



HANDBOOK-VOLUME II DIGITAL SYSTEMS VALIDATION

CHAPTER 11 AIRCRAFT ELECTROMAGNETIC COMPATIBILITY (GUIDELINES TO ASSESS EMC DESIGNS)



PREPARED BY:

**COMPUTER RESOURCE MANAGEMENT, INC.
950 HERNDON PARKWAY, SUITE 360
HERNDON, VIRGINIA 22070**

PREPARED FOR:

**FEDERAL AVIATION ADMINISTRATION
TECHNICAL CENTER
ATLANTIC CITY INTERNATIONAL AIRPORT, NEW JERSEY 08405**

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PREFACE

This document is based on work performed by the Boeing Commercial Airplane Company, P.O. Box 3707, Seattle, Washington 98124, under the National Aeronautics and Space Administration (NASA) contract NASA-12261 for the Federal Aviation Administration (FAA), NASA-Ames Research Center, Moffett Field, California 94035. The contract was firm fixed price, level-of-effort term from September 1985 to September 1986 with an extension to June 1987. Contracting officers were W. C. Botts, Boeing and A. N. Johnson, NASA. The FAA contracting officer's technical representative (COTR) was William E. Larsen. Deliverables were an Interim Report (Draft) and an Interim Report (Final). The program manager was Robert D. Force, and principal investigator, Clifton A. Clarke.

Bob Force, who helped put the program together, played a central part in managing the planning and organization of the total document. The section 2.1, "Existing Systems," is derived from the valuable "Active Controls Technology" report which Bob co-authored. Bill Larsen provided extensive expertise taken from his own experience, and through constructive source material. He also provided sound and highly regarded recommendations for content and organization. Dale R. Reed collaborated on the wiring-induced voltages and worked some of the in-depth computations. He offered very profitable perspectives on the approach to aircraft engineering analysis and design; his forbearance and tenacity are appreciated.

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This document would not be possible without the unparalleled IEEE Symposium Records and the periodical "ITEM", R & B Enterprises, whose presentations were a source of valuable data applicable to an aircraft.



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EXECUTIVE SUMMARY

The Aircraft Electromagnetic Compatibility guidelines document deals with electromagnetic compatibility in a commercial transport aircraft including the specifications, the activities, the design, and the tests to verify and validate compatibility.

Objectives are to view architecture, equipment and wiring location, material properties, circuit susceptibilities, and environment as seen from the electromagnetic compatibility design perspective of balanced circuits, filters, electrical bonding, grounding, and shielding.

Even today, digital electronics are much more common in aircraft. Automated flight controls of future aircraft will operate under the control of digital clocks, data buses, switching regulators, pulse width modulated power, and radio frequency transmitters on the one hand, and on the other, sensitive analog and digital instrumentation.

Safe and efficient flight will depend on the performance of electronics. It will be important to understand the electromagnetic interference types and the electromagnetic interference paths (figure E-1).

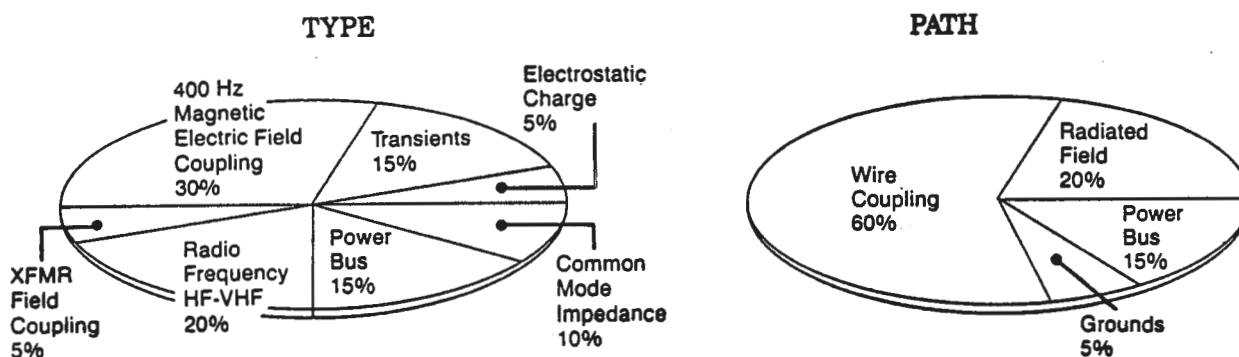


FIGURE E-1. POSTULATED PERCENTAGES

The Electromagnetic Interference (EMI) types are set forth in this document by showing a profile of the magnetic and electric fields from power lines; some military, urban, and rural radio frequency field strengths; and the properties of transients. Significance of the wire circuit return, the balanced circuit, grounding, shielding, and software highlight the protection techniques of a layered design which will block paths of electromagnetic interference and maintain interface signal quality. Design specifications, activities, and reviews are proposed to help set up guidelines for equipment verification and aircraft validation.



1. INTRODUCTION

1.1. Background and Scope

1.1.1. A Case of Engine Shutdown

Captain Hoag thought briefly of the moment when he left home last night. His son, daughter, and wife were all there. They had joked about their planned upcoming vacation, their first together in two years.

A voice broke into his short reverie: "Flight 211, you're low and to the left - please maintain 023 - you have a cell at 2 o'clock."

Hoag said to the first officer, John Pearson: "John, push it forward and bring 'er up."

John said: "Gotcha covered. I flew one of these new ones two weeks ago into Loridan International - they sure do handle smoothly."

"Okay - uh huh." "Gear down."

Flight 211, Atlantic Air, was on approach to Keithrode International Airport (KIA). It was 16:45 on October 16. Two hundred and twenty-seven passengers were on board. The weather had been partly cloudy with thunderstorms predicted and cell activity in the proximity of the air terminal.

Tower: "Flight 211, you are cleared on Runway 3. You are still low."

Captain Hoag: "John, bring 'er up."

A lightning flash occurred off to the left. Then, instantly, a blinding flash, an overwhelming shudder - the aircraft metal structure and body seemed to vibrate under a massive pressure and energy wave.

Hoag: "We've lost number 1! Push it all the way forward!"

Pearson: "Okay"

Hoag: "All the way forward - all the way forward - oh."

The plane hit the earth with a great screech, scrape, a shower of sparks, and a grinding of metal. It then rose again, lumbering and awry, as if struggling to be airborne - struggling - then it smashed again to the ground. The tail buckled. Flames broke out.

Fifty-six people were killed - including the crew. A number of people were injured.

HYPOTHESIS NO. 1: The piercing lightning strike to the left-hand engine caused a large atmospheric pressure wave. This wave traveled through the engine intake into the main engine chamber, snuffing out the flame, and thus the engine power.

HYPOTHESIS NO. 2: The lightning strike to the engine established a large electrical current flow in the engine structure and cowling. Electrical circuits connected to the structure experienced voltage transients causing valve malfunction and leading to an engine shutdown.

This case is dramatic. It is awesome. It commands attention. We recognize that we must protect against this type of event, model it, and develop reiterative computations and tests to uncover the boundaries of transient energies invading important electrical circuits.

This case is given to illustrate the contrast between a very visible and threatening electrical upset or damage phenomenon and the usual run of invisible Electromagnetic Interference (EMI) unfamiliar to most airline passengers. Normal electromagnetic interference environmental problems ordinarily have not carried with them the drama of the above case. Aircraft have been constructed with controls and electrical apparatus having EMI problems but not having any influence on safety.

Knowledge of and protection against induced noise voltages are necessary today. They will be even more necessary in future aircraft where vital and critical control functions are being taken over by avionics interconnected by digital data buses. These data buses that could impair safety if beset by electrical noise. We need access to knowledge of the various types of noise. There is a growing awareness of electromagnetic interference and Electromagnetic Compatibility (EMC).

1.1.2. Electromagnetic Compatibility

Electromagnetic interference could cause a flight delay or endanger the operation of an aircraft at 30,000 feet. Generators of electromagnetic interference for aircraft (figure 1.1-1) take on several forms:

- Transmitters of radio frequencies that may be installed on the aircraft itself, such as High-Frequency (HF) or Very High-Frequency (VHF) communication links, or high-energy sources located on the ground such as our everyday frequency modulated (FM) radio or HF-VHF-UHF broadcast stations.
- The aircraft power line 400-Hz electric and magnetic fields.
- The computer and avionics microprocessor timing and control clock signal circuits that generate radio frequencies of one MHz or higher.
- The aircraft power switching regulators which are used to convert from one level of power to another.

Electrical switching transients sparked by the turn on and off of aircraft lights, fans, and engines or by the operation of control surfaces, ailerons, slats, and flaps.

Electrostatic discharges including lightning.

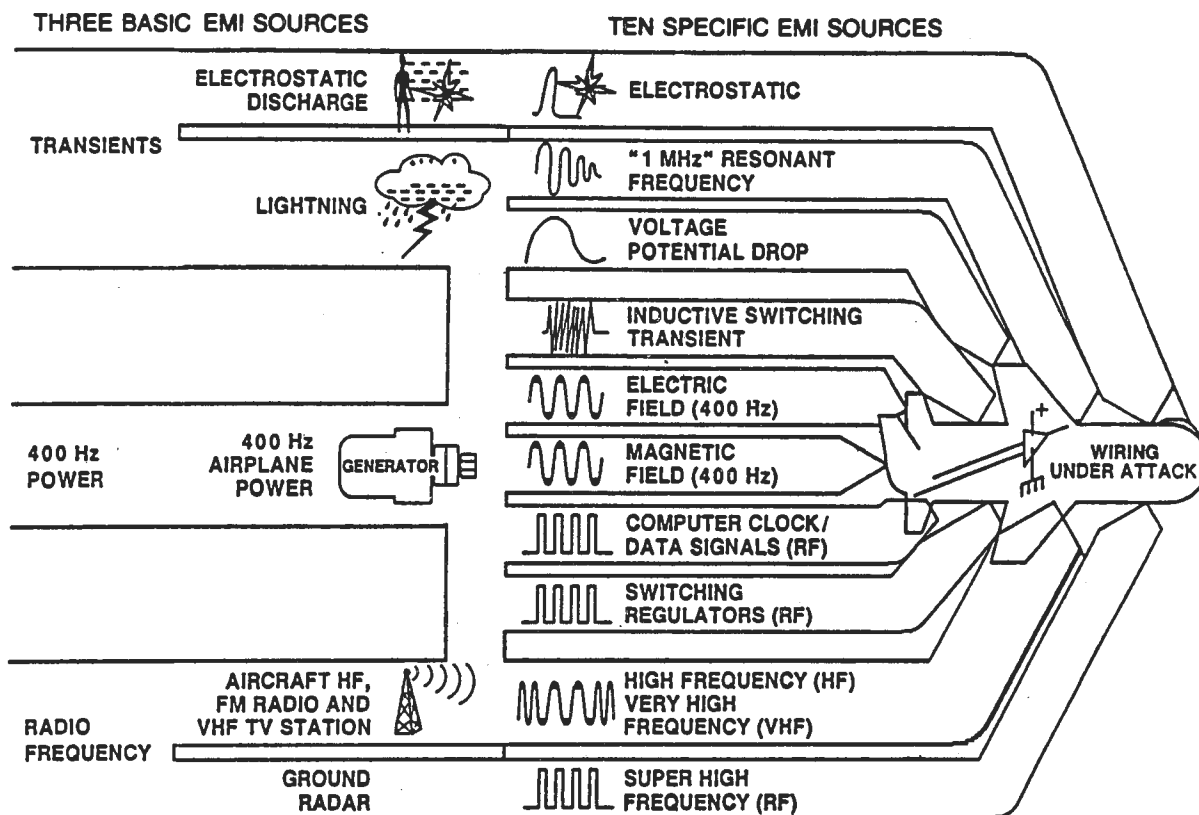


FIGURE 1.1-1. REPRESENTATIVE EMI SOURCES

These transients and electromagnetic waves may transfer into wiring and cause "electromagnetic interference" to microcircuits inside electrical equipment and avionics. This interference could result in a trifling disorder in a flight deck display or, more seriously, an engine shutdown.

The conductive paths of electrical wiring provide an avenue to usher electromagnetic interference directly to airplane avionics and signal inputs. Eliminate wiring, and electromagnetic interference almost vanishes. Wiring is

the most important factor in electromagnetic interference and electromagnetic compatibility. Of much less importance is the electromagnetic interference path through the avionic equipment metal housing or case. Wiring is the electrical interface and connection between avionic equipment. Its designated job is to transfer avionics signals, data, and information. In that function, it can often act to transfer electromagnetic interference energy to other wires in a wire bundle. It also sprays or radiates like a transmitting antenna and very efficiently receives radio frequency energy like a receiving antenna.

Recently, an experienced electromagnetic compatibility engineer, "an old hand," was asked, "What's the number one requirement for an electromagnetic compatibility design program that will rule out electromagnetic interference problems?"

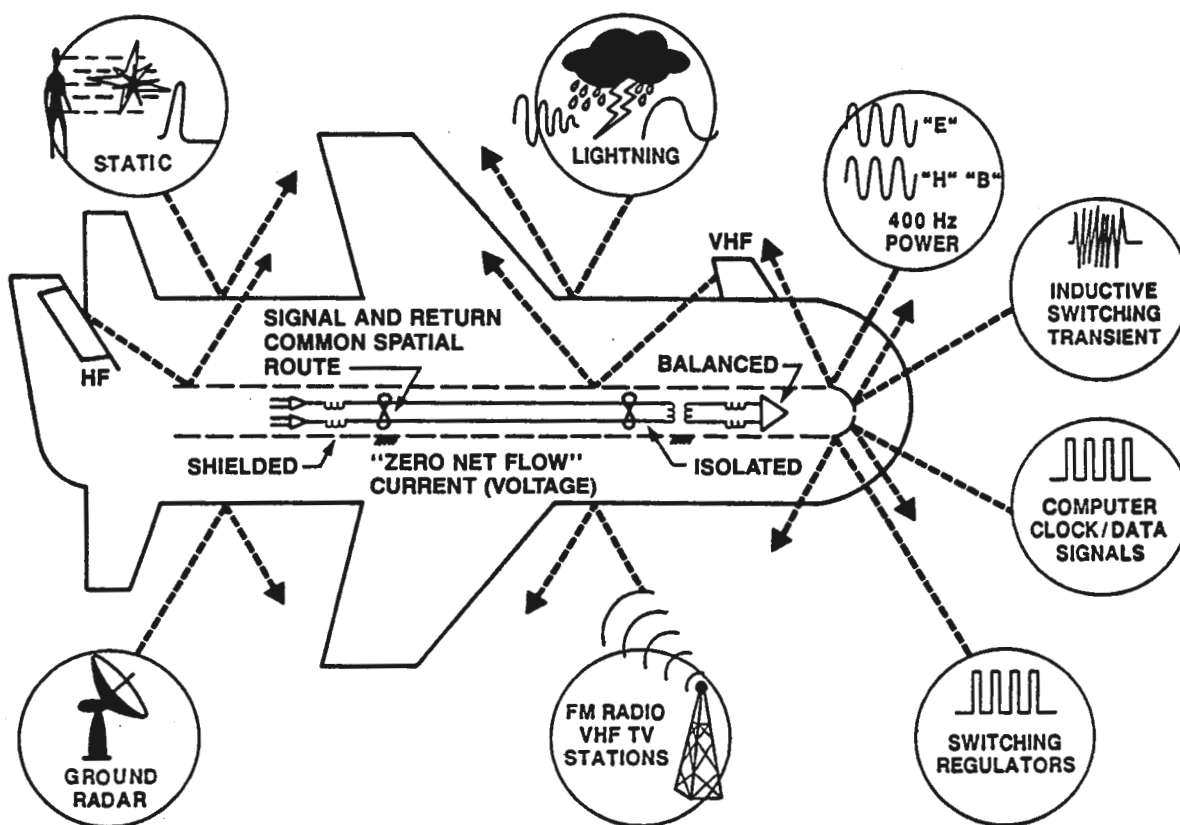


FIGURE 1.1-2. EMI ENVIRONMENT IN AIRCRAFT

His answer, "Zero net current flow in a shielded, balanced, isolated circuit." In other words, always route the signal wire and return or the power wire and its return together using twisted pair, coax, or shielded pair, which means that aircraft basic structure is not used as the return path for the circuit (figure 1.1-2). This will almost eliminate the three electromagnetic interference avenues of entry: common mode impedance paths, magnetic field coupling, and electric field coupling. (Common mode impedance conditions exist when two circuits share a portion of the same electrical path.)

So, that's it: use a return wire and use a shield for each circuit.

Without good shielding, the results are predictable. Removing the shield from the circuit and separating the return from the signal wire not only destroys the efficiency of the circuit, it also opens up the circuit to intervention from stray electric and magnetic fields. These fields provide an avenue directly to the micro-processor memories, computers, and controllers that are needed for aircraft operation and for processing and control of flight deck displays and instruments. A recent estimate of airplane problems (figure 1.1-3) indicates that power line electric and magnetic field coupling (30%), radio frequency fields (20%), transients (15%), and common mode impedance paths (10%) make up a total of 75% of deficiencies. These can be largely corrected with proper wiring design. Electromagnetic interference may occur in many of the aircraft subsystems (figure 1.1-4). If uncontrolled, it appears as radio tones, static, or 400-Hz hum on the passenger entertainment systems. It can show up as flight deck display distortion or illegibility, impaired data transmittals, computer memory loss, and may even result in suspension of equipment operation. Radio frequencies are becoming more of a concern. Today, the predominant radio frequency fields that impair avionic equipment operation fall into the HF-VHF radio frequency spectrum (figure 1.1-5). In future aircraft, the range may vary. (See section 10, Bibliography, ECAC Study.) Electromagnetic compatibility requirements encompass almost every subsystem on an aircraft.

The field of electromagnetic compatibility is not only a discipline in itself, it is also the cornerstone in the proper application of other engineering technologies. Good engineering design techniques employed in the areas of avionics, wiring and cabling, electrical bonding and grounding, lightning protection, and others, go a long way in achieving electromagnetic compatibility. Electromagnetic fields are produced by the generation, transmission, and utilization of electrical energy. Stray electromagnetic energy is generally not desired, and quite often interferes with the operation of electrical/electronic equipment, hence, the name electromagnetic interference. Control specifications for electromagnetic interference generation and electromagnetic interference susceptibility are required to achieve electromagnetic compatibility. With proper wiring, shielding, and application of voltage limiters in a good electromagnetic compatibility design, there is increased confidence in equipment operation which converts directly to the bottom line of on-time dispatch for the airlines and their passengers.

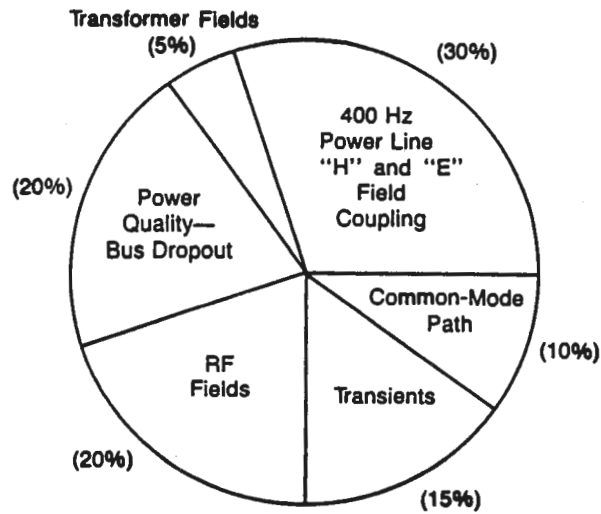


FIGURE 1.1-3. PERCENT TROUBLESHOOTING-A/C

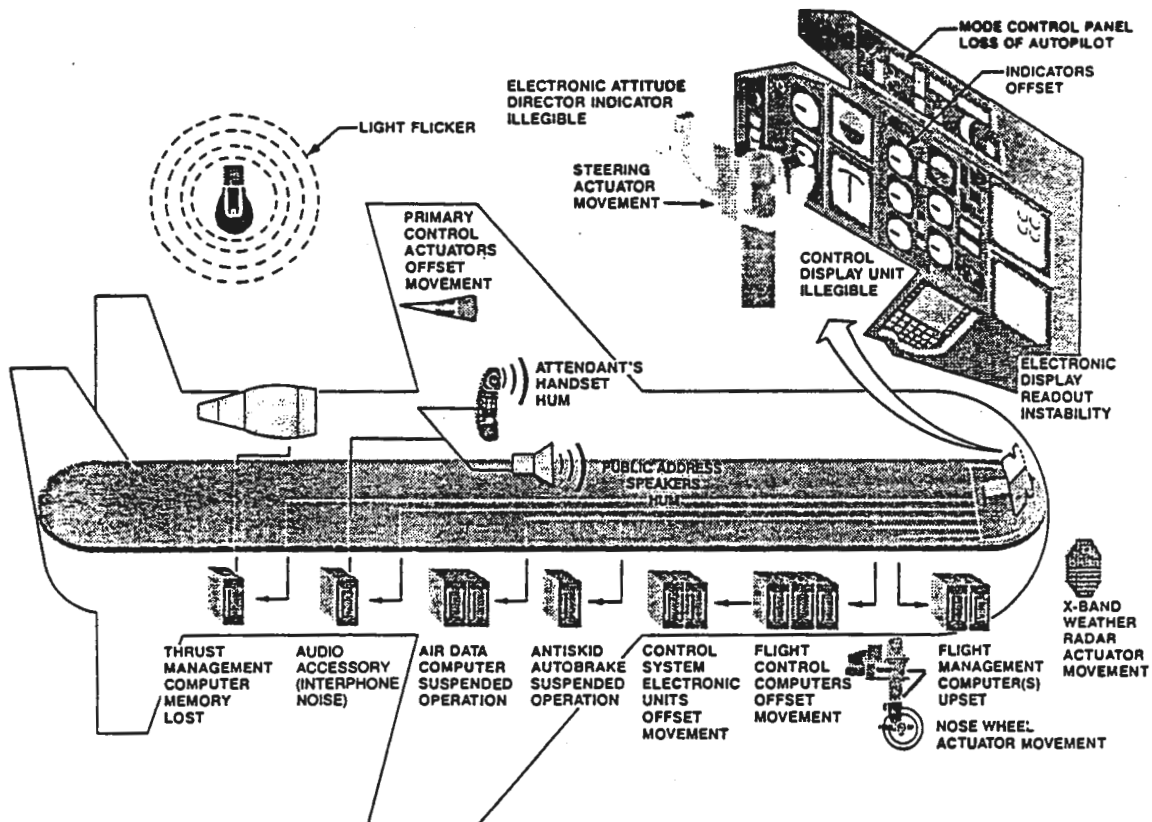


FIGURE 1.1-4. ELECTROMAGNETIC INTERFERENCE EFFECTS

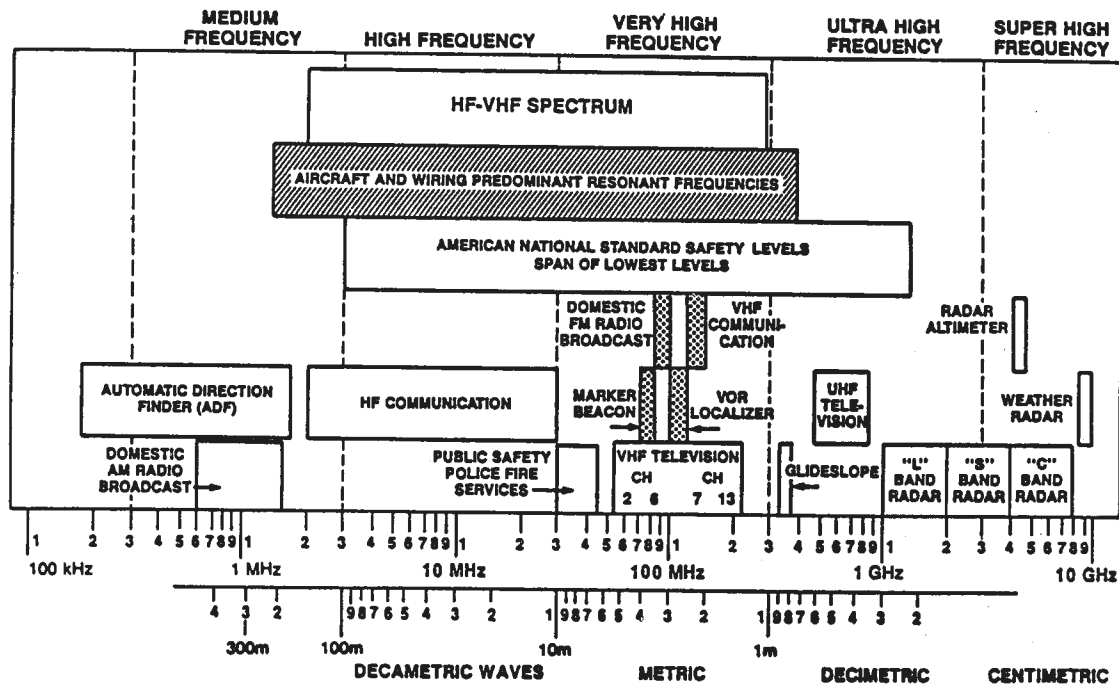


FIGURE 1.1-5. RADIO FREQUENCY RANGE

The steps to reach electromagnetic compatibility in an airline are basically threefold: (1) procure equipment and wiring according to EMC specifications, (2) package the equipment and wiring in the aircraft to obtain protection from structure, and (3) measure and test the equipment and wiring for all aspects of EMI during the program to guarantee verification and validation of EMC (see section 7, Verification and Validation).

EMC specifications for the commercial airplane are covered by Federal Aviation Regulations (FAR) and are specified in terms of system operation. FAR, Part 25.1353, Paragraph A, covers electrical equipment: "Electrical equipment controls and wiring must be installed so that operation of any one unit or system of units will not adversely affect the simultaneous operation of any other electrical unit or system essential to safe operation."

FAR, Part 25.1431, Paragraph C, covers electronic equipment: "Radio and electronic equipment, controls and wiring must be installed so that operation of any one unit or system of units will not adversely affect the simultaneous operation of any other radio or electronic unit or system of units, required by this chapter."

Environmental and emission control specifications for electronic equipment intended for installation in aircraft are documented in the current revision of Radio Technical Commission for Aeronautics (RTCA) Environmental Specification DO-160. (See sections 5 and 9.)

