems for commercial transport aircraft (i.e., 10<sup>-9</sup> per hour and more) result in the incorporation of a high degree of redundancy and redundancy management. The flight control computer system which Douglas Aircraft Company/Honeywell, Inc. (DAC/HI) has developed includes sixteen (16) digital processors. This level of redundancy is without consideration of secondary redundancy.
- B. It is anticipated that any near term supplier of FBL/FBW systems will incorporate N-version techniques to yield a level of protection from certain types of generic faults. Honeywell, Inc. (HI) was the first organization to use 3-version techniques in a production aircraft (i.e., MD-11, 3-version Hardware/Software (HDW/SW) in the flight control computer/LSAS). The DAC/HI developed FBW/FBL flight control computers use 3-version techniques.
- C. Another technical advancement developed for the DAC/HI flight control computers is transparent recovery, to provide enhanced protection from the electromagnetic environment (EME) while providing a means to, in effect, bypass "soft faults".
- D. Various architectures, gathered from literature searches, would be analyzed to ascertain the relative merits with regard to complexity, reliability, maintainability, cost, etc. This task should be completed assuming Fiber Optic Architecture (FOA) technology.
- E. All fiber optic architecture will require input and output (I/O) interface electronics since the current electronic systems are unlikely to be overtaken by a new technology in the near future. Standard interface components (fibers, sizes, sensors) are crucial to the success of an optical architecture. Standard interfaces will reduce maintainability and installation costs and complexity.

TABLE 4.3-1  
Projected FBL Architecture Evolution

| <u>COCKPIT COMMANDS (MANUAL CONTROLS)</u>   | <u>DISPLAYS</u>  | <u>AIRCRAFT SENSORS</u>  | <u>FLIGHT AND PROPULSION CONTROL</u>  |
|---|--|--|---|
| <ul style="list-style-type: none"> <li>- Electronic Sensors and Switches</li> <li>- Dedicated Fiber Optic Data Links</li> </ul>                     | <ul style="list-style-type: none"> <li>- Dedicated Fiber Optic Links</li> </ul>  | <ul style="list-style-type: none"> <li>- Electronic Sensors and Switches</li> <li>- Dedicated Fiber Optic Data Links</li> </ul>                      | <p><b>NEAR TERM</b></p> <ul style="list-style-type: none"> <li>- Dedicated Fiber Optic Links</li> <li>- Electronic Feedback Sensors/Switches</li> </ul> |
| <b>MEDIUM TERM</b>  |  |  |   |
| <ul style="list-style-type: none"> <li>- Fiber Optic "Star" and/or "Tree" Data Buses</li> <li>- Electronic Feedback Sensors / Switches</li> </ul>   |  |  |   |
| <ul style="list-style-type: none"> <li>- Fiber Optic "Star" and/or "Linear" Data Buses</li> <li>- Electronic Feedback Sensors / Switches</li> </ul> |  |  |   |
| <b>LONG TERM</b>  |  |  |   |
| <ul style="list-style-type: none"> <li>- Fiber Optic Sensors and Switches</li> <li>- Dedicated Fiber Optic Data Links</li> </ul>                    | <ul style="list-style-type: none"> <li>- Fiber Optic Sensors and Switches</li> <li>- Dedicated Fiber Optic Data Links</li> </ul> | <ul style="list-style-type: none"> <li>- Fiber Optic "Star" and/or "Linear" Data Buses</li> <li>- Fiber Optic Feedback Sensors / Switches</li> </ul> |   |
| <ul style="list-style-type: none"> <li>- Optically Multiplexed Fiber Optic Sensors, Switches, and Data Links</li> </ul>                             |  | <ul style="list-style-type: none"> <li>- Optically Multiplexed High-Speed Data Bus</li> </ul>  |   |
| <b>21ST Century</b>   |  |  |   |
| <p>Smart Skins (Including Structural Health/Failure Monitoring)</p>   |  |  |   |

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4.1.1 Fly-By-Wire/Fly-By-Light Flight Control Computer Internal  
Architecture (Continued)

- F. The first generation FBL data bus will be a linear, multi-terminal (one flight control computer and 9 to 24 actuators) network. The flight control computer will transmit position commands to the actuators and each actuator will provide the flight control computer with status and feedback data. An ARINC 629 protocol will be used.
- G. A second generation FBL data bus could evolve with the maturity of fiber optics and fault tolerant computing. All aircraft subsystems may communicate on a single high speed bus. Such a system would most likely employ active networking to accommodate the large number of terminals.

4.2 Fly-By-Light Data Bus

- A. Fiber optic flight control systems need to be developed to address such issues as higher immunity requirements for EME effects, enhanced survivability and reliability, increased data transmission capacity, and reduced aircraft weight. As EME immunity requirements are being raised, the increased use of composite structures on aircraft is decreasing the natural EM shielding heretofore provided by all metal aircraft skins. This situation creates some very real problems, especially for FBW aircraft. While it is feasible to shield the line replaceable units (LRUs), shielding all the data busses and sensor systems in remote areas of the aircraft would be costly and add excessive weight. Because of their inherent EME effects immunity, fiber optic systems provide a potential solution to this problem. Integrated fiber optic flight control systems use fiber optics for most data communications, as well as for sensor and sensing subsystems.
- B. Fiber optic data busses are the cornerstone of integrated fiber optic flight control systems. Linear and/or star types will be used for the avionics and communications data busses, and tree types will be used for FBL data busses.
- C. The linear and star types will be used to interconnect the major LRUs. These data busses should be capable of supporting 32 or more bidirectional terminals. In general, star configurations have a better power budget than their linear counterparts. However, a single point failure may incapacitate the entire bus.

#### 4.2 Fly-By-Light Data Bus (Continued)

D. For a linear-type data bus, asymmetric couplers and nonintrusive taps are currently the most attractive techniques to couple the terminals to the bus fiber. There are three basic linear data bus configurations. The dual-fiber approach has the best power budget and the least-impacting single-point failures, but it requires twice the number of transceivers and terminations as the single-fiber loop configuration as well as a dual-fiber installation. The single-fiber configuration has the worst power budget and requires twice the number of transceivers and terminations as the single-fiber loop configuration. While a single failure has potentially more impact on a single-fiber configuration than on a dual-fiber approach, it cannot bring down the entire single-fiber bus. Presumably a single-fiber configuration would be easier to install than a dual-fiber configuration. The single-fiber loop configuration uses the fewest transceivers and terminations, has a better power budget than the single-fiber configuration, and would be relatively simple to install, but a break at the loop would result in a failure of the entire bus.

E. If the star coupler fails in any of its configurations, then the entire bus fails. The transmissive star configuration has a better power budget than the reflective star configuration, but requires twice as many fibers and terminations.

NOTE: The 1x2 couplers in the reflective star data bus would be incorporated in the transceiver.

F. The bidirectional passive tree type FBL data busses are used to communicate commands from the flight control computers to the actuators, and status and position feedback from the actuators to the flight control computers. In this configuration, the actuators do not communicate with each other. A failure in the coupler closest to the flight control computer or the intervening interconnection will cause the entire bus to fail.

#### 4.3 Fiber Optic Aircraft Architectures

Table 4.3-1 summarizes the projected evolution of FBL aircraft architectures.

#### 4.3 Fiber Optic Aircraft Architectures (Continued)

In the near term, fiber optics will directly replace existing electrical data busses and links. The current architectures will retain the electronic feedback sensors and switches by using opto-electric converters to interface with the fiber optic data links. In the long term, DAC/HI will develop fiber optic bus architectures with "star", "tree", and eventually "linear" configurations. The electrical sensors will be replaced by fiber optic sensors and switches which will be simple, passive devices that are lighter and more EMI immune than their electrical counterparts. Eventually, all optical aircraft systems will be optically multiplexed for very high speed data busses to reduce the wiring required and allow more channels on the same bus. Ultimately, the aircraft fiber optic control system will be integrated with the fiber optic filaments embedded in "smart skin" composite structures to monitor structural health and failure.

As Configuration 1 (Figure 4.3-1) shows, the first step to implementing FBL is to directly substitute fiber optic links into current discrete LRU FBW systems. In this example, quad redundant flight control computers have dedicated data links to every flight control surface and engine. Similarly, the triple redundant secondary flight control computers each have a dedicated link to each flap, slat, and spoiler surface. This is a typical FBW architecture and has many wires. By substituting fiber optics for wires, the aircraft architecture will show weight savings and increased EMI immunity. The displays, propulsion controls, and sensors will all have dedicated data links as they do now.

Configuration 2A (Figure 4.3-2) shows the next evolution of a FBL aircraft. The aircraft sensors and video displays will still have dedicated fiber optic links but now the primary and secondary controls will each have their own integrated "tree" or "star" fiber optic data bus. This will take advantage of the fiber optic medium's ability to handle higher data rates and to handle many channels on the same bus. Here each branch or arm of the data bus connects to a control surface. For the flight control computers, each branch connects to a rudder, elevator, or aileron surface. For the secondary flight control computers, each branch of the "star" fiber optic data bus connects to a flap, slat, or spoiler. This architecture reduces the amount of wiring or fiber optic links from the previous architecture.

Configuration 2B (Figure 4.3-3) further simplifies the flight control architecture after the "linear" fiber optic data bus has been developed. In this configuration, all the flight control computers primary surface and engine controls are multiplexed onto a single linear data bus for each flight control computer and duplicated for redundancy. Similarly, each secondary flight control computer has a double redundant pair of linear data busses

FLIGHT AND  
PROPULSION CONTROL

DEDICATED FIBER  
OPTIC LINKS  
ELECTRONIC FEEDBACK  
SENSORS/SWITCHES

COCKPIT COMMANDS  
(MANUAL CONTROLS)

ELECTRONIC SENSORS  
& SWITCHES  
FIBER OPTIC DEDICATED  
DATA LINKS

DISPLAYS

DEDICATED FIBER  
OPTIC VIDEO LINKS

AIRCRAFT SENSORS

ELECTRONIC SENSORS  
& SWITCHES  
FIBER OPTIC DEDICATED  
DATA LINKS

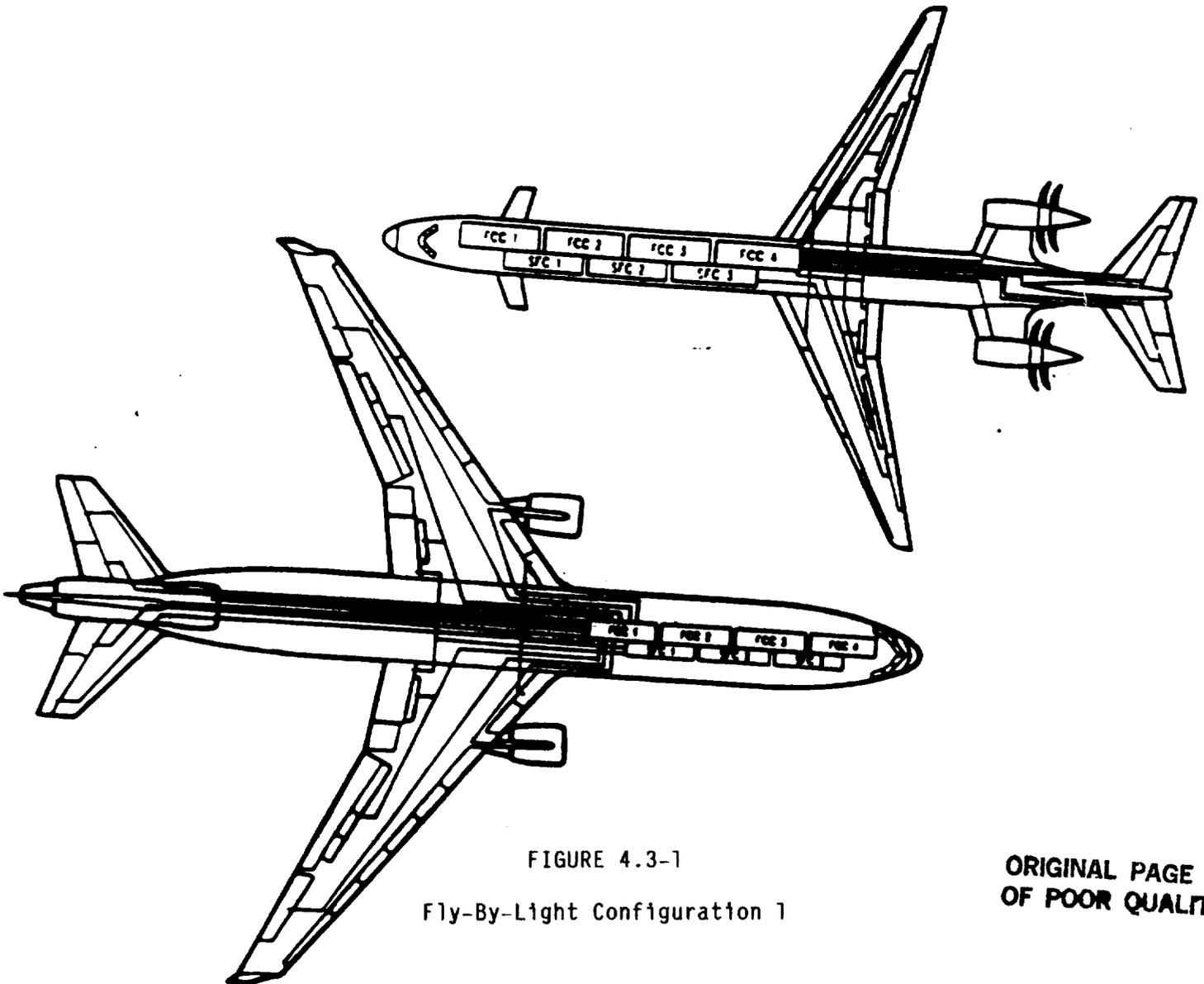


FIGURE 4.3-1

Fly-By-Light Configuration 1

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FLIGHT AND  
PROPULSION CONTROL

FIBER OPTIC STAR  
AND/OR TREE DATA BUSES  
ELECTRONIC FEEDBACK  
SENSORS/SWITCHES

COCKPIT COMMANDS  
(MANUAL CONTROLS)

ELECTRONIC SENSORS  
& SWITCHES  
FIBER OPTIC DEDICATED  
DATA LINKS

DISPLAYS

DEDICATED FIBER  
OPTIC VIDEO LINKS

AIRCRAFT SENSORS

ELECTRONIC SENSORS  
& SWITCHES  
FIBER OPTIC DEDICATED  
DATA LINKS

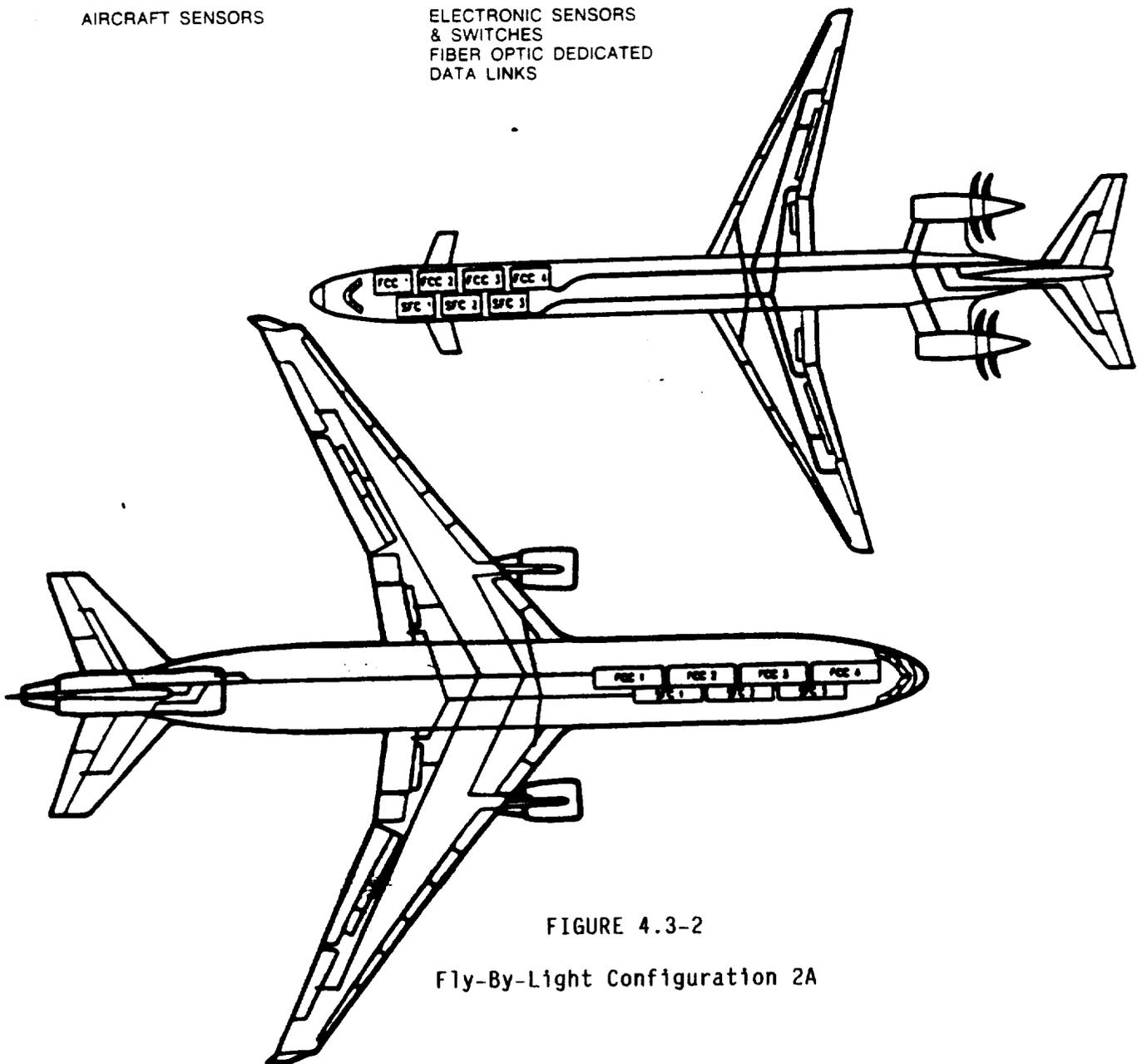


FIGURE 4.3-2

Fly-By-Light Configuration 2A

FLIGHT AND  
PROPULSION CONTROL

FIBER OPTIC 'STAR'  
AND/OR 'TREE' DATA BUSES  
ELECTRONIC FEEDBACK  
SENSORS/SWITCHES

COCKPIT COMMANDS  
(MANUAL CONTROLS)

ELECTRONIC SENSORS  
& SWITCHES  
FIBER OPTIC DEDICATED  
DATA LINKS

DISPLAYS

DEDICATED FIBER  
OPTIC VIDEO LINKS

AIRCRAFT SENSORS

ELECTRONIC SENSORS  
& SWITCHES  
FIBER OPTIC DEDICATED  
DATA LINKS

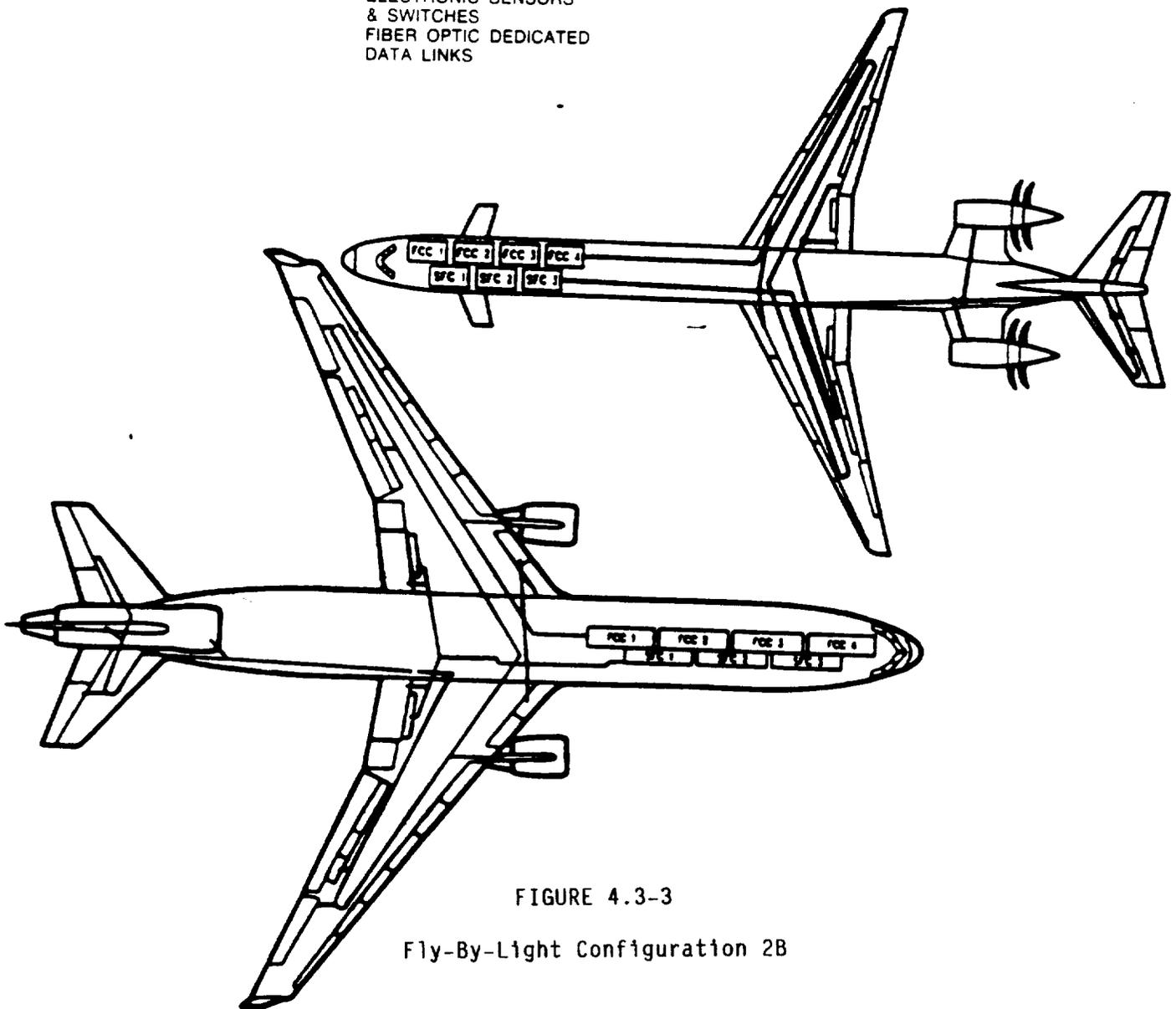


FIGURE 4.3-3

Fly-By-Light Configuration 2B

#### 4.3 Fiber Optic Aircraft Architectures (Continued)

routed throughout the plane to the secondary surfaces. Although the displays and sensors still have dedicated data links, the engine Full Authority Digital Engine Control will be integrated with the primary flight controls in the flight control computers and all the surfaces' data links will be reduced. This saves manufacturing time and weight.