1.2. Electromagnetic Compatibility Priorities

1.2.1. Responsibilities and Policies

A new airplane presents a challenge of equipment location, packaging, assessment of environment, and resolution of new technology problems. The airframe manufacturer takes on the responsibility in the electromagnetic compatibility design to outline the operational-temporal-spatial anatomy of the airplane early in the conceptual stage. It is a difficult and extensive task. (See sections 4.2 and 4.3.)

Equipment location, wiring, transmitters, receivers, 400-Hz power, lightning diversion, and static dissipation are all familiar items to be pursued, tracked down, identified, and recorded. They absorb many labor hours.

Even with the rapid advance of technology, many components, characteristics, and parameters on a new airplane do not change. The electromagnetic compatibility engineer has a responsibility to know the off-the-shelf equipment. A productive policy is to seek out and rely on existing specifications and designs to the greatest extent possible, thereby sidestepping duplication of effort in analysis, scheduling, and testing. EMI test levels documented in the current issue of the RTCA Document DO-160, in many instances, represent today's environment. (They do not take into account the effects of new composite structures.) The test levels have been developed over the years. Airframe manufacturers and subcontractors can take good advantage of existing specifications and test data.

So, the first priority of the airframe manufacturer is to take on the task of defining the electromagnetic compatibility anatomy - existing equipment, new equipment, new environment, and locations.

Location often sets the electromagnetic compatibility requirements for avionics. Electronics or lack of it influences design requirements for equipment. (See section 6.)

The first priority of the avionics supplier or subcontractor (including in-house suppliers) is to know the electromagnetic interference environment and then identify, design, and protect each power input and signal input to guarantee that the equipment will operate within performance standards in that environment. The designer must shoulder the responsibility of knowing the interface wiring. Often the designer is concerned with operational requirements and must make a special effort to recognize the noise requirements. DO-160 is the basic industry standard. (See section 3.4.2 and section 9.)

The airframe manufacturer deals with new materials and properties, wiring, electrical bonding, shielding effectiveness, and definition of environment.

The avionics supplier deals with input/output circuit protection, internal grounding, circuit card layout, and interface wiring. The supplier must observe the board-level noise thresholds in a real-time, interactive setting to ensure no "state changes" and to ensure adequate containment of electromagnetic

interference at the box level. The equipment engineer designs devices to be tolerant to magnetic fields from 400-Hz power (400-mV induced), 400-Hz electric field (up to 1,000V induced in test), radio frequencies (1V induced), transients from coils and lightning (600V), and electrostatic discharge (10,000V or higher). Future aircraft having critical fly-by-wire systems may require higher levels, especially for radio frequency fields. The designer must also install filters to control and contain emissions from the oscillator and switching regulator clock and harmonic radio frequencies. Interface wiring must be designed, agreed upon, and documented in an interface control drawing, otherwise hardware would require multiple electromagnetic interference designs.

Along with the size of a new program and the extent of its new technology, three pivotal factors influence the level of effort and are sine qua non to success: (1) management support, (2) the electromagnetic compatibility engineer's product experience, and (3) the electromagnetic compatibility experience of other participating engineers on the program. Management support means setting the priority for early resolution of requirements before formal document release of vendor technical specifications and statements of work. Product experience means minimum duplication and maximum productive effort. Experienced participating engineers means satisfactory coverage of subcontractor requirements, and good electrical bonding and wiring practices.

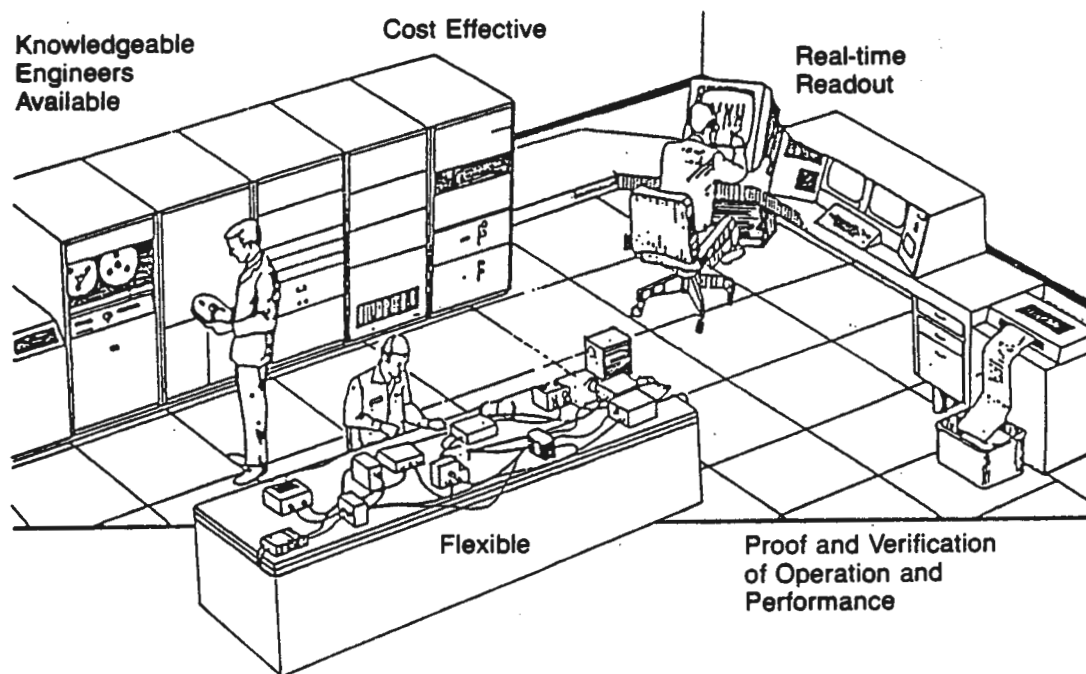


FIGURE 1.2-1. COST-EFFECTIVE SUBSYSTEM TESTING

On a recent airplane development program, extensive subsystem testing was performed on engineering models and production models with these three recognized benefits: (1) proof and verification of equipment performance, (2)

good diagnostic testing, and (3) knowledge of equipment operation. Diagnostic/trouble-shooting tests can be run in cooperation with the subcontractor. Subsystem testing is becoming a key element in a successful program (figure 1.2-1). (See section 7.)

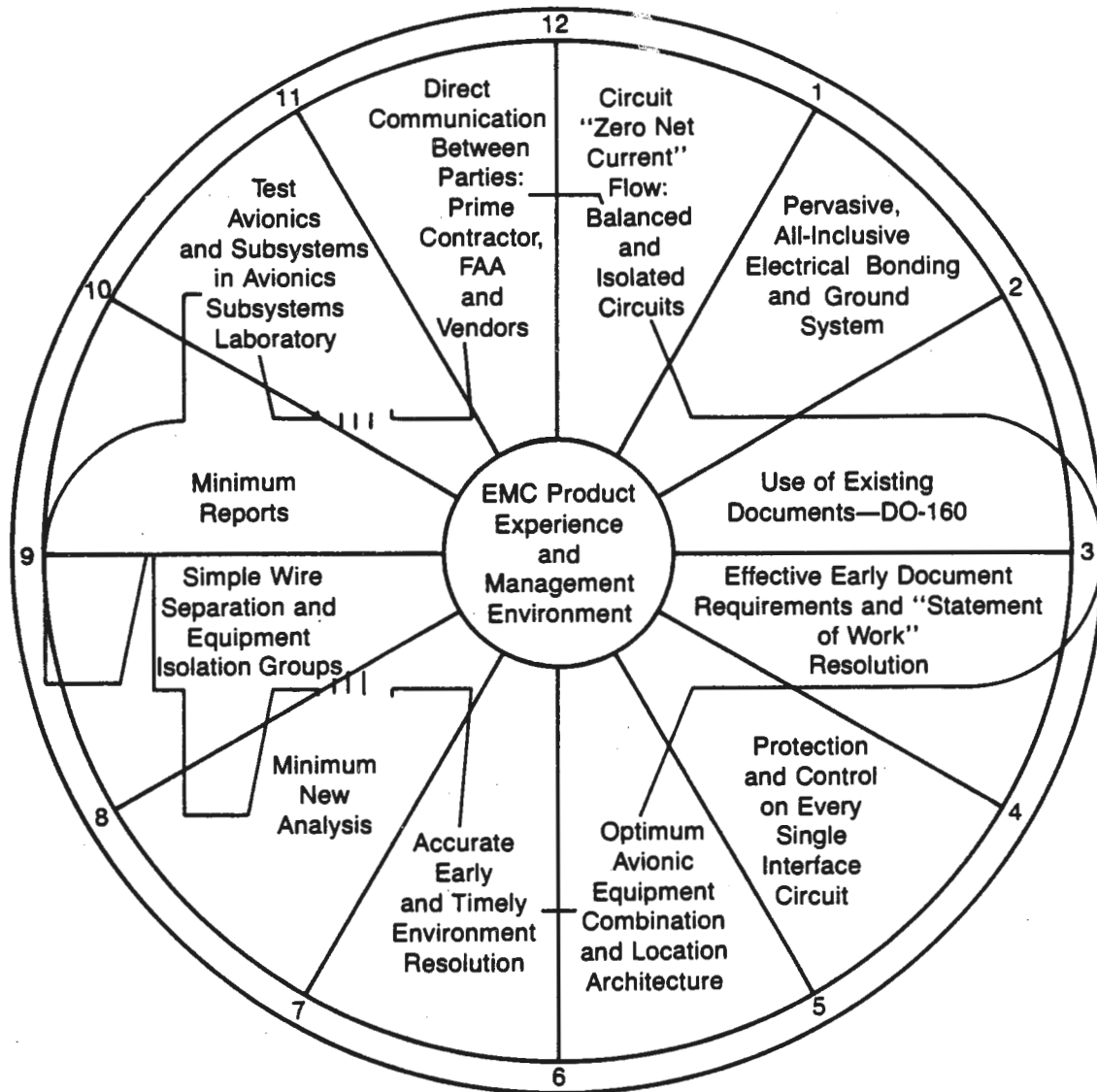


FIGURE 1.2-2. KEY EMC FACILITATORS

Entering into a new program incurs a commitment effort (figure 1.2-2), a commitment sized by the new technology and environment, and a commitment that dissipates with the settling of each subcontractor requirement, each design, each test, and each verification. Early adoption and documentation of known

steps for successful electromagnetic compatibility are established on these baseline technical priorities: (1) zero net current flow in a balanced, isolated, shielded circuit, (2) total, all-inclusive electrical bonding of every structure and every detail (including conductive paint on external dielectric surfaces), and (3) optimum avionic equipment in-line design and location making maximum use of structural shielding.

1.2.2. Documentation

To provide a foundation, baseline, reference, and continuity, there are these key documents: system level specification (possibly a paragraph or two), electromagnetic compatibility plan (may be brief), electromagnetic compatibility requirements (extensive and detailed), procurement specifications (with statements of work), interface identification, equipment test (procedure and report), and airplane test (procedure and report). (See sections 5 and 6.)

Procurement specifications must be finalized before contract formal approval.

The fact that minimum documentation leads to a more productive program is as fundamental as the fact that there are certain program top documents that must be instituted to define scope and intent and to reference industry specifications. The documentation of many of the tasks, analyses, and tests on a program can be covered by brief individual memorandums or reports. Some programs issue design notes, a practice that is an efficient method of recording and disseminating design requirements, rationale, and information.

1.3. How to Use This Document

The purpose of this Aircraft Electromagnetic Compatibility document is to digest, unify, highlight, and give perspective to the substantive aspects of electromagnetic compatibility applied to a commercial transport aircraft. The material in this document is not new. A very important resource to the electromagnetic compatibility engineer is the electromagnetic compatibility knowledge of other individuals associated with a program. Beliefs exist that graphite-epoxy is an insulator; that box-to-box radiation is important (it is the interface wiring that is the key to EMC); that shield tie (pigtail) length is not critical; that every single interface line does not need to be analyzed for protection and emission control; that ground planes are not required; that single-point grounding is always good (it is good for power, but not good for digital circuitry); and that extensive verification procedures can be ignored and are not essential.

The 1970s and 1980s have seen a striking rise in the quality and extent of EMC/EMI design engineering knowledge that has important consequences for EMC. This document attempts to collect and apply that information to the airplane. Derivations and fundamentals of EMC are not elucidated. This document is limited in that sense. Reference can be made to the bibliography for some excellent articles.

EMC information not found herein:

- Fundamentals or basics of EMC.

- Formulas, models, derivations.
- Antenna-to-antenna coupling.
- Power system quality.
- Lightning.

EMC information included:

- Aircraft-applied EMC.
- Architecture, equipment layout.
- Dominant EMI environments.
- Circuit susceptibilities.
- Bonding, grounding, shielding.
- Wiring design.
- Verification, validation.

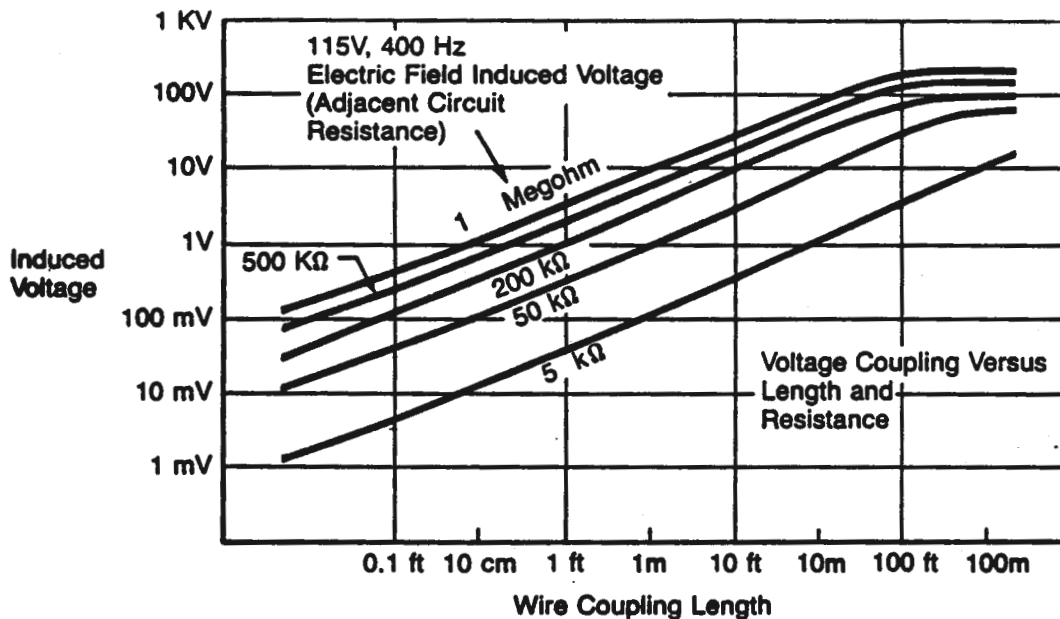


FIGURE 1.3-1. REPRESENTATIVE "E" FIELD COUPLING

It is recommended that this document be read thoroughly before concentrating on specific sections. This document is not a design document or a "design cookbook." In no way can it replace specific analysis and design effort. It is a set of guidelines to outline and scope deficiencies and qualities in present day aircraft that may aid in the approach to future designs. The information and data is illustrative and advisory to provide definition and make comparisons of parametric properties and behavior, and it is not for use or adaptation to specific designs. For example, figure 1.3-1 maps the expected voltages induced in a single aircraft wire circuit (having resistive loads) from an adjacent 115V, 400-Hz power wire where both have their returns in aircraft structure. This figure illustrates the significance of length of coupling and resistive loads and gives a rough estimate of amplitudes. But there are other

interacting parameters that might be considered; ergo, each circuit type in the aircraft must be evaluated separately.

Much emphasis on wiring design exists in this document. The soul of EMC is a balanced, isolated, shielded interface circuit. It closes the door on EMI. It rejects transients, radio frequencies, and 400-Hz fields. It is practically impervious to conductive, inductive, and capacitive transfer of energy.

Linearity rises as one of the elegant attributes of electromagnetics, giving simplicity to variations in the electromagnetic dependent and independent parameters: length, height, resistance, voltage, time, over much of their range. These parameters often vary on a 1:1 ratio (20 dB per decade) or an exponential ratio, possibly 40 dB per decade. The linear relationship breaks down or changes at corner frequencies, 3 dB points, and resonant nodes where the dominance of electrical parameters make a transition from one to the other.

Ratios and the decibel relationship will be used throughout. The decibel, abbreviated dB, is a unit expressing the ratio between two amounts of power, P_1 and P_2 , existing at two points. By definition, the number of dB equals $10 \log$ to the base 10 (P_1 is divided by P_2). For special cases where P_2 equals 1 mW or 1W, the dB ratio is defined as "dBm" or "dBW." For power, a factor of 10 equals 10 dB. Since power P equals V^2 divided by R , or I^2 times R , decibels can be used to express voltage and current ratios where the voltages and currents are measured at places having identical impedances. By definition, dB equals $20 \log$ of (V_1 divided by V_2), and dB equals $20 \log$ (I_1 divided by I_2). For convenience, V_2 or I_2 are often chosen as 1 μ V and 1 μ A, and the dB ratio defined as dB above a microvolt or dB above a microamp. Also for convenience, these ratios are more often used whether or not they are referenced to identical impedances. A factor of 10 equals 20 dB. Memorize these voltage-current ratios: 6 dB = 2X, 10 dB = 3X, 12 dB = 4, 20 dB = 10, 40 dB = 100, and 60 dB = 1000.



2. AIRPLANE AVIONICS AND CRITICALITIES

2.1. Existing Systems

The well-known hallmarks of a commercial transport aircraft (figure 2.1-1) are: one-hundred and fifty paying passengers or more, flight attendants, airline competition, scheduled dispatch, fixed cost, aisles, lavatories, galleys, and video entertainment.

The necessary control of capital cost and running expenses, the need for quick and easy equipment maintenance, and the desire for on-time dispatch: these goals urge the airframe manufacturer to focus on a well-designed, well-planned electrical architecture, including electromagnetic compatibility. Standardized avionic Line Replaceable Units (LRU) are motivation for low cost and competition among subcontractors (figure 2.1-2). Cost, airworthiness, and safety are the critical drivers for avionics.

Most passengers are unaware of the safety built into the interface wiring and electrical/electronic systems (figure 2.1-3). Passenger and crew safety, with regard to protection from hazards of high voltages, has been guaranteed historically by the ubiquitous aluminum housings, spars, supports, and structure. The high-quality structural aluminum grounding paths are inherently the electrical return or an electrical reference for digital signals, shield ties, motor power current, and fault currents (figure 2.1-4). This structure bypasses the need for separately installed wires or buses to fulfill those functions (figure 2.1-5). Also enhancing the electrical sinking and conducting properties of structure are the extensive air-conditioning, water, and hydraulic systems that form a skeleton of metallic and composite materials throughout the flight deck, cabin, cargo bay, wheel wells, and wing leading and trailing edges. Many of these shielding and sinking properties today help to contain or divert electromagnetic interference from electronic game signals, electrostatic discharge, lightning transients, and high-energy broadcast radio frequencies (figures 2.1-6 and 2.1-7).

Aluminum alloys form the wing, fuselage, and empennage structure, but these metal alloys must mate with materials like fiberglass, Kevlar, and moderately conductive materials like graphite-epoxy. The interfaces at fairings, doors, ducting, and fasteners open up possibilities of apertures and gaps. These mating surfaces must be electrically bonded so that power currents, fault currents, electrostatic charge, and lightning currents flow (figure 2.1-8).