Standard Integrated Modular Avionics (IMA) will provide the technology to produce the architecture in Configuration 3 (Figure 4.3-4). Here, all the aircraft systems are integrated into one avionics rack which contains standard processor modules that perform specific aircraft functions. The Line Replaceable Modules (LRMs) differ only in the software that has been downloaded to them. The IMA rack is replicated four times for redundancy. Fiber optic data busses also interconnect the racks and modules to allow the processors to share tasks. All the primary and secondary flight controls are integrated as well as the sensors, displays, and radio systems. All the systems communicate along a single fiber optic data bus routed throughout the plane and duplicated for redundancy. The technology for IMA should eventually become standard through the ARINC 651 program or the USAF "Pave Pillar" program. The IMA architecture will allow reduced maintenance cost, increased fault tolerance, increased self healing, and reduced weight, space, and wiring. Also, by then, the technology for fiber optic sensors and switches will be developed. These will allow completely fiber optic systems to be EME effects immune and will eliminate the need to route electrical power for sensors.

Finally, by the 21st century, FBL should evolve to Configuration 4 (Figure 4.3-5). Here, the fiber optic technology uses optically multiplexed high speed data busses and smart skins. Future technology will take full advantage of the high bandwidth of fiber optic busses to accomplish high speed optical multiplexing. This will greatly reduce the number of data links required and will save weight and space. The fiber optics embedded in the composite skin will aid manufacturing while the composite is being processed and can sense skin temperature, pressure, strain, and failure of the structure skin while the plane is in service. This can be integrated with the control laws for active surface control.

FLIGHT AND  
PROPULSION CONTROL

FIBER OPTIC 'STAR'  
AND/OR 'TREE' DATA BUSES  
ELECTRONIC FEEDBACK  
SENSORS/SWITCHES

COCKPIT COMMANDS  
(MANUAL CONTROLS)

ELECTRONIC SENSORS  
& SWITCHES  
FIBER OPTIC DEDICATED  
DATA LINKS

DISPLAYS

DEDICATED FIBER  
OPTIC VIDEO LINKS

AIRCRAFT SENSORS

ELECTRONIC SENSORS  
& SWITCHES  
FIBER OPTIC DEDICATED  
DATA LINKS

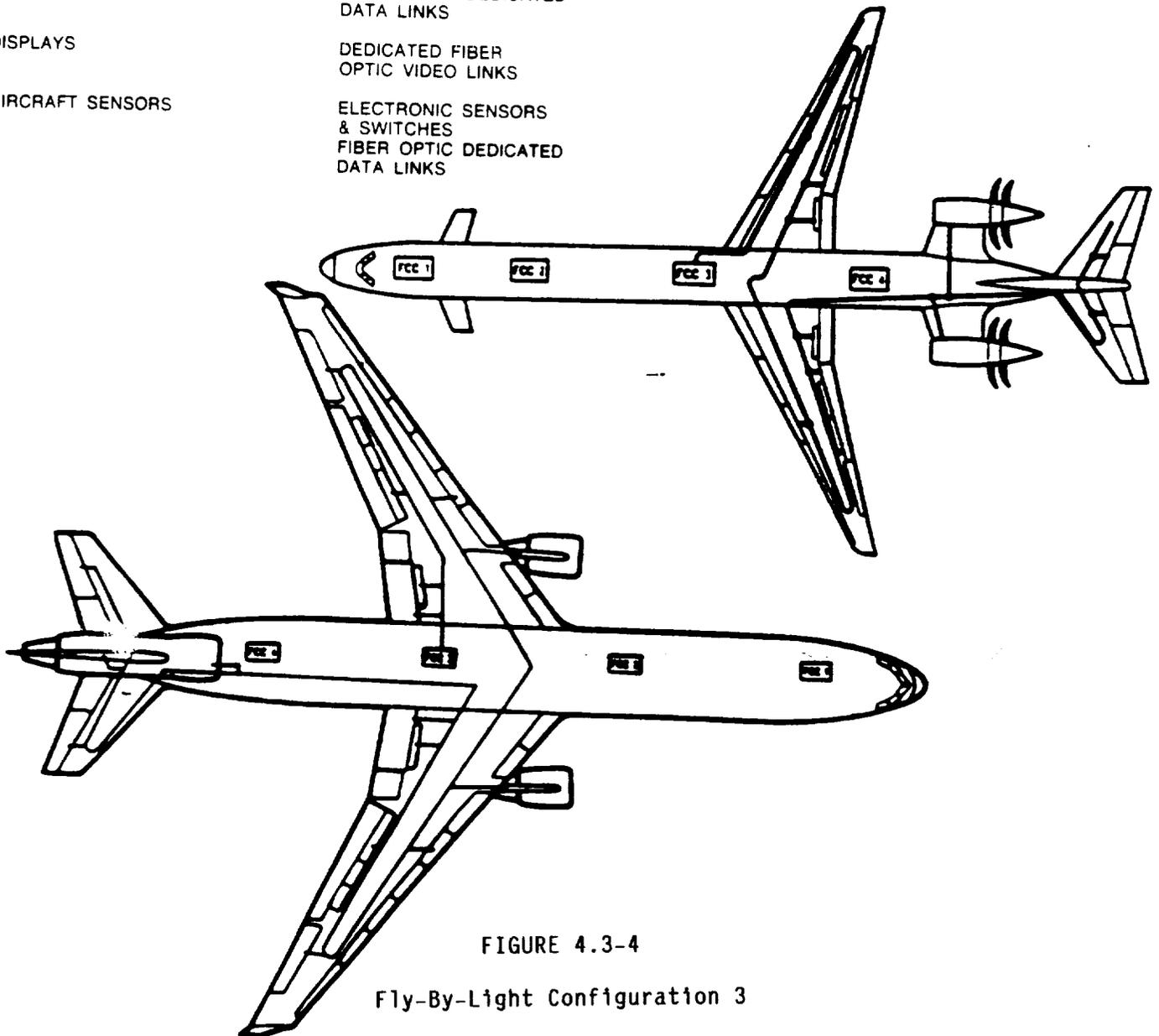


FIGURE 4.3-4

Fly-By-Light Configuration 3

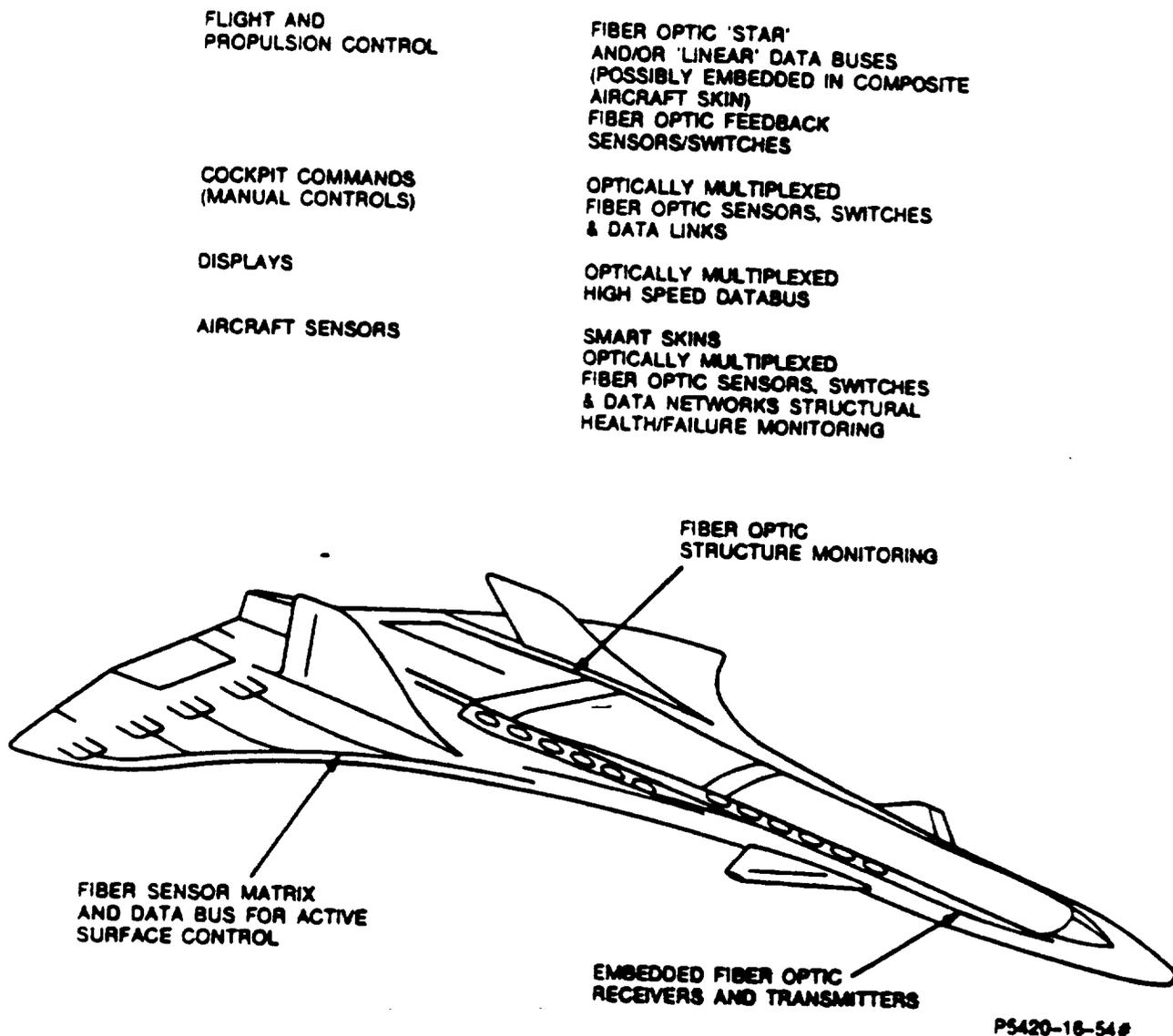


FIGURE 4.3-5

Fly-By-Light Configuration 4

#### 4.4 Fiber Optics Subsystem Design

##### 4.4.1 Avionics Data Bus Subsystem

The avionics data bus subsystem entails the development and validation of a multi-terminal (32 to 50) bidirectional optical data bus. Both linear and star configurations need to be developed for this application. ARINC 629 would be used for the bus protocol in this subsystem, which is designed to interconnect the main aircraft avionics/flight control boxes.

A diagram of a typical Avionics Bus subsystem is shown in Figure 4.4-1.

The optical technology areas required for the development of this subsystem are:

Linear Data Bus - low loss taps, low loss couplers, transceivers.

Star Data Bus - Transmissive or reflective stars, low loss couplers, transceivers.

##### 4.4.2 Fly-By-Light Actuator Data Bus Subsystem

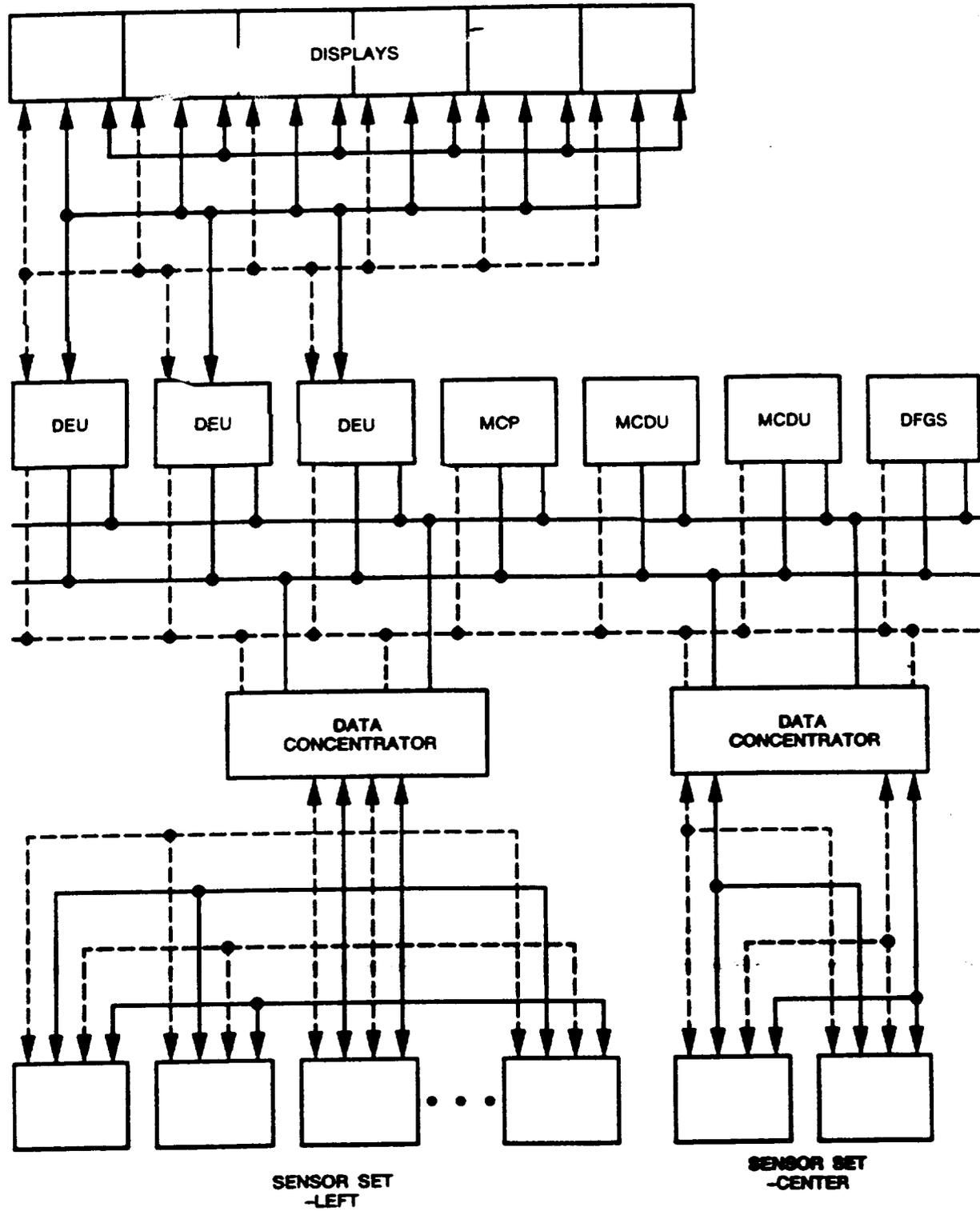
The fly-by-light actuator data bus subsystem entails the development and validation of a multi-terminal (one flight control computer and nine Actuators) bidirectional optical tree data bus. This subsystem is designed such that the flight control computer transmits position commands to the actuators and each actuator provides the flight control computer with status and feedback data. Communication among actuators is neither desired nor permitted in this configuration. ARINC 629 would be used for the bus protocol in this subsystem. A possible configuration for this subsystem is shown in Figure 4.4-2. Also a more detailed diagram is shown in Figure 4.4-3.

##### 4.4.3 Sidestick Command Optical Sensing Subsystem

The sidestick command optical sensing subsystem entails the development and validation of optical position or pressure/force sensors in a conventional sidestick controller. An example of a possible sidestick is shown in Figure 4.5-4. This sidestick would require two optical sensors, one each in pitch and roll. The two optical sensors are multiplexed on a single trunk fiber with the command position data being decoded at a remote electronics unit.

Fly-By-Light  
Technology Development Plan

FOLDOUT FRAME



FOLDOUT FRAME

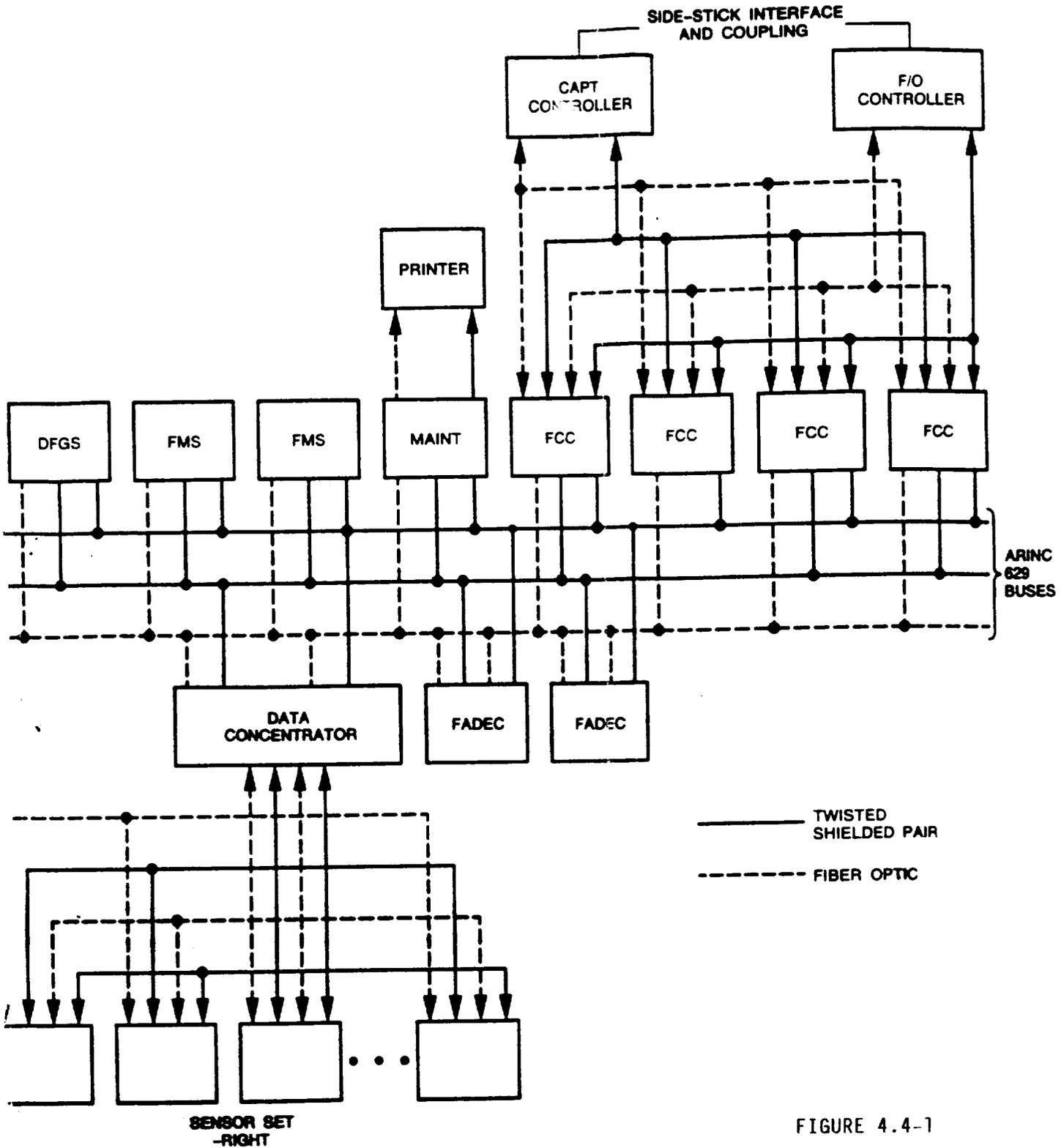


FIGURE 4.4-1

Avionics Data Bus Subsystem Diagram

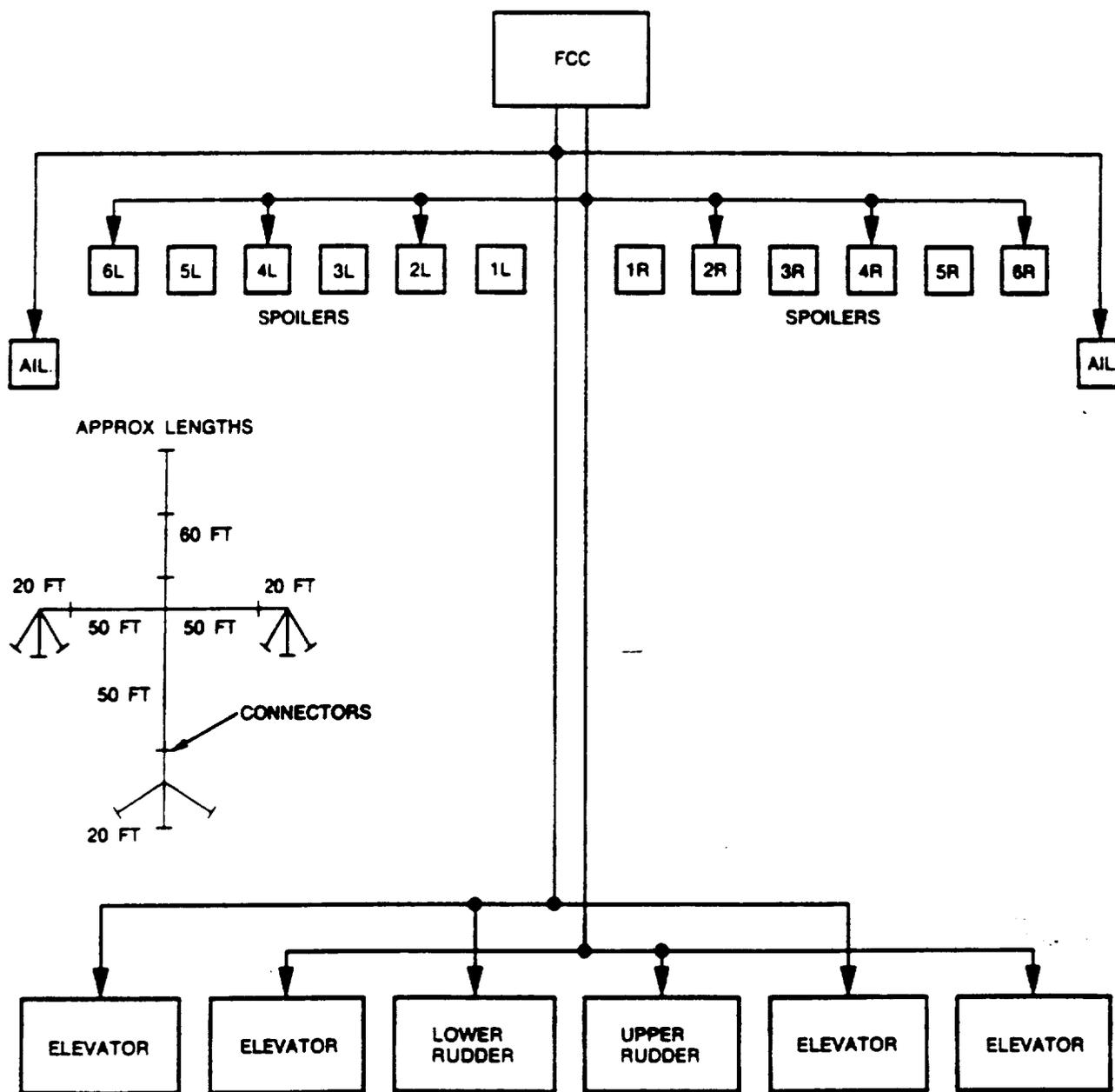


FIGURE 4.4-2

Fly-By-Light Actuator Data Bus Subsystem

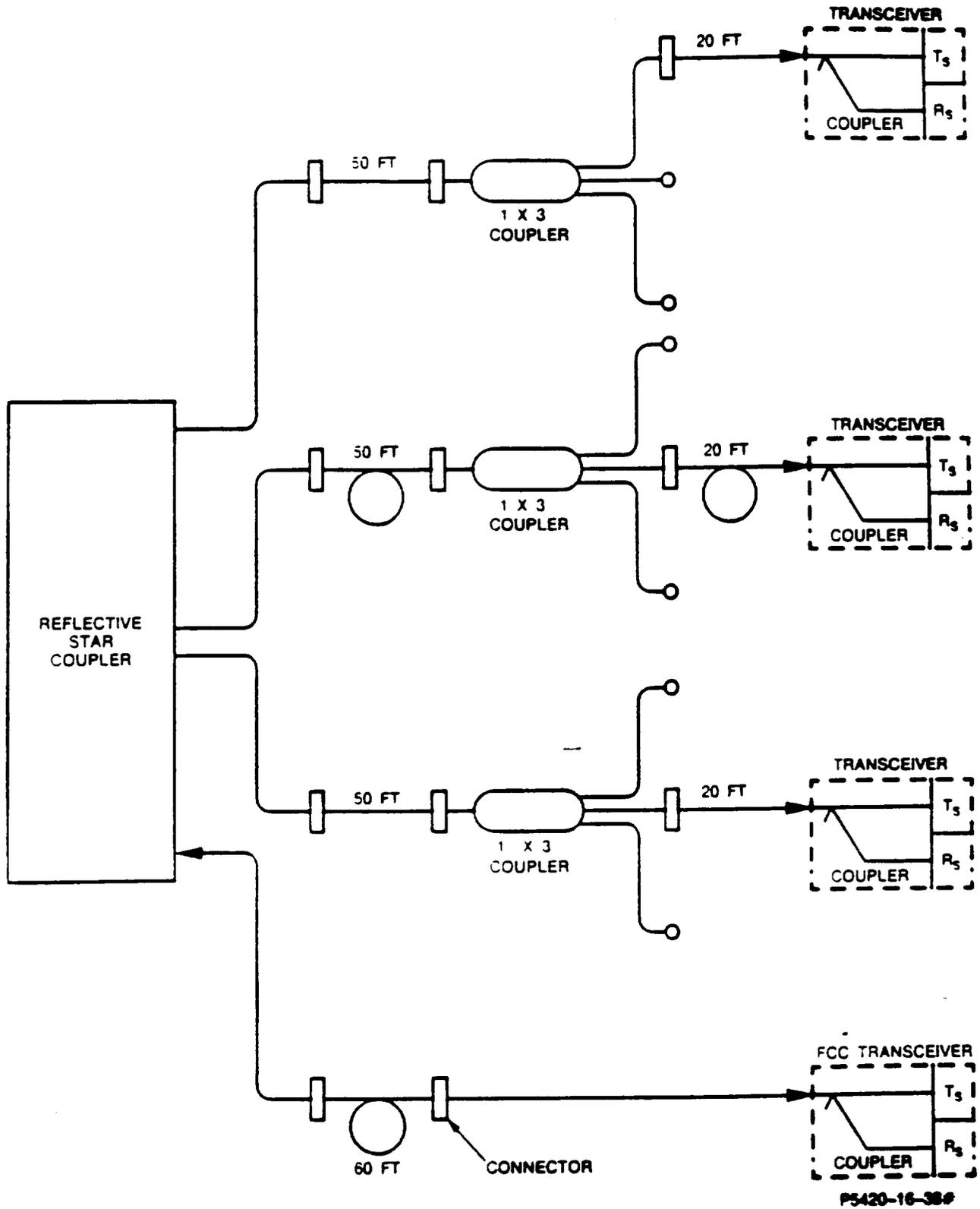


FIGURE 4.4-3

Fly-By-Light Actuator Data Bus Subsystem Detailed Diagram

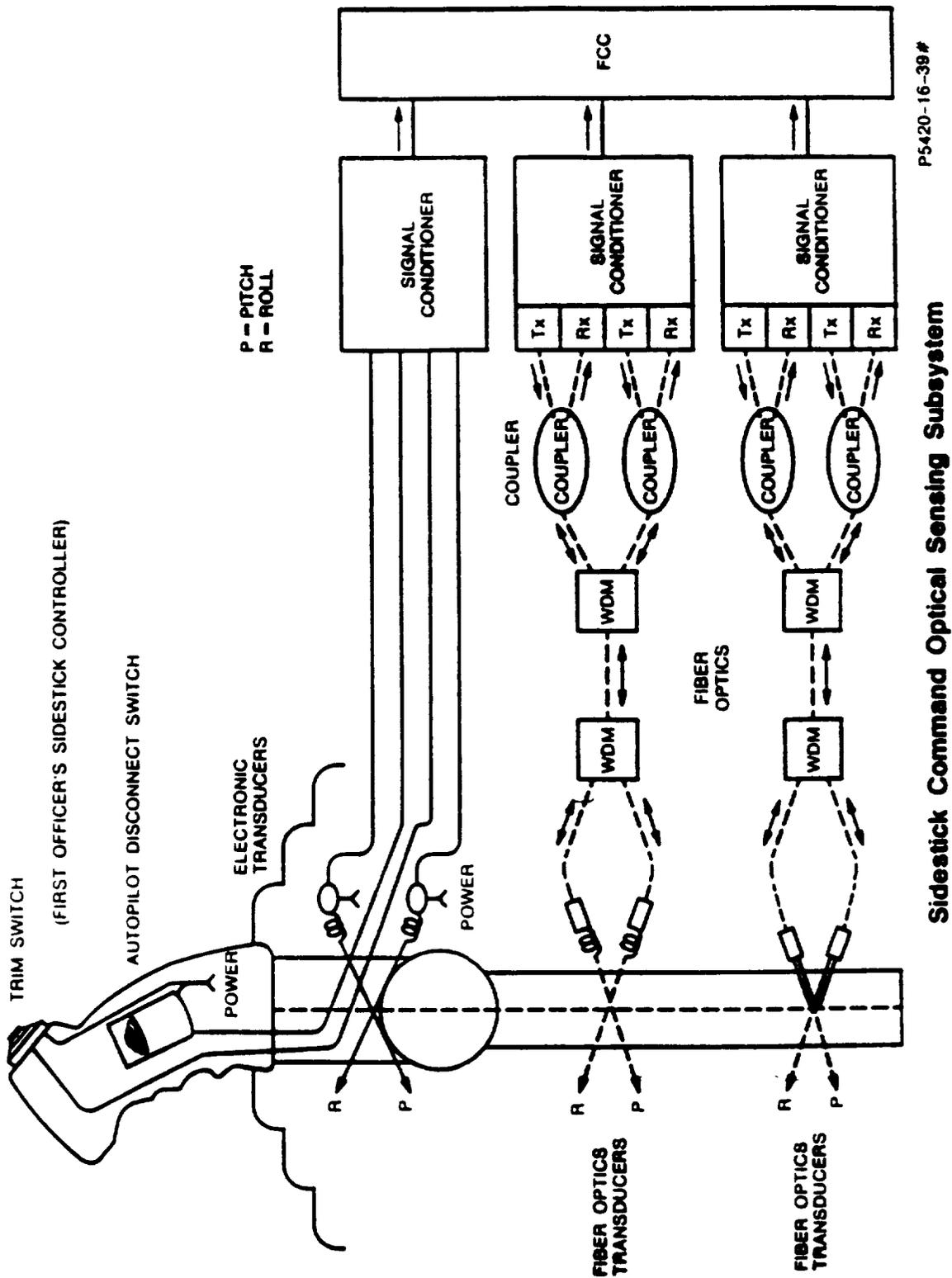


FIGURE 4.4-4

Sidestick Command Optical Sensing Subsystem

#### 4.4.4 Rudder Command Optical Sensing Subsystem

The rudder command optical sensing subsystem entails the development and validation of optical position or pressure sensors to interface with conventional rudder pedals. The subsystem will require one optical sensor for each rudder pedal. If both rudder pedals are optically interfaced, optical multiplexing can be incorporated into the subsystem design. In Figure 4.5-5 is a rudder command optical sensing subsystem that DAC is developing for an IRAD project.

#### 4.4.5 Optically Powered Flap and Slat Position Indication Subsystem

The optically powered flap and slat position indication subsystem entails the development and validation of ultra-low power optically powered sensors and a multiplexing scheme. This subsystem optically powers seven sensors via a single trunk fiber and returns position and status data to the microprocessor unit over the same fiber. The position data from the sensors is interpreted by a microprocessor unit which in turn drives the appropriate displays. This subsystem is shown in Figure 4.4-6; also a more detailed diagram is shown in Figure 4.4-7.

#### 4.4.6 Autothrottle Optical Sensor Subsystem

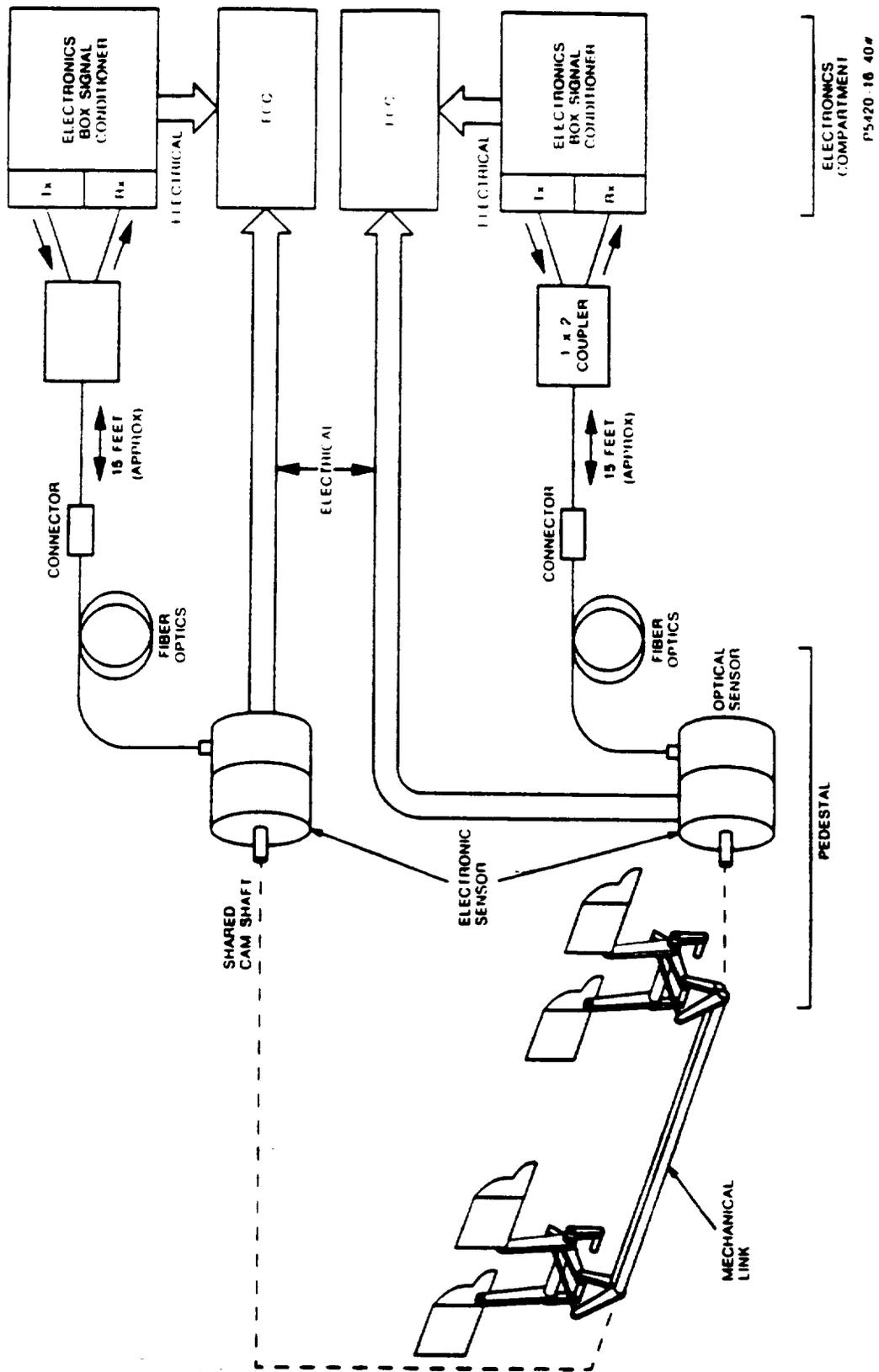
The autothrottle optical sensor system entails the development and validation of optical accelerometers, position sensors, and data links. This subsystem is designed to permit different portions to be implemented incrementally. A diagram of an autothrottle optical sensor subsystem that DAC is developing for an IRAD is shown in Figure 4.4-8.

#### 4.4.7 Throttle Lever Command Optical Sensing Subsystem

The throttle lever command optical sensing subsystem entails the development and validation of an optical position sensor that is interfaced with a conventional throttle quadrant. The subsystem will require one optical sensor for each throttle lever. Optical multiplexing can be incorporated into the subsystem design. A typical throttle subsystem is shown in Figure 4.4-9.

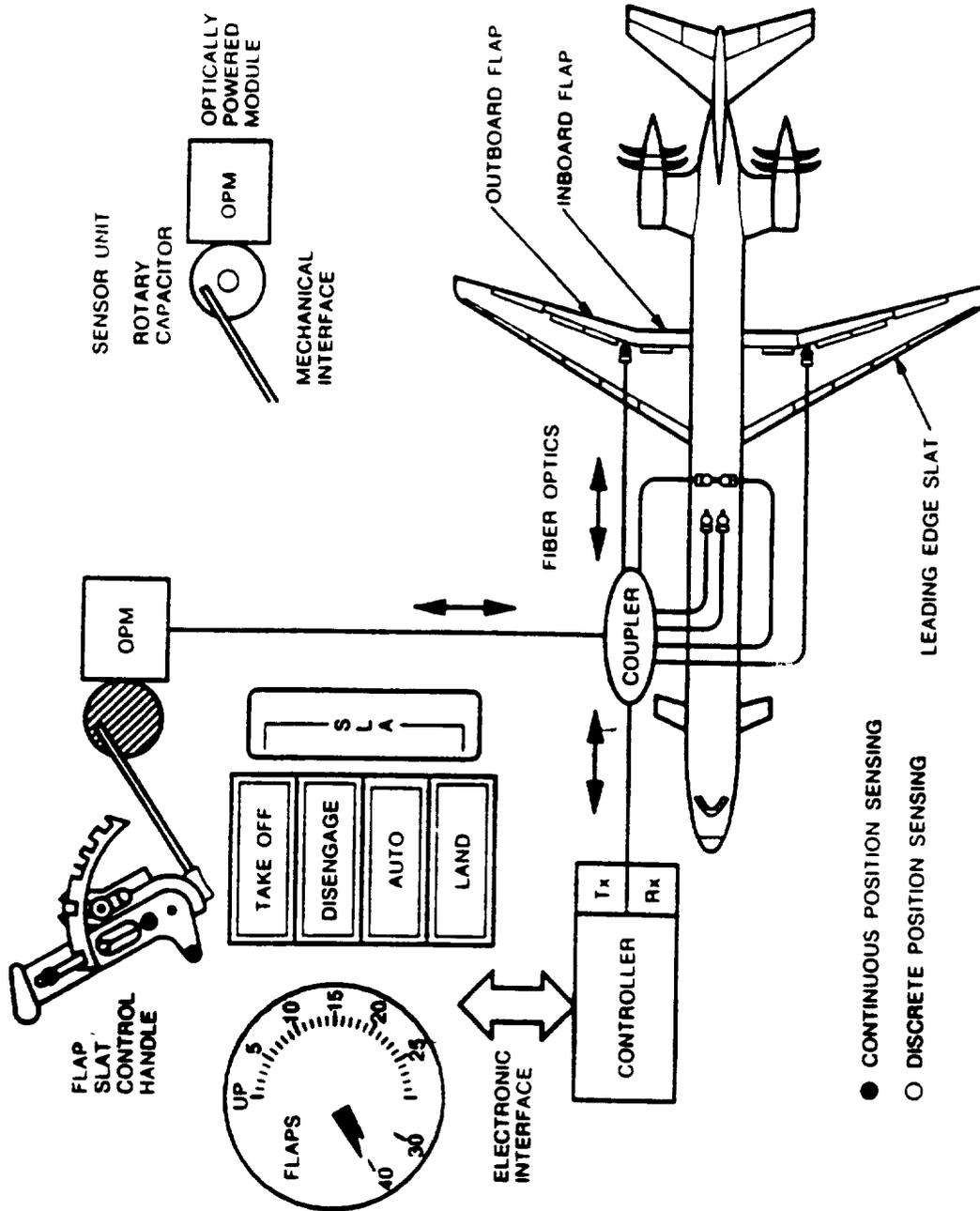
#### 4.4.8 Autothrottle Optical Switch Sensing Subsystem

The autothrottle optical switch sensing subsystem entails the development and validation of optical limit switches to be installed on the throttle quadrant to provide autothrottle disconnect, TO/GA, low limit, and reverse thrust inputs to the flight control computers. An autothrottle switch subsystem is shown in Figure 4.4-10.



Rudder Command Optical Sensing Subsystem Diagram

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FIGURE 4.4-6

Optically Powered Flap and Slat Position Indication Subsystem

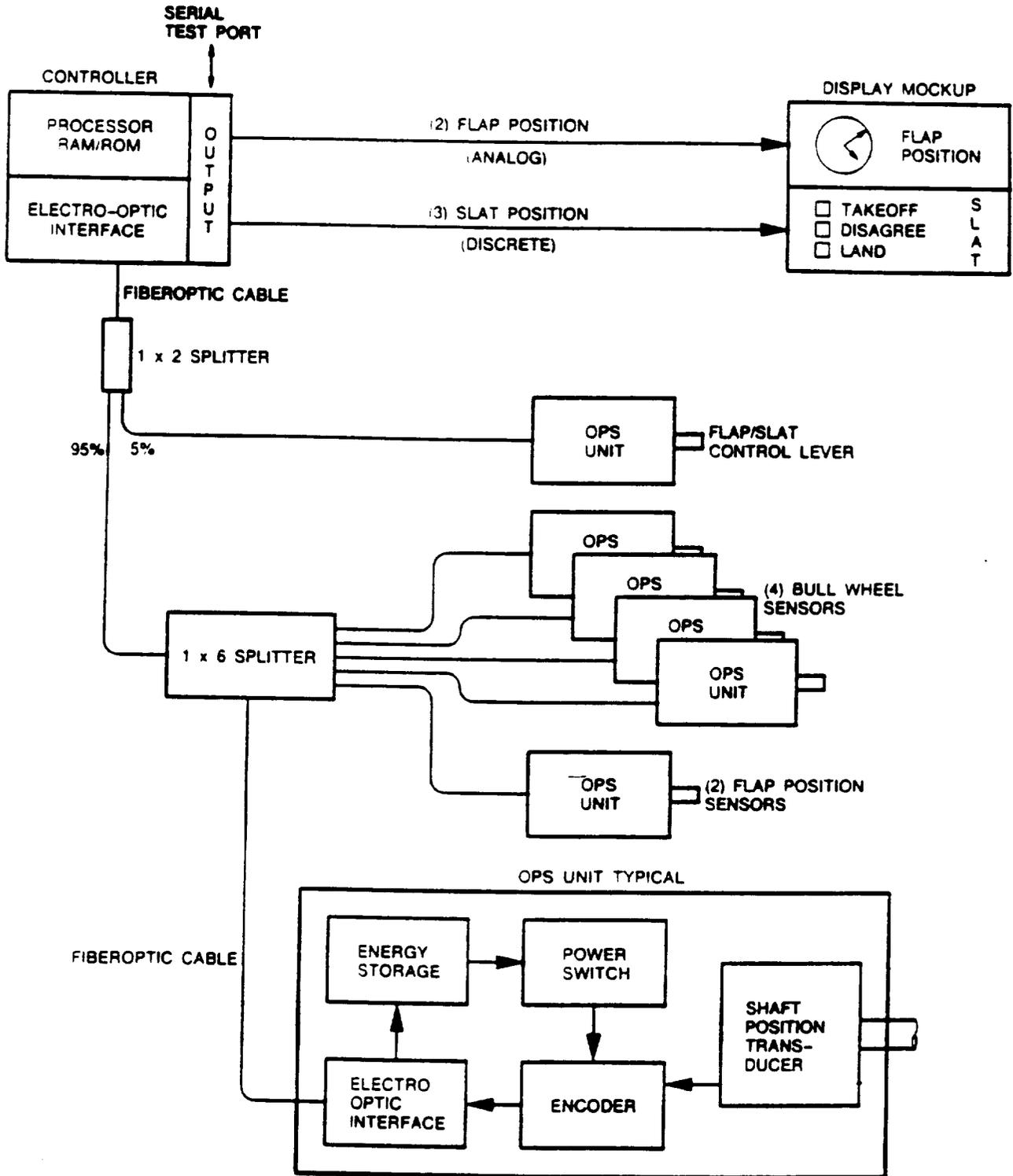
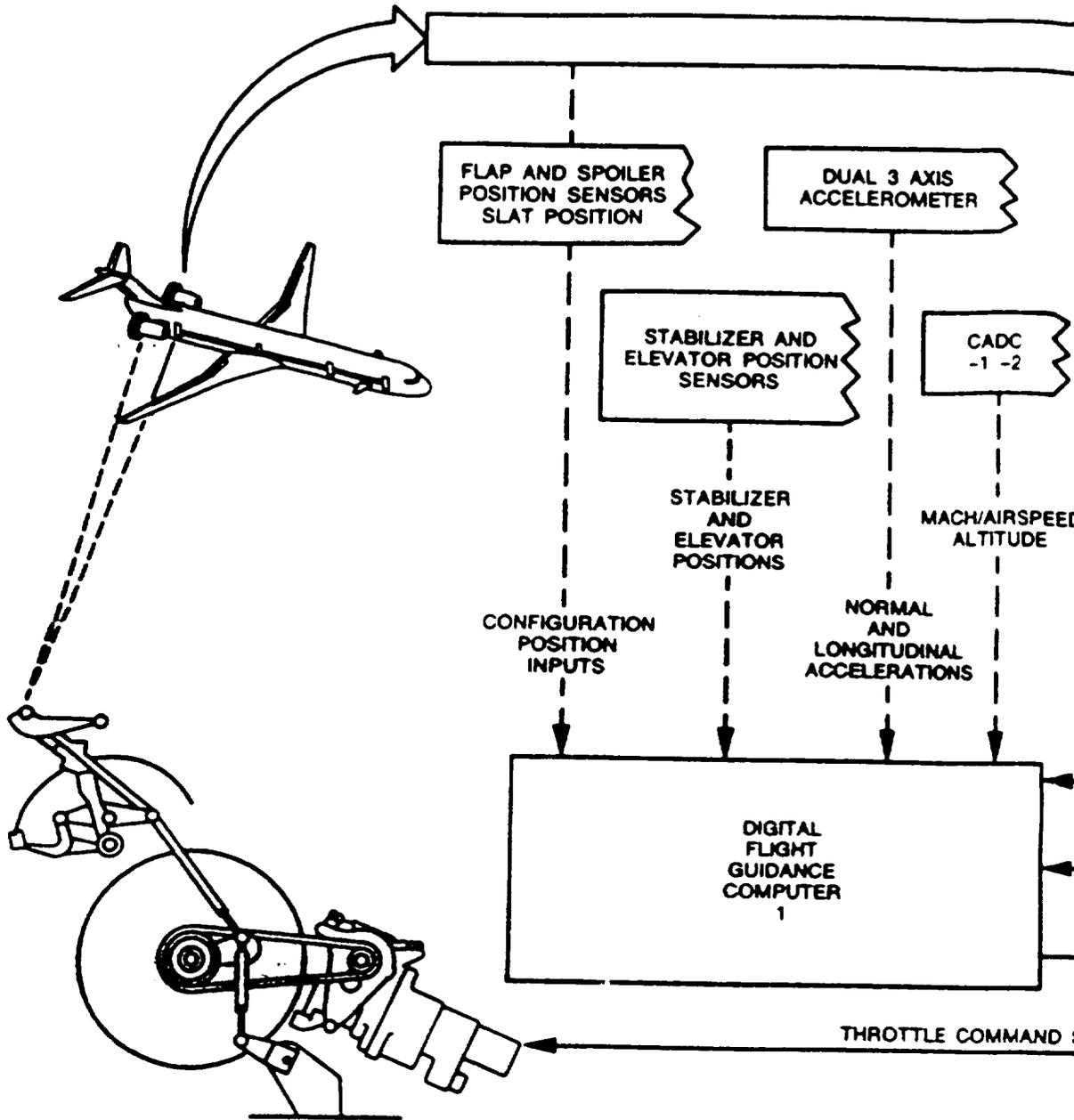