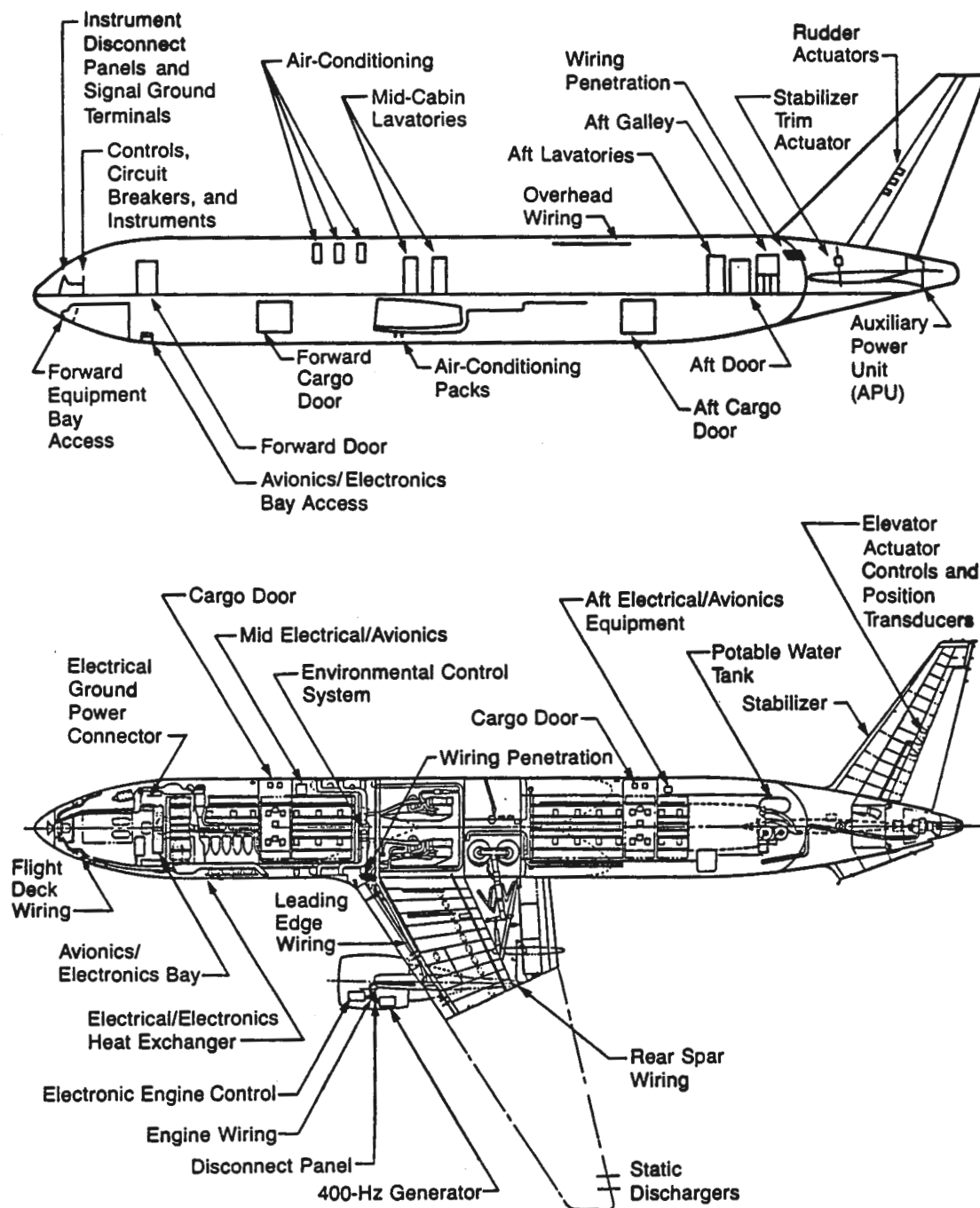


FIGURE 2.1-1. PRESENT-DAY AIRCRAFT SYSTEMS

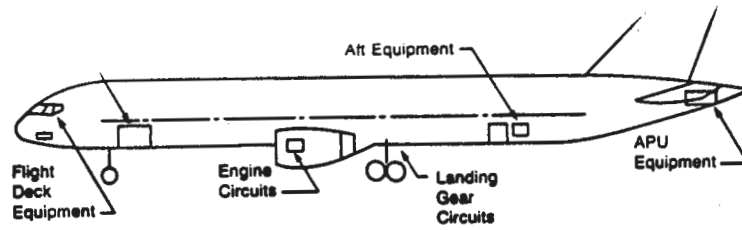


FIGURE 2.1-2. AVIONICS BAYS

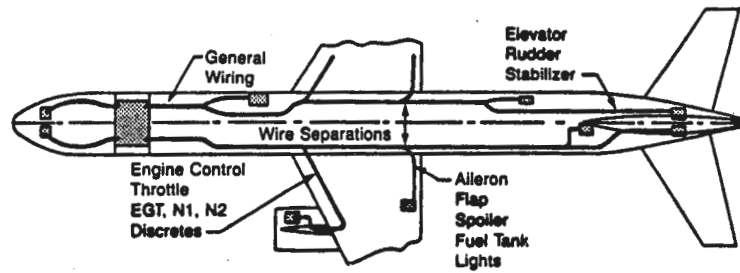


FIGURE 2.1-3. WIRING

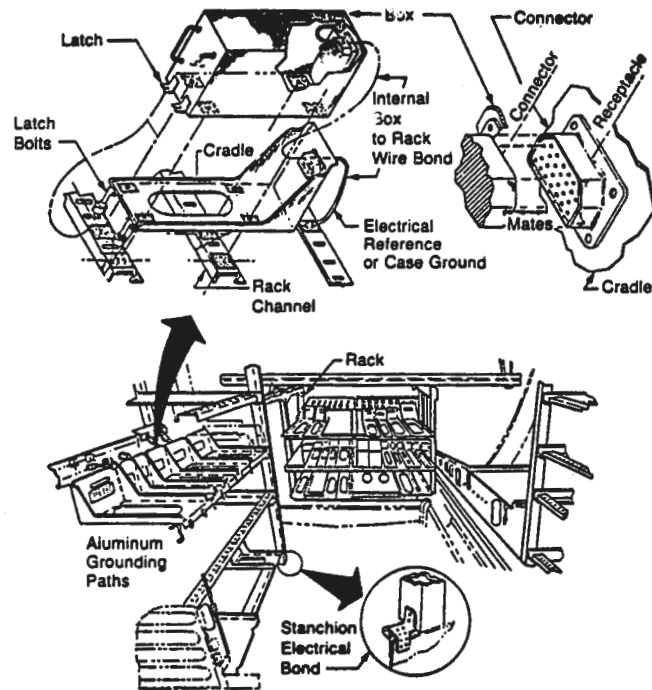


FIGURE 2.1-4. MAIN BAY ALUMINUM STRUCTURE

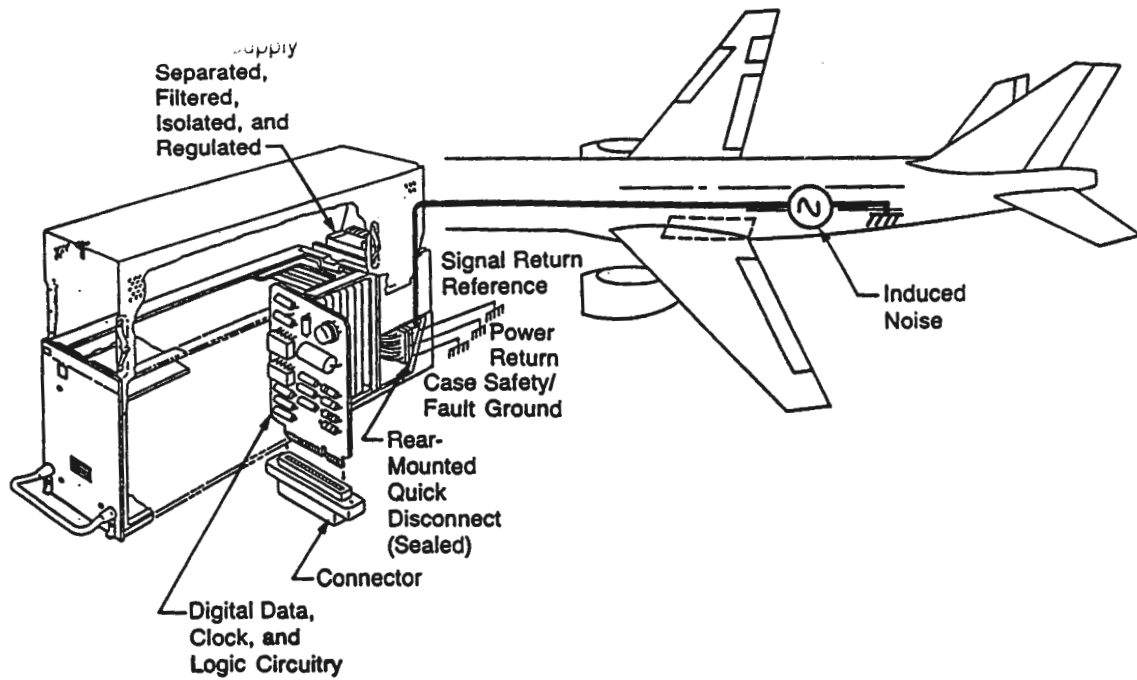


FIGURE 2.1-5. STRUCTURE RETURN

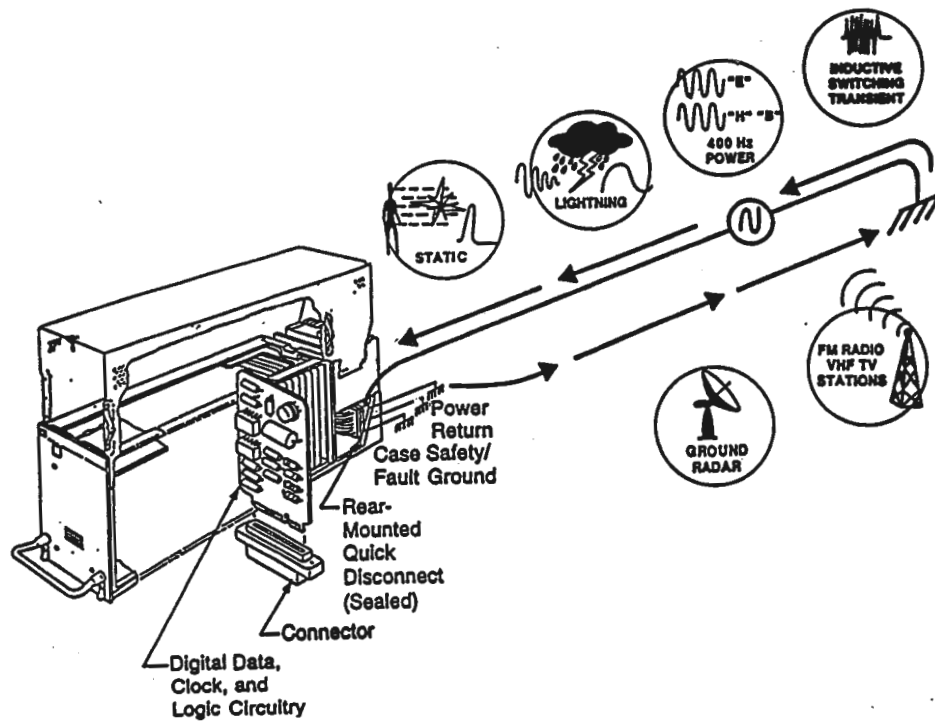


FIGURE 2.1-6. INDUCED NOISE

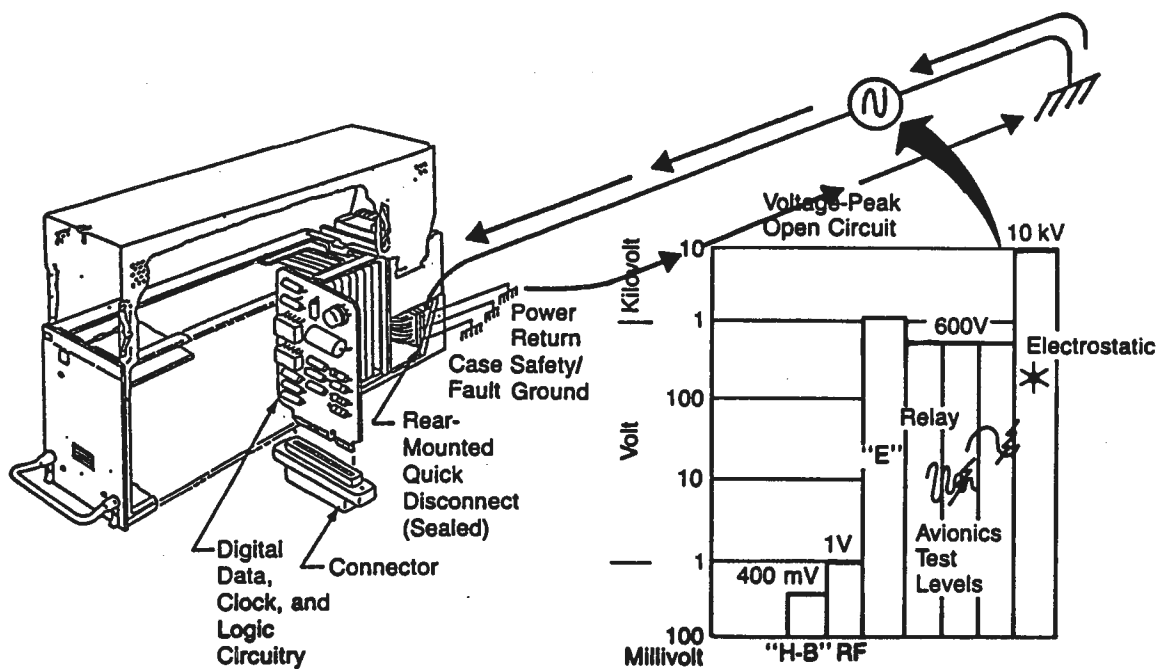


FIGURE 2.1-7. NOISE VOLTAGES

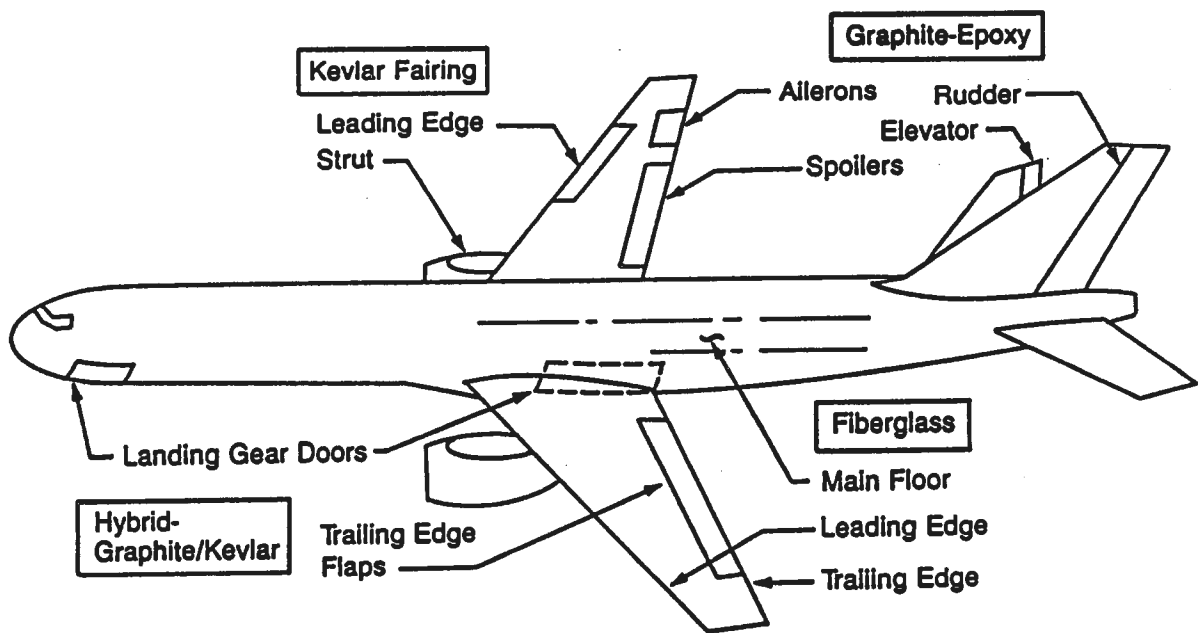


FIGURE 2.1-8. NONMETALLIC INTERFACES

With each new program, we build on the safety and electromagnetic compatibility practices of past experience, and as the program progresses, we purchase and install units of avionic equipment which were designed, built, and tested years ago - existing equipment, "off-the-shelf" equipment. So, years of in-service experience with existing equipment are brought together with the new technology designs on the new program. We cannot overlook the fact that electromagnetic compatibility and safety standards have been well established. Hence, this document does not hold any revealing secrets or creative innovations. Many of the requirements for electromagnetic compatibility are very well known having been spawned, tried, and steadily refined over the years.

With an experienced eye on airworthiness and reliability, traditional electrical and electronic functions have been carefully shaped by the airframe manufacturer and subcontractors to provide and enhance engine instrumentation (safety), communication and navigational aids (safety and convenience), autopilot equipment (pilot workload), and aircraft utilities (safety and passenger comfort). Redundancy of avionics functions and components has become a de facto standard. Avionics equipment in today's aircraft (1986) strongly contribute to the desired levels of performance and safe flight through separation and duality (figure 2.1-9). Any one single part or unit in a fully operational and functioning aircraft is not vital and crucial to continued flight. Triple redundant flight control computers and data buses, for instance, practically guarantee continuous operation and provide a confident reliance on the automatic pilot system (figure 2.1-10). Electromagnetic compatibility is important and must be "designed in" to be cost effective or the avionic equipment will not work, but electromagnetic compatibility has not been critical to aircraft safety.

Engine monitors or sensors are becoming critical, especially on the new electronic engine controls (figure 2.1-11). Left engine circuits are separated from the right for safety. Selected circuits on each engine are separated from each other. Oil pressure transducers, engine temperature thermocouples, exhaust gas temperatures (EGT), and speed sensors (N1, N2) provide information on performance and operational boundaries and status. Knowledge of the status of hydraulics, oil, and fuel systems helps avert critical situations. Throttle lever angle position must be known, and of course, fire detection is mandatory. Electromagnetic interference must be designed out of these circuits early in any program.

Communication receivers and transmitters are the key in today's flight schedules and goals of smooth flight and fuel economy (figure 2.1-12). Noise must be at a minimum to keep from degrading receiver thresholds. Broadband noise will reduce communication range. Narrow band noise will induce unwanted tones. The flight crew does not want to hear static, 400-Hz hum, or popping, even though this electromagnetic compatibility problem may only be a nuisance.

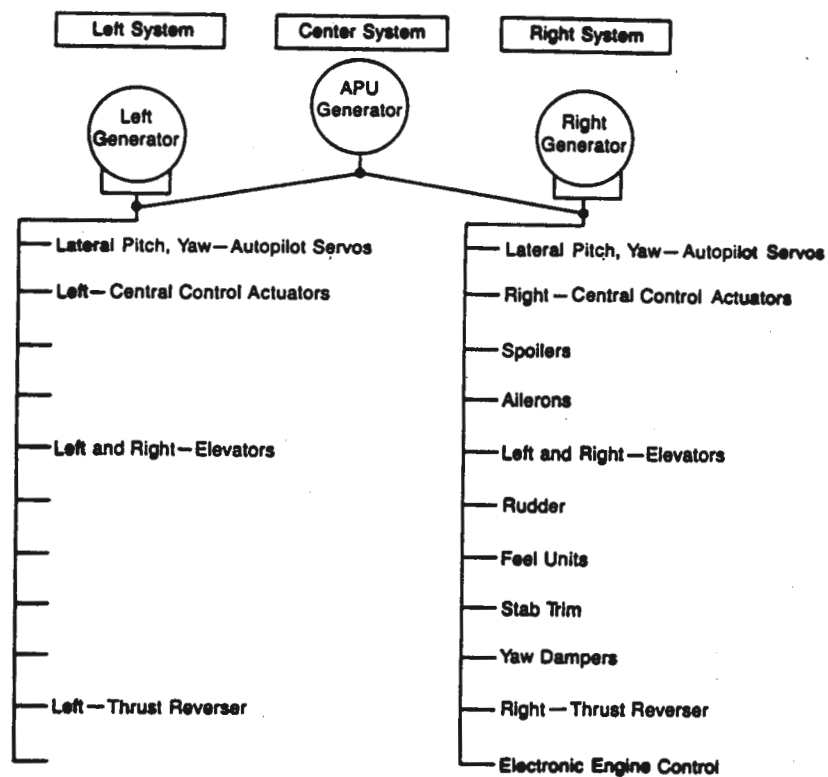


FIGURE 2.1-9. SEPARATION EXAMPLE

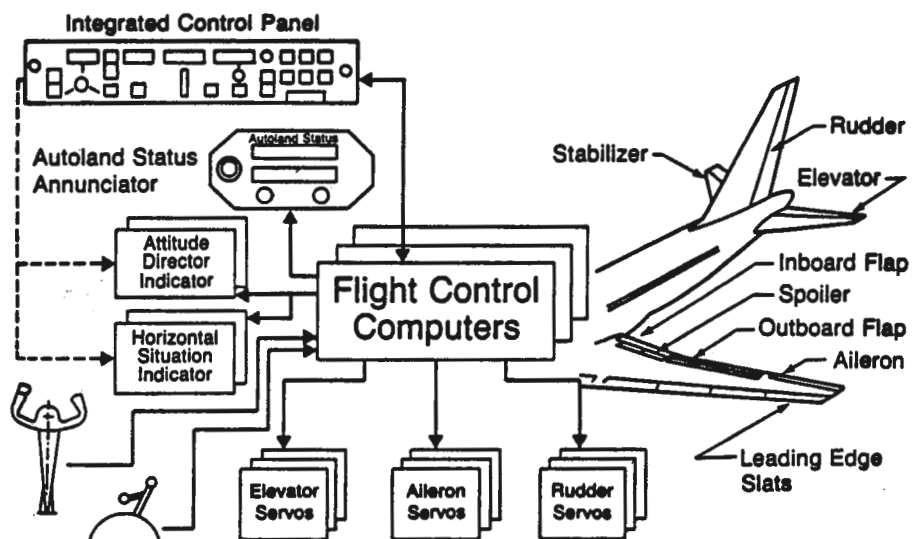


FIGURE 2.1-10. FLIGHT CONTROL AUTOPILOT REDUNDANCY

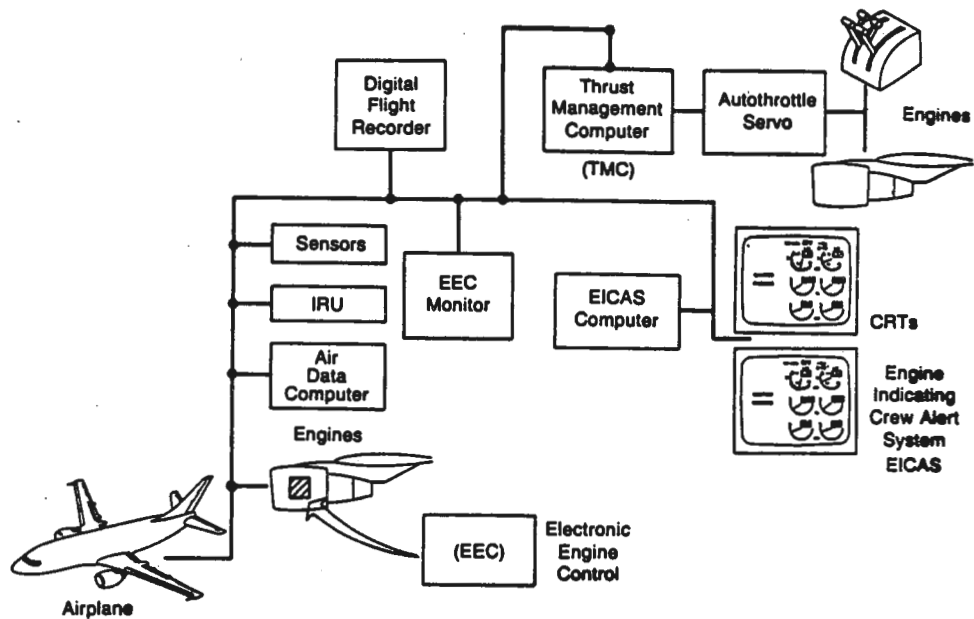


FIGURE 2.1-11. ELECTRONIC ENGINE CONTROL

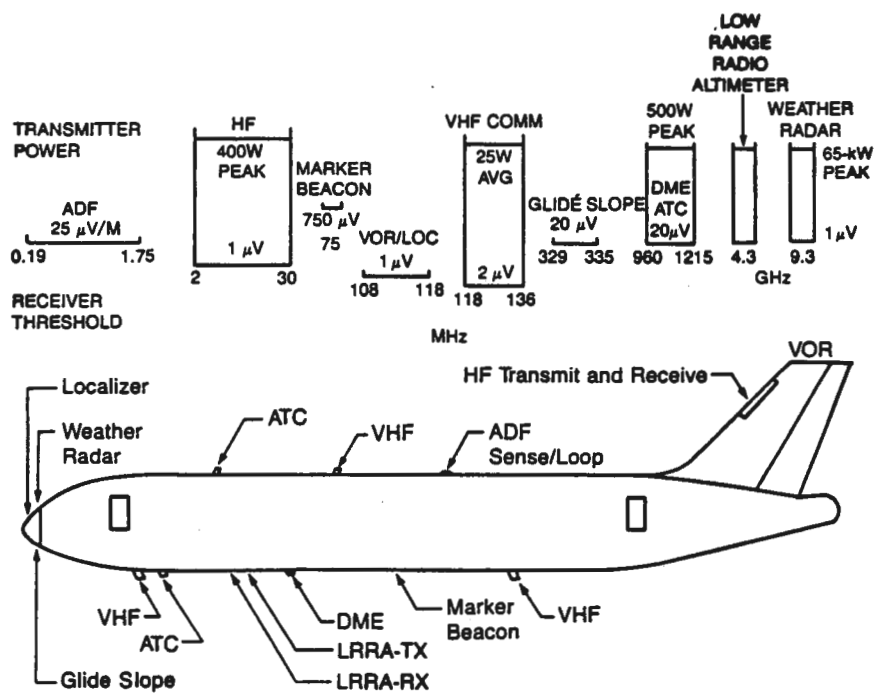


FIGURE 2.1-12. TRANSMITTERS-RECEIVERS

Automatic controls reduce pilot workload and assist in long flights and optimum flight profiles (figure 2.1-10). We still see the old-fashioned compass installed in the instrument panel on the flight deck, but the "horizontal situation indicator" is now a Cathode Ray Tube (CRT) (which, by the way, is very susceptible to 400-Hz magnetic fields) and is designated as the electronic horizontal situation indicator (EHSI). There are two EHSIs to rely on besides the old-fashioned backup compass (figure 2.1-13). Future aircraft will see flat panel displays that must be protected from radio frequencies and transients that induce "snow," "banding," "lines," or "ripple" on the screen.

Today's professional pilot "flies" the plane with the steering column having steel cable strung from the column to hydraulic actuators which then amplify the force and operate control surfaces (figure 2.1-14). The engine throttle is operated by a steel cable. A steel cable does not recognize electromagnetic interference.

Navigation equipment with ground systems that locate and track have grown more and more sophisticated, accurate, and reliable (figure 2.1-15). Specifications and requirements have been developed over the years (table 2.1-1). VHF and HF communications are used over land and sea, all along the airways, and in high-volume areas (figure 2.1-16). Communication avionic units are stand-alone components that can be individually replaced in the electronic equipment bay or on the flight deck (figure 2.1-16). Because of the excellent flexibility of unit replacement, ease of maintenance and substantially lower unit costs, there has been little or no historical incentive for large-scale integration of functions on present day airplanes (figure 2.1-17). Most equipment today is basically digital, but some is still analog (figure 2.1-18). Future aircraft will probably see almost total use of digital circuitry.

Aircraft utilities, of course, are basic to commercial transport operation. Fluorescent lights in the cabin are noisy. Pressurization controls, cooling fans, and window heaters contribute to aircraft magnetic fields. Cargo handling and windshield wiper motors, fans, and galley heaters inflict broadband noise on digital and analog circuits. Future systems may have pulse width modulated voltages that will contribute to noise.

Today's interface circuits span the systems from engine instrumentation to navigation to flight control to utilities, but electromagnetic compatibility engineers do not deal with flight systems. They deal with circuit types, and basically those types can be counted on two hands. The types can be boiled down to digital data, radio frequency signals, analog signals, discrete state changes, and 400-Hz 115V or 28V-dc power and in the future pulsed dc power.

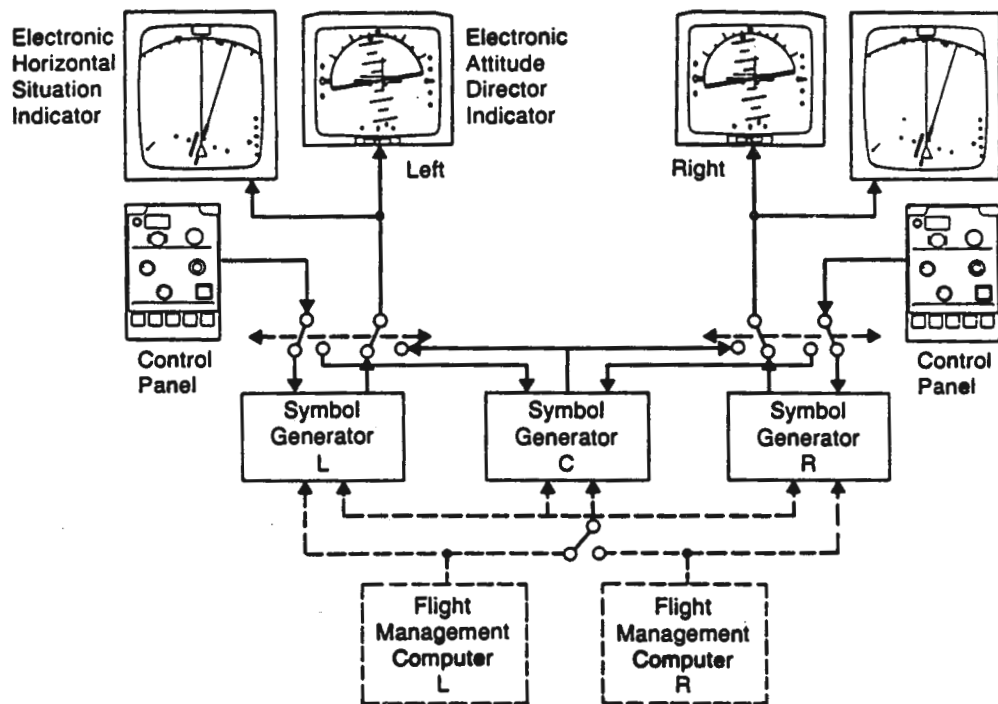


FIGURE 2.1-13. FLIGHT INSTRUMENT SYSTEM

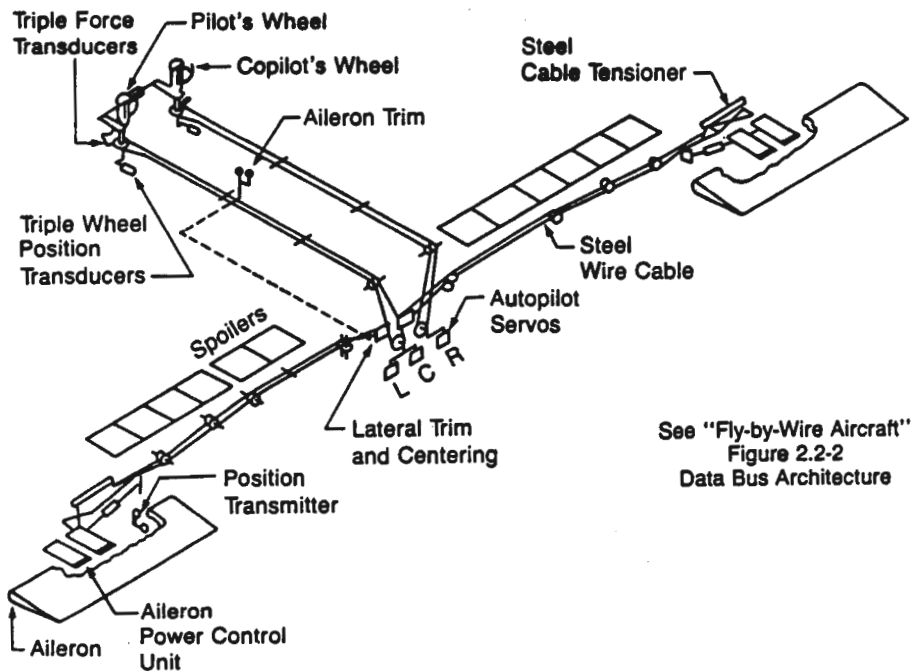


FIGURE 2.1-14. ROLL CONTROL SYSTEM

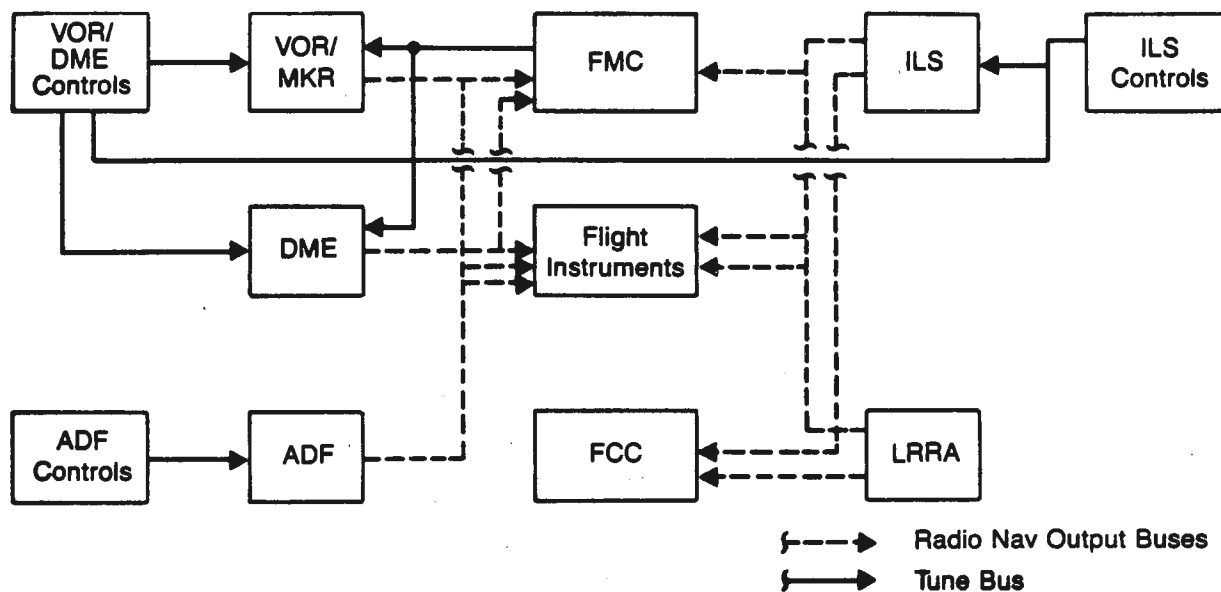


FIGURE 2.1-15. NAVIGATION

TABLE 2.1-1. NAVIGATION EQUIPMENT DOCUMENTATION

Navigation Equipment	Federal Aviation Regulation (FAR)	Technical Standard Order (TSO)	Advisory Circular	Number Required
VOR	121.349a,e	C40a	90-45A	2
DME	121.349c	C66a	90-45A	1
LOC/GS	121.349a	C34b	120-28A	1
		C36b	120-29	—
MB	121.349a	C35c	—	1
ADF	121.349b	C41b	20-63	1
INS/ISS	121.355	—	25-4	2
	121, App. G	—	121-13	—
RNAV	—	—	90-45A	—
Omega	—	—	120-31	—