FIGURE 4.4-7

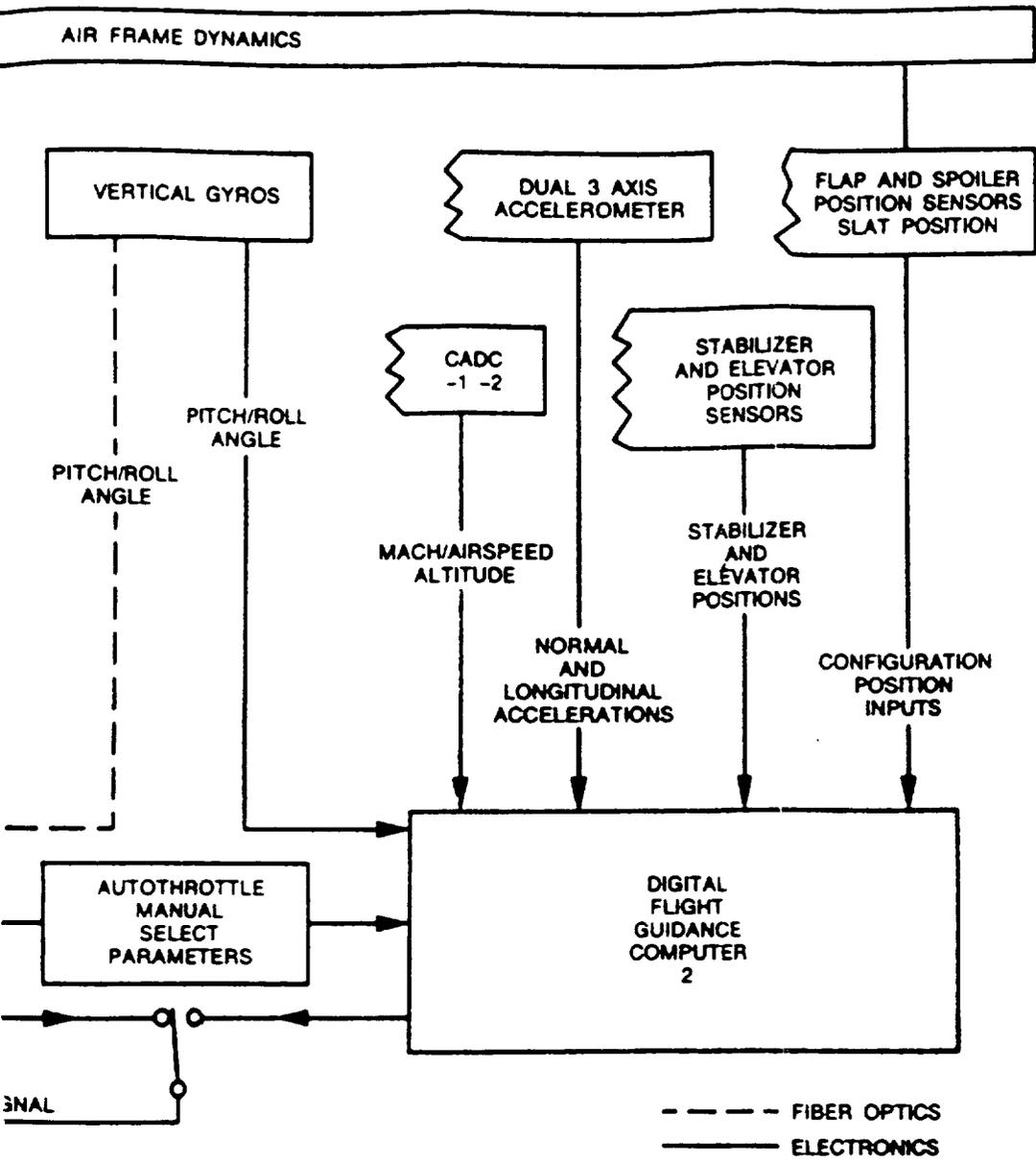
Optically Powered Flap and Slat Position  
Indication Subsystem Detailed Diagram

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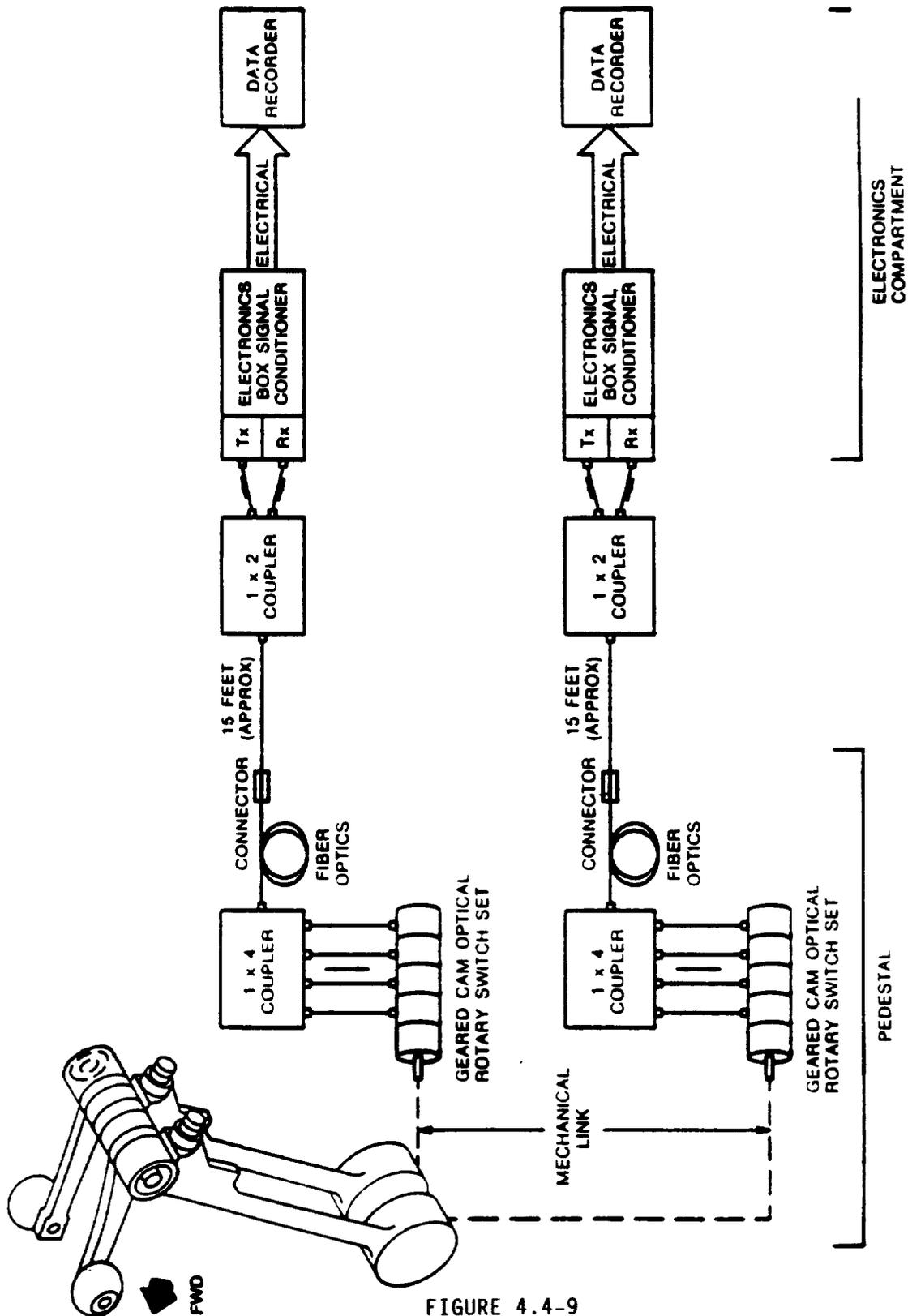
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FIGURE 4.4-8

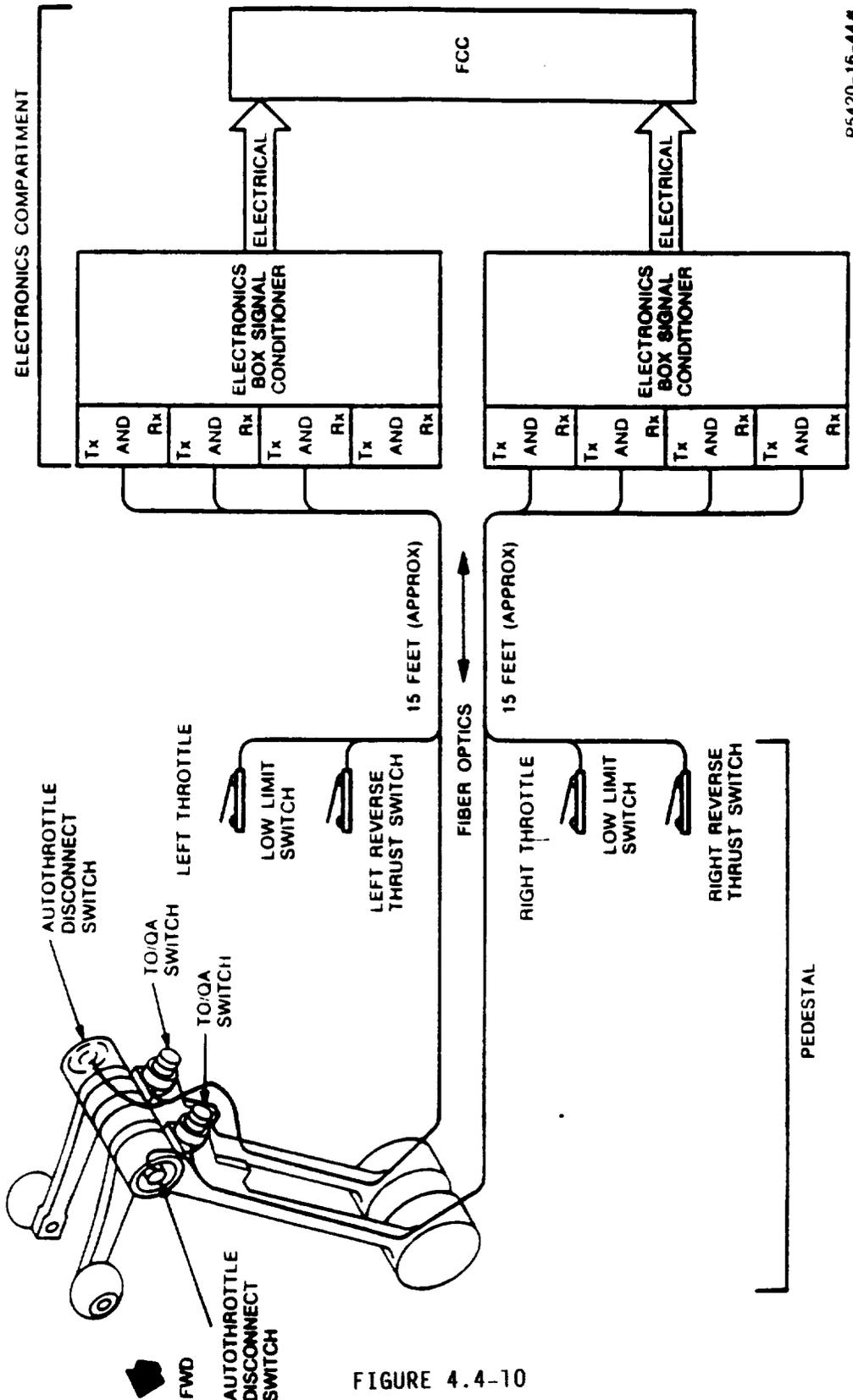
Autothrottle Optical Sensor Subsystem Design



P5420

FIGURE 4.4-9

Throttle Lever Optical Command Position Sensing Subsystem



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FIGURE 4.4-10

Autothrottle Optical Switch Sensing Subsystem Diagram



## 5.0 IMPACT OF FLY-BY-LIGHT (BENEFIT ANALYSIS)

### 5.1 Background

In this section, the weight and cost benefits of fly-by-light (FBL) will be discussed for two classes of commercial transport aircraft, a narrowbody and a widebody.

#### 5.1.1 EME Immunity

Immunity to the effects of the EME, weight savings, and high bandwidth currently drive the development and implementation of fiber optics on aircraft.

Both military and commercial aircraft users have experienced "soft faults" or erroneous system failure messages resulting from interference by the EME of their avionic systems. The long runs of electrical signal wire and data busses act like an antenna running the length of the plane. These pick up electrical noise from the environment and from each other. The airlines blame soft faults for 40% of their aircraft downtime and claim that 60% of the avionics LRUs removed come back with "no fault found" messages attached and the military suspects such interference causes certain unexplained accidents. Since fiber optics are inherently immune to EM fields, they will reduce or eliminate interference caused by electromagnetic noise. The military customers need the fiber optic components performance benefits. Specifically, the military requires the high data transmission rate provided by fiber optics, the weights savings they provide, and their immunity to the EME for electronic warfare protection.

In addition, the commercial customer will benefit from fiber optics through reduced manufacturing costs and a greater return on their investment from the aircraft. Many manufacturing steps are simplified with fiber optics. Also, aircraft fiber optic data lines exhibit the performance advantages over their electrical counterparts. Note that additional safety benefits result from the optical fiber's inherent immunity to short circuits and sparks.

#### 5.1.2 Weight Savings

Directly substituting fiber optic data links for wire signal paths would significantly reduce weight on aircraft. Furthermore, since fiber optics have a higher bandwidth than wire, fiber optic data busses have much higher data transmission rates than electrical data busses. Video display drivers, backplane data busses, and other high speed applications require these high transmission rates. Also, the higher bandwidth allows equivalent electrical data bus runs to be replaced with one tenth the amount of fiber optic wire runs to greatly reduce aircraft weight and production cost.

## 5.2 Narrowbody Aircraft

For the narrowbody, the MD-88 will be used as the baseline and a series of evolutionary changes will be compared involving the flight control and engine control systems. This will encompass mechanical, fly-by-wire (FBW), and near and medium term FBL. Both sensor and data distributions are incorporated in this analysis.

### 5.2.1 Narrow Body Comparison

To estimate the weight savings expected for a narrowbody aircraft, we calculated the wire weights for an MD-88 aircraft and scaled these as appropriate to equivalent fiber optic weights. Figure 5.2-1 shows the dimensions and general features of an MD-80 derivative aircraft and all the primary and secondary flight control surfaces which would have fiber optic lines routed to them in a FBL configuration. Table 5.2-1 summarizes the immediate weight savings realized by direct substitution of fiber optics for wire on the aircraft in the current configuration. This shows a 352 lbs weight savings in flight controls and 804 lbs for the aircraft. Table 5.2-2 shows a long-term FBW configuration giving a 750 lb weight savings over the current mechanical system. The advanced FBL configuration would save 1200 lbs. The long-term configuration would be designed to take full advantage of the FBW technology.

Table 5.2-3 compares the three configurations in terms of engineering, development and recurring cost and compares the three maintenance life cycle costs. Notice that conventional electrical cable requires maintenance for its shielding. Fiber optics do not require this shielding and can save the cost of its maintenance. Table 5.2-3 then totals the Direct Operating Cost and Return on Investment for the three configurations. It shows a 7.5% reduction in direct operating costs for FBW and a 10% reduction for FBL. There is a 9.6% improvement in return on investment for FBW and a 20% improvement for FBL. Therefore, implementing fiber optics on aircraft would greatly reduce the cost of ownership of aircraft and increase the profit of operating them.

## 5.3 Widebody Aircraft

For the widebody, a concept aircraft similar to the MD-11 (mechanical control with full time stability augmentation) will be used as the baseline. For this study, to exemplify the benefits achieved by converting long wire runs in data distribution systems to fiber optics, only the flight control computers to actuation system are converted to fiber optics.

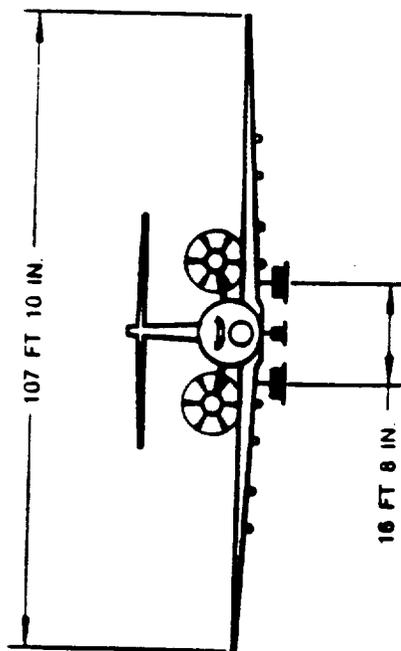
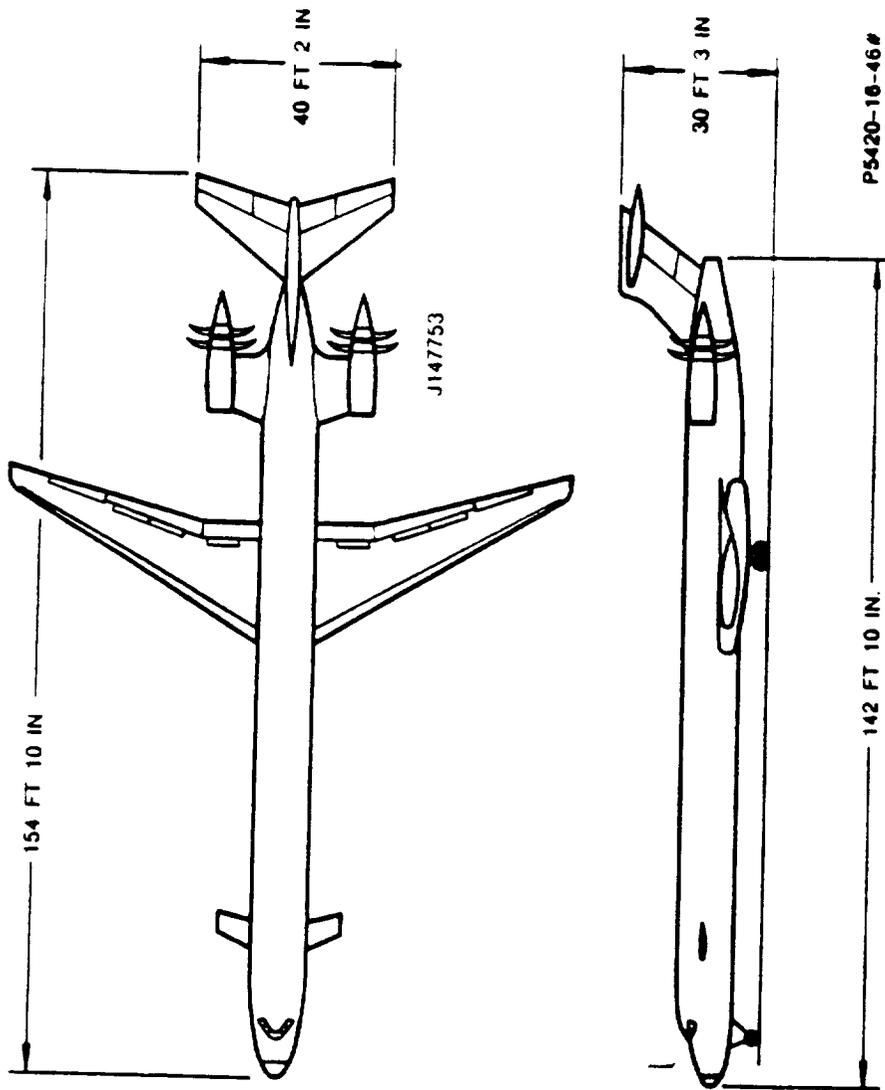


FIGURE 5.2-1

Narrowbody Baseline Aircraft Dimensions

**Direct (signal to signal) Conversion of Flight Control  
Signal Wires to Fiber Optics for a MD-80 Type Aircraft**

|                               | <u>Flight Control</u> | <u>Aircraft</u> |
|-------------------------------|-----------------------|-----------------|
| Total Signal Path Length      | 41,040 ft             | 93,540 ft       |
| Signal Wire Weight            | 418 lbs               | 954 lbs         |
| Signal Fiber Optic Weight     | 66 lbs                | 150 lbs         |
| <b>Approx. Weight Savings</b> | <b>352 lbs</b>        | <b>804 lbs</b>  |

- \* wire 20 gauge shielded/jacketed
- \* fiber 100/140 micron jacketed
- \* wire ground return loops eliminated for fiber optics
- \* new electro-optics (Tx & Rx) added

TABLE 5.2-1

**Narrowbody Weight Comparison**

|  | <u>Mechanical<br/>Baseline</u> | <u>Fly-By-Wire</u> |                  | <u>Fly-By-Light</u> |                  |
|--|--------------------------------|--------------------|------------------|---------------------|------------------|
|  |                                | <u>Near Term</u>   | <u>Long Term</u> | <u>Near Term</u>    | <u>Long Term</u> |
| Cockpit Commands<br>(MANUAL CONTROLS)        | 105                            | 94                 | 94               | 90                  | 85               |
| Flight and Propulsion<br>Control Avionics    | 270                            | 380                | 200              | 380                 | 200              |
| Avionics                                     | 1223                           | 823                | 790              | 823                 | 790              |
| Electrical Power                             |                                | 100                | 150              | 100                 | 150              |
| Flight and Propulsion<br>and Display Control | 558                            | 310                | 85               | 70                  | 18               |
| Aircraft Sensors                             | 380                            | 426                | 426              | 140                 | 105              |
| <b>TOTAL</b>                                 | <b>2,536</b>                   | <b>2,133</b>       | <b>1,745</b>     | <b>1,603</b>        | <b>1,348</b>     |