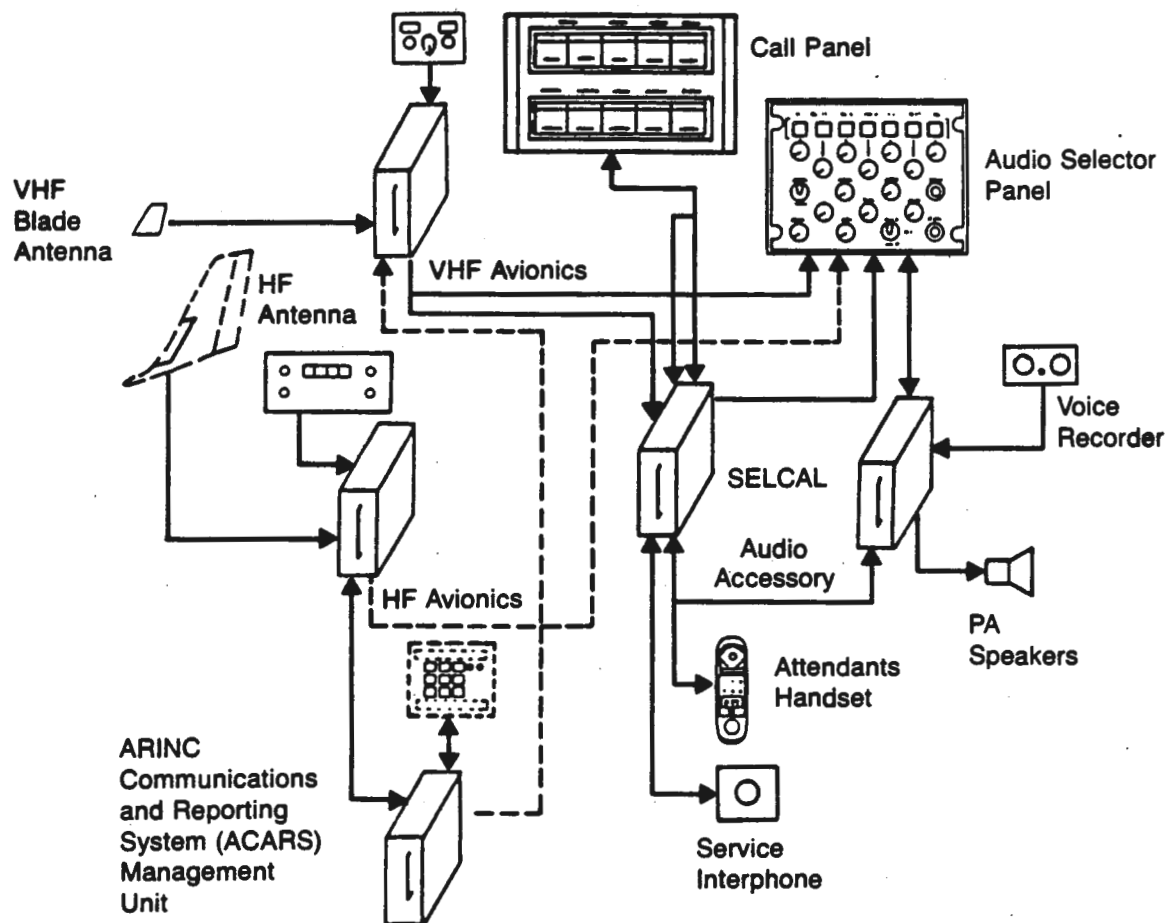


FIGURE 2.1-16. COMMUNICATION

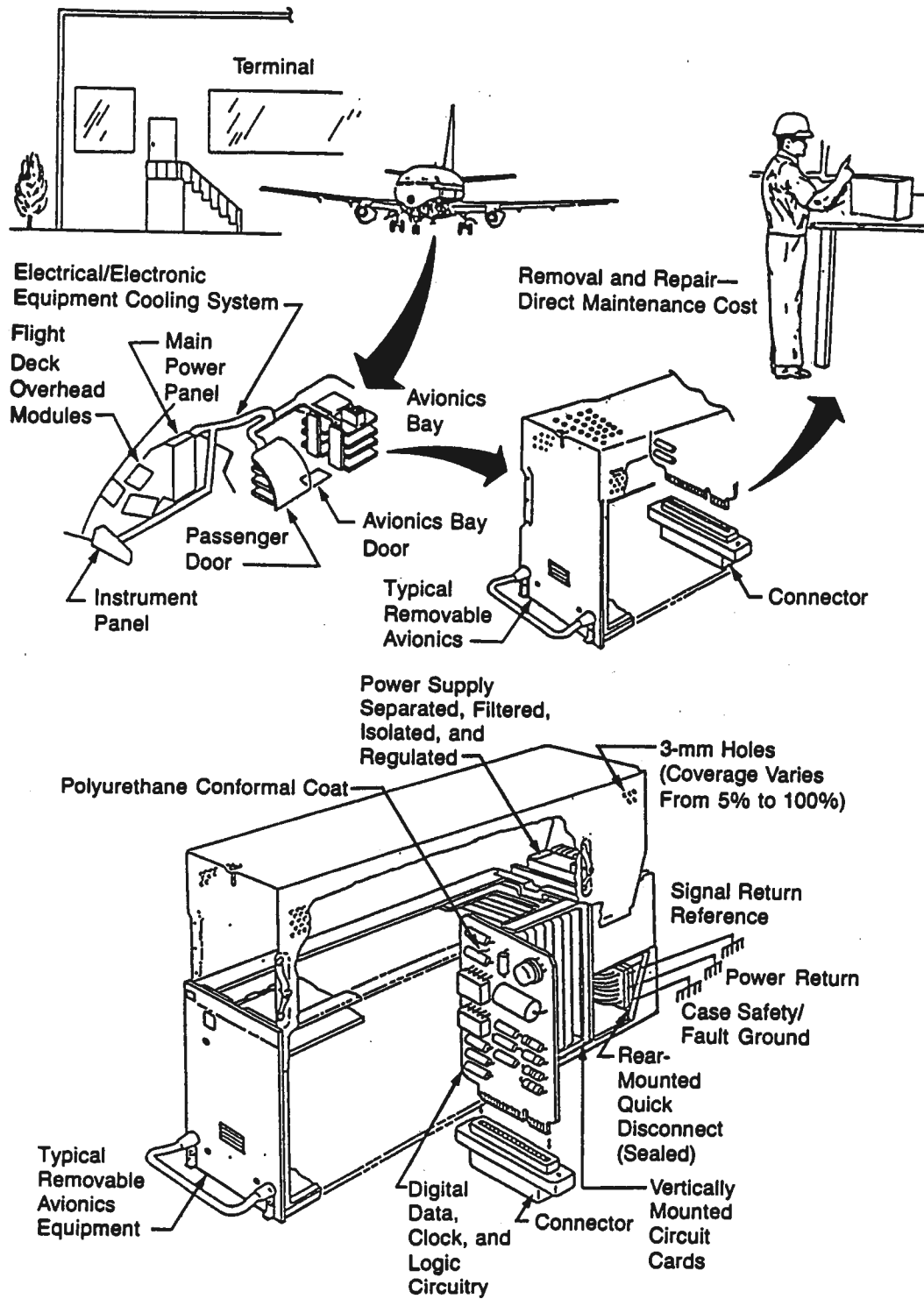


FIGURE 2.1-17. LINE REPLACEABLE UNIT

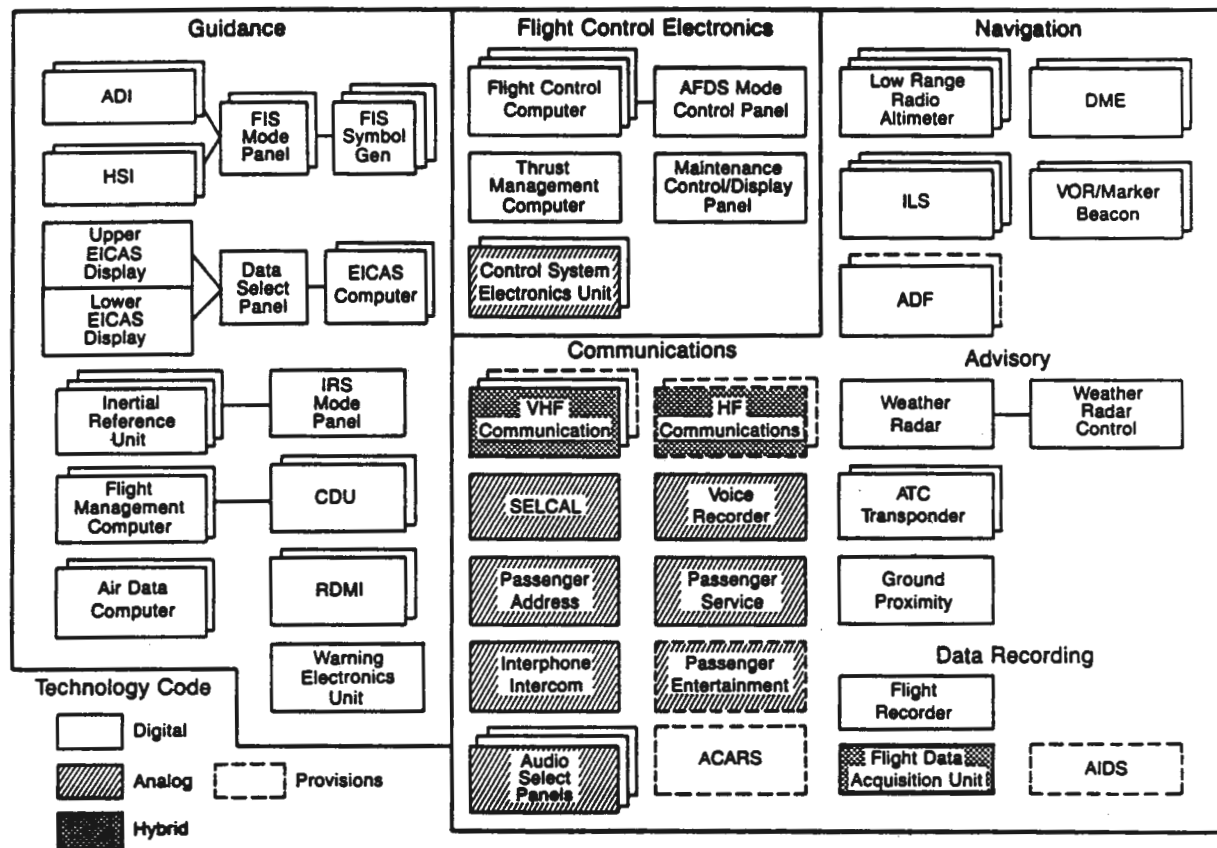


FIGURE 2.1-18. FLIGHT MANAGEMENT SYSTEMS

If one were to look at the specific types of interconnecting circuits, here are some examples of the electrical/electronic characteristics and parameters:

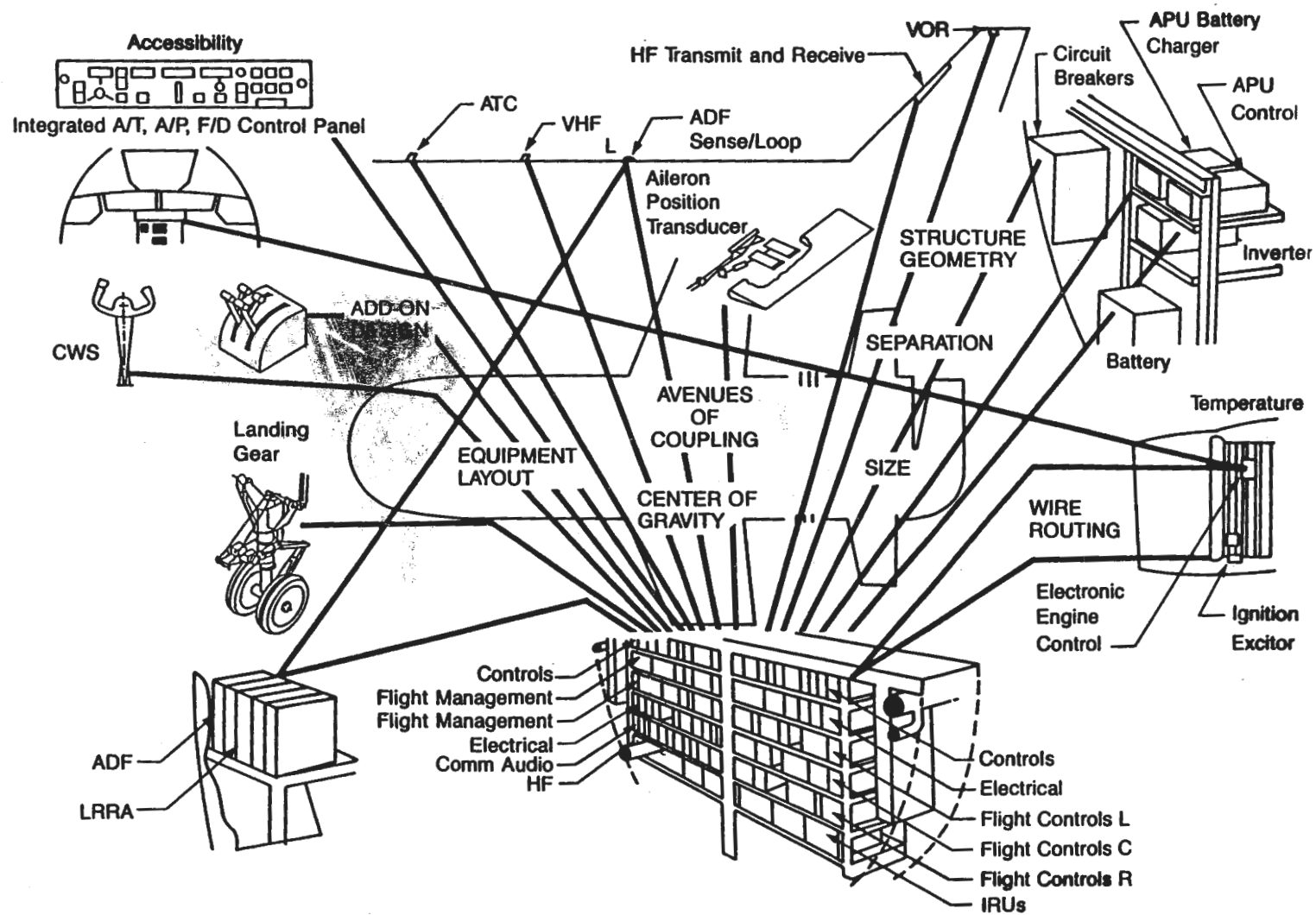
EXAMPLES

CIRCUIT	LEVEL	RATE/FREQUENCY	THRESHOLD
(1) Digital data:	+5V	12 kHz	2.5V
(ARINC 429 transmitter/receivers: information transfer, altitude, airspeed, direction, positions)			
(2) Pulse circuits:	+50V pulse	1 pps	mV
(Fuel flow; engine speed)			
(3) "Discrete"	+30V	Event ON/OFF	10V
(Status: Pumps, valves, relays, heaters)			
(4) Analog	0V to 10V	Continuous	mV
(Position indicators)			
(5) Power	28V dc	(sw. reg. 10 to 100 kHz)	
	115V ac	400 Hz and harmonics	
(Avionics, pumps, lights, motors, generator feeders)			
(6) RF	Volts	Variable	μ V

The electromagnetic compatibility engineer helps to simplify and forge the layout of the equipment and routing of the wires (figure 2.1-19). He pays particular attention to radiated emission from wiring and radio frequency fields imposed on wiring (figure 2.1-20). Equipment location has been settled usually in the past by greater attention to and emphasis on temperature characteristics, size, separation, accessibility, and center of gravity factors rather than electromagnetic compatibility. And, rightly so. Wiring, wire clamps, shield ties to structure, and routing of wire-bundle installations have been shaped by the space available and structure geometry. They have been installed as an "add-on" while all the time being given the appellation of weight penalty. Conduit or metallic ground planes have not been needed. Shielding could be added any time.

We and our predecessors seldom created and documented substantive and comprehensive input/output interface circuit drawings to provide an analysis on a systematic basis of the noise sources and the major avenues of coupling. Electromagnetic interference sometimes has been most elusive and difficult to analyze, model, and predict.

FIGURE 2.1-19. EQUIPMENT/WIRING LOCATION COMPLEXITIES



Avionic equipment units have been designed and tested to industry electromagnetic compatibility specifications and then, along with their interconnecting wiring, installed directly in the aircraft. They almost always perform properly. This method has been cost-effective and not at odds with safety. Fixes to equipment or to the airplane can sometimes be made fairly easily and inexpensively, although they are often done in a hurry and with the ever-present specter of the very expensive "retrofit."

As we look back, it is recognized that the electromagnetic compatibility effort has been reactionary, not anticipatory.

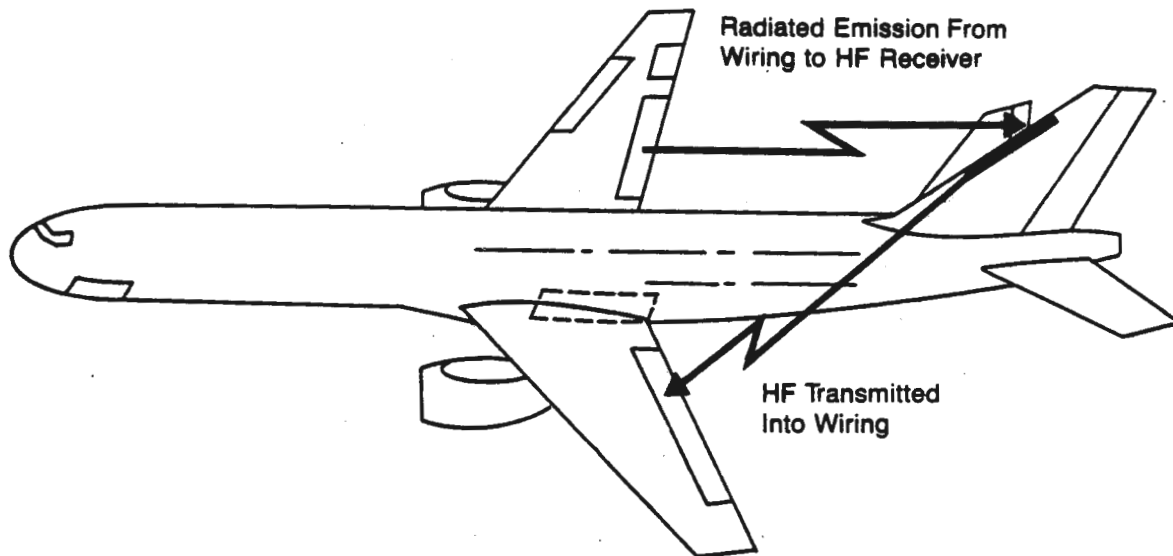


FIGURE 2.1-20. RADIATED EMISSION

2.2. Future Aircraft

Turn now and look ahead to a hypothetical future aircraft.

A view of a system avionic architecture might focus on the technologies of digital buses, electric actuators, and composite structural materials as having the greatest interest for the electromagnetic compatibility engineer or airworthiness specialist (figure 2.2-1). A system of communication and control designed around fly-by-wire digital data buses and electric actuators will have an impact on analysis tasks; that is, greater effort will be needed to assure absence of electromagnetic interference in the data bus and control of electromagnetic interference from the actuators and pulsed dc.

Connection of autonomous electronics through a multiple access two-way data bus will remove dependence on a centralized controlling processor (figure 2.2-2).

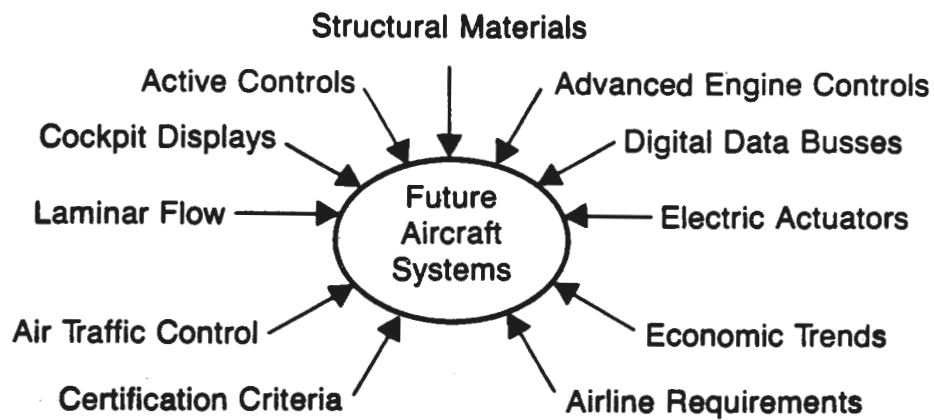


FIGURE 2.2-1. EMC AND AIRCRAFT SYSTEMS DEVELOPMENT

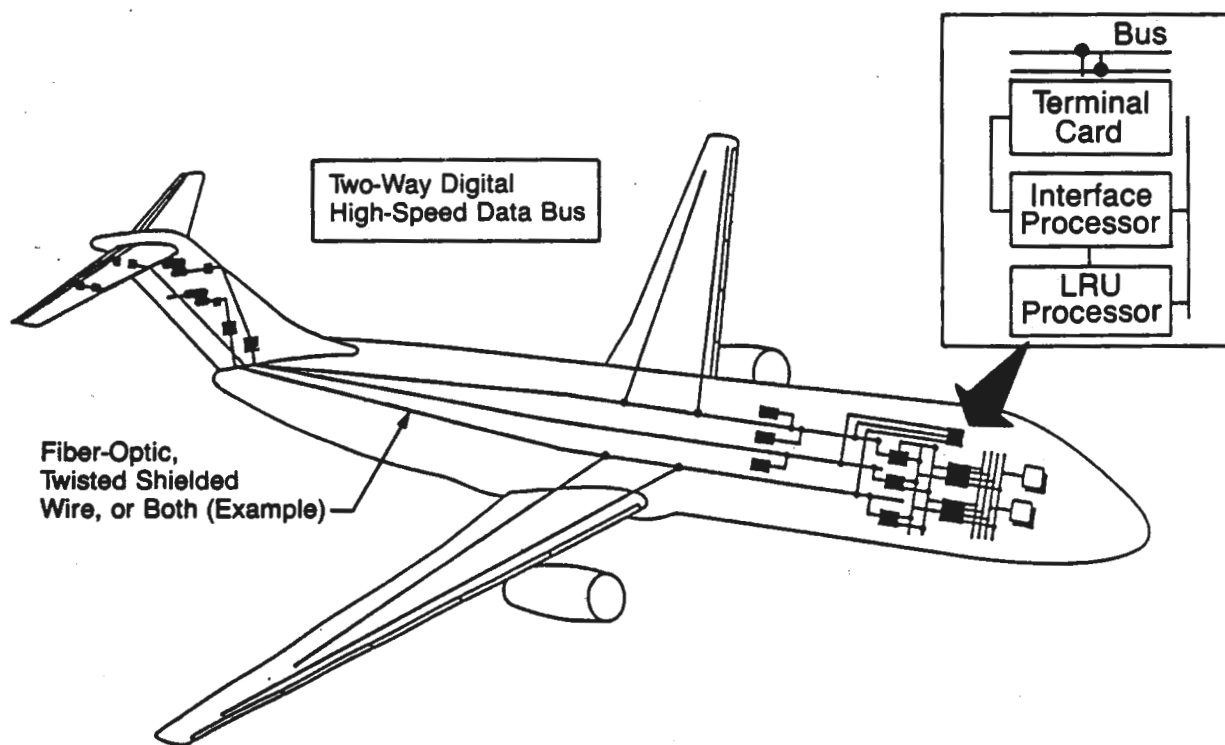


FIGURE 2.2-2. DATA BUS ARCHITECTURE

There will be a number of new aircraft technologies that will moderate problems but will mean new areas of expertise and study:

- All-electric airplane (electric actuators), pulsed dc.
- Autonomous, multiple-access data bus with decentralized computers (dual redundant).
- Self-diagnostics, self-test, error control, and record keeping.
- Electronic keyboard actuation of power and avionics.
- Side-stick controllers.
- Single point-of-entry program loading.
- Flat-panel and head-up displays.
- Voice control.

What will be the impact of these systems on electromagnetic compatibility? More inductive switching transient control? Electrostatic transient control? Better high-energy radio frequency fields rejection and control?

Flat panel displays collocated with their own microprocessor avionics on the flight deck, depending on design, will mean less wiring and fewer coupling paths, less susceptibility, less electromagnetic radiation and electromagnetic interference. An improved flight management system will integrate navigation, communication, guidance, and energy management. Moreover, there will be expanded management capability of all of the systems (figure 2.2-3).

Multi-function keyboards will permit speed, flexibility, and maximum accessibility of options and data selection initiated by voice (figure 2.2-4). Side-stick controllers, electronic throttle, and flap control will not only facilitate operation and open up the instrument panel viewing area, but will also demand guarantees of safe operation and freedom from noise. Head-up displays and electronically-controlled relays and actuators will help clear and simplify instrumentation and add more power and versatility to aircraft control on the flight deck.

One of the key elements in a future aircraft is the data bus, where wire and weight savings will be dramatic. The number of interface circuits will decrease (figure 2.2-5). Research on digital buses will be required to better define susceptibilities to high-energy radio frequencies and transients. It is expected that some form of multiple access bus will be available for future aircraft (figure 2.2-6). Commercial transport planes are now using the ARINC 429 bus, a 12-kHz or 100-kHz, unidirectional, twisted pair, shielded bus that operates from each transmitter to multiple receivers with a binary-coded decimal

format. The military bus is MIL-STD-1553: a 1-MHz, bidirectional, twisted pair, shielded, transformer-isolated bus totally run from a centralized 1553 bus controller with a serial, Manchester II, bi-phase format. (See section 10, Bibliography, ARINC 429 and MIL-STD-1553 bus system.) New systems are being developed to provide multiple access and decentralized processing. One such is called DATAC, a 1-MHz, bidirectional, twisted pair or fiber-optic bus with controlled specifications of protocol to facilitate transmission and reception.

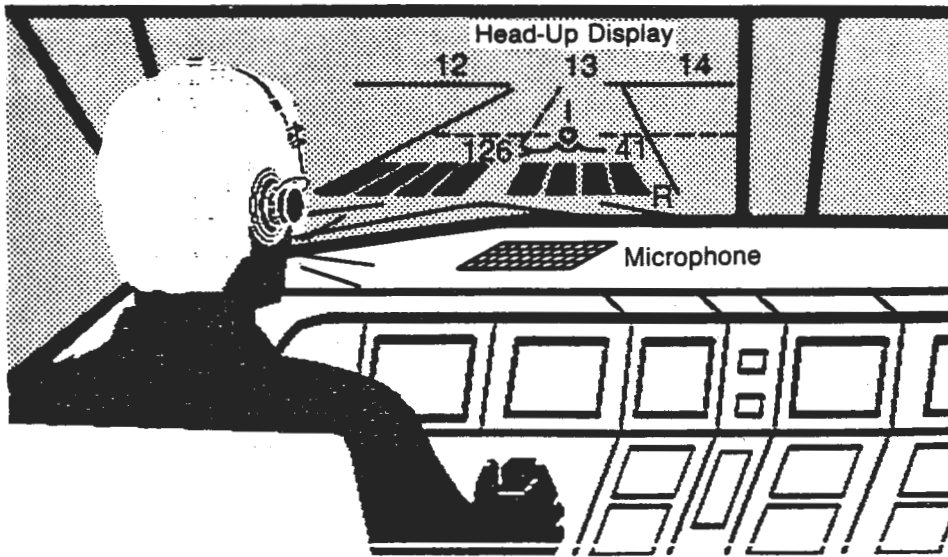


FIGURE 2.2-3. LINE-OF-SIGHT DISPLAY AND VOICE CONTROL

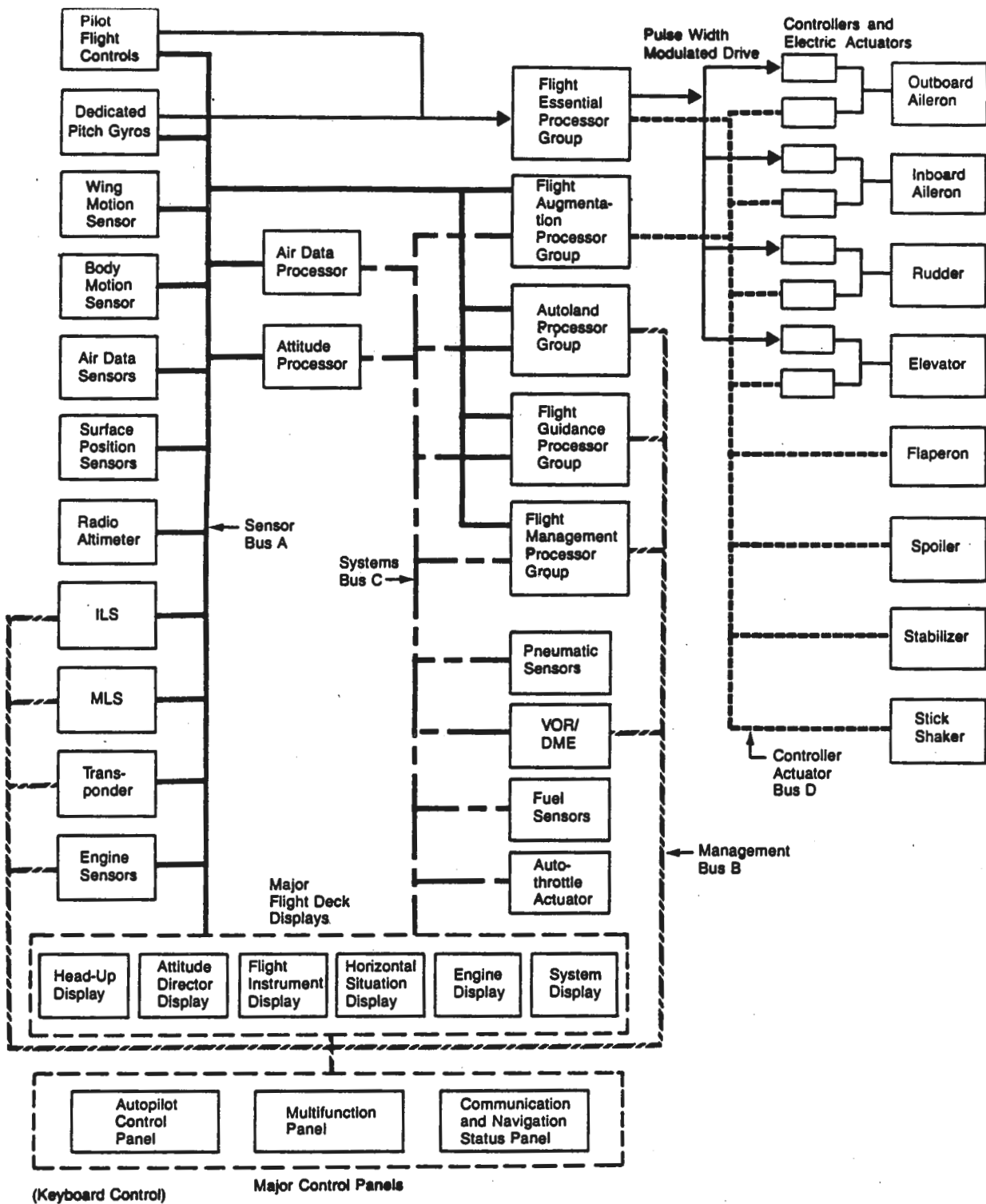


FIGURE 2.2-4. HYPOTHETICAL ARCHITECTURE

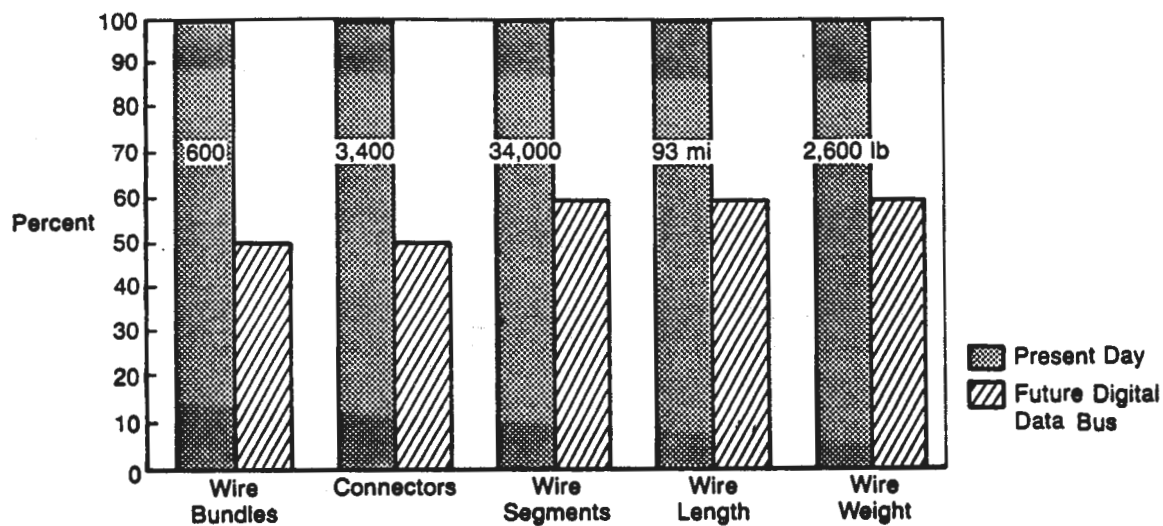


FIGURE 2.2-5. DATA BUS COMPARISON

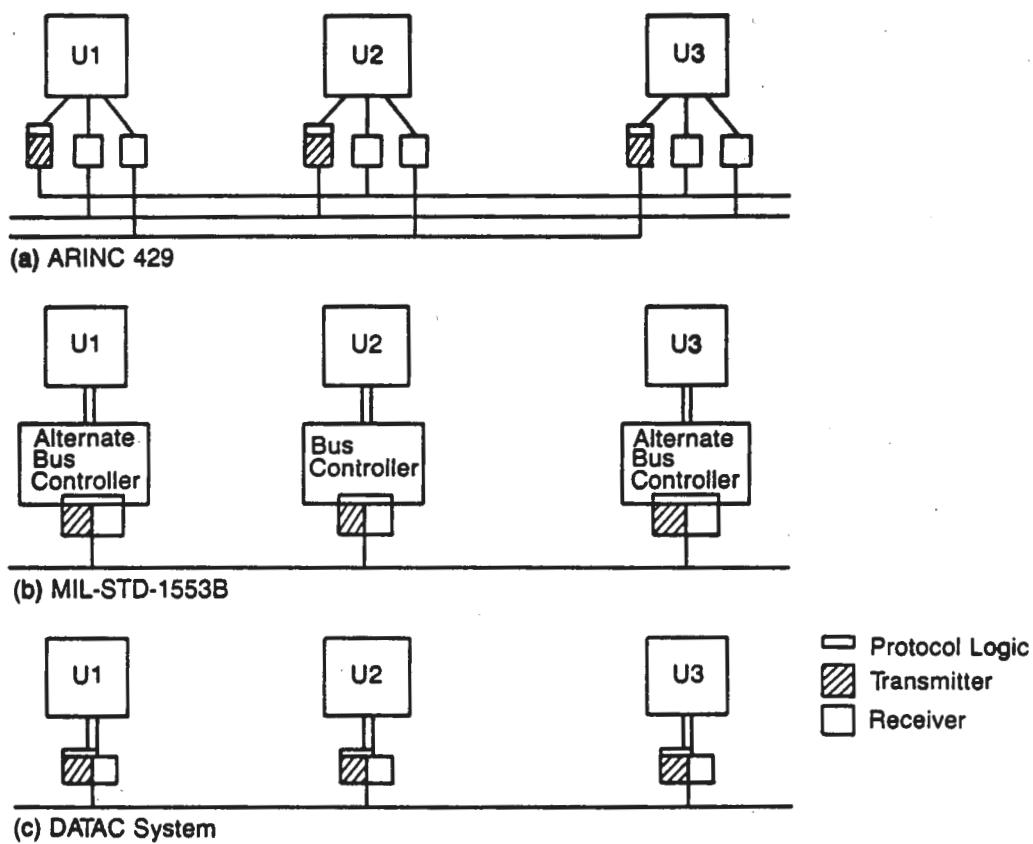


FIGURE 2.2-6. DIGITAL DATA BUSES

Possibly one of the most revolutionary changes to come about for electromagnetic compatibility will be the increased introduction of self-test and self-diagnostic capabilities in avionics allowing real-time readout or a history of equipment susceptibilities to noise (figure 2.2-7). Microprocessor evolution is speeding the development of these systems, and dramatic changes will occur in the immediate future (figure 2.2-8 and figure 2.2-9).

Hydraulic actuators powering control surfaces and other aircraft items such as landing gears and doors may be replaced by all-electric actuators now being developed. The actuator controllers contain "pulse width modulated" signals driving stepper motors. They are relatively high-voltage devices, are noisy, and the pulses need to be contained.

New composite and metallic alloy structural materials will mean greater efforts in the evaluation of the cornerstones of electromagnetic compatibility - the grounding and bonding designs that maintain current paths and stable electrical references (figure 2.2-10).

Many existing systems are being advanced and improved (table 2.2-1 and figure 2.2-11).

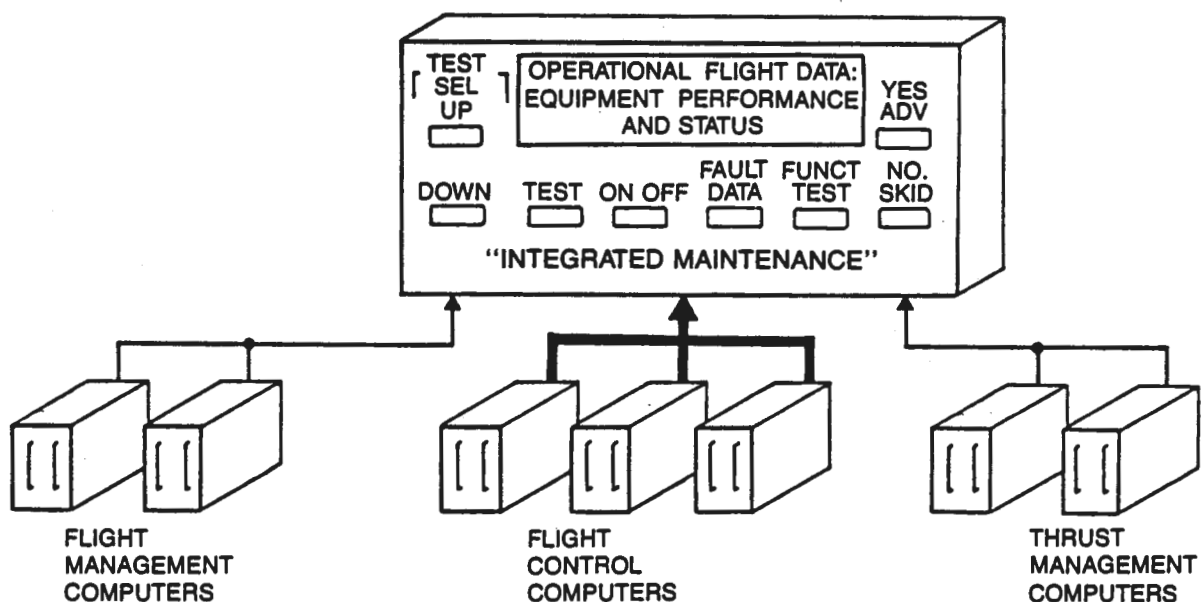


FIGURE 2.2-7. PERFORMANCE AND STATUS MONITORS

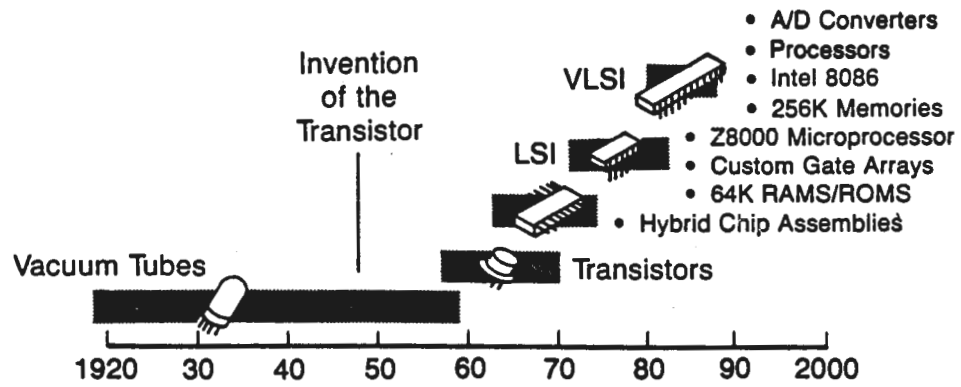


FIGURE 2.2-8. MICROPROCESSOR EVOLUTION

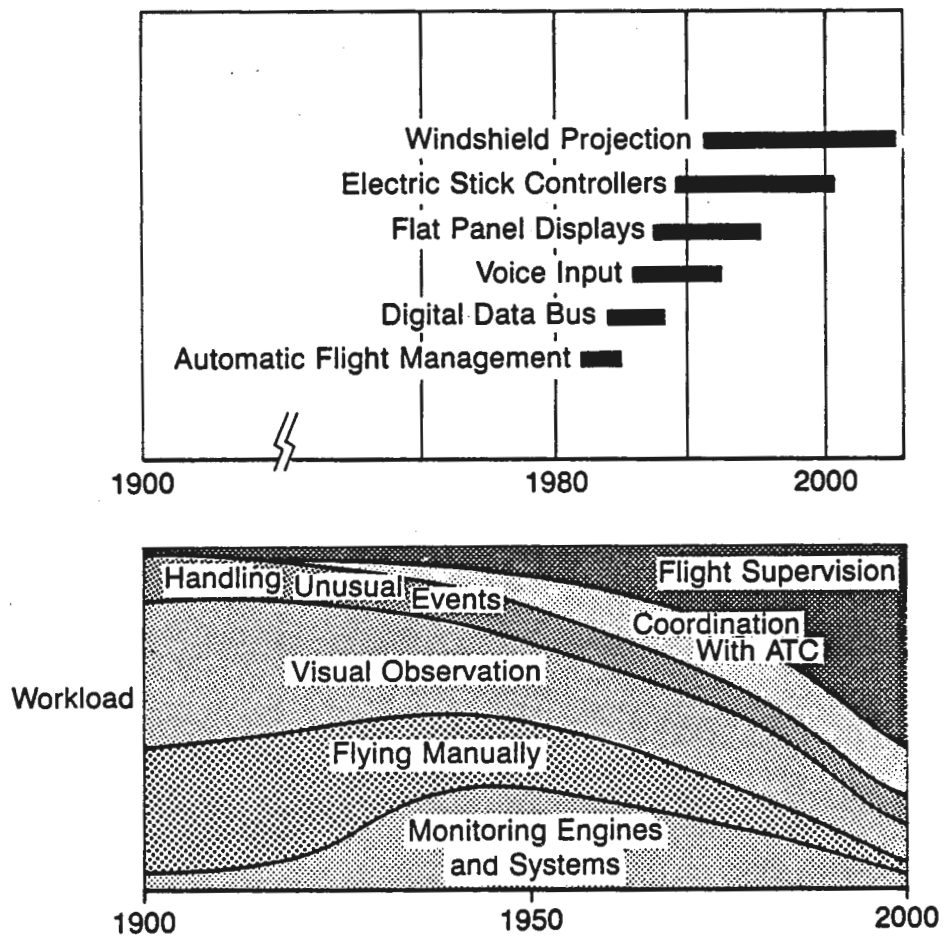


FIGURE 2.2-9. AVIONICS TECHNOLOGY PROGRESS

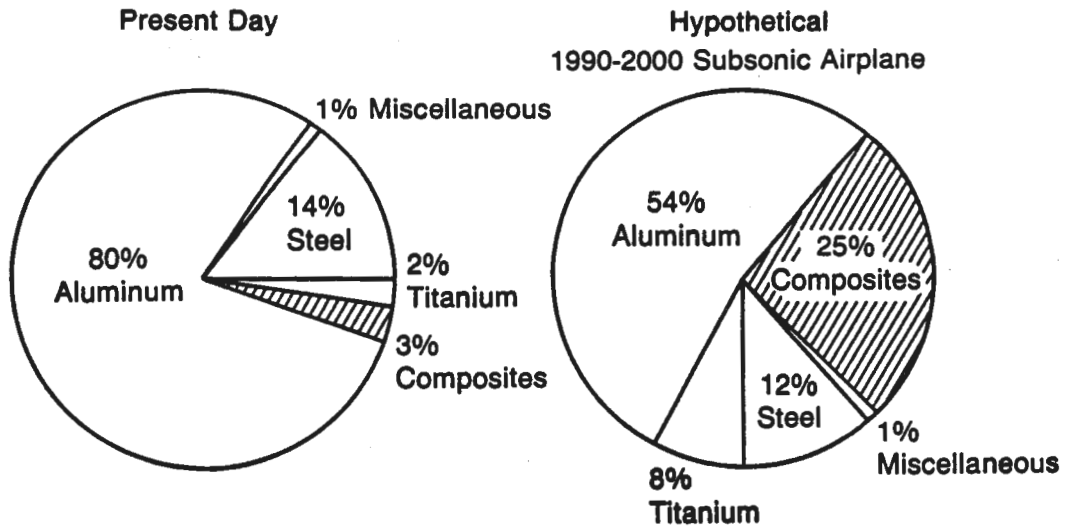


FIGURE 2.2-10. MATERIALS DISTRIBUTION

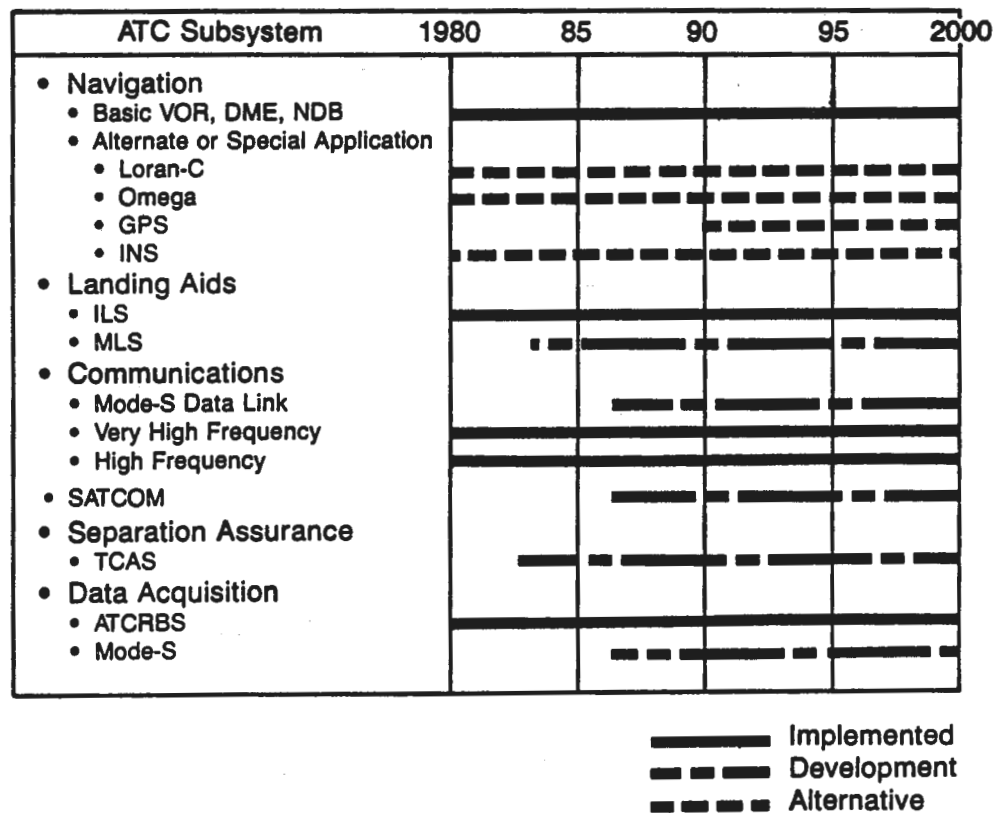


FIGURE 2.2-11. 1990'S ATC IMPLEMENTATION

TABLE 2.2-1. 1990's ATC IMPLEMENTATION

SYSTEM	APPLICATION	AVIONICS
• DATA ACQUISITION • ATRBS WITH MODE-S	• Mode-S-Equipped Transponder Required on Air Carrier Aircraft	• Mode-S Transponder
• SEPARATION ASSURANCE • TCAS	• Required for All Air Carriers	• Interrogator, Controls, and Displays
• COMMUNICATIONS • VHF VOICE • MODE-S D/L • HF SSB • SATCOM	• Most U.S. Domestic and Foreign ATC Operations • High-Density U.S. Airspace • Overocean and Lesser Developed Overland Air Routes • Overocean Voice and Data	• VHF Transceiver • Mode-S Data Link Modem and I/O Devices • HF SSB Transceiver • Data Unit and RF Unit
• NAVIGATION AIDS • VOR • DME • NDB • INS • OMEGA • GPS	• Required for Short-Range Navigation • Required for Short-Range Navigation • Needed for Navigation and Approach Guidance in Some Areas • Used for Long-Range Navigation Independently or With Other Systems (e.g., for Position Fixing) • Used for Long-Range Navigation Independently or To Position-Fix INS in Either VLF or Omega Modes • May Find use for Either Short- or Long-Range Navigation or To Position-Fix INS	• Receiver • Interrogator • Automatic Direction Finder One or More Types Needed for Long-Range Navigation; INS Installation Must Be at Least a Dual System
• LANDING AIDS • ILS • MLS	• Required Until 1995 or Until All Destination Runways Have MLS • Required After 1995 but Needed Before To Obtain Improved Landing Guidance Available at Runways Where Implemented	• ILS Localizer, Glideslope, and Marker Beacon Receivers • MLS Receiver

The air traffic control radar beacon system (ATCRBS) operation (based on a ground interrogation of an airplane transponder to assess flight identity, altitude, range, and azimuth) is being upgraded with a new beacon mode, called Mode-S, which is a two-way digital link, to selectively address each aircraft. Surveillance en route and at terminals during approach starts at about 6000 ft (1830m) above ground level.