- \* all weights in pounds
- \* flight and propulsion control avionics include primary and secondary flight control as well as engine and actuator control electronics
- \* mechanical or other backup systems not included
- \* mechanical baseline includes dual SCAS

TABLE 5.2-2

**Narrowbody Weight Comparison**

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Narrowbody Flight Controls Development and Production Cost Comparisons

|                               | <u>Mechanical Baseline</u> | <u>Fly-By-Wire</u> | <u>Projected Fly-By-LIGHT (RQM)</u> |
|-------------------------------|----------------------------|--------------------|-------------------------------------|
| <b>Non-Recurring</b>          |                            |                    |                                     |
| Engineering and Development   | \$8,400,000                | \$34,420,000       | \$49,570,000                        |
| <b>Recurring Per Aircraft</b> |                            |                    |                                     |
| Material Cost                 | \$53,000                   | \$30,000           | \$32,000                            |
| Avionics Assembly Co          | \$42,000                   | \$21,000           | \$17,000                            |
| Equipment Cost                | \$831,000                  | \$848,000          | \$870,000                           |
| Handling/Installation         | \$194,000                  | \$156,000          | \$180,000                           |
| <b>TOTAL</b>                  | <b>\$1,120,000</b>         | <b>\$1,055,000</b> | <b>\$1,099,000</b>                  |

Narrowbody Maintenance Life Cycle Cost Comparison

|                 | <u>\$/HR</u> | <u>\$/Life</u>     | <u>\$/HR</u> | <u>\$/Life</u>   | <u>\$/HR</u> | <u>\$/Life</u>   |
|-----------------|--------------|--------------------|--------------|------------------|--------------|------------------|
| Autoflight      | 3.3          | \$200,000          | 2.8          | \$168,000        | 2.2          | \$132,000        |
| Communications  | 1.3          | \$ 78,000          | 1.0          | \$ 60,000        | 0.8          | \$ 48,000        |
| Flight Controls | 1.2          | \$ 72,000          | 0.8          | \$ 48,000        | 0.5          | \$ 30,000        |
| Instrumentation | 1.1          | \$ 66,000          | 0.9          | \$ 54,000        | 0.7          | \$ 42,000        |
| Navigation      | 9.8          | \$588,000          | 7.9          | \$474,000        | 6.3          | \$378,000        |
| <b>TOTAL</b>    | <b>16.7</b>  | <b>\$1,004,000</b> | <b>13.4</b>  | <b>\$804,000</b> | <b>10.5</b>  | <b>\$630,000</b> |

Narrowbody Flight Controls Life Cycle Cost Comparison per Aircraft

|                                       |                      |                      |
|---------------------------------------|----------------------|----------------------|
| Non-Recurring                         | \$32,000             | \$51,000             |
| Recurring                             | (\$65,000)           | (\$21,000)           |
| Maintenance (Labor Equivalent Revenue | (\$200,000)          | (\$374,000)          |
| (relative to baseline)                | (\$2,642,000) nt     | (\$6,117,000) nt     |
| <b>TOTAL</b>                          | <b>(\$2,875,000)</b> | <b>(\$6,461,000)</b> |

-----

**DIRECT OPERATING COST**            7.5% reduction            10.0% reduction

**RETURN ON INVESTMENT**            9.6% improvement            20.0% improvement

- \* utilization 8 hrs/day
- \* 800 aircraft production run
- \* ave life = 20 yrs
- \* 60,000 flight hrs
- \* "( )" additional profits above the mechanical baseline aircraft
- \* all type include flight guidance system
- \* ave speed = 320 mi/hr
- \* ave yield = 80.10/passenger mi.
- \* load factor = 0.7
- \* "nt" near term weight savings
- \* costs estimated in 1989 dollars
- \* averages for fly-by-wire and fly-by-light configurations
- \* mechanical baseline includes flight engineer

TABLE 5.2-3

Narrowbody Flight Controls Development, Production, Maintenance, and Life Cycle Cost Comparisons

### 5.3.1 Dimensions

Figure 5.3-1 shows the dimensions for a typical DC-10 aircraft used to estimate the cost and weight savings for a widebody class aircraft. For simplicity, no flight control computer cross channel conversion was considered. Tables 5.3-1 through 5.3-3 repeat the same analysis for a DC-10 baseline widebody aircraft. Here, the long-term FBW configuration saves 1290 lbs and the FBL configuration saves 1760 lbs for near term direct substitution of fiber for wire, for a direct signal wire to fiber conversion we save 1120 lbs for flight controls and 2615 lbs for aircraft. The life cycle analysis shows a 4.5% DOC reduction for FBW and 7.4% for FBL. The analysis estimates a 5.1% increase in return on investment for FBW and a 10.2% improvement in direct operating cost for FBL. Hence, implementing fiber optics on a widebody aircraft results in even greater cost savings and profit increases than on narrowbody aircraft.

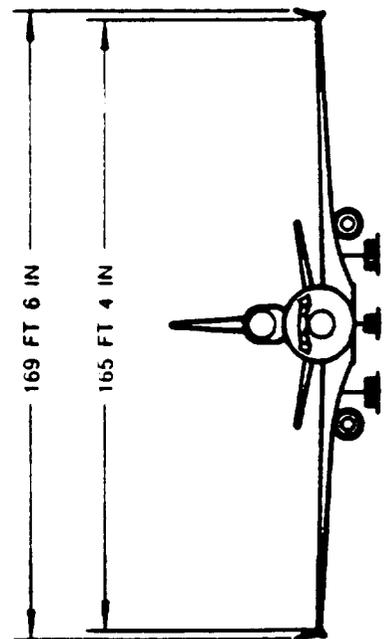
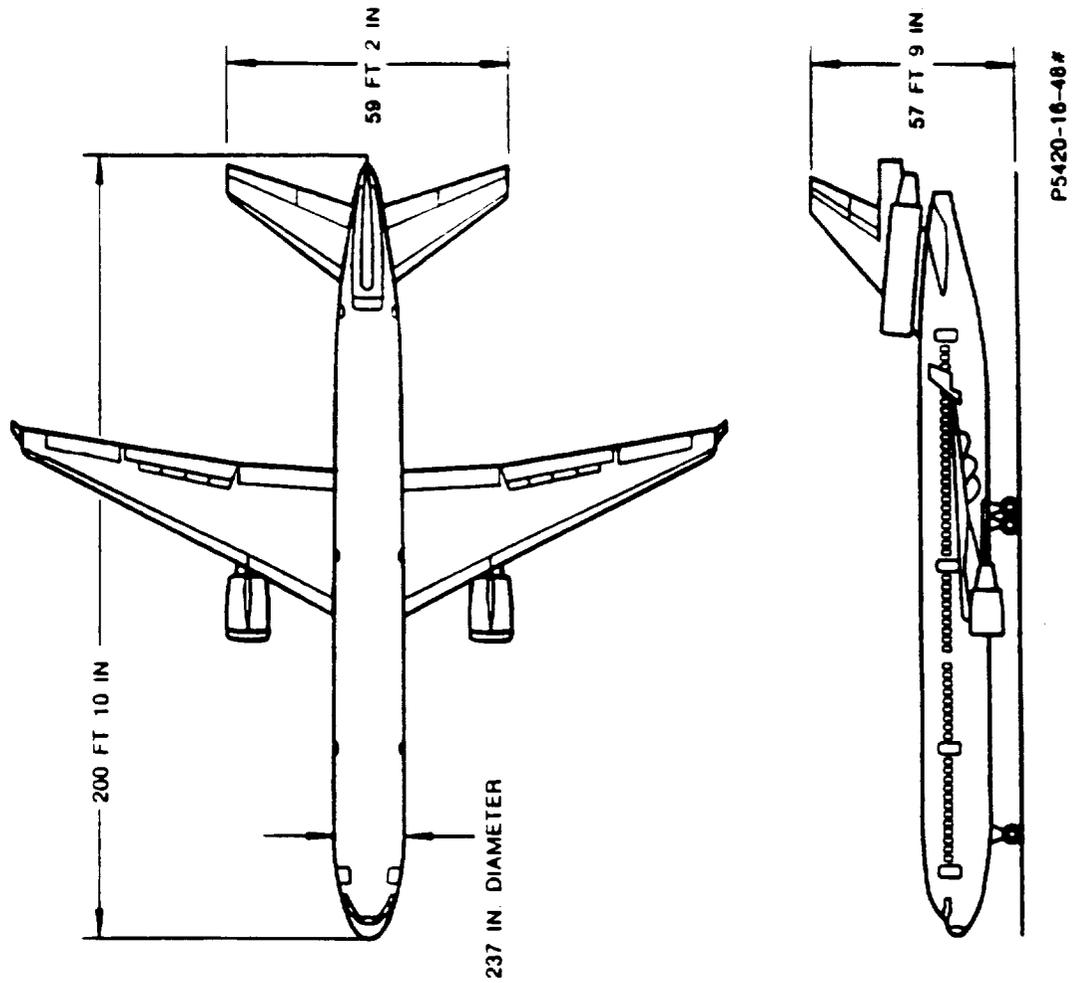


FIGURE 5.3-1

Widebody Baseline Aircraft Dimensions

TABLE 5.3-1

Widebody Weight Comparison

Direct (signal to signal) Conversion of Flight Control  
Signal Wires to Fiber Optics for a MD-11 Type Aircraft

|                               | <u>Flight Control</u> | <u>Aircraft</u>  |
|-------------------------------|-----------------------|------------------|
| Total Signal Path Length      | 130,500 ft            | 303,200 ft       |
| Signal Wire Weight            | 1,330 lbs             | 3,100 lbs        |
| Signal Fiber Optic Weight     | 210 lbs               | 485 lbs          |
| <b>Approx. Weight Savings</b> | <b>1,120 lbs</b>      | <b>2,615 lbs</b> |

- \* wire 20 gauge shielded/jacketed
- \* fiber 100/140 micron jacketed
- \* wire ground return loops eliminated for fiber optics
- \* new electro-optics (Tx & Rx) added

TABLE 5.3-2

Widebody Weight Comparison

|  | <u>Mechanical<br/>Baseline</u> | <u>Fly-By-Wire</u> |                  | <u>Fly-By-Light</u> |                  |
|--|--------------------------------|--------------------|------------------|---------------------|------------------|
|  |                                | <u>Near Term</u>   | <u>Long Term</u> | <u>Near Term</u>    | <u>Long Term</u> |
| Cockpit Commands<br>(MANUAL CONTROLS)        | 180                            | 110                | 110              | 110                 | 100              |
| Flight and Propulsion<br>Control Avionics    | 300                            | 420                | 200              | 420                 | 200              |
| Avionics                                     | 1800                           | 1350               | 1000             | 1350                | 1000             |
| Electrical Power                             |                                | 300                | 400              | 300                 | 400              |
| Flight and Propulsion<br>and Display Control | 950                            | 450                | 150              | 180                 | 40               |
| Aircraft Sensors                             | 420                            | 500                | 500              | 200                 | 150              |
| <b>TOTAL</b>                                 | <b>3,650</b>                   | <b>3,130</b>       | <b>2,360</b>     | <b>2,560</b>        | <b>1,890</b>     |

- \* all weights in pounds
- \* flight and propulsion control avionics include primary and secondary flight control as well as engine and actuator control electronics
- \* mechanical or other backup systems not included
- \* mechanical baseline includes dual SCAS

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Widebody Flight Controls Development and Production Cost Comparisons

|                               | <u>Mechanical Baseline</u> | <u>Fly-By-Wire</u> | <u>Projected Fly-By-Light (RQW)</u> |
|-------------------------------|----------------------------|--------------------|-------------------------------------|
| <u>Non-Recurring</u>          |                            |                    |                                     |
| Engineering and Development   | \$14,000,000               | \$38,000,000       | \$60,000,000                        |
| <u>Recurring Per Aircraft</u> |                            |                    |                                     |
| Material Cost                 | \$90,000                   | \$55,000           | \$63,000                            |
| Avionics Assembly Co          | \$70,000                   | \$43,000           | \$35,000                            |
| Equipment Cost                | \$950,000                  | \$1,000,000        | \$1,030,000                         |
| Handling/Installation         | \$250,000                  | \$210,000          | \$220,000                           |
| <b>TOTAL</b>                  | <b>\$1,370,000</b>         | <b>\$1,308,000</b> | <b>\$1,348,000</b>                  |

Widebody Maintenance Life Cycle Cost Comparison

|                 | <u>\$/HR</u> | <u>\$/Life</u>     | <u>\$/HR</u> | <u>\$/Life</u>   | <u>\$/HR</u> | <u>\$/Life</u>   |
|-----------------|--------------|--------------------|--------------|------------------|--------------|------------------|
| Autoflight      | 3.7          | \$222,000          | 3.2          | \$192,000        | 2.6          | \$156,000        |
| Communications  | 1.8          | \$108,000          | 1.5          | \$ 90,000        | 1.1          | \$ 66,000        |
| Flight Controls | 1.7          | \$102,000          | 1.0          | \$ 60,000        | 0.7          | \$ 42,000        |
| Instrumentation | 1.6          | \$ 96,000          | 1.2          | \$ 72,000        | 0.8          | \$ 48,000        |
| Navigation      | 10.4         | \$624,000          | 8.3          | \$498,000        | 6.9          | \$414,000        |
| <b>TOTAL</b>    | <b>19.2</b>  | <b>\$1,152,000</b> | <b>15.2</b>  | <b>\$912,000</b> | <b>12.1</b>  | <b>\$726,000</b> |

Widebody Flight Controls Life Cycle Cost Comparison per Aircraft

|   |                      |                      |
|---|----------------------|----------------------|
| Non-Recurring   | \$30,000             | \$57,500             |
| Recurring   | (\$62,000)           | (\$22,000)           |
| Maintenance (Labor Equivalent Revenue (relative to baseline)) | (\$240,000)          | (\$426,000)          |
|   | (\$3,409,000) nt     | (\$7,146,000) nt     |
| <b>TOTAL</b>  | <b>(\$3,681,000)</b> | <b>(\$7,536,500)</b> |

---

|                              |                         |                          |
|------------------------------|-------------------------|--------------------------|
| <b>DIRECT OPERATING COST</b> | <b>4.5% reduction</b>   | <b>7.4% reduction</b>    |
| <b>RETURN ON INVESTMENT</b>  | <b>5.1% improvement</b> | <b>10.2% improvement</b> |

- \* utilization 8 hrs/day
- \* 800 aircraft production run
- \* ave life = 20 yrs
- \* 60,000 flight hrs
- \* "( )" additional profits above the mechanical baseline aircraft
- \* all type include flight guidance system
- \* ave speed = 320 mi/hr
- \* ave yield = 90.10/passenger mi.
- \* load factor = 0.7
- \* "nt" near term weight savings
- \* costs estimated in 1989 dollars
- \* averages for fly-by-wire and fly-by-light configurations
- \* mechanical baseline includes flight engineer

TABLE 5.3-3

Widebody Flight Controls Development, Production, Maintenance, and Life Cycle Cost Comparisons



## 6.0 VERIFICATION, VALIDATION, AND CERTIFICATION

### 6.1 Introduction

It could be argued that designing and building a fly-by-wire (FBW)/ fly-by-light (FBL) system is easy compared to the associated validation, verification, and Federal Aviation Administration (FAA) certification problems. The selected VVC approach must address the following major areas:

- A. Electromagnetic Effects Protection
- B. System Integrity
- C. Software
- D. Flight Test
- E. Direct FAA FBL Program Participation

Proving that a FBW/FBL commercial aircraft is fully protected against hazardous effects associated with EME requires the ability to fully test the aircraft while it is exposed to the threat. The state-of-the-art technology in this area is generally regarded as inadequate. In addition, analysis which can accurately predict the effects of the EME propagation inside a complex airframe structure is far from mature and yet is deemed essential to FBW/FBL technology by most experts. SAE Committee AE4R is struggling with these problems now.

System design integrity has, in the past, been demonstrated using bench testing, fault insertion, and analysis. Many organizations tasked with development of FBW/FBL systems, including Douglas Aircraft Company and Honeywell Inc., believe the complexity and criticality of commercial transport FBW/FBL Systems will likely render such methodologies inadequate.

A combination of Flight Test and FAA involvement will yield an understanding of how to quantify technical, cost, and certification risks associated with U.S. built and certified transport aircraft.

Direct FAA involvement in this program is essential to maximizing the benefits to U.S. industry.

## 6.1 Introduction (Continued)

Within the context of the discussion in this section, validation, verification, and certification are defined as follows:

- A. Verification: Establishing the truth of correspondence between a product and its specification; i.e., "Are we building the product right?" (Boehm, 1981). Within the context of the system life cycle, this means establishing that the evolving system satisfies all of its requirements. Verification should be performed in stages during full scale development of the system.
- B. Validation: Establishing the fitness or worth of a product for its operational mission; i.e., "Are we building the right product?" (Boehm, 1981). Within the context of the system life cycle, this means establishing that the system requirements are correct and complete. Validation should be performed prior to full scale development of the system.
- C. Certification: Obtaining regulatory agency approval for a product by establishing that it complies with all applicable government regulations.

## 6.2 Electromagnetic Effects Protection

### 6.2.1 Introduction

The modern trend to develop aircraft with nonmetallic composite materials and FBL digital flight control systems (FCS) requires a basic and fundamental understanding of the EME on such systems. These flight-critical control systems must have failure rates which are the same as other flight critical systems, on the order of one failure in  $10^9$  flight hours. Although FBL systems have eliminated EM coupling effects on signal lines, what remains are the problems associated with coupling to power cables and through the box enclosures. The challenge that the aerospace community now faces is how to evaluate and mitigate EME effects and achieve these high reliabilities. Meeting this challenge is a significant task, because of the scope and complexity of the issues involved. For example:

- A. The frequency range of interest is from DC to 40 GHz. This requires a wide variety of analysis, hardening, and test techniques.
- B. Aircraft are mechanically (and therefore electromagnetically) complex in shape, materials, and construction techniques.
- C. Digital systems are also complex, having many millions of logic states, some of which are known and some of which are unknown.

### 6.2.1 Introduction (Continued)

- D. There are significant problems associated with verifying these high reliability rates.

The overall objective in the EME area is to develop a national resource of applicable EME assessment technology. The overall scope is to develop aircraft EME models and methods that are experimentally validated.

As an integral step in an aircraft assessment using such a national resource, EME threat levels inside the commercial aircraft would also be defined. In addition, methods and approaches for verification and validation of system immunity, and for certification of the aircraft will be developed.

The flow chart shown in Figure 6.2-1 illustrates the complementary and iterative nature of an analysis and test cycle appropriate for aircraft designs involving significant departures from traditional practices (e.g., materials, geometries, flight critical electronic control systems). The processes shown in Figure 6.2-1 provide the characterization of the various aircraft EM responses to an aircraft EME interaction. Such responses are needed for electrical/electronic system designs that must deal with protection against the effects of the EME threat.

The interplay between analysis and test will be present throughout the design, verification, validation, and certification cycle. Estimates of the aircraft EM responses, based upon predictions derived from analytic models, could be used to help formulate the test philosophy (e.g., fixturation, measurement points, measurement levels). Test measurement data will be used to refine analytic models.

### 6.2.2 Technical Approach

Consistent with the top level system position that the EME threat issue occupies for flight critical electrical/electronics systems, the EME assessment process would involve a comprehensive system (top down) approach as depicted in Figure 6.2-1.

The design process is the primary means that ensures that an aircraft system will be immune to the EME threat. The design analysis data identifies the measures implemented and the corresponding worst case assessments that establish the degree of immunity expected.

Ideally, the design process would apply the capabilities made available by the national resource assessment technology in a top down approach that is in harmony with the system topology perspective.

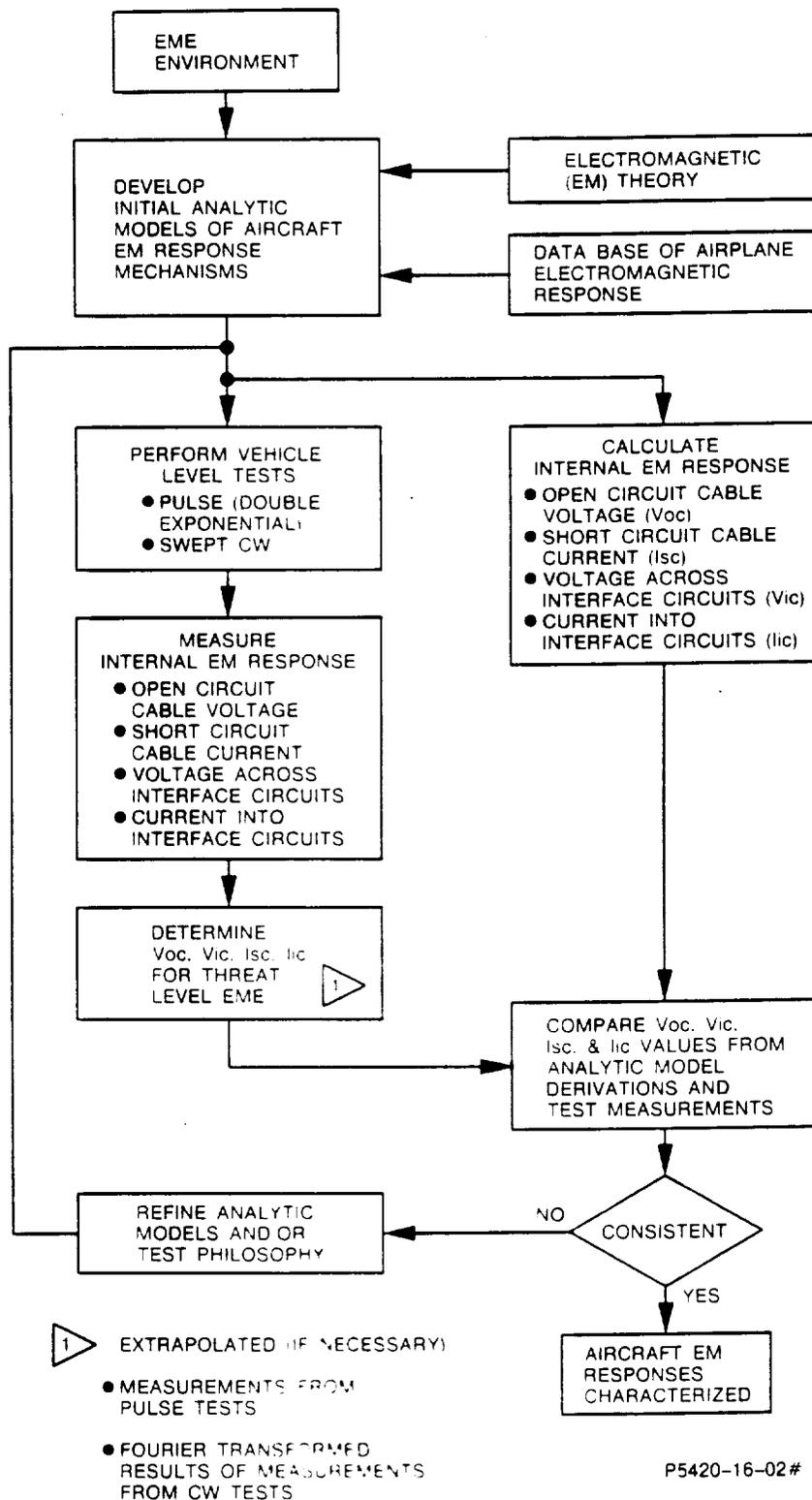


FIGURE 6.2-1

EME Analysis/Test Process

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## 6.2.2 Technical Approach (Continued)

The design process produces the data that provides the analytic and empiric assessment foundation for the verification, validation and certification basis and that enables the bottom up empirical (spot testing) checking to be effective and meaningful. The empirical measurements associated with the verification, validation and certification process complement the data from the design process and completes the data package needed for verification, validation and certification. The empirical activities identified for this program include equipment testing, in-situ cable current injections, aircraft simulated lightning current injection and aircraft HIRF illumination.