The airport surface detection system (ASDS), a primary radar system, is being implemented to transmit a pictorial presentation of the terminal surface area in order to expedite traffic.

The Instrument Landing System (ILS) consists of a 108- to 112-MHz localizer for horizontal, a 328- to 335-MHz glide slope for vertical, and two 75-MHz marker beacons for distance to provide position starting at 33 km from the runway (18 nmi). The system is used extensively and will be augmented with a new Microwave Landing System (MLS) to provide landing in zero visibility utilizing a ground-transmitted signal to the aircraft to establish elevation, azimuth, and distance and is now collocated with the MLS until replacing it.

Navigation systems will expand in extent and capability. The Traffic Alert and Collision Avoidance System (TCAS) requires a transponder trace, and interrogator to detect, track, and compute aircraft flight path projections and initiate selected levels of warning and avoidance maneuver advisories.

The Very High-Frequency Omnidirectional Range (VOR) establishes a magnetic bearing, is a short-range aid, and is generally collocated with a Distance Measuring Equipment (DME) station, which provides for more precise distance measurement from the VOR.

The Automatic Direction Finder (ADF) uses nondirectional beacons (200 to 415 kHz) to provide a bearing (± 3 deg) at a range of 18 to 650 km (10 to 350 nmi).

Loran-C covers about 1850 km (1000 nmi) with pulses at 100 Hz with a positioning accuracy of 0.46 km (0.25 nmi), repeatable to 18m to 90m (60 to 300 ft) and is operated by the Coast Guard along the contiguous U.S. and Alaska, but is presently not being installed by scheduled air carriers.

Many of these systems will be supported or replaced by the Global Positioning System (GPS) which in future years will provide worldwide, accurate, all-weather positioning.

Communication systems will see unparalleled improvement in reliability and flexibility.

Very High-Frequency (VHF) voice communication, the continental U.S. air traffic control system to airport towers and control centers, operates between 116 and 136 MHz with projected bandwidth spacing of 25 kHz at a receiver sensitivity of around a microvolt.

High-frequency (HF) voice communication, for use over water, operates at 2 to 30 MHz with a receiver threshold of about $2\mu\text{V}$ and transmitter output of possibly 400W peak effective power.

2.3. Criticalities

2.3.1. Measures and Definitions

By any measure, airplanes are safe. Airplane avionics are a part of that safety picture. And electromagnetic compatibility is a part of the proper operation of avionics.

Although safety records have been built and established over the years, the criticality of avionics and their interconnecting wiring is not an unchanging situation. Of course, mathematically finite possibilities of avionics performance errors or malfunctions always exist, and it almost goes without saying that absolute, definitive resolution of the exact probabilities and reliability of critical circuits is elusive.

"The simultaneous failure of two reliable independent systems, each of which has dual redundancy," states FAA Advisory Circular AC No: 25.1309-1, "is expected to be extremely improbable." But still, the performance of critical avionics equipment cannot be allowed to be affected by the noise in the airplane.

Critical circuits need evaluation. Evaluation is needed on non-essential systems for their effect on critical circuits. Flight crews must not be given misleading information. Noise effects can be barred with shielding, rejected by balanced circuits, diverted and contained with filtering, or neutralized by software.

Table 2.3-1 itemizes some equipment criticality categories and definitions, and figure 2.3-1 charts a general relationship of probability and consequence.

TABLE 2.3-1. CATEGORIES OF CRITICALITY

Category	Function	Impact	Probability of Occurrence
A	Critical	Prevent safe flight	Extremely improbable $\leq 10^{-9}$ (per 1 hr.)
B	Essential	Impaired ability to cope	Improbable $\leq 10^{-5}$ to $> 10^{-9}$
C	Nonessential	No significant degradation	May be probable $> 10^{-5}$

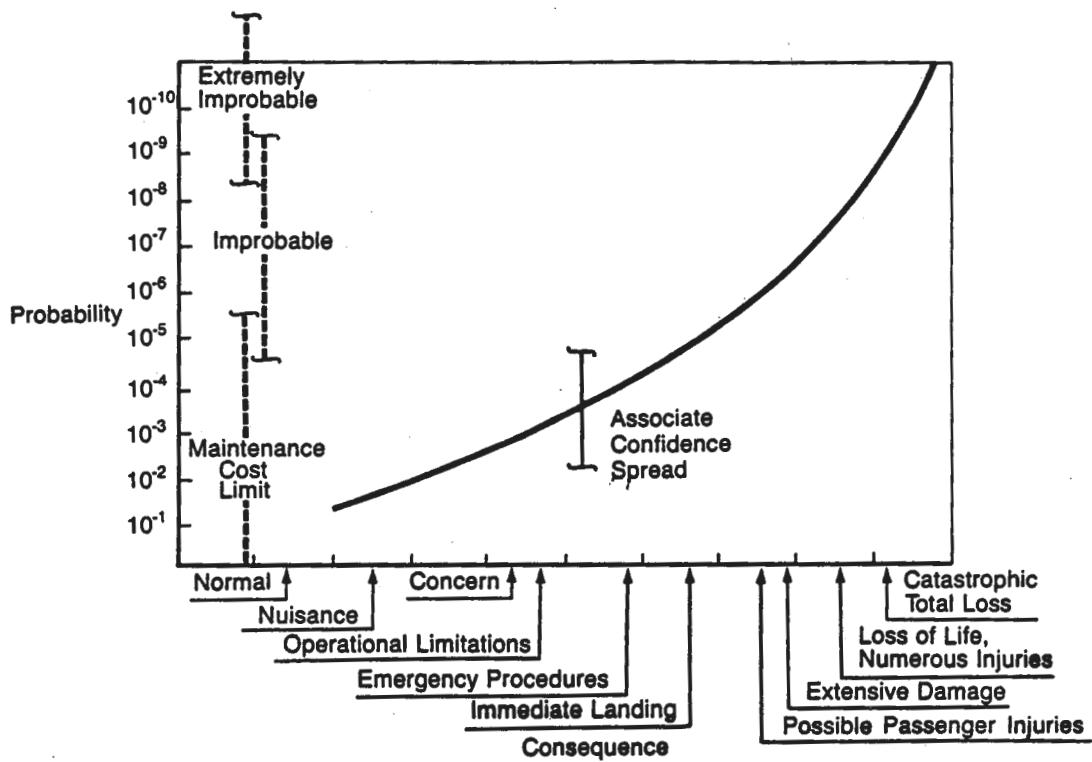


FIGURE 2.3-1. PROBABILITY VERSUS CONSEQUENCE

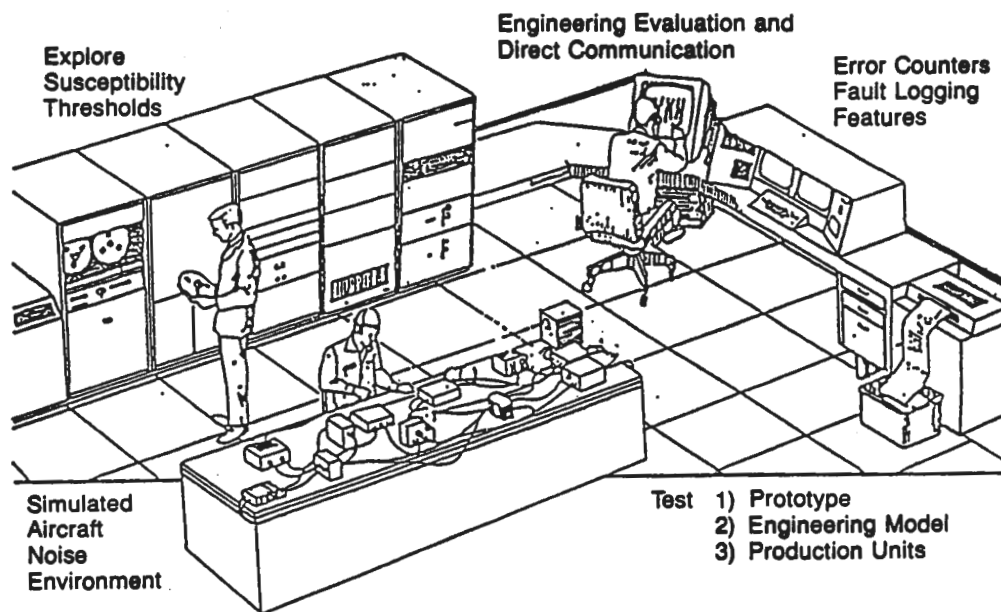


FIGURE 2.3-2. THREE-STAGE SUBSYSTEM TESTING

Toward the end of an airplane program, malfunctions or upsets can practically be brought to zero by repeated subsystem testing by the airframe manufacturer in conjunction with the avionics manufacturer of, first, prototypes, then engineering models, and finally production units. The avionics operation in a simulated noisy environment will be understood, and verified, and the data and experience will help contribute to aircraft validation (figure 2.3-2).

2.3.2. Critical Equipment

What makes equipment critical? What conditions influence criticalities? Here are some defining generalities:

Function:

- Does the unit support safety of flight or is it for convenience, comfort, work relief, or economy?
- Is the unit employed to maintain flight, or is it needed to proceed to the nearest airport, or to continue to destination?

Redundancy:

- Is the unit triple or quadruple redundant?
- If the first unit fails and the second unit fails, can loss of aircraft be averted?

History:

- What is the history, condition, and age of the unit or the aircraft?
- Are all units operating at flight dispatch?

Flight conditions:

- Is criticality based on phase of flight: takeoff, climb-out, cruise, landing?
- Is criticality based on type of flight: deficient or lack of navigation facilities, instrument flight, heavy weather, long flight, over water, nighttime operation?
- Is flight safety dependent on automated electronic controls, instruments and sensors?

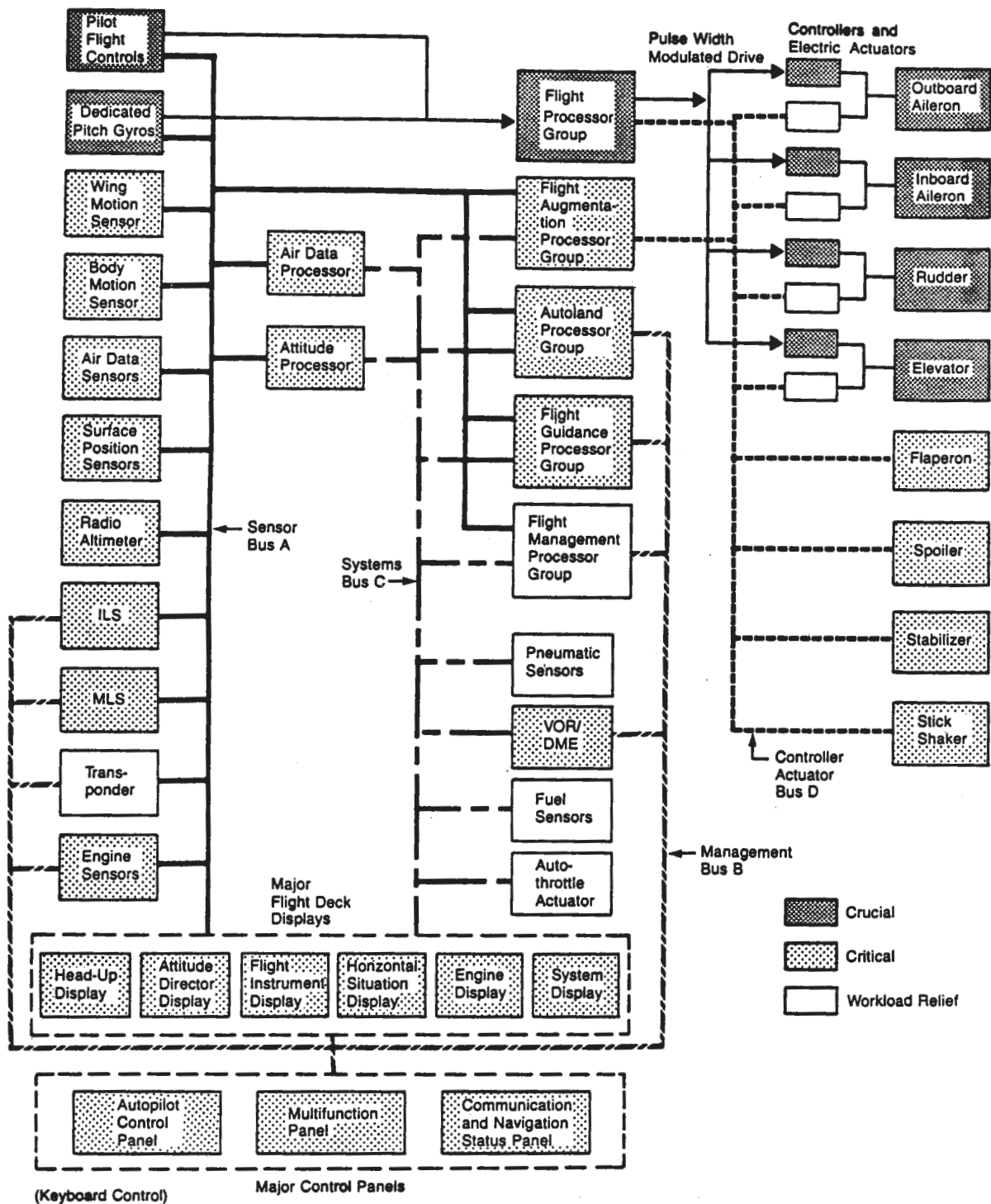


FIGURE 2.3-3. PROJECTED CRITICALITIES

What circuits are classified as critical? Certain circuits or functions must be extracted and given special attention for electromagnetic compatibility. Circuits that might be considered include these illustrative examples:

- Flight control/flight management: control surface actuators, displacement transducers, servo valves, position indicators, switches/valves; electric controllers/actuators; negative stability controllers; augmentation controllers; air data, attitude, altitude, airspeed, and situation indicators, displays, control panels and their backups; automated landing system.
- Navigation/communication: VHF transceivers, voice recorder, tape recorder, ADF, radio altimeter, instrument landing system, marker beacon.
- Power: standby power, instrument lights, standby instruments.
- Advisory flight instrumentation: actuators, position indicators, displays, test circuits, pressure, temperature, quantities.
- Fire detection systems.
- Landing gear: antiskid control.
- Engine: controller actuators, computers, displays, temperatures, speeds, pressures, quantities, restart/shutdown, thrust reversers.

Figure 2.3-3 delineates a block diagram of some hypothetical categories of future aircraft systems.

2.4. Packaging and Architecture

The aircraft is an electrical/electronic package. Right from the start, the aircraft packaging design must include a layout and topology that optimizes electromagnetic compatibility. Here are the dominant EMC desired designs:

- Major subsystems are grouped together for an in-line equipment design, input to output, to draw out the shortest possible wire bundle routing.
- Major incompatible wiring groups are separated: power feeders from electronics; analog (with single point ground) from digital; high voltage/high frequency from digital; low-level sensitive from power.
- The aircraft has a designed system-level shielding barrier combined with a designed, controlled, equipotential ground plane system.
- Every interface circuit, electronics and power, is filtered and protected. Controlled transmission line design techniques are employed for susceptible and digital interface circuits.
- Every installed unit of equipment meets the EMC qualification test requirements.

All of these designs and technologies need early conceptual consideration, planning, and lay-out. A mockup is invaluable. In-line design is the best. Locate equipment to minimize wiring lengths. Electromagnetic coupling increases with length. Make wire connections short. It's good for weight. It's excellent for electromagnetic compatibility.

Here is a more detailed checklist.

Grounding:

- Ground connections cleaned and burnished, no paint, must verify.
- Ground wires evaluated for resonance, capacity, and continuity.
- Avionics case grounded with a low resistance (and verified).
- Grounding paths and references designated on drawing.
- Trays, conduits, metal liners, foils designed for return currents.
- Graphite-epoxy not used as structure return.
- Analog circuits are single point grounded or isolation provided.
- Each avionic unit return/ground identified.
- Engine circuit return/ground analyzed for wire length and impedance.

Bonding:

- Structure electrically bonded and aperture bonds confirmed.
- Electrostatic conductive paints applied to external dielectric or nonconductive surfaces.
- Equipment bonding to structure verified.
- All isolated metal objects bonded.

Shielding:

- 360 degree, peripheral shield connections used.
- Backshell continuity defined and checked.
- All circuits evaluated for shielding.
- All low-level, sensitive circuits shielded.
- Shield noise not carried inside case confirmed.

Wiring:

- Landing gear wiring installed in conduit and flexible overbraid.
- No wiring routed under unshielded fairings.
- Wiring installed close to structure.
- Shielded, balanced, isolated interface circuits employed.
- Power feeders given a power line and a wire return.
- Signal lines given a signal line and a wire return.
- Twisted pair used for audio circuits.
- Oscillator harmonic frequencies emanating from clocks, switching regulators, pulse width modulators, and digital data lines shielded. Conducted and radiated emissions, close to receivers, may need to be controlled below DO-160 limits.

Packaging/installation:

- Test data dug out to verify equipment complies with specification.
- Equipment tested to up-to-date requirements.
- Transformers separated from CRTs and audio circuits.
- Environmental control, flight control/electric actuators, flight management, radio transmitters and receivers, power, and engine control systems unearthed, defined, grouped, and aligned.
- Mockup constructed to settle on equipment topology.
- Functional or subsystem grouping provided for environmental control electronics, switching units, valves, fans; flight control, electric controllers located next to stepper motors; air data, caution advisory computers next to their transducers; radio frequency units, power supply, electronics, amplifier, antennas; heavy power generators, converter regulators, transformer rectifiers, switching contactors, power switching unit installed together, and separated from avionics; power "single point grounds" positioned next to power regulator or distribution unit; power switching unit centrally located; dedicated avionic power supplies installed next to the unit they supply; electronic engine controllers positioned to shorten and minimize wiring to engine sensors and controls.
- All circuits evaluated for filtering.
- Equipment input and output wiring not doubled back upon itself.
- All circuits evaluated for categories and separation.

These are not all the answers but they will help.

2.5. Issues

2.5.1. An Early Start

The whole EMC scenario rises up and needs resolution at the concept of the program. This is an issue. The scope and depth are resolved first - is it a proposal? - proof of concept? - or full production program? What are the funding boundaries?

Environmental assessment quickly follows, all of the environment inside and outside of the airplane. This is an issue.

Early in the program, light must be shed on the "tailored" design requirements or waivers for subcontractors. New systems demand definition. Policy and schedule come into focus in a brief EMC plan. These are large tasks and they are all at issue. Bringing two or three people on board early is an added expense. It seems unnecessary. Electromagnetic compatibility design does not start at the design stage, it starts at the concept stage.

2.5.2. Design

2.5.2.1. Shielding

Shielding is one of the best ways, the most all-around effective ways, to protect a circuit. Shielding diverts almost all of the energies of noise. Shielding stops transients as well as radio frequencies. It protects against electric fields. It helps to maintain controlled stripline impedance. It guards against arcing and sparking. But, it adds weight to the aircraft, requires maintenance, and costs more. Shields require shield ties - difficult to install, particularly for panel-mounted equipment. With adequate modeling and design effort, shielding can be judiciously applied.

2.5.2.2. Power and Signal Returns and Wire Grounds

Sometimes a wire is good, sometimes bad. Many of today's circuits or case housings are referenced to structure ground with a wire, which may resonate. Many of the interface circuits between pieces of equipment do not have a wire return, which will open the circuit up to noise. Both of these practices are detrimental to EMC, but they save on weight. Wiring is good as a signal return and bad as a circuit reference.

2.5.2.3. Dielectrics

By specification, wiring and connector insulations have dielectric withstand strengths at sea level of 1500V 400-Hz steady state. That 1500V weakens as one goes up in altitude. It can be down around 300V its lowest point, at an altitude of 250,000 ft. In the future, as power line voltages rise and aircraft altitudes increase, dielectric strengths and corona will become more of a concern.

2.5.3. Environment and Test

2.5.3.1. High-Energy Radio Frequency Fields

Magnitude, pattern, frequency, polarization, modulation, and geographical location of high-energy radio frequency and radar transmitters are needed. To evaluate aircraft circuit immunity, a systems approach must be implemented to study shielding, induced voltages, circuit protection, and software correction techniques. An industry susceptibility test specification is needed. In areas such as this, compliance to DO-160 may not guarantee system-level compatibility. Efforts are underway to provide an update of high-energy radio frequency field environments and protection techniques. (See section 10, Bibliography, ECAC study on Electromagnetic Environment.)

2.5.3.2. Fields From 400 Hz and Transients

Some say that the power system 400-Hz is the most troublesome electromagnetic environment on the aircraft causing 50% or more of the shortcomings in EMC quality. Audio 400-Hz "hum" derives from magnetic ("H") field. Electric ("E") fields can trigger a comparator circuit that has high-input impedance, and it is well known that powerline transients cause logic upset resulting in lockup or even equipment shutdown until the flaws are found and rooted out. Through better definition, analysis, and modeling techniques, avionics designers must be made aware that their equipment is operating in this environment. Higher design and test levels are appropriate in some instances, coupled with more attention to finding interface circuit thresholds during test. It is controversial.

2.5.3.3. Test Conditions

In the laboratory during development or qualification testing, it is expensive, difficult, and meaningless to strive to recreate actual aircraft installation or production configurations. There are wire lengths, resonant conditions, and test coupling conditions that may not adequately simulate the aircraft and, because of this test deficiency, may ultimately lead to an upset occurring on the aircraft. Possibly one answer is to verify that test levels are high enough with an adequate safety margin. Subsystem tests offer information on EMI characteristics to help moderate this dilemma. Aircraft tests authenticate the final EMC design.

2.5.3.4. Emission Variances

Today's conducted and radiated emission limits are restrictive under some conditions (see section 9). Computers, with digital clocks and switching regulators, sometimes emit harmonics in the HF megahertz region that defy total containment and consequently emissions may be a few dB above limits. Cases exist where it is uneconomical and unnecessary to go to extraordinary efforts to filter those emissions if it can be shown that they will not in any way radiate to local receiving antennas. Infringements may be approved and deviations granted without adverse effects on neighboring circuits. The present radiated emission limit in the VHF range is not too restrictive and, in fact,

2.5.3.5. Subsystem Testing

Subsystem testing is different in intent from equipment qualification testing. Subsystem testing, usually held at the airframe manufacturer's facility with support from the avionics supplier, provides insight into the susceptibility thresholds and emission levels while simulating noise environments of the airplane.

The tests are for engineering evaluation and they provide significant information on computer processing performance and interface data quality. Test software exercises the processing functions and the interfaces between avionics units, and further, it monitors processing, memory, and transmissions of data for any abnormal conditions. Error counters or fault logging features examine operation in real-time and provide capability for hard copy printout. Formal pass or fail standards for susceptibility and emission do not apply. Completion of the specific test procedures and the investigation is the gauge for determining success.

The features are that many interface circuits are per manufacturer's configuration, software is up-to-date, and observation is real-time. At this stage of the program, the equipment avionics engineer is available to provide rapid evaluations and judgments. Re-evaluation and decisions on rework or test changes are easy and flexible. Often these tests can be performed on a "non-interference" basis using informal procedures. They establish standards for the airplane test. Costs are low.

2.5.3.6. Aircraft Testing

Aircraft testing is the final authentic proof. It offers first hand "real-world" validation. Wiring, equipment, and installations are the final design. It is expensive, however. It is expensive beginning with the test procedure (step-by-step development and approvals), then the aircraft test itself (a labor intensive operation compounded on top of a costly unit of equipment), then continuing with the formal documentation of unplanned events during the test, and ultimately ending with an elaborate final report.

2.6. Variances in Electromagnetic Compatibility

2.6.1. Diagnostics and Troubleshooting

Flight test phoned project. Project called flight deck instrumentation staff. Then staff contacted the electromagnetic compatibility engineers.

A new aircraft on the flight line had a discrepancy in the left engine fuel flow reading. The fuel flow digital readout on the display screen of the criticality advisory system (CAS) was variable and erratic whenever the 400-Hz power to the engine Mach probe heater was energized. What was happening and what was the cause? A work authorization was quickly approved; time on the airplane scheduled; and laboratory test support called in.

On the airplane, in cramped quarters next to the electronics bay, investigators used an oscilloscope to troubleshoot the problem on the circuits running from the fuel flow meter on the left wing engine to the CAS computer located in the electronics bay. Connectors were hard to reach and remove with care. Knuckles got scraped. Equipment was difficult to move. Engine run time was expensive. As a part of the investigation, the left engine wire bundle was disconnected from the left CAS computer and reconnected to the right CAS computer to see if the problem would "follow" the bundle. It did, and that showed that the CAS computer itself was not totally at fault and that 400-Hz, 115V power in a wing wire bundle was being coupled to the fuel flow circuit.

The 400-Hz, 115V power wire to the mach probe heater, traced out on the wiring diagrams, starts in the electronics bay and runs out to the engine; but then the 115V heater return wire is taken from the engine back onto the engine strut; the wire is there connected to structure with structure being used as return back to the power source in the electronics bay. So the "high side" of the 115V power wire runs in the same bundle in the leading edge for 100 ft next to the unshielded, fuel flow circuit. There is 100 ft of "electric field" capacitive coupling.

On the other hand, the fuel flow circuit is three wire balanced, has the meter wires isolated on the engine, and has a high-resistance input to an operational amplifier in the CAS computer protected at the input by 5 k Ω resistors and diodes to ground. The two, 350-mV pulses generated by the fuel flow meter have a working period approximating the period of the 400 cycles per second. The 115V 400-Hz noise was induced right on the 350 mV.

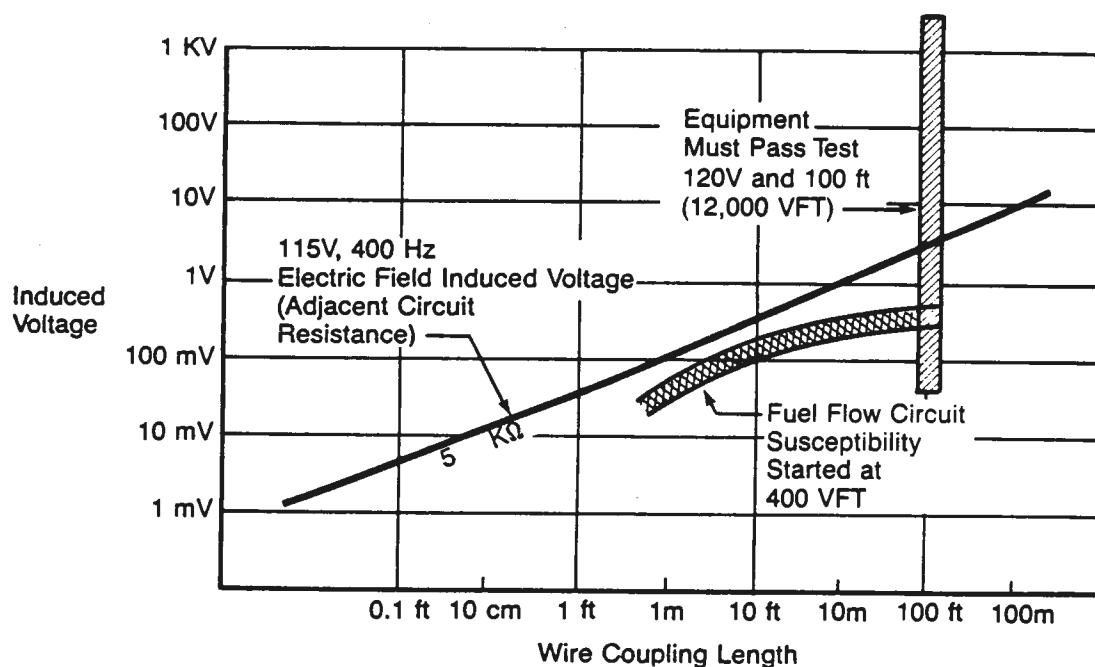


FIGURE 2.6-1. A DISCREPANCY

The fuel flow circuits were rapidly set up and simulated in the laboratory and they malfunctioned under the DO-160 power line "electric field" test just as had occurred on the aircraft. It took only 40V at 10 ft (400 Vft), equivalent to 4V at 100 ft, of coupling to cause upset (figure 2.6-1). The test requirement is 120V at 100 ft (12,000 Vft). Other DO-160 tests caused no upset. Once the simulated circuits with the proper pulses had been developed and the thresholds and boundary characteristics of the upset outlined, the test was scheduled, set up, and run in an avionics laboratory with a production configuration CAS system. With management approval, communication was established through Project with the subcontractor. He was able to duplicate the condition.

The long, 100-ft wiring run, all the way from the electronics bay through the wing pressure seal, then along the wing leading edge to the engine strut and down into the engine, had provided an extensive opportunity for electric field coupling into the operational amplifier wiring. What will correct electric field coupling? A low resistance or shield will.

Wire shielding, installed on the airplane fuel flow circuit, provided an excellent barrier against the electric field and was a quick solution.

The operational amplifier design in the CAS computer, although a balanced circuit, did not offer proper noise rejection to stop the upset. It turned out that that circuit design was inherently difficult to balance, partly because of resistor mismatch, partly because of capacitor imbalance, and possibly because of a phase shift occurring in the signal return.

This "variance" from electromagnetic compatibility on the aircraft took months to resolve. Flight test, project, manufacturing, management, and subcontractors: all were involved. Work authorizations, reviews, justifications, and documentation were invoked, processed, approved, and completed. A necessary, but costly, effort.

2.6.2. Table of Variances

2.6.2.1. Commercial

Diagnostics and troubleshooting of electronics are continuously demanded because of avionic circuit susceptibility to transients, 400-Hz electric ("E") and magnetic ("H") fields, radio frequencies (HF-VHF) including clock and switching regulator harmonics, and power quality. Knowledge of past lessons may help save design and troubleshooting expense (table 2.6-1). The bottom line is adequacy of specifications. In table 2.6-1, under the DO-160 column, recommendations are made on possible increases in test levels to improve aircraft electromagnetic compatibility.

2.6.2.2. Air Force

The Air Force has documented troubleshooting experience. "The coupling of fields through the unintentional antennas formed by aircraft wiring," Zenter, the author, says, "is the cause of many interference problems." (See table 2.6-2).

TABLE 2.6-1. VARIANCES IN EMC

EMI CLASSIFICATION	SOURCE	EQUIPMENT RECEPTOR	PATH	SYMPTOMS	REMEDY	DO-160	
Static:	1 windshield 2 covers/pan	captr VHF rcv	dsch "	dis captr aud	gnd windshield bnd covers	add test	ant-antenna aud-audio hum or tones cab-cabin cap-capacitor captr-computer CRT-cathode ray tube del-change or movement
	3 duct 4 ant cover	ADF rcv "	" "	" "	gnd duct paint covr		dis-disengage dsch-discharge "E"-electric field eq-equipment FF-fuel flow
Sw.trnsnt:	1 powerline	captr	wc	del rud/ dis captr	lgnd	inc trnsnt	del-change or movement dis-disengage dsch-discharge "E"-electric field eq-equipment FF-fuel flow
	2	"	"	lock up	cap/softwr		fil-filter gl-ground loop lgnd-single point ground "H"-magnetic field HUD-head up display
	3	"	"	false cad	"		inc-increase ind-indicator change intrphn-interphone intrpt-interrupt iso-isolate lt-light N2-engine speed prx-proximity switch rad-transceivers rcdrs-recorders rcv-receiver rdr-radar RE-radiated emission sep-separation shld-shield sup-suppression txf-transformer TP-twisted pair trnsnt-transient tx-transmitter wc-wire coupling wx-weather
400Hz "E" field:	1 powerline	captr	"	ind- FF	shld	inc "E" field level	
	2	N2 ind	"	ind- N2	"		
	3	captr	"	ind-F/qty	"		
	4	aud cir	"	aud	"		
400Hz "H" field:	1 powerline	intrphn	gl	"	lgnd	inc "H" field level	
	2	headset VHF rcv	"	"	"		
	3	videocoax	"	del CRT	Tfx iso,lgnd		
	4	CRT	wc	"	TP & sep		
	5	tape head	"	aud	"		
	6	aud cir	"	"	fil,sep		
	7	"	"	"	TP,sep,lgnd		
Clock hrncs:	1 wx rdr	ILS rcv	"	"	sep	inc RF level	
	2 portble rad/rcdrs	omega rcv	"	ind del: spd,course	shld		
Sw.reg.hrncs:	1 captr	VHF rcv	"	aud	sep, shld	inc RF level	
HF-VHF freq:	1 VHF tx	handset	RE	"	fil micrphn	inc RF level	
	2 VHF tx	headset	"	"	txf iso		
	3 HF tx	captr	wc	cab press	fil, cap		
	4 "	aud cir	"	aud	dubshld		
	5 "	"	"	ind	"		
	6 HF eq	VHF eq	"	aud	"		
"Crosstalk":	1 VHF, DME	aud cir	"	"	sep,lgnd		
Radar:	1 airport rdr	prx sw	RE	ind lts	shld		
Power qualty:	1 powerline	captr	28vac	ind	fil		
	2	"	115v	no-land	"		
	3	"	"	pwr intrpt	cap, softwr		
	4	"	"	del CRT	softwr		

TABLE 2.6-2. AIR FORCE VARIANCES

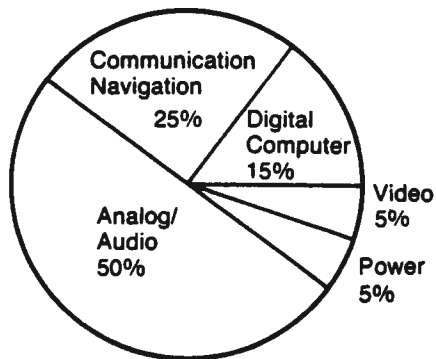
EMI CLASSIFICATION	SOURCE	EQUIPMENT RECEPTOR	PATH	SYMPTOMS	REMEDY	MS461
Static:	1 canopy 2 " 3 ant 4 "	UHF rcv " " "	dsch " " "	aud " " "	sep bnd,gnd " "	add test
Sw.trnsnt:	1 powerline 2 3 4 5 6 7 8	captr intrphn	wc "	ind-rdr warn IFF code tx ind-HUD cam op bomb disarm ind-flare del-captr memory ind-teran fol rdr aud	shld,fil diode sup, softwr	add test
400Hz "E" field:	1 powerline 2	omega rcv VLF rcv	RE	ind-del aud	sep fil	add test
400Hz "H" field:	1 powerline 2 3	intrphn captr "	wc	aud del CRT ind oxy	ign sep "	inc "H" field level
Clock hrncs:	1 captr	ant	RE	aud	fil,shld	
Sw.reg.hrncs:		--none				
HF-VHF-UHF:	1 tx 2 3 4 5 6 7 8 9 10 vid sig 11 strb sig 12 dig sig	captr wiring " " " " " " " ant rcv " "	RE	del-cntrl surfce del-steer gear del-rdr ant del-winch del-eng speed ind-altimeter aud-intrphn del-A/C heading del-nav flag aud aud,del-stick aud	fil,bnd,shld	inc RF level
Radar:	1 rdr 2 HF tx	prx sw captr	RE	lose antiskid ind-lts	fil,bnd,shld	inc RF level
Power qualty:	1 powerline 2	captr "	26v 115v	del captr dis captr	fil,bnd,shld	

2.6.3. Corrective Action and Modeling

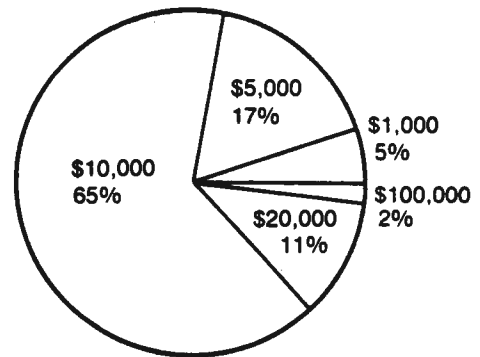
A comprehensive study of troubleshooting is needed before accurate corrections to the aircraft system specifications or the RTCA DO-160 equipment specification can be recommended. Documented, statistical groupings of troubleshooting experience concerning historical EMI characteristics and cost factors do not exist, but one might postulate some possible categories and hypothetical percentages as shown in the accompanying pie charts (figure 2.6-2).

A study of categories such as these might reveal that analysis and modeling of analog/audio circuit susceptibility is most beneficial, or possibly better models of aircraft wire coupling might help, or even modeling and documenting of the most popular fixes.

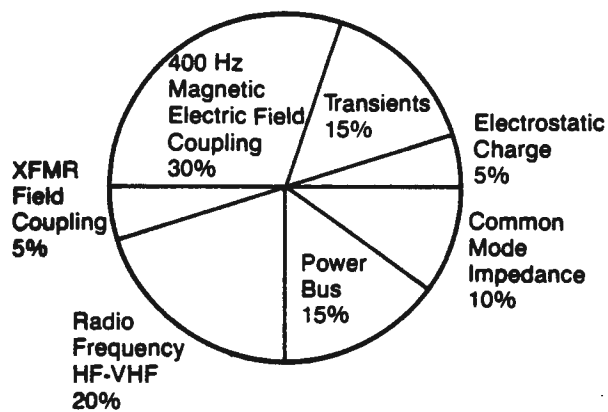
EQUIPMENT/CIRCUIT TYPE



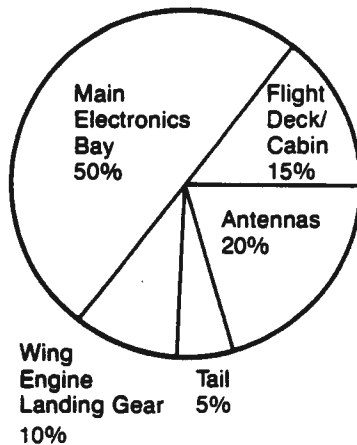
RANGE OF EXPENSE



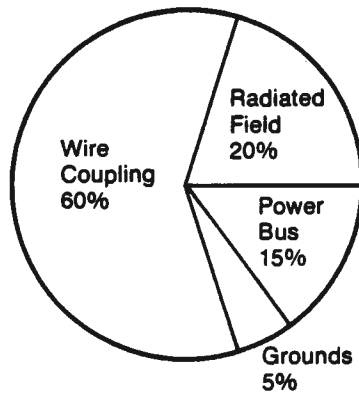
EMI TYPE



EMI LOCATION



EMI PATH



FIXES

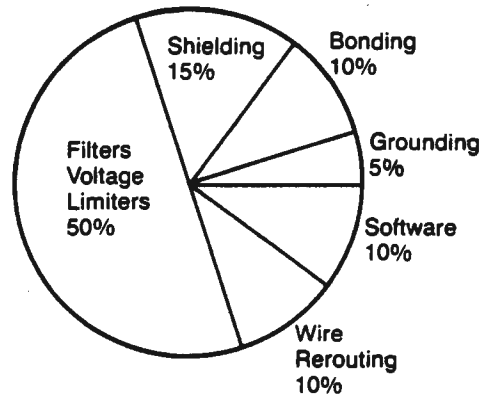


FIGURE 2.6-2. POSTULATED HALLMARKS OF VARIANCES



3. AVIONICS THRESHOLDS AND PROTECTION

3.1. Hardware Tolerance

Recognizing the conditions of proper interface wiring design, circuit protection, and the contributions of aircraft structural shielding protection is of course important in a "top-down" electromagnetic compatibility design of an aircraft, but these recognitions must also be tied closely to a knowledge of the noise tolerances and noise thresholds of transistors, micro-circuits, and logic. The lines can be drawn: environment, protection, threshold.

Airplane environmental noise voltages flourish far above the transistor and micro-circuit thresholds of damage, upset, or offset, which means that micro-circuits demand protection in every case. Transistor-Transistor-Logic (TTL) gates and microprocessors do not have even a modest tolerance to the run-of-the-mill aircraft noise types, such as electrostatic pulses, lightning induced transients, inductive switching transients, or even some high-energy radio frequency signals.

A TTL logic gate will "change state" at a threshold of about 800 mV when radio frequencies are injected starting at low frequencies and on up into the megahertz range, but in the tens to hundreds of megahertz the threshold rises to greater than 5V before upset occurs. Figure 3.1-1 maps that threshold. Whereas a gate will operate or change state at 800 mV it may be damaged if subjected to much greater than 10V at low frequencies and can usually survive about 100V transients that are of microsecond or nanosecond duration depending on thermal dissipation. As shown in the figure, a transistor or semiconductor "PN" junction will detect power levels as low as 100 μ W under certain conditions.

Microprocessor chips are high density and high speed. Microprocessor circuits may be upset, change state, or change performance when signals with noise power levels down to 10 μ W are injected on signal, address, or clock lines. As frequency is increased beyond the operational range of a microprocessor, the power required to cause upset and damage increases. These levels change with conditions such as loading, circuit geometry, radio frequency paths, and sometimes software design. Operational factors such as address or memory or timing or process changes affect the definition of upset. Under the onslaught of steady state, low-frequency signals, microcircuits can be damaged at power amplitudes of less than 1W, but high frequencies or fast narrow pulses require much higher wattage levels - ten to hundreds of watts.

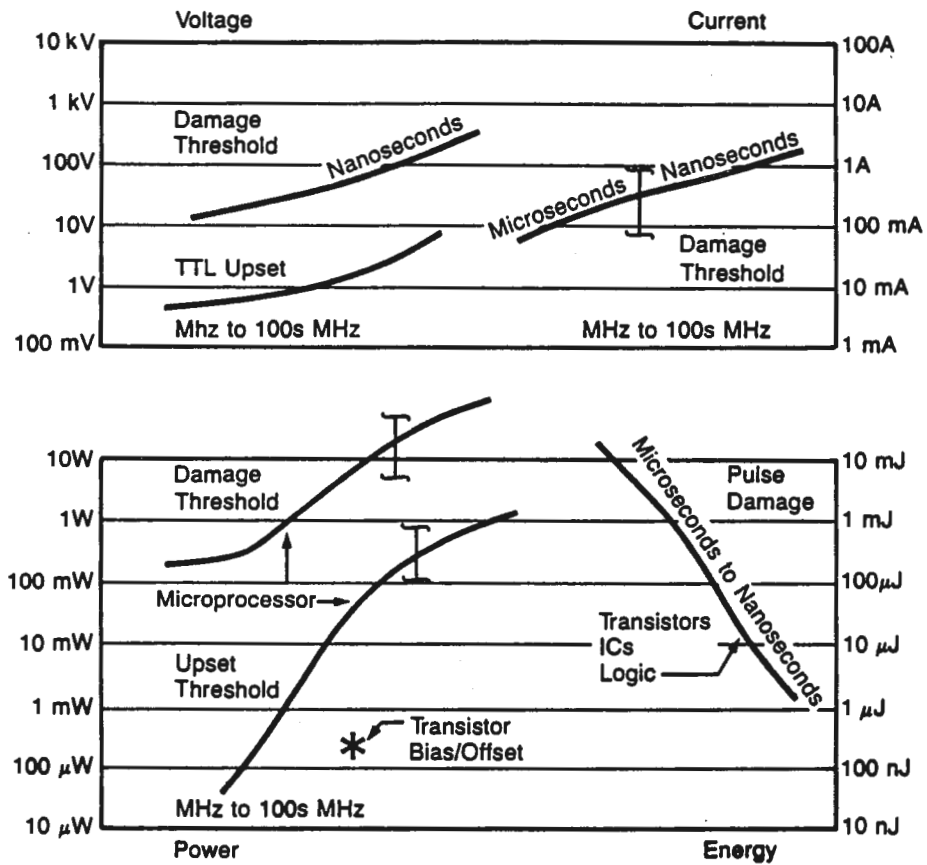
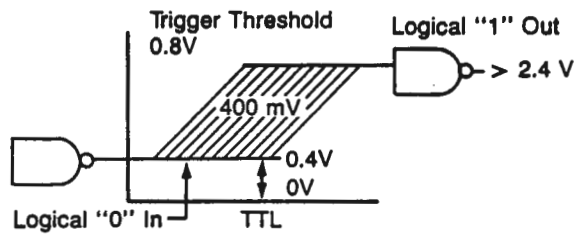


FIGURE 3.1-1. MICROCIRCUIT TOLERANCE



	Transistor-Transistor Logic TTL	Emitted Coupled Logic ECL	High-Threshold Logic HTL	Complementary Metal Oxide Logic CMOS
Average	1.2V	100 mV	7.5V	2.2V
Minimum	400 mV	—	5V	1.5V

FIGURE 3.1-2. NOISE MARGIN

Individual pulses, measured by their energy content, require anywhere from 10 mJ to 1 μ J to cause damage as shown in figure 3.1-1. Pulse durations are microseconds to nanoseconds. Awareness of these levels is important when designing the shielding for the aircraft or avionics wiring to protect against HF transients or resonances.

Shielding protection or voltage limiting must be provided to reduce transient noise signals below the damage threshold. Protection must also be designed to keep continuous wave radio frequencies below the TTL or integrated circuit operational thresholds (figure 3.1-2). Software can be designed to correct for random, nonrepetitious transients.

Extensive effort in testing, modeling, measuring, and graphing operational upset and damage levels has been documented in the industry. Minimum and maximum spreads are available. Voltage damage amplitudes may vary by a factor of two or three, say from 100V to 300V from one manufacturer to another, or from one lot to another. The threshold of operation may vary from 800 mV to 1.5V from one transistor to another. Damage thresholds change by a factor of ten or more with the number of transient pulses and the rate of applying them. The spreads and averages are interesting in the study of susceptibilities, but at the bottom line is this: protection must be built in to account for the minimums - for example, 100V for transient damage, 800 mV or less for TTL gate state change, and 10 μ W for microcircuit performance alteration. It is necessary to constrain radio frequency power and stored transient energy access to avionics by diverting and/or blocking the noise with balanced circuits, filters, or shields on every input-output interface circuit.

When two conductors are spaced 100 mils (2.5 mm, 2500 μ m) apart, and the voltage between them is slowly increased, an arc will start (at sea level, atmospheric pressure) and establish itself at around 2000V or 3000V. If there are only 20 mils of spacing (0.5 mm, 500 μ m) between the conductors, the arc will start at 1000V. Circuit card conductors have these close spacings. So it is easy to understand when microcircuits, chips, and thin-film devices with substrate circuit separations of a few microns fail at 100V.

3.2. EMI and Software

It is difficult to know where to start in the treatment of electromagnetic interference relative to software in an aircraft context. What is special about an aircraft? How does aircraft electromagnetic interference uniquely relate to software?

Not by component failure or damage: damage may appear anywhere, anytime - broken parts - vibration - faults - mishandling.

Not by errors or deficiencies in software itself: this is the purview of the software designer; he must compensate for these regardless of aircraft electromagnetic interference. And also, not especially by the internally generated noise from power and switching regulators, or clocks and data lines: noise sources found in any electronic package.

The aircraft associated electromagnetic interference arises from power line 400 Hz, radio frequencies, electrical transients, and power bus momentary interruptions. They are very different in their characteristics, their occurrence, and their threat of upset. Two of these noise threats must be eliminated from software concerns right away.

First, 400 Hz is simply 400 cycles per second of a noise voltage from the power line imposed on a neighboring circuit. Cycle duration time is 2.5 ms; with zero to peak being 625 μ s; the positive risetime repeats every 2.5 ms. If a balanced circuit is slightly unbalanced and responsive for any reason, the 400 Hz will trigger operation of the balanced circuit continually or, much worse, in an intermittent fashion. If there is an error or disorder from 400 Hz, then there has been an error in the original design that must be fixed. Four-hundred Hertz must be controlled and kept out of interface circuits.

And second, radio frequencies can also be eliminated from software concerns. Aircraft wiring cannot be allowed in a radio frequency field stronger than the original design specification. If a digital circuit sees an overlay of an unwanted radio frequency (not damaging, but causing loss of data), the software will not be able to correct upset. The radio frequency environment must be known, documented, and immunity designed in to the interface circuits. Transmitter radio frequency noise and digital data are in the same frequency range, the most important being HF-VHF, 1 to 300 MHz.

Transients are the problem. Electrical transients have existed in the past, they exist today, and will continue to exist on the airplane as well as in the laboratory. Electrostatic discharge, lightning, or powerline inductive switching transients are usually of large magnitude (hundreds to thousands of volts). The amplitudes are reduced below circuit damage level (100V; a few amps) by design; but transients are sometimes not totally rejected, and they result in short-term destruction of data words. It is the responsibility of the EMC engineer to supply threat transient levels and repetition rates to hardware/software designers to assure proper programming of fault tolerance and detection. What, then, are the time characteristics of transients relative to data words that the software designer must know to override noise or make software tolerant to noise?

Some inductive switching transients have ringing frequencies of 10 MHz recurring at a 1-MHz repetition rate and lasting for around 1000 μ s (figure 3.2-1). During test, the transient is repeated every two seconds. (DO-160 specifies 8-10 pulses per second for 10 seconds.) A 12-kHz, ARINC 429 signal has a bit width of 80 μ s, and the 32-bit word has a duration of 256 μ s. A 1000 μ s, inductive switching transient can decimate a 256 μ s data word.

An electrostatic discharge, on the other hand, is a very fast 100-ns event (figure 3.2-1) and probably does not recur until after sufficient time, possibly 4 or 5 sec, to recharge the object originally collecting the charge. This transient might only affect one bit.

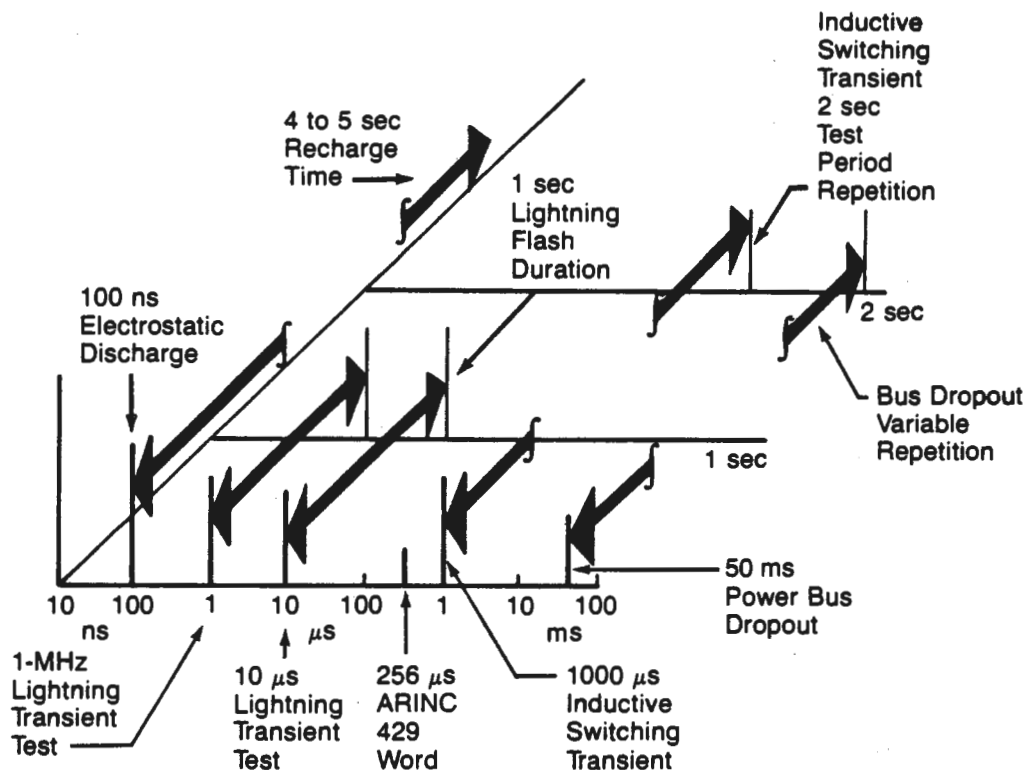


FIGURE 3.2-1. TRANSIENT EVENTS

Lightning transients have low frequencies, 10 μ s or longer, and ringing high frequencies, for instance at 1 MHz, 3 MHz, and 7 MHz, set up by the electrical resonant lengths of an aircraft and damped out in a few microseconds; but then they may recur again and again under multiple strokes of a total lightning flash lasting for possibly one second (figure 3.2-1). A lightning flash might result in disorder in a number of words.

Of important concern is the momentary interruption of the power bus, referred to as a "bus switching" or "dropout" transient, where power drops or decays to zero for up to 50 ms (figure 3.2-1) when the power supply is transferred from ground power to engine power or from one engine generator to another. (DO-160 has a test requirement of a 200 ms interrupt for ac equipment, and a 1-second interrupt for dc equipment.) Software control may be required to ensure a graceful shutdown, proper data storage, or even continued processing.