### 6.2.2.1 Equipment Testing

Previous activities (analyses, measurements) have determined both aircraft interior EM fields levels and bulk cable current response levels for the various transient and CW environments. These internal environments can then be translated into equipment testing procedures which will involve simultaneous bulk cable current injection techniques and field illumination. These procedures will be applied to aircraft systems equipment on the bench. The performance of these systems will be monitored and verified during this exposure.

### 6.2.2.2 In-Situ Cable Current Injection

This activity involves the EM stimulus of aircraft systems as installed in the aircraft. The objective here is to validate that the bench test results are applicable to as-installed equipment. Cable current injection and local illumination will be done on critical systems as installed on the aircraft to verify that in-situ performance is the same as that of the bench test. The systems will be cycled through all pertinent operational modes for verification.

### 6.2.2.3 Aircraft Simulated Lightning Current Injection

The objective here is to validate the lightning coupling results from scale model data. Low level simulated lightning currents will be injected into the vehicle and cable responses will be monitored and compared to previous data. These injected currents may be either transient or swept CW. High level currents will then be injected into the aircraft with all systems operating to verify scaling and system performance. It is anticipated that the airplane level lightning simulation testing identified for this program will verify the effectiveness of design analysis and subsystem/equipment testing, thus negating the need for airplane level testing, for the verifications validation and certification process.

#### 6.2.2.4 Aircraft HIRF Illumination

The objective is to validate performance in the HIRF environments. The aircraft will be subject to threat level environments. Measurements of internal fields and cable currents will be done to validate the scale model efforts and the bench testing environments. System performance in this environment will also be validated. It is anticipated that the airplane level HIRF illumination testing identified with this program will verify the effectiveness of design analysis and subsystem/equipment testing, thus negating the need for airplane level testing, for the verifications validation and certification process.

#### 6.2.3 Certification Procedures

The design and verification/validation activities will provide the experience and large data base which can be used to develop the certification basis and associated procedures. The objective here is to develop inexpensive requirements such as analysis and bench testing procedures so that certification can be done without having to fully illuminate or stimulate an entire aircraft. These procedures will be well defined, documented and justified.

### 6.3 System Integrity (System Modeling)

#### 6.3.1 Introduction

Extensive system fault effects modeling has, in general, not been viewed as essential to proper development of previous generation commercial transport systems such as CAT IIIB autoland. The introduction of FBW/FBL will likely require such modeling. The complexity of this type system makes it essential that the modeling work at the National Aeronautics and Space Agency (NASA), LaRc Airlab Facility (e.g., SURE, CARE III, GGLOSS, HARP), be enhanced and expanded to accommodate the complexities of systems targeted for near term application to FBW/FBL.

The limiting factor for this technology will likely be available computing resources. Both gate level and system level techniques must be further developed and expanded. Many new approaches need to be explored.

#### 6.3.2 Technology Issues

The technologies with greatest impact on fault tolerant V&V are broadly categorized into six areas:

- o System and Component Level Modeling
- o Circuit Simulation Software and Hardware
- o System Reliability Software
- o Integrated Computing Environment
- o Rapid System Prototyping Environment
- o Technology Acquisition and Transfer

#### 6.3.3 Strategies

The strategy will be to develop the six technologies listed above to a level where they are user-friendly, cost-effective, and comprehensive.

##### 6.3.3.1 Modeling

The approach will define modeling methodology at the system, board, and gate levels. At the system level, DAC/HI will capitalize on the existing modeling tools available in the marketplace and at NASA, and HI's modeling experience to develop system prototype models to demonstrate new concepts for FBW/FBL control systems verification and validation. At the board and gate level modeling, the strategy will be to develop or procure modeling

### 6.3.3.1 Modeling (Continued)

technologies compatible with simulation systems to provide the means to detect faults at the gate level. The results of the lower level simulations will be used at the system level for failure modes and effects analysis.

- o Develop modeling techniques to be able to transition up or down in hierarchy: starting at one end of the spectrum with gate level circuit simulation, and transitioning that information in a series of steps to the system level reliability prediction programs. Obviously, there are many steps in between with the reformatting of data and the moving of a data base from one platform to another.
- o Utilize modeling technology to accelerate the development of new architectures for FBW/FBL control system. With user-friendly CAE systems, new configurations of FBL/FBW can be designed and tested in a much more timely fashion than conventional methods.

#### 6.3.3.1.1 System Level Modeling

System level analysis will incorporate data and results from lower level simulation and failure mode and effect analysis to determine the ultimate consequence of injected faults on the system/aircraft. The starting points for this analysis could be Mentor's Quick-Sim circuit simulation. Going through the various steps of gate level/chip simulation, then module simulation, and possibly even going to system circuit simulation (assuming there is enough computing power available). The ultimate goal is to predict reliability and failure mode effects.

- o Initial efforts will be targeted at modeling one lane of an eight processor flight control system for circuit simulation. The processor will be modeled at the functional bus level primarily because the gate level model for the processor is not available. Semiconductor vendors consider this information proprietary. The approach then is the logic and all the monitoring circuitry around the microprocessor will be modeled such that faults can be injected anywhere. To obtain realistic simulation results, small pieces of application software will be transformed into test vectors and simulated in the circuit.
- o Input a FBW/FBL prototype model consisting of an 8-processor system into SURE to evaluate SURE in its ability to predict the reliability or large systems using single-point failures. In parallel, the same model will be input into HARP and evaluated in its ability to detect transient and near-coincident failures. The size of the model will be expanded from 8 to 16 processors, and both SURE and HARP will be evaluated on their suitability to accurately perform reliability prediction of realistic FBW/FBL flight control systems.

#### 6.3.3.1.1 System Level Modeling (Continued)

- o Advance the development of system fault injection and statistical analysis techniques to characterize the fault recovery of fault tolerant systems. Determine arrival time distribution for transient and near-coincident faults. Reliability modeling assumes that detection times of faults are exponentially distributed. This assumption and current sampling practices have shown to introduce significant variations in system reliability estimation. Joint efforts with NASA's AIRLAB will be pursued to determine new sampling techniques and statistically robust parameter estimation.

#### 6.3.3.1.2 Module Card Level Modeling

The card level modeling simulates at the component level faults can be injected at the pin I/O.

The information will be passed to a high level (system level) simulation to manifest its system consequences.

- o Develop/procure models to perform functional and fault simulation of module with faults inserted at the pin level.
- o Develop the capability to generate Behavioral Language Models (BLM) which can aid in our simulations of complex microprocessor-based modules. The capability can then be used to input faults at the pin I/O level for failure mode effects analysis and higher-level simulation.
- o Develop libraries to support functional fault simulation of large microprocessor-based boards. Explore the advantage of hardware verification models in terms of "bus cycles."
- o Evaluate the transition to VHDL (Very High Level Description Language) as a standard modeling methodology for the system and subsystem levels.

#### 6.3.3.1.3 Gate Level Modeling

Gate level modeling is the functional circuit description of large components (LSI, VLSI, and ASIC) using small scale integration type of logic devices to describe their behavior. It has been classically at this level that fault simulation of complex ICs have been accomplished. Generic NAND gates, nor gates, inverters, etc., have been typically used for this type of modeling.

- o Develop/procure gate level models to make possible gate level fault simulation of processor cards on fault simulation accelerators or high-speed general-purpose computers.

#### 6.3.3.1.3 Gate Level Modeling (Continued)

- o Develop gate level models to perform fault simulation using GGLOSS.
- o Develop a generic gate level library to support the generation of other libraries for fault simulation.

#### 6.3.3.1.4 Modeling Tools Integration

These are software programs to aid in the generation of both gate level and behavior language models and the conversion of libraries using standard-data formats such as Electronic Design Interchange Format (EDIF), Very High Level Description Language (VHDL), etc. The tools developed should integrate the overall modeling system to enable the interfacing for one level of hierarchy to the next.

Influence the development of integrated system, board, and gate level modeling tools. Pursue the development of smart knowledge-based models similar to Logic Automations' Behavior Language SMARTMODELS to facilitate the debugging of system models.

#### 6.3.3.2 Simulation Technologies

Simulation technologies are critical tools in assessing system performance and behavior. The approach uses very high-speed fault simulation to provide fault coverage at the pin and gate levels. Failure mode effect simulations give a measure of the system architecture's detection and tolerance of transient and near-coincident faults at system levels.

- o Define methods for mapping fault coverage at the gate level to system levels to provide an automated interface from low-level fault simulation to system level models. Also, develop a method for communicating results of a failure mode fault analysis run at low-level to a higher-level simulation to obtain the ramifications.
- o Benchmark NASA's GGLOSS fault simulator using a single-processor board executing software. Since GGLOSS is a gate level fault simulator, the SDP-185-based processor board is a good candidate for gate level fault simulation since HI has the gate level model. The SDP-185 design and libraries will be netlisted and transferred to the GGLOSS system for fault simulation on a VAX computing environment. If satisfactory, attempts will be made at simulating systems with multiple microprocessors. The SDP-185 is a custom Honeywell-developed 2901-based processor circuit.

### 6.3.3.2 Simulation Technologies (Continued)

- o Investigate the capabilities of available and developing fault simulators and their suitability to fault tolerant V&V. Explore gate level hardware fault simulators versus software-based fault simulators on high-performance computers. The initial effort will be to fault simulate the SDP-185 board on the existing MACH-1000 simulation accelerator, and compare the fault coverage to the results from the GGLOSS fault simulation.
- o Influence simulation vendors to modify their firmware to handle very-high-speed failure modes to affect the simulations of processors running real-time software.
- o Explore the advantages of mixed-mode (both gate level and behavioral language models) concurrent simulation on hardware accelerators and high-speed workstations. This innovative system uses shared computing resource where a hardware accelerator simulates at the gate/transistor levels while coupled with a general-purpose high-speed workstation simulating at the behavioral model level.
- o Select/procure software simulators implemented in a computing environment for deployment of a full-scale FBW/FBL control system verification development.
- o Develop software tools to sift and auto-compare simulation results at gate levels to study the propagation of inserted faults to the higher levels in the system.
- o Develop user-friendly simulation tools designed to increase productivity of system engineers.
- o Develop software tools to expedite the conversion of application software into test vectors at the lower levels. Test vectors have historically been a manually intensive effort. Tools should be developed to aid this effort.

### 6.3.3.3 Reliability Prediction

To obtain the overall system reliability (probability of failure/mission) and for parameter sensitivity studies, analytical techniques such as Markov modeling must be used. A Markov reliability model calculates the probability of a system being in various states as a function of time. A state in the model represents the system status as a function of the failed and unfailed components and the system's redundancy management strategy. Given the system architecture and reconfiguration rules, the system reliability can be calculated. Other inputs to the model will include fault data for sequence dependent failures, correlated failures, near-coincident failures, the coverage of transient and permanent faults, and fault arrival rates.

#### 6.3.3.3 Reliability Prediction (Continued)

The starting point for the system reliability modeling will be the software tools, SURE (Semi-Markov Unreliability Range Evaluator) and HARP (Hybrid Automated Reliability Predictor), which were developed by NASA. Initially, one-processor then four-processor, (one box) and then eight-processor (two boxes) configurations of the FBW/FBL Flight Control System will be modeled for permanent failures in parallel with SURE and HARP. This should help in assessing the capabilities and limitations of SURE and HARP in modeling large systems. Once this is accomplished, other types of failures and fault handling models will be added to the models. Later, the model will be increased to 16-processor (four boxes) and then generalized to any number.

#### 6.3.3.4 Integrated Computing Environment

The integration of the various computer resources into a common environment for modeling and reliability prediction is critical for fault tolerant V&V. At the present, modelling and reliability predicting time, lack of adequate processor capability (throughput) and processor availability are the largest obstacles identified. To simulate the amount of electronic circuitry in a FBW/FBL flight control system on today's workstation networks or a VAX cluster is not realistically feasible. Hardware accelerators, or a clever division of tasks, or very efficient models will be needed to make real progress at system analysis. Some of the associated resources that will be needed include:

- o Parallel computer architectures with high processing capabilities, such as large parallel processor mainframes or parallel super workstations.
- o Hardware simulation acceleration.
- o Concurrent processing.
- o Fault tolerant software engineering.
- o Computer-aided engineering/network support personnel.
- o Standard networks and protocols.

#### 6.3.3.5 Rapid Prototyping Technologies

The success of fault tolerant verification will depend on the close technical cooperation between developers, customers, and users. The traditional phase-oriented approach to system design assumes a strictly linear ordering of development steps. This approach is not only very costly but fails to include important elements of communication and feedback in the development process, making system evolution a lengthy process. Rapid prototyping technologies used within the system development context allows the

#### 6.3.3.5 Rapid Prototyping Technologies (Continued)

design and building of prototypes (models) to demonstrate the feasibility of systems or the evaluation of alternative system designs using early customer or user requirements. The resulting process provides a means of mapping system requirements into a sequence of cycles; re-design, re-implementation, and re-evaluation, which allows new customer requirements to be quickly captured, implemented, and evaluated before the target system is built.

The most important techniques used in rapid prototyping are: modular design, interactive human interface, and simulation. Emerging development tools promote the use of these techniques at various levels of the system design process. Our approach will advance the integration of these tools with expert systems and data base management to provide the environment tailored for prototype development.

- o Introduce Computer Aided Software Engineering (CASE) tools as a rapid prototyping environment for fault tolerant software development. Integrate expert systems and data base management tools to the CASE system environment to facilitate the creation, management, and automated conversion of rapid prototypes into target systems.
- o Develop a rapid prototype for a fault tolerant test-bed system. This prototype facility will be used to demonstrate feasibility of fault tolerant V&V concepts and techniques.
- o Develop a rapid prototype for the generation of self-test programs suitable for automated validation and verification of fault tolerant systems.
- o Introduce VHDL as the standard modeling and simulation tool for the design and verification of fault tolerant v7v systems.
- o Influence the adoption of design for testability guidelines as a standard design methodology to facilitate the generation of testable systems.

#### 6.3.3.6 Technology Acquisition and Transfer

This strategy emphasizes the importance of forming effective relationships with a variety of government, industry, and educational organizations. Development of a national resource capability for verification of FBW/FBL control systems technical expertise in various disciplines is imperative. This expertise will be acquired through teaming with industry technology groups, government research centers, universities, and industry organizations.

6.3.3.6 Technology Acquisition and Transfer (Continued)

The critical disciplines are identified as modeling, simulation, and supporting technologies.

6.4 Software

6.4.1 Evaluation of RTCA/DO-178A

DAC/HI will work with the FAA to evaluate the adequacy of RTCA/DO-178A with regard to flight crucial systems. Specifically, DAC/HI will attempt to show whether the Level 1 development approach is sufficient to yield a system that can meet reliability requirements in the  $10^{-9}$  range.

6.4.2 Verification and The Software Development Cycle

DAC/HI will identify and evaluate verification tools and methods that can be applied to each of the software development activities depicted in Figure 6.4-1. Note that Figure 6.4-1 represents a baseline life cycle paradigm (similar to those suggested in DOD-STD-2167 and RTCA/DO-178A).

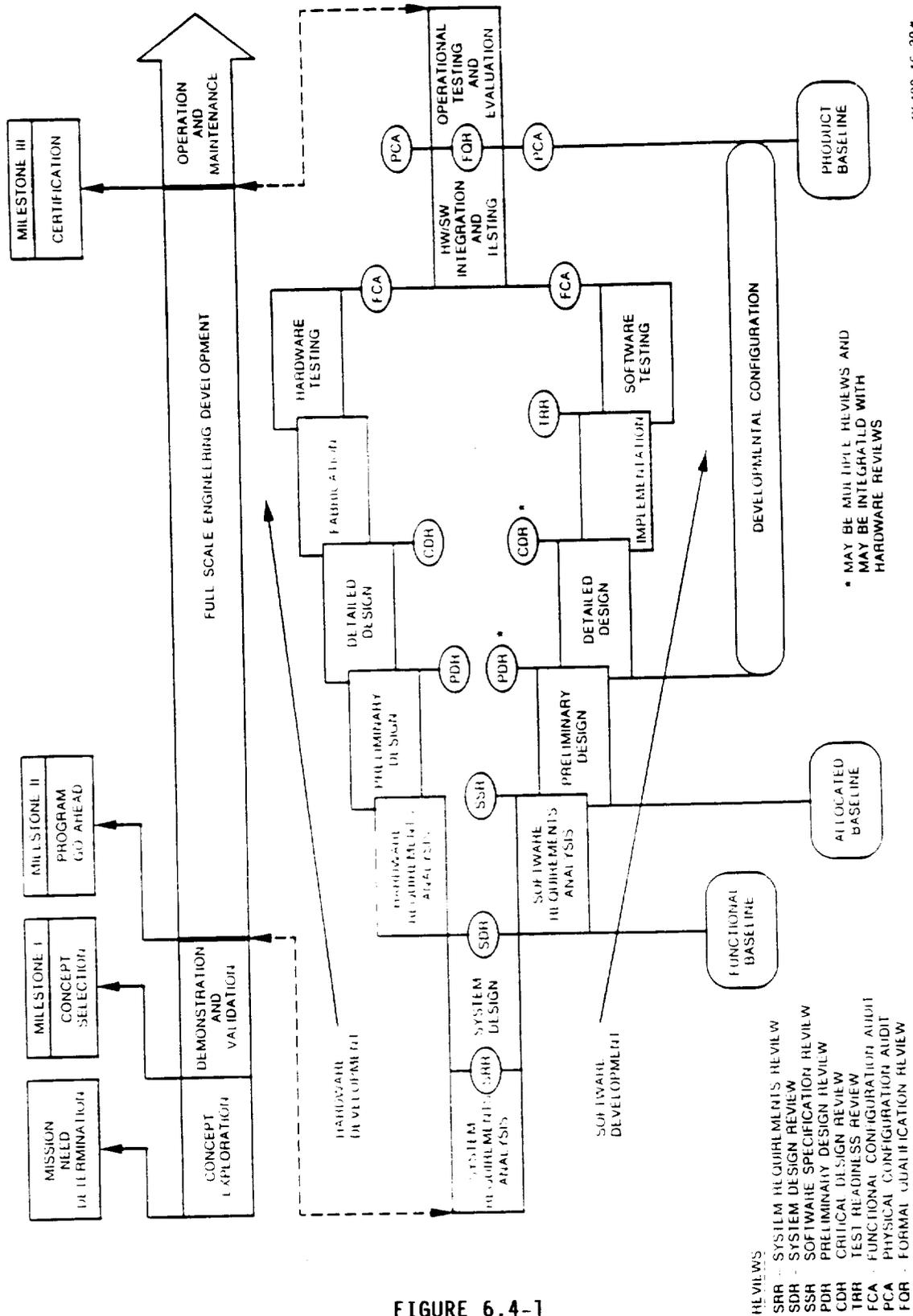
6.4.2.1 Software Requirements Verification

DAC/HI will identify and evaluate tools and methods that can be used to verify that the software requirements are an adequate translation of the system requirements allocated to software and that implementation is feasible. Current state-of-the-art tools and techniques include:

- A. Structured analysis tools with built-in consistency checking (e.g., Excelsior and Teamwork).
- B. Traceability matrices.
- C. Walk-throughs and reviews.
- D. Requirements-based testing.

6.4.2.2 Software Design Verification

DAC/HI will identify and evaluate tools and methods that can be used to verify that the software design (top level architecture approach and detailed design) represents a clear, consistent, and accurate translation of the software requirements, adequately addresses all issues peculiar to real-time embedded software design, and that the key algorithms are performed with the required precision and accuracy. Current state-of-the-art tools and techniques include:



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Life Cycle: Time Phasing

FIGURE 6.4-1

Life Cycle: Time Phasing

#### 6.4.2.2 Software Design Verification (Continued)

- A. Modeling tools that support top level architecture design and simulate execution.
- B. Structured design tools with built-in consistency checking.
- C. Design diversity (multiversion software development).
- D. Traceability matrices.
- E. Walk throughs and reviews.
- F. Structure-based testing.

#### 6.4.2.3 Software Implementation Verification

DAC/HI will identify and evaluate tools and methods that can be used to verify that the software implementation (code) effectively and efficiently represents a clear, consistent, and accurate translation of the software detailed design and that it adequately addresses all issues peculiar to the target environment. Current state-of-the-art tools and techniques include:

- A. Automatic code generation tools that accept textual (PDL) or graphic detailed design information and produce target code.
- B. Implementation diversity (multiversion software development).
- C. Traceability matrices.
- D. Walk-throughs and reviews.
- E. Structure-based testing.

#### 6.4.2.4.1 Overview of Mutation Testing for System Verification Validation

Software testing attempts to provide a partial answer to the following question: if a program is correct on a finite number of test cases, is it correct in general? Several techniques have been devised for generating test cases, including input space partitioning, symbolic testing, and functional testing. Because software testing is insufficient to guarantee program correctness, these techniques do not attempt to establish absolute program correctness but to provide the tester with some level of confidence in the program.

Although each of these techniques is effective at detecting errors in the program, mutation analysis goes one step further by supplying the tester with information in the absence of known errors. This unique ability helps the tester predict the reliability of the program and indicates quantitatively when the testing process can end. Mutation analysis has been shown

6.4.2.4.1 Overview of Mutation Testing for System Verification Validation  
(Continued)

analytically and experimentally to be a generalization of other testing methodologies. Thus, a mutation analysis tool gives a tester the capabilities of several other test techniques as well as features that are unique to mutation. The following discussion describes the mutation approach to software testing.

A. Theory of Mutation Analysis

Mutation analysis is a powerful technique for software testing that assists the tester in creating test data and then interacts with the tester to improve the quality of the test data. Mutation analysis is based on the "competent programmer hypothesis" - the assumption that the program to be tested has been written by a competent programmer. Therefore, if the program is not correct, it differs from the correct program by at most a few small errors. Mutation analysis allows the tester to determine whether a set of test data is adequate to detect these errors. The first step in mutation analysis is the construction of a collection of mutants of the test program. Each mutant is identical to the original program except for a single syntactic change (for example, replacing one operator by another or altering the value of a constant). Such a change is called a mutation. Each mutant is then executed, using the same set of test data each time. Many of the mutants will produce different output than the original program. The test data is said to kill these mutants; the data was adequate to find the errors that these mutants represent. Some mutants, however, may produce the same output as the original program. These live mutants provide value information. A mutant may remain alive for one of two reasons:

- o The test data is inadequate. The test data failed to distinguish the mutant from the original program. For example, the test data may not exercise the portion of the program that was mutated.
- o The mutant is equivalent to the original program. The mutant and the original program produce the same output, hence no test case can distinguish between the two.

Normally, only a small percentage of mutants are equivalent to the original program; these are usually easy to locate and remove from further consideration. more test cases can be added in an effort to kill non-equivalent mutants. The adequacy of a test set of test cases is measured by an adequacy score; a score of 100% indicates that the test cases kill all non-equivalent mutants.

6.4.2.4.1 Overview of Mutation Testing for System Verification Validation  
(Continued)

B. Mutation Testing System

A mutation-based testing system allows a tester to perform mutation analysis on a program (or sub-program). The tester supplies the program to be tested and chooses the types of mutations to be performed. The tester also supplies one or more test cases. The testing system executes the original program and each mutant on each test case and compares the output produced by the two programs. If the output of the mutant differs from the output of the original program, the mutant is marked dead. Once execution is complete, the tester can examine any mutants that are still alive. The tester can then declare a mutant to be equivalent to the original program, or the tester can supply additional test cases in an effort to kill the mutant (and possibly other live mutants as well). Some mutation systems are able to detect automatically certain kinds of equivalent mutants.

Several mutation systems have been built over the past ten years, including PIMS, EXPER and FMS.3, which supported mutation analysis for Fortran programs, and CMS.1, which handled COBOL. The most recent mutation-based system is the Mothra, an integrated software testing environment under development at Purdue University's Software Engineering Research Center.

Experience with Mothra and earlier systems has shown mutation analysis to be a powerful tool for program testing. Mutation-based testing systems have a number of attractive features:

- o Mutation analysis includes - as special cases - most other test methodologies. Statement coverage and branch coverage are among the methodologies that mutation analysis subsumes.
- o A mutation based system provides an interactive test environment that allows the tester to locate and remove errors.
- o Mutation analysis allows a greater degree of automation than most other testing methodologies.
- o Mutation analysis provides information that other test methodologies do not. In particular, the mutation score for a particular program indicates the adequacy of the data used to test the program, thereby serving as a quantitative measure of how well the program has been testing.

#### 6.4.2.4.1 Overview of Mutation Testing for System Verification Validation (Continued)

A potential problem of mutation based testing systems is the amount of computer resources (both space and time) required for the testing of large programs. The number of mutants generated for a program tends to grow quadratically with the number of names in the program. Storing huge numbers of mutants can be difficult on many computer systems; executing that many mutants is an even larger problem. Fortunately, there are several ways to overcome the problem of limited resources:

- o Mutant sampling. Mutation systems allows the tester to specify random sampling of a certain percentage of mutants. Often, sampling even a small percentage of the possible mutants is enough to reveal inadequate test data.
- o Selective application of mutant operators. Mutation systems also allow the tester to specify that only certain kinds of mutations are to be performed. Thus, the tester can select mutations that are likely to have a high payoff relative to the amount of time they require.
- o Use of high performance computers. Computationally intensive tasks of mutation testing can be performed on high-speed computer systems such as parallel architectures and vector machines.

#### C. Evaluation of Mutation Testing

Mutation testing covers the class of errors known as blunders. Blunders are errors in which the software did not do what the programmer intended. This contrasts with design errors in which the software does what the programmer intended, but the intended functionality is not correct. Blunders include typographical errors, the "off-by-one" error, the use of wrong variables, etc. This type of error is easier to find than design errors. Most blunders are caught either by compilers that translate strongly typed languages, such as Ada; or by existing methodologies used in the development of highly reliable software. The much more difficult problem of testing for design errors still needs to be addressed.

#### 6.4.2.4.2 Overview of Architectural Design and Assessment System (ADAS)

ADAS is a set of computer-aided engineering tools which support a methodology for the architecture level design and analysis of software algorithms and their hardware implementations. Through an iterative design process, software and hardware designs are analyzed and refined to approach a final design.

6.4.2.4.2 Overview of Architectural Design and Assessment System (ADAS)  
(Continued)

ADAS uses directed graphics to model software and hardware systems. An ADAS directed graph consists of node and arcs. Nodes represent individual software operations or hardware functional elements. Arcs represent the flow of data or control from one node to another. Both nodes and arcs have name and other attributes associated with them. The attributes provide the information necessary for modeling the system under study. A graph can contain a cycle, which represents a flow of data that returns to the node that originates it. There is nothing intrinsic to the graph to indicate whether it represents a software or hardware model. Some associated ADAS graph attributes are defined as follows:

queue size is the maximum number of tokens allowed on the arc.

firing delay is the length of time the node is busy after it is enabled to fire (input, resource, and output conditions are all met).

ha-module is the hardware resource to which the node is mapped. If two nodes are enabled to fire at the same time, and the nodes have identical hardware resources, then their firing is serialized.

priority may be used when there is resource contention, to allow deterministic sequencing of the nodes. If the default is used, ADAS schedules the nodes according to an internal algorithm.

threshold defines the number of tokens which must be on a given arc in order for the node to fire.

consume defines the number of tokens consumed from the arc when the node fires.

produce is the number of tokens the node produces when it is done firing.

In addition, nodes and arcs have many other attributes including those which determine graph appearance, such as size, color and position. There are also attributes which can point to files which contain behavioral models or other user defined simulation data.

In order for an ADAS node to fire, certain criteria must be met. There must be sufficient tokens on the input arcs to satisfy the threshold and consume values of one or more arcs. The hardware resource must be available. Finally, there must be sufficient space on the output arcs to satisfy the produce values.

#### 6.4.2.4.2 Overview of Architectural Design and Assessment System (ADAS) (Continued)

When all the conditions are met for a node to fire, it first consumes the tokens from its input arcs. If a behavioral model is present, it is executed. Then the node delays for a specified time equal to its firing delay, locking out its resource from other nodes. When the appropriate time has passed, the resource is released and tokens are produced and placed on the output arcs.

If models more complicated than the basic ADAS model are required, behavioral models written in Ada or C may be used. These allow complex interaction between nodes, stochastic values of attributes such as [/*node user file name*] may be used to point to a file containing any information the user defines. This could include implementation technology (e.g. GaAs, Si), degree of parallelism available in the node, type of fault tolerance used, etc. The behavioral model would then be designed to use this information in an appropriate fashion.

ADAS tools provide detailed simulation results for the performance analysis and evaluation of a design. Statistics such as utilization and latency may be collected for hardware modules and software nodes. Flow rates and access frequencies may be collected for arcs. ADAS uses both simulation-based and analytical techniques to calculate performance characteristics.

A menu-driven graphic editor is used to construct and modify the software and hardware graphs. Other tools analyze the performance of the models, allocate software tasks to hardware graph components, and provide functional simulation capabilities. Attributes associated with the model components are used to check for design inconsistencies.

#### 6.4.3 Software Verification Effectiveness Measurement

##### 6.4.3.1 Quantifying Software Reliability

Software reliability is the probability that the software will meet requirements for a given period of time in a specified environment. Software reliability measurements are based on the frequency with which problems (i.e., design faults, operational errors, etc.) occur. Unrecognized design defects are the primary failure source for software, unlike hardware reliability where physical causes and wear play important roles. Models are either based on the external behavior of a system or on the system's structure. The models represented here are all based on the system's external behavior. The values of the parameters in the modeling systems must be estimated sometimes. Different inference procedures have been devised to calculate the parameters optimum value. These prediction systems will not be discussed here.

#### 6.4.3.1 Quantifying Software Reliability (Continued)

In order to describe these software models, some terms need to be defined. A failure is defined as the condition where software does not meet its requirements. A fault is a specific defect in the software that, under certain circumstances, will cause a failure. Time to Failure (TTF), Mean-Time-To-Failure (MTTF), Rate of Occurrence of Failure (ROCOF), and failure intensity, (number of failures/unit time) are terms used to describe software reliability.

The following is an overview of models used in software reliability analysis:

##### A. Fault Manifestation Models

1. The product change statistical model (Currit, et al) calculates the MTTF as a product of the fractional improvement, from each change, of the MTTF raised to the power of the number of changes introduced and the estimated initial MTTF.
2. Deterministic Exponential Order Statistical (DET/EOS) models, used for a uniform failure manifestation rate and a fixed unknown number of failures, include the Jelinski-Moranda model. The DET/EOS model calculates a likelihood function for TTF data as a probability density function of observed time to next failures.
3. Doubly-Stochastic Exponential Order Statistical (DS/EOS) Models.
  - a. Independent Identically Distributed Order Statistic (IIDOS) models, used for a manifestation rate derived from order statistics from a probability distribution function and a finite unknown number of faults, include the Littlewood Stochastic Reliability Growth Model. This model uses a gamma distribution for the individual fault manifestation rates to calculate the ROCOF.
  - b. Non-Homogeneous Poisson Process (NHPP) models, used for a manifestation rate derived from an NHPP and an infinite number of faults or a Poisson distributed number of faults, include the generalized gamma DS/EOS Model.
  - c. Distribution-free model (Giammo) calculates the manifestation rate using all possible distributions bounded by the data.

#### 6.4.3.1 Quantifying Software Reliability (Continued)

##### B. Interfailure Time Models

1. The basic execution time model calculates the failure intensity for relatively uniform operational profiles using the initial failure intensity at the start of execution, the average number of failures experienced at a given point in time, and the total number of failures over all time as parameters.
2. The Littleton-Verrall Model calculates the ROCOF based on the shape parameter of a gamma distribution, and a family of scale parameters for that distribution.
3. The logarithmic Poisson execution time model (Musa and Okumoto) defines the failure intensity as a function of the initial failure intensity multiplied by the exponential function of the product of the average number of failures experienced at a given point in time and a failure intensity decay factor. This failure intensity decay factor is the magnitude of the derivative of the natural log of the failure intensity with respect to the average number of failures experienced. Musa demonstrates that this model is useful for highly nonuniform operation profiles.

#### 6.4.3.2 Relating Software Reliability to Software Verification Techniques

An area of required technology development will be to investigate the feasibility of using parametric modeling techniques (analysis of various software verification techniques and correlation to historical reliability data) to develop a model that will predict reliability as a function of the software development process.

#### 6.5 Flight Test

Flight test is the ultimate stage in the verification of design process. It is the final proof of concept activity for any new aircraft system approach and is an indispensable step in such situations. All other stages of design verification (laboratory simulation and iron bird studies) can only be approximate models of the full up operational aircraft (see Section 7 for the description of the Flight Test Program). However, flight test is limited, it cannot examine every flight control system state.

#### 6.6 Direct FAA/FBL Program Participation

The lack of established mature U.S. certification basis and procedures for FBW/FBL stands as an impediment to U.S. introduction of advanced technology commercial transport aircraft into the world marketplace. This program, which includes participation of the FAA, NASA, DoD, and several major aircraft

6.6 Direct FAA/FBL Program Participation (Continued)

manufacturers, is defining the development path for advanced technology certification criteria which are critical to the U.S. manufacturers competitive posture with foreign manufacturers.

Currently digital systems immune to soft faults, tolerant to EM environment upset, and demonstrating MTBF exceeding the life of the airframe structure and without backup are being proposed for design, yet the existing certification methods and criteria have not addressed issues particular to the complexity of these systems. In light of this, over the past one and a half (1-1/2) years, DAC/HI has had many discussions with FAA personnel relative to a certification basis for FBW/FBL systems.

Most recent discussions have centered specifically on how FAA specialists would directly participate in this FBL Program. These discussions are continuing at the detail stage with and within the FAA. Current FAA plans call for direct participation by the following organizations:

- A. Los Angeles Aircraft Certification Office
- B. Seattle Aircraft Certification Office
- C. Transport Standards Office
- D. FAA Technical Center via an inter-agency agreement with NASA-Langley (current in place)

FAA direct involvement is perceived to be of major importance. Without established procedures, certification risks, financial, schedule, and technical cannot be accurately quantified.

During the last half of 1989, the role of the FAA should be clarified. Further, it is anticipated that the FBL Program will heavily influence the content of a new certification "Advisory Circular" (AC) on the subject of FBW/FBL. Related workshops are also anticipated.



7.0 FLIGHT TEST

7.1 Background

Douglas Aircraft Company and Honeywell, Inc. (DAC/HI) are currently involved in developing and verifying/validating fly-by-wire/fly-by-light (FBW/FBL) concepts for application to future transport aircraft. First delivery from HI to DAC of a set of FBW units occurred in June, 1989. Upgrades are scheduled for the next two years during which numerous technical issues must be resolved.

7.2 Deliverables/Characteristics of Flight Control Computer System

To facilitate Verification and Validation (V&V) aspects of the FBL program including flight test. DAC/HI suggest that they supply one (option for two) set(s) of FBW/FBL flight control computers to NASA. Early program results will directly influence the capabilities built into these units.

At the same time, DAC/HI suggests that they supply a fixed-base test facility capable of full-fidelity on-ground verification of the FBW/FBL flight control computers prior to flight test. This would be integral to an "iron bird" facility. The simulation quality would be adequate to validate aircraft handling and performance requirements. This facility would be resident at the Langley flight test facility and be the property of the National Aeronautics and Space Agency (NASA).

7.2.1 Flight Control Computer Physical Characteristics

The proposed basic flight equipment consists of four flight control computers which are line replaceable units (LRU) with the following characteristics:

- A. 41.2 lbs. per flight control computer
- B. 128 watts per flight control computer
- C. 10 MCU ARINC 600 chasis

During the course of the program certain techniques will be evaluated and those with a high degree of merit and feasibility will be incorporated into the flight control computers. These techniques may be associated with the following areas of investigation: Electromagnetic effect (EME) hardening, transient fault protection, extended availability through secondary redundance, etc.

7.2.2 Flight Control Computer System Deliverables

Deliverables associated with the program include the following:

- A. Four flight control computers (one ship set) and one spare flight control computer

7.2.2 Flight Control Computer System Deliverables (Continued)

- B. One (1) closed-loop validation facility
- C. System Specification Document
- D. Software Requirements Document
- E. Flight control computer hardware/software updates as required

On-site support will be provided continuously from initial delivery until program completion. The type and magnitude of on-site support will be as dictated by program needs.

7.3 Basic Flight Control Computer Architecture/Design Overview

7.3.1 Basic Architecture

The basic aircraft configuration consists of four flight control computers which are dual-lane fail-passive, configured to provide a fail-operational Primary Flight Control (PFC) System. These four flight control computers provide system generic fault protection through three-version techniques. The basic redundancy management architecture was originally developed as a Honeywell, Inc. (HI) IR&D Project for application to FBW systems. This architecture is now being adapted to the MD-11 flight control computer/LSAS currently in development.

7.3.2 Generic Fault Protection

This system has been in development at HI for eight years. N-version techniques have been developed and refined during the 737-300 and MD-11 systems developments as well as through continuing research and development efforts, including joint research programs with University of California, Los Angeles (UCLA). Currently, the base MD-11 equipment with modifications made for a completely digital FBW system with ARINC 629 bidirectional bussing is being evaluated at DAC in their FBW/FBL R&D Program.

7.3.3 Flexible Input/Output Interface Capability

The flight control computer architecture was designed specifically to allow for substantial Input/Output (I/O) signal composition changes with minimal to no changes to the hardware. Generic I/O card connector interfaces allow additions/subtractions/swaps of I/O card types to increase or decrease the quantity of a given signal type easily.

This flexibility will be used to match the flight control computers to the other avionic subsystems onboard the flight test aircraft, as necessary thus reducing the risk of an interface mismatch and subsequent schedule delay.

#### 7.3.4 Partitioning of Software by Criticality

The flight control computer includes the capability to partition (relative to fault effects) sections of the overall software package that execute concurrently on a single processor. The mechanism for achieving software-fault-effects partitioning is being used on the DAC MD-11 aircraft program and, as such, will have a certification precedent. For most commercial digital avionics systems, the entire software package associated with a specific processor must be certified to the criticality of the most critical software element. The problem this presents relative to integration is obvious.

To facilitate partitioning, HI has developed a specialized integrated circuit. The specific methodology is patented.

#### 7.3.5 Electronic Design Practices

The electronic design rules adopted for the flight control computer resulted from the experience gained through previous digital LRU production programs at HI. This previous experience has demonstrated the importance of many design precepts, including the following:

- A. Distinct electronic separation of the two lanes that make up the dual flight control computer.
- B. Design guidelines that maximize immunity to noise and external Electromagnetic (EM) disturbances. Standardization of parallel digital data bus and I/O interface protocols.
- C. Design guidelines that provide effective Built-In-Test (BIT) for fault detection and isolation.

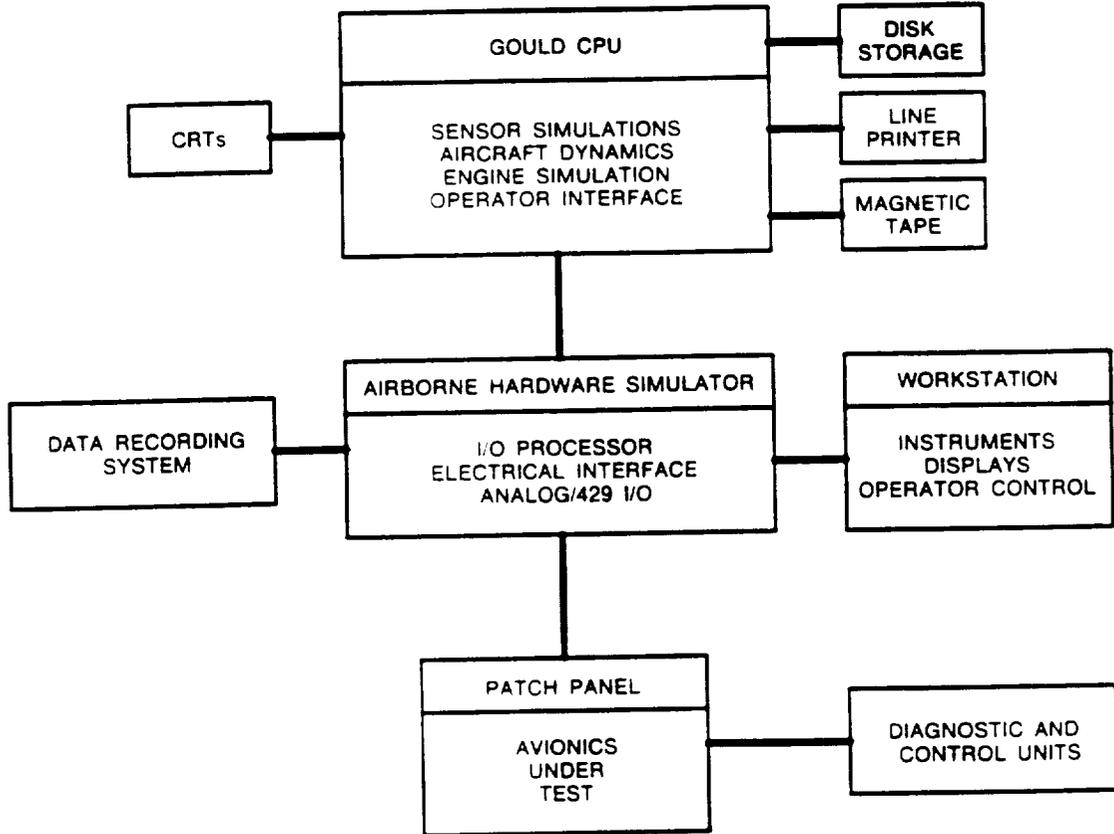
The design approach used for the flight control computer is that of clear and distinct separation of redundant elements. This results in easily understood and highly visible hardware partitioning, including separate processors, memories, power supplies, and I/O conversion.

Multilayer circuit boards and motherboards provide high integrity ground and power systems within the flight control computers. Logic clocking signals required between circuit boards are transmitted differentially and are the only edge sensitive signals communicated between circuit boards.

### 7.4 Closed-Loop Validation Facility Overview

#### 7.4.1 Rationale for Closed-Loop Validation Facility

In order to test and verify the FBL/PFC system a sophisticated closed-loop validation facility will be required to provide the capability of both a high fidelity dynamic aircraft environment simulation and a static simulation for flight control computer validation.



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**A Typical VALFAC System Configuration**

FIGURE 7.4-1

A Typical VALFAC System Configuration

#### 7.4.1 Rationale for Closed-Loop Validation Facility (Continued)

This facility will augment the empirical portion of the V&V process of the FCS, providing a means of testing and validating the system prior to actual flight test. The facility is essential for the flight test phase of the program for providing a means of verifying any and all changes made to the flight control computers before succeeding flights. A typical VALFAC system configuration is shown in Figure 7.4-1.

It will be used to simulate real-time aircraft and engine dynamics, air data, navigation, and autoflight system, as well as to provide a pilot interface and a Closed-Loop Validation Facility operator's interface. It will provide all electrical and optical interfaces to the flight control computers and have the capability of injecting faults onto the electrical and optical interfaces for testing the control system responses to different failures.

The Closed-Loop Validation Facility could also be used during EME environment testing using an iron bird mock-up of the FBL system for providing monitoring capability of the system and actuator buses of the flight control system.

#### 7.4.2 Simulation Capability/Description

The fundamental purpose of the validation facility is two-fold as outlined below.

- A. To support the testing and validation of the FBL integrated system by providing simulations for the needed functions/subsystems, and to provide a means of validating the entire integrated system, including all incremental changes, prior to each test flight.
- B. To support the testing and validation of the FBL/flight control computer suite during its development and after incremental changes by providing a simulation of its entire environment.

In order to accomplish the above, the validation facility will provide the following:

- A. Aerodynamic and engine simulations.
- B. Simulation of all avionics interfacing to the flight control computer suite.
  - 1. Inertial reference system
  - 2. Rate gyro system
  - 3. Air data system
  - 4. Auto flight system
  - 5. etc.

#### 7.4.2 Simulation Capability/Description (Continued)

##### C. Simulation controls.

1. Pilot inputs
2. Dynamic testing mode
3. Static testing mode

The following sections describe the above requirements.

##### 7.4.2.1 Aerodynamic Simulation

The computational capability of the computer system used for the aerodynamic simulation will be sufficient to simulate the dynamics of aircraft motion to the extent necessary to realistically simulate flight of the aircraft throughout its operating envelope as it would be seen by the flight control computer suite.

A realistic aerodynamic simulation will require the following:

- A. A full six-degree-of-freedom aerodynamic model with an update rate fast enough to generate all aircraft characteristics of concern.
- B. Experiment data from the test aircraft to model the aircraft engines and aerodynamic effects due to all surface deflections.

##### 7.4.2.2 Avionics Simulations

In addition to the flight control aerodynamic simulation, the validation facility must also provide a simulation for all the avionics systems that interface with the flight control computers. The below paragraphs give a brief description of the main systems that will be simulated.

- A. Inertial Reference System (IRS): This simulation outputs the necessary set of IRS data as required by the overall simulation. Coordinate transformations must be made as necessary to determine accelerations, velocities, attitude, and position relative to the earth's coordinate system. In some cases, the IRS data will need to be computed in double precision to achieve the accuracy necessary for long distance navigation. The effects of simulated winds must be included in the calculation of velocities and positions. Multiple outputs will need to be provided to simulate multiple IRSs. Each output will need the capability of being separately manipulated for the simulation of errors such as drift, noise, and relative time skew.
- B. Air Data Computer (ADC): This simulation calculates the airspeeds, pressures, temperatures, and altitude of the aircraft. These computations must be based upon the definition of the international standard atmosphere. The model should have the capability of accepting barometric corrections as needed.

#### 7.4.2.2 Avionics Simulations (Continued)

- C. Thrust Management Computer (TMC): This simulation provides the necessary automatic throttle control to couple with the flight control computer. The appropriate thrust commands are computed by this model for use by the engine model according to mode and command information received from the autoflight system model.
- D. Autoflight System (AFS): This simulation provides the autoflight functions of the validation facility. When engaged, this simulation drives the flight control computers as well as the TMC simulation.

#### 7.4.2.3 Avionics Interfaces

The specifications for the interfaces between the flight control computer suite and the other aircraft avionics will be defined according to the configuration of the test aircraft. These interface specifications will describe exactly how the flight control computer suite will interface with each of the other avionic systems by describing bus type, bus electrical characteristics, and the bus data formats and update rates.

The validation facility will emulate the avionic interfaces as detailed in the above mentioned specifications accurately so as to validate compatibility between the flight control computer suite and the test aircraft avionics as development proceeds.

#### 7.4.2.4 Simulation Controls

The following sections describe the various means by which the operation of the Validation Facility may be controlled.

##### 7.4.2.4.1 Pilot Station

The pilot station will provide the full instrumentation necessary to accurately assess the functionality of the flight control computers. Simulated or actual flight instruments, plasma panels, CRT's, and other types of displays may be used for generating visual outputs. Flight control panels and radio management panels are typically installed in this workstation to allow for user selection of aircraft flight modes and annunciations as well as radio frequencies. Engine and aircraft information is also provided at the pilot station. Active pilot station control inputs should include the following:

- A. Sidesticks
- B. Speedbrake
- C. Rudder controls
- D. Throttle

7.4.2.4.1 Pilot Station (Continued)

- E. Flap/slat controls
- F. Mode control panel
- G. Pitch-trim controls

An alternative to supplying the above cockpit controls would be to interface the validation facility directly to the test aircraft, using the cockpit controls of the test aircraft to drive the facility. Although this may prove to be a difficult task, the potential benefits may warrant further investigation.

7.4.2.4.2 Facility Operator Station

The Facility Operator Station will consist of the following equipment:

- A. Simulation status displays
- B. Control development terminals
- C. Strip chart recorders for real-time data recording
- D. Printers for outputting non-real-time data

Between the operator station and the pilot station, control of the facility will be maximized.

The operator station will provide the means to monitor the status of the simulations as well as to enter special modes of operation. The two main modes of operation are dynamic test mode and static test mode as outlined in the following two sections.

A. Dynamic Test Mode

The validation facility will support a dynamic test mode that can be initiated through the operator station. During dynamic mode testing, the aerodynamic model is enabled. During operation in the dynamic mode, the operator will have the capability to perform any of the following functions.

1. Fly the simulated aircraft to perform tests on the integrated FBL system.
2. Record outputs from both the simulation of the flight control computer suite.
3. Insert error offsets noise or wind into sensor output.
4. etc.

#### 7.4.2.4.2 Facility Operator Station (Continued)

The operator station will allow either initialization of or intervention into the simulation during dynamic test mode operation. During initialization, the capability of selecting either default initial conditions or entering new initial conditions prior to trimming the aircraft will be provided.

Some additional functions may be added as the integration validation facility will be used to support EME environment testing of the FBL system on the iron bird.

#### B. Static Test Mode

The validation facility will also support a static test mode that can be initiated through the operator station. During static test mode operation, the aerodynamic simulation is disabled. The only functions supported in this mode are the following:

1. I/O to the flight control computer suite.
2. Modification of the data being transmitted to the flight control computers including any error code fields.
3. Examination of data being received from the flight control computers.
4. Strip chart and printer setup for recording test results.

The static test mode is more flexible in a development environment than the dynamic mode in that each variable output to the flight control computers is individually controllable.

#### 7.4.2.5 Maintainability

It is important to keep the closed-loop validation facility fully functional during all phases of program development. Measures must be taken to ensure a high degree of integrity and maintainability of the facility. The system should have the following characteristics.

- A. All I/O and control cards should be based on proven designs. Spares must be available for immediate replacement should the need arise.
- B. Additional computer processor and control cards must be stored as spares should a subsequent replacement be necessary.
- C. Built-In Test (BIT) features should be included in the design. Computer diagnostics and any other simulator hardware should have test programs that are capable of detecting faults to the card level.
- D. A software backup system must be provided.

#### 7.4.2.5 Maintainability (Continued)

- E. Preventive and corrective maintenance must be provided according to maintenance agreement.
- F. All documentation must accurately depict the configuration of the system, and all software and hardware shall be maintained under configuration control.

#### 7.4.2.6 Documentation and Configuration Control

The closed-loop validation facility designs and implementation methodologies will be documented in detail. A full definition of the facility as a system will be generated. Each subsystem and hardware function will be thoroughly described. Schematics, wire lists, assembly drawings, and other written documentation will be maintained. Overall software development will follow DO-178 Guidelines. Software functions will be designed using structured techniques. Detailed source language listings will also be generated. Software and documentation will be archived on computer readable media.

#### 7.4.2.7 Certification

The closed-loop validation facility will be designed, built, inspected, and tested to levels required by the Federal Aviation Administration (FAA) and NASA. Certification of the closed-loop validation facility will not be required, but the three basic steps of conformity, acceptance test, and system test will be taken.

### 7.5 Flight Test Issues

#### 7.5.1 General Flight Test Issues

##### 7.5.1.1 Handling Qualities and Crew Workload Evaluation

Flight test is used to evaluate both the handling qualities of the aircraft and crew workload during different flight regimes. The evaluation typically takes into account the ergonomics of the cockpit and the responsiveness of the control system to pilot inputs in the maneuvering of the aircraft.

For the fly-by-light aircraft flight test, the focus will be on the response of the system rather than the ergonomics of the cockpit since the purpose of flight test is to prove the concept of the FBW/FBL system.

#### 7.5.1.2 Vehicle Aerodynamic Evaluation

Flight test is also typically used to evaluate the aerodynamics of the test aircraft. Since the test aircraft is of a proven aerodynamic design, this particular part of the evaluation will be focused on the testing and evaluation of the control law software that has been programmed into the flight control computers. This evaluation will determine if the flight control system adequately responds according to the established performance criteria set for the aircraft type.

#### 7.5.1.3 Shock/Vibration and Other Operational Problems

This is crucial part of the flight test evaluation for the avionics equipment to ascertain any observable operational problems, such as shock and vibration effects on the system components or any other mechanical type problems.

#### 7.5.2 Flight Test Issues Specific to Fly-By-Light Aircraft

##### 7.5.2.1 Proof of Concept

Flight test is critical for the evaluation of the fully integrated FBW/FBL system. Variation induced by actual flight will deviate from the simulation slightly. Therefore, the flight test provides the confidence in the control system that it perform safely and functions correctly in the actual circumstances it was intended to operate in.

##### 7.5.2.2 Shock/Vibration and Environmental Evaluation of Fiber Optic Connectors

An important aspect to the FBW/FBL flight test is the actual vibration effects on fiber optic connectors known as Vibro-Mechanical Interference (VMI). In order to adequately assess if VMI is a problem, the fiber optics must experience long term vibration and environmental aging. This will require many flight hours to be logged with the FBW/FBL system.

Also another concern for fiber optics is the exposure to the harsh environment of the aircraft. Such as the different temperature extremes, other mechanical stresses, and the contaminants (hydraulic fluid, etc.) present in the fuselage of the aircraft. Flight test will help assess these potential problems.

##### 7.5.2.3 Evaluation of Fiber Optic EME Protection Effectiveness

This part of the flight test evaluation should help answer the following questions:

- A. Does fiber optics provide sufficient protection against the EME threat?
- B. What is the protection effectiveness level that fiber optics provide to the system?



8.0 RECOMMENDATIONS

- A. Establish a national resource for system modeling to include redundancy management.
- B. Establish a national resource for electromagnetic effect (EME) assessments/analytical predictions.
- C. Establish a national resource for EME full threat testing capability.
- D. Survey and document industry experience with the application of N-version techniques.
- E. Based upon experience gained in both industrial applications and research experiments, one or more development methodologies for N-version techniques should be documented. It is believed essential that formal mature methodologies be clearly established.
- F. Flight test validation via flight test of FBW/FBL flight control system.
- G. Optical data transmission validation.
- H. Optical sensor development, post first generation fly-by-light.
- I. Evaluation of life cycle maintenance of FBL (extended availability).
- J. Validation of transient tolerant architectures, (system recovery techniques).
- K. Extension of software methodologies/validation appropriate for full-time flight-critical commercial transport avionics systems.



9.0 OVERVIEW OF FLY-BY-LIGHT TECHNOLOGY DEVELOPMENT PLAN

9.1 Outline of Phases (reference Table 9.0-1)

- A. Phase I: Fly-By-Light (FBL) Architecture Development
- B. Phase II: Verification and Validation (V&V) of FBL
- C. Phase III: Integration of a FBL System
- D. Phase IV: Flight Test of a FBL System

9.1.1 Description of Phases

9.1.1.1 Phase I: Fly-By-Light Architecture Development

This phase encompasses the definition and design of representative new technology subsystems, their validation and integration as needed for a first generation FBL/flight control system which incorporates an optimized blend of electronic and optical technologies; and the development, prototyping, acquisition, and testing of new technologies, hardware, and subsystems that make up the FBL system.

9.1.1.2 Phase II: Verification and Validation of Fly-By-Light

This phase encompasses the establishment of the criteria and approach for the certification of a FBL/flight control system; the development of design/optimization tools; V&V tools for fault tolerant operation at the subsystem and system levels; the development of electromagnetic (EM) propagation models to predict coupling levels, and effects through the aircraft and its systems; and document proof of relevant correlation between the models and "real" systems via an extensive hardware test and analysis program.

A major goal of this phase is to develop and demonstrate the use of tools for certification purposes by performing V&V via models rather than exhaustive and expensive physical testing of hardware and software. In association with this program the Federal Aviation Administration (FAA) will directly participate in deriving a FBL certification basis.

9.1.1.3 Phase III: Integration of a Fly-By-Light System

This phase encompasses the bench test plan development; the design and construction of a FBL "iron bird" integration and test facility; the installation and integration of FBL system; and the hot bench testing of the FBL/flight control system.

9.1.1.4 Phase IV: Flight Test of a Fly-By-Light System

This phase encompasses the development of the flight test aircraft installation designs and the flight test plan; the aircraft modification; the aircraft installation of the FBL/flight control system, and the flight test.

9.2 Major Aspects

- 9.2.1 Phase I: Fly-By-Light Architecture Development (30M)
- A. Identify/define methodologies with potential application on next generation aircraft. (1M)
  - B. Contrast methodologies to provide subsystem extended availability; i.e., effective mean-time-between-failure (MTBF) equal to or greater than 100,000 hours. (2M)
  - C. Contrast methodologies for subsystem fault tolerance. (2M)
    - 1. Hard faults.
    - 2. Soft faults.
    - 3. Generic faults.
  - D. Contrast potential next generation sensor methodologies. (3M)
    - 1. Power-by-light.
    - 2. Passive optical.
    - 3. Traditional sensors.
  - E. Contrast next generation actuation technology. (2M)
    - 1. Smart versus simple.
    - 2. Fault protection.
    - 3. Interface.
  - F. Determine optimum data sharing/transmission methodologies. (5M)
    - 1. Optical.
    - 2. Electrical.
    - 3. Hybrid.
  - G. Develop viable fly-by-light integrated architectures for near and long term applications. (15M)

|         |   |       |
|---------|---|-------|
| 9.2.2   | Phase II: Verification and Validation of Fly-By-Light   | (46M) |
| 9.2.2.1 | Electromagnetic Effect (EME) Verification and Validation (V&V).   | (30M) |
|         | A. Develop a national resource for analytical prediction of EMEs.   | (17M) |
|         | 1. On surface of aircraft.  |       |
|         | 2. Inside structure.  |       |
|         | 3. On cables.   |       |
|         | 4. Within line replaceable units (LRUs).  |       |
|         | 5. Within card assemblies.  |       |
|         | 6. Within electronic devices.   |       |
|         | 7. Predict results of EME tests/test configurations.  |       |
|         | B. Develop technology necessary for limited full threat EME testing of aircraft and associated subsystems.  | (8M)  |
|         | C. Validate EME prediction resource.  | (5M)  |
| 9.2.2.2 | Fault Tolerance V&V   | (16M) |
|         | A. Evaluate available means for modeling fault effects (e.g., Semi-Markov Unreliability Range Evaluator [SURE], CARE III, Hybrid Automated Reliability Predictor [HARP]). | (1M)  |
|         | B. Define capabilities required to fully validate fault handling capability/characteristics.  | (1M)  |
|         | 1. Functional modeling.   |       |
|         | 2. Circuit modeling.  |       |
|         | 3. NAND-gate-equivalent modeling.   |       |
|         | C. Develop/enhance analytic modeling tools to provide needed capabilities through the year 2005.  | (8M)  |
|         | D. Validate modeling/analysis tools.  | (6M)  |
| 9.2.2.3 | FAA Participation   |       |
|         | A. Integral FAA involvement will provide for development of cost-effective certification procedures for fly-by-wire/fly-by-light (FBW/FBL) aircraft built in the USA.     |       |

- 9.2.3 Phase III: Integration of a Fly-By-Light System (65M)
  - A. Build new flight hardware. (40M)
  - B. Develop "iron bird" integration/validation facility. (25M)
    - 1. Simulator.
    - 2. Subsystem/component installation.
    - 3. Evaluation testing.
    - 4. Aircraft modeling.
    - 5. Subsystem/component test facilities.
  
- 9.2.4 Phase IV: Flight Test of a Fly-By-Light System (31M)
  - A. Aircraft modification to accept new subsystems. (9M)
  - B. Design/build/installation of in-flight data gathering facilities for each major subsystem to be evaluated. (6M)
  - C. Subsystem/component installation. (7M)
  - D. Ground check test. (2M)
  - E. Test plans. (1M)
  - F. Test flights. (6M)

This plan includes \$172M total industry and government funding.

# FLY - BY - LIGHT TECHNOLOGY DEVELOPMENT PLAN

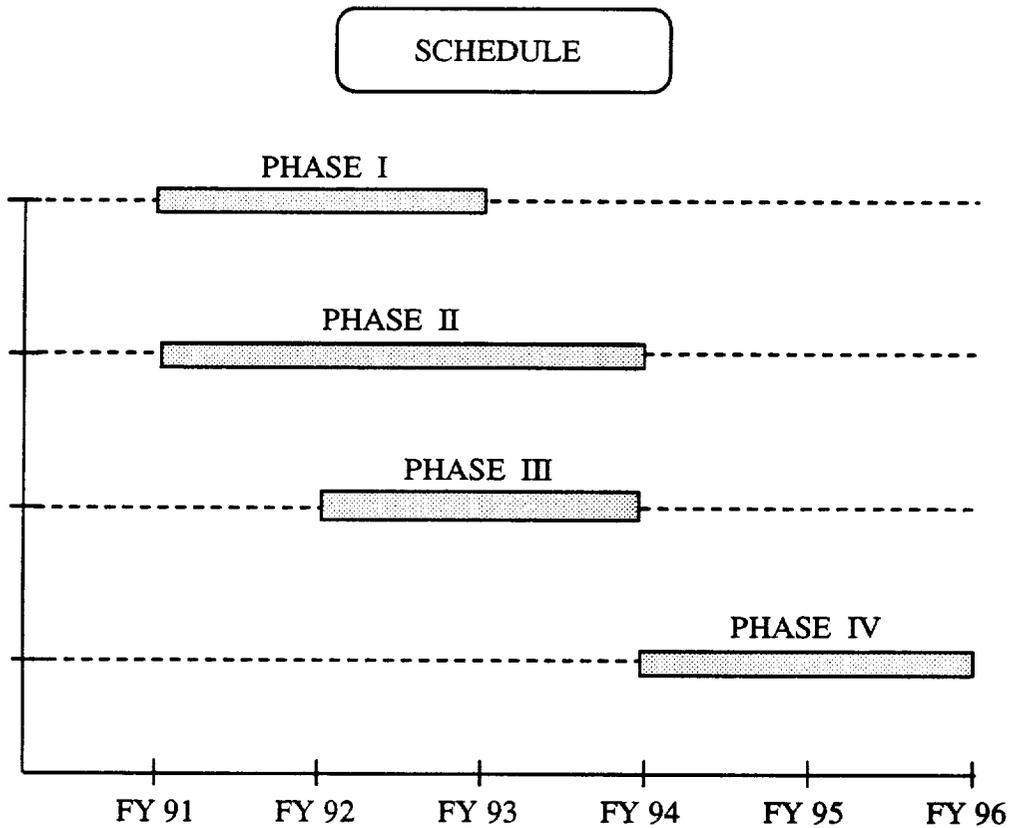


TABLE  
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**FLY-BY-LIGHT TECHNOLOGY DEVELOPMENT PLAN**

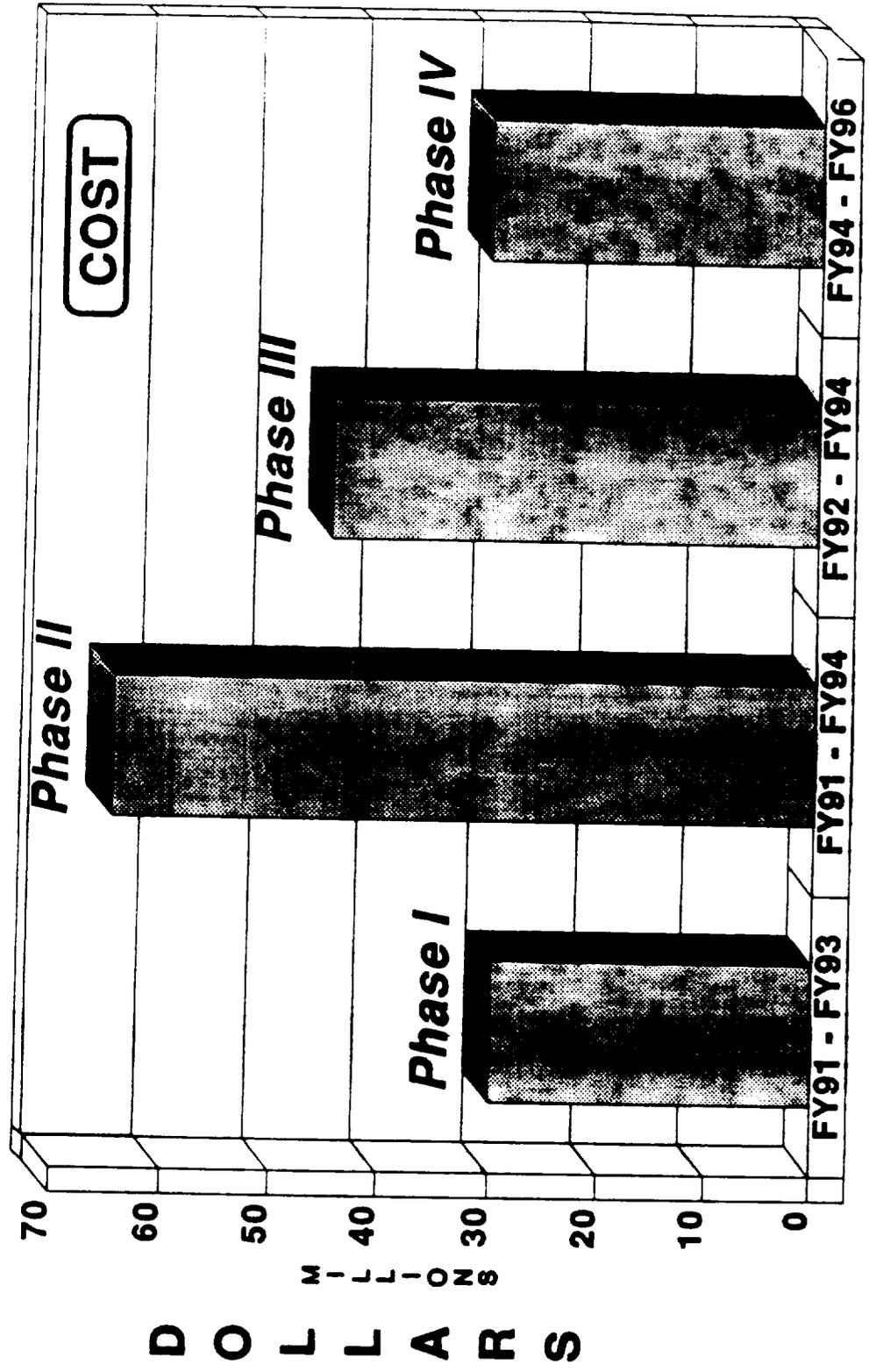


TABLE  
9.0-2

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# Report Documentation Page

|   |  |  |   |  |           |
|---|--|--|---|--|-----------|
| 1. Report No.<br>NASA CR-181954   |  | 2. Government Accession No.                          |   | 3. Recipient's Catalog No.   |           |
| 4. Title and Subtitle<br><br>Fly-by-Light Technology Development Plan   |  |  |   | 5. Report Date<br>August 1990  |           |
|   |  |  |   | 6. Performing Organization Code  |           |
| 7. Author(s)<br>J. R. Todd  |  |  |   | 8. Performing Organization Report No.  |           |
|   |  |  |   | 10. Work Unit No.<br>506-46-21-05  |           |
| 9. Performing Organization Name and Address<br>McDonnell Douglas Corporation<br>Douglas Aircraft Company<br>3855 Lakewood Blvd.<br>Long Beach, CA 90846   |  |  |   | 11. Contract or Grant No.<br>NAS1-18028  |           |
|   |  |  |   | 13. Type of Report and Period Covered<br>Contractor Report                             |           |
| 12. Sponsoring Agency Name and Address<br>National Aeronautics and Space Administration<br>Langley Research Center<br>Hampton, VA 23665-5225  |  |  |   | 14. Sponsoring Agency Code   |           |
|   |  |  |   | 15. Supplementary Notes<br>Technical Monitor: Cary R. Spitzer<br>Final Report - Task 9 |           |
| 16. Abstract<br><p>This report describes a national plan to make Fly-by-Light an effective technology leap for the U.S. commercial aircraft industry in the late 1990's. The driving factors and developments which make this technology viable are discussed and documentation, analyses, and recommendations needed to facilitate the introduction of commercial Fly-by-Light aircraft are provided. To accomplish this goal a unified "national" effort, coordinated by NASA, is proposed.</p> |  |  |   |  |           |
| 17. Key Words (Suggested by Author(s))<br>Fly-by-Light<br>Fly-by-Wire<br>Flight Controls<br>Fault Tolerance<br>Optical Sensors  |  |  | 18. Distribution Statement<br><br>[REDACTED] (REVIEW FOR GENERAL RELEASE<br>2 YEARS FROM PUBLICATION DATE)<br><br>Subject Category 08 |  |           |
| 19. Security Classif. (of this report)<br>Unclassified  |  | 20. Security Classif. (of this page)<br>Unclassified |   | 21. No. of pages<br>182  | 22. Price |