So the software designer institutes techniques to control equipment operation, override errors, and make software tolerant to noise when transients and power shutdown occur.

There are a number of methods employed for data correction and resetting, such as:

- Microprocessor reset to initial state ("backward")
- Microprocessor forced to known state ("forward")
- Functionally equivalent, but dissimilar backup system
- Data word repetition or redundant data supply

There are also a number of means of detecting errors of data, flow, or hardware operation, such as:

For data checks -

- Read after write on output data line
- Data bus activity, reasonableness check, data averaging
- Bits per word, parity, status bits
- Check sums: averages, spreads, maximums, minimums

For flow checks -

- Out of sequence, out of loop
- Excess or deficient time
- Event record of activity with respect to total program execution ("state activity")

It may be postulated that the aircraft has five layers of protection: (1) structure shielding, (2) circuit shielding, (3) balanced circuit, (4) voltage/current limiting, and (5) software. Software is the last line of defense.

3.3. Digital and Discrete Circuits

3.3.1. ARINC 429 Drivers and Receivers

The ARINC 429 bus, a digital transmission interface system, fans out from the main equipment bay to the flight deck and to external sections in the wing, engine, or empennage. With two or more 429 circuits per unit, there could be two- to three-hundred individual buses. They may reach 100 ft (30m) in length, and are routed in bundles where they encounter transients, radio frequencies, and power line noise. The ARINC 429 "MARK 33" Digital Information Transfer System (DITS) offers immunity to noise by using a well designed combination of a balanced circuit, a relatively high-trigger threshold, a "high-resistance" input, and finally, wire braid shielding. Parity, status bit, and "bits per word" software checks help to extend protection even further.

The output signal of the 429 transmitter measures at the high level, $10 \pm 1V$ "line-to-line" and at the low level, $0 \pm 0.5V$. At the other end of the line, the receiver must operate with an input signal at a high level of 6.5V to 13V and be at a low below 2.5V. The margin from 2.5V to 6.5V is undefined.

The trigger threshold of operation, therefore, can be 2.5V.

The transmitter driver output resistance is 70Ω to 80Ω "line-to-line" and the receiver input resistance is $12 K\Omega$ or greater in each line so that there is at least $12 K\Omega$ resistance in the circuit from transmitter to receiver. That is an important resistance for protection against transients. The high resistance provides immunity. Just ignoring the shield for a moment, if a 600V transient occurs on the wire and appears at the $12-K\Omega$ input, the resulting current amounts to only 50 mA, a very low "transient" current and not enough energy for damage.

The ARINC specification does not define the circuit ground or case ground. For shields, the specification states that: "The circuit should be twisted pair shielded from data source to sink with the shield grounded at both ends at an aircraft ground close to the rack connector." Shield tie length is not defined. The 429 system has been subjected to the RTCA DO-160 electromagnetic interference tests and passed. Three factors in implementing this digital bus design demand extra effort in manufacture to ensure quality: (1) electrical tolerances of components (avionics supplier responsibility), (2) shield tie to ground (airframe manufacturer and supplier), and (3) case ground (airframe manufacturer and supplier). Future digital buses may have similar noise characteristics and rejection capabilities.

The 429 is a balanced circuit, and balanced circuits are important; the installation of shields is also important, but grounding decides effectiveness of receiver immunity: the grounding of the box, the grounding of the circuit, and the grounding of the shields.

3.3.2. Circuit and Shield Grounds

Noise on a single wire in space with no connection to a ground plane cannot be measured, and no current flows for an "electrically short" wire.

Connect one end to a ground plane and an induced noise voltage in the wire of 1V can be measured at the other end, the ungrounded end. This is a single wire over ground (figure 3.3-1a) and is the technique used to install the "discrete" circuits on the aircraft. It is susceptible over the entire frequency range.

If the ungrounded end is left unconnected, practically no current flows and no energy is dissipated. That's an incredibly significant fact when analyzing for protection against damage. If no current flows, components cannot be damaged. Where circuits are exposed to high-level transients on the wing, isolate them at one end if possible.

Now, two wires over a ground plane with resistors between them at each end to form a circuit and one end connected to ground, say the source end, sets up the same condition - all the voltage will be measured at the load end relative to ground, with practically no voltage across the load resistor at low frequencies.

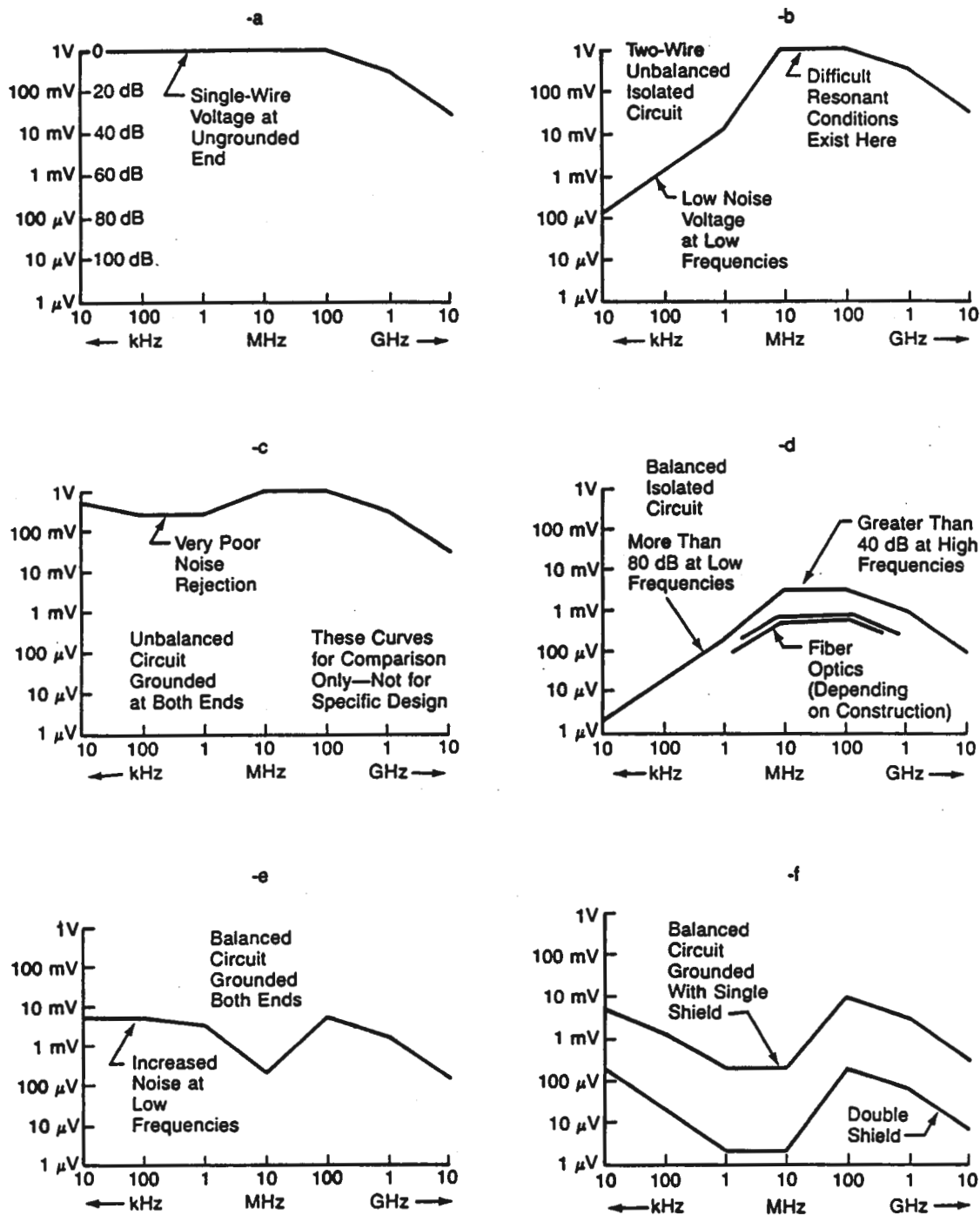


FIGURE 3.3-1. CIRCUIT NOISE REJECTION (A B C D E F)

This is a two-wire, unbalanced circuit (figure 3.3-1b). At the higher frequencies, resonance exists and voltages appear.

A two-wire, unbalanced circuit grounded at both ends, source and load, offers practically no noise rejection, maybe 10 dB or so (figure 3.3-1c). The noise rejection is lost at low frequencies.

Make the circuit a balanced circuit, as the 429 is, and it can establish more than 40 dB of noise rejection for "differential mode" line-to-line voltages (figure 3.3-1d) at the higher frequencies. The line-to-ground noise, called "common mode," remains the same as the unbalanced circuit.

In an aircraft installation, it is optional whether or not the balanced circuit is grounded. Grounding loses some of the low frequency protection (figure 3.3-1e). The balanced, isolated design can offer 80 to 100 dB at low frequencies, just about equivalent to fiber optics or transformer isolation under practical installation conditions.

Figure 3.3-1f shows a comparison of one shield (like 429) and two shields grounded at both ends with the circuit and equipment case grounded. If either the circuit or case were lifted from ground, the circuit would have much more low frequency isolation and protection.

A balanced circuit (either grounded or isolated) that is double shielded with shields tied to the case connector and the base surface of the enclosure case grounded is almost always the best. The circuit can be isolated with a transformer, optical device, or fiber optics. This is the optimum practical design for low and high frequencies.

3.3.3. Discretes

A "discrete" circuit is designed and installed to indicate events, such as on or off, engaged or open. Sometimes it is constructed using a single wire (figure 3.3-1a) from the electronics bay out to a unit on the wing where a switch, to provide indication, grounds it to structure using the structure as the return. Their thresholds are high and, therefore, radio frequency noise is usually not a problem. But these circuits can be hit with the full force of transients and need to be protected with shielding, voltage or current limiting, or increased power ratings.

3.4. Equipment/Wiring Isolation and Separation

3.4.1. Quality of Wiring Design

If there are no wires, there is no electromagnetic interference. Wiring takes on different characters in its role as a conductor of signals. It is a signal conductor, driver to receiver. It acts as a transmitting radiating antenna or a very efficient receiving antenna; it is a party in transformer action to participate in voltage-current-energy transfer when in a cable bundle; and it also acts as an intermediary or a transfer agent to import electrical energy, then retransfer that energy to other wiring - sometimes called secondary coupling.

The wiring design is a fabric, a multi-weave electrical mosaic. There is a mosaic of conditions: wire size, grounding, returns, shields, shield ties, separation, and wire length. There is a mosaic of properties: metals, dielectrics, resistance, inductance, capacitance, impedance, and nonlinear effects. Careful analysis and development of the design will bring about an acceptable level of electromagnetic interference and a cost-effective electromagnetic compatibility.

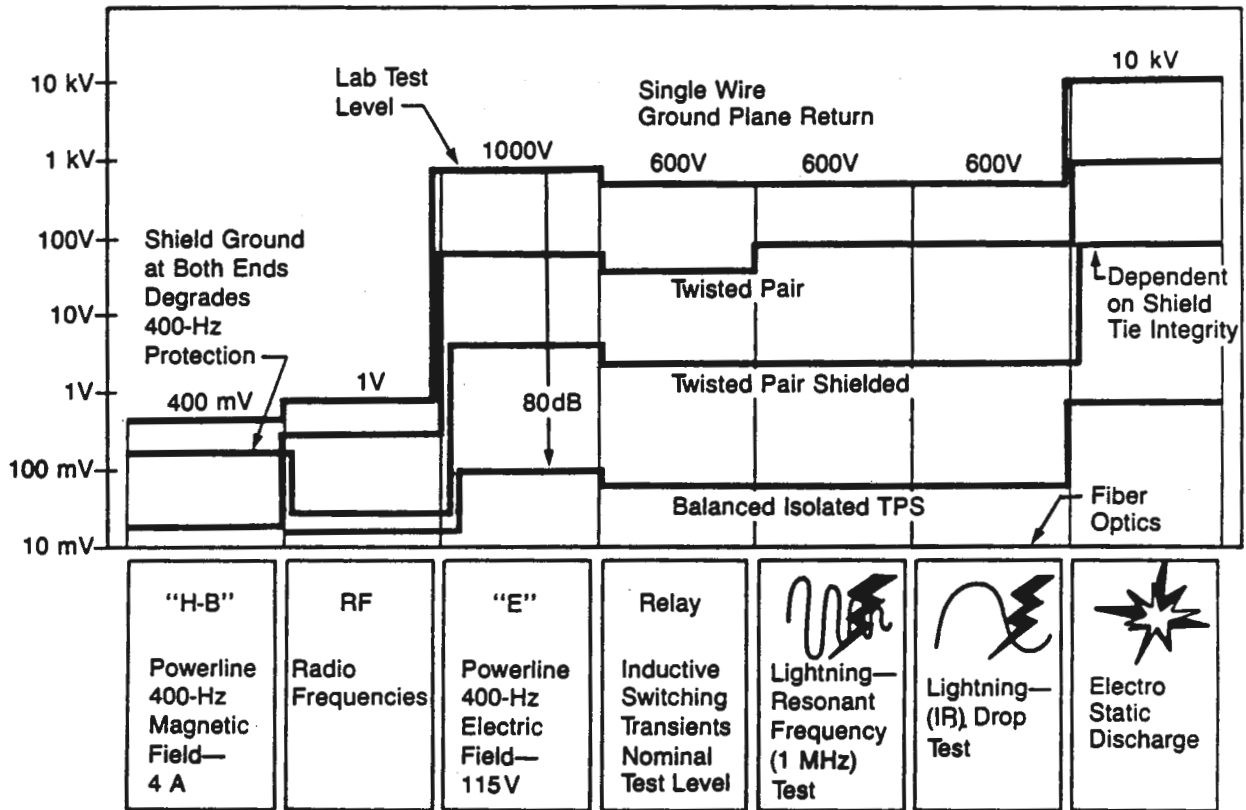


FIGURE 3.4-1. CIRCUIT RESPONSE TO EMI

The complexities of the mosaic could be largely dispelled through the simple use of a twisted-pair-shielded (or coax), balanced-isolated circuit design that reduces noise, and sustains signal quality. See in figure 3.4-1 how the twisted pair, shielded, balanced circuit has reduced noise to an acceptable level. This design provides a winning combination. It is the king of avionic interface design, boasting a possible noise rejection of 80 dB (10,000 times) and finding widespread use in digital communication or control circuits where stability, quality, and signal fidelity guarantee performance and confidence. Figure 3.4-1 was constructed to illustrate the general levels of improvement for comparison, but is not appropriate to be used or adapted to specific designs.

The building blocks to signal wiring quality are threefold: twisted pair for minimum magnetic field induction, down by 60 dB; shield for minimum electric fields, down by 30 dB or more; and balanced isolated design for a minimum common impedance, down by 40 dB. For future aircraft, it will be necessary to accurately model types of circuits on an aircraft to understand the induced noise levels and assure desired signal quality. Determination of the actual voltage, current, power, or energy on the wire circuit is the goal. Current and power become important when designing the rating or sizing of circuit protection.

There is no easy answer to avionic interface wiring design. Interface circuits can be built (by careful engineering evaluation, assessment, and construction) to offer optimum circuit compatibility through four fundamental approaches:

- Equipment and wiring location/grouping/organization design.
- Connector choice and wiring assignment.
- Return current rule.
- Coupling and modeling analysis.

Definition of the total circuit is required: the driver, the wiring, the receiver, operating frequency, and intercircuit connections. The following tasks are applicable: identification of all identical circuits; identification of all common circuits, common returns, every ground connection, capacitor, or resistor connection; determination of output and input impedances, balanced and unbalanced impedances to ground. A check of other current paths that are not intentionally designed is often appropriate, such as, circuit paths from ground wires through mutual capacitances (leakage capacitance), through other ground conductors or transformers, shields, and circuit card grounds.

Identical or common circuits are best assigned to the same connector. Circuit currents should exit and return in the same connector. Low-frequency signals, less than 1 kHz, may return in a well-signed aluminum or copper grounding structure where transmission line design techniques are not necessary.

These rules are constructive:

- Separate power and signal (if power must be in the same connector as signal, separate power and signal by ground pins).
- Separate families of circuits in individual connectors by frequency: audio, digital, pulse width modulation (PWM), video.
- Position wires for shortest practical route to other equipment.
- Assign connectors for optimum wire routing to other equipment.

Current paths returning in structure are designed to follow immediately adjacent to the cable (an image path). Currents should not be forced to take a wide path

through distant connectors and structure. This is sometimes called the return current rule.

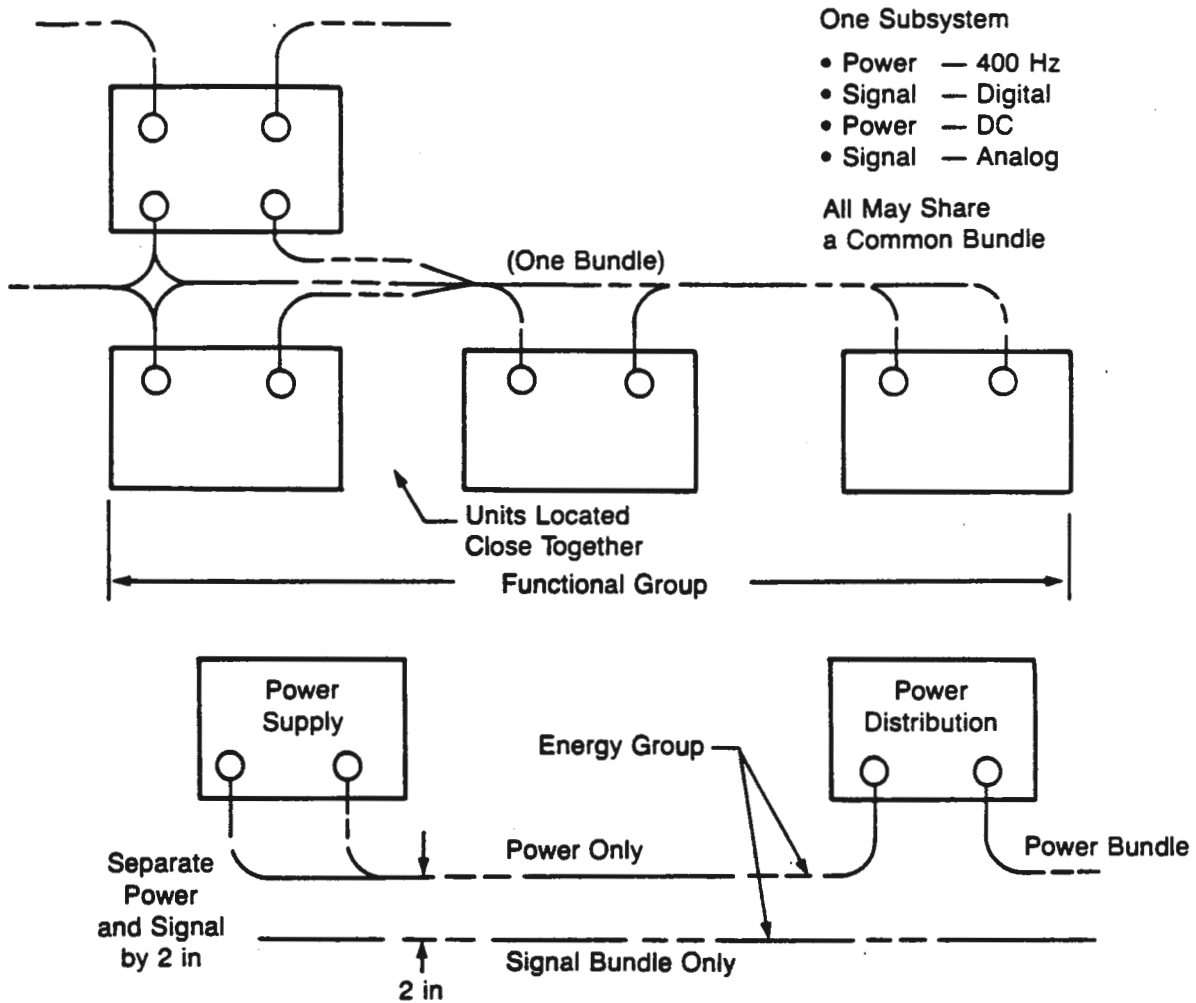


FIGURE 3.4-2. WIRING CATEGORIES

A simple way to achieve electromagnetic compatibility in wiring or in equipment is by functional or subsystem grouping (figure 3.4-2) - that is, keeping units or equipment of the same subsystem close together. An early definition and visibility of the system design, configuration, and function must be obtained. Equipment and subsystems and their locations must be known:

- Primary power, secondary power, distribution boxes, heavy switching (solenoids), lighting.
- Electronic flight control, electronic flight instrumentation, engine electronic controllers.

- Navigation, VOR, ILS, and DME.
- Radio transmitter/receivers, HF, VHF.

An early developmental wiring mockup is a requirement for good wiring design. The best design is grouping the subsystems close together in an in-line design. Generator to receiver, input to output, and source to load. Individual connectors can be dedicated with common groups of circuits. Functional grouping can often be extended in the aircraft between units that are widely separated. Here, engineering judgment and analysis is important. Where functional grouping is used, bundles are more easily separated to comply with redundancy and safety requirements or a separation of wiring to protect against physical damage to redundant critical equipment such as engine-mounted electronic controllers.

On long wire runs, functional grouping may result in excessive coupling between noisy lines and susceptible or sensitive lines. Where space is available, it is recommended that wiring be separated by "signal/energy" categories as follows:

- Ac feeder power bus; large ac control circuits; heavy current dc or secondary ac switching circuits valves, motor, or actuator drives; large inductive loads.
- Standard 115V and 28V power; signal circuits and regulated power circuits.
- Low-level, sensitive circuits such as audio, analog, dc reference, or dc secondary power.
- High-power radio frequency circuits (coax) or high-level pulse width modulated (PWM) circuits.

Wire separation is not the best way to reduce noise between circuits, but it is often necessary. Separation is very effective when used for isolation, redundancy, and safety (figure 3.4-3). Independent computer controls, instruments, power sources, or standby instrumentation are isolated to increase reliability. Left and right engine power lines and circuits, as well as engine indication signals, are separated and isolated. Communication, EICAS systems, ILS, LRRA, DME, and computer control devices, FMC, ADC, are separated and isolated. Primary controls (for example, roll, pitch, yaw, and stabilizer or spoiler controls as well as position sensors) are isolated.

3.4.2. Power and Energy Levels

A twisted pair shielded, balanced, isolated circuit, properly designed into the system as an interface circuit, minimizes the possibilities of noise-induced damage and establishes a basic high-quality design. But not all circuits are balanced and use a return wire; some interface circuits on the airplane have a ground return (structure return).

It is helpful and instructive in the initial design stage of a program to have a concept of the magnitudes of not only the noise voltage amplitudes but also the power and energy levels of radio frequency signals, 400-Hz power, and

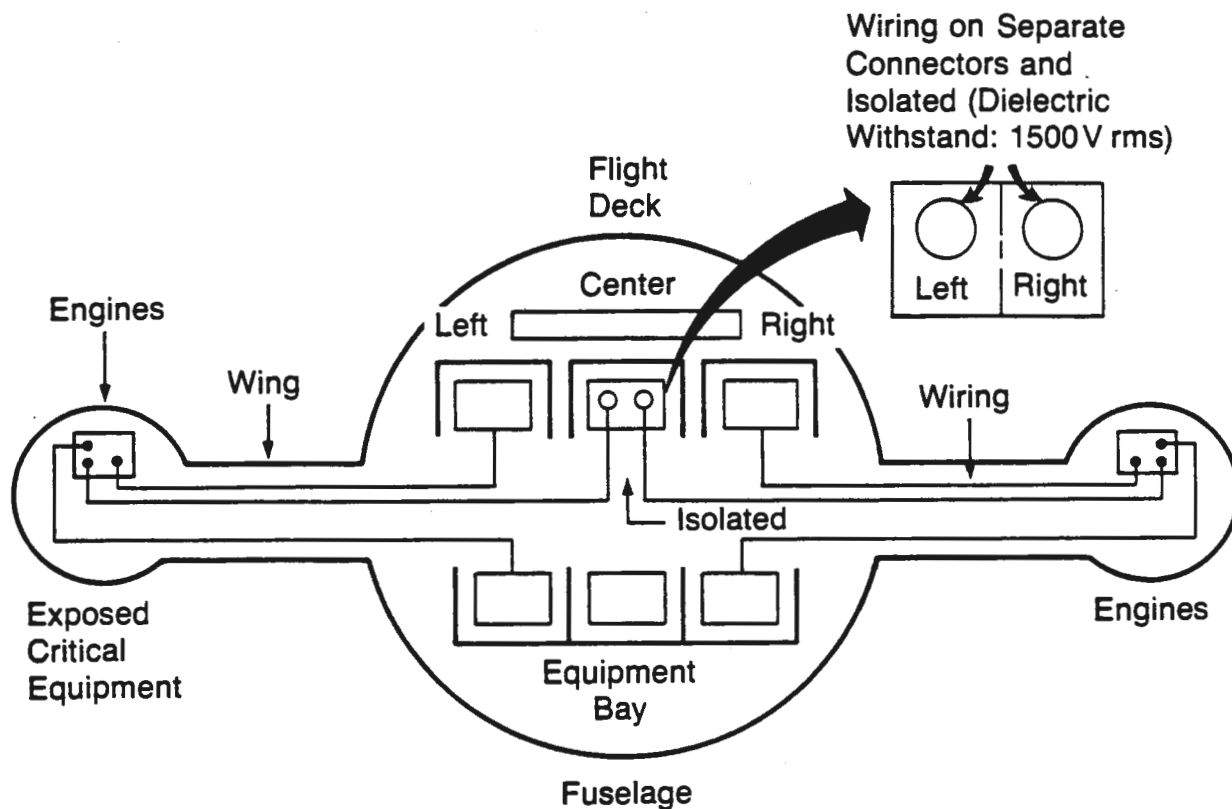


FIGURE 3.4-3. CRITICAL CIRCUITS WIRE SEPARATION

transient noise that can be imposed on a circuit which uses structure as the return path. It is important to analyze and define the power level (for continuous signals) and the energy levels (for transients).

One way to do this is by looking at selected electromagnetic interference laboratory test levels (figure 3.4-5). The test levels shown here are the avionic equipment test levels seen by equipment itself and are not the actual environmental magnitudes that may exist on the aircraft that the airframe manufacturer must consider, such as, higher levels of radio frequency signals or lightning transients.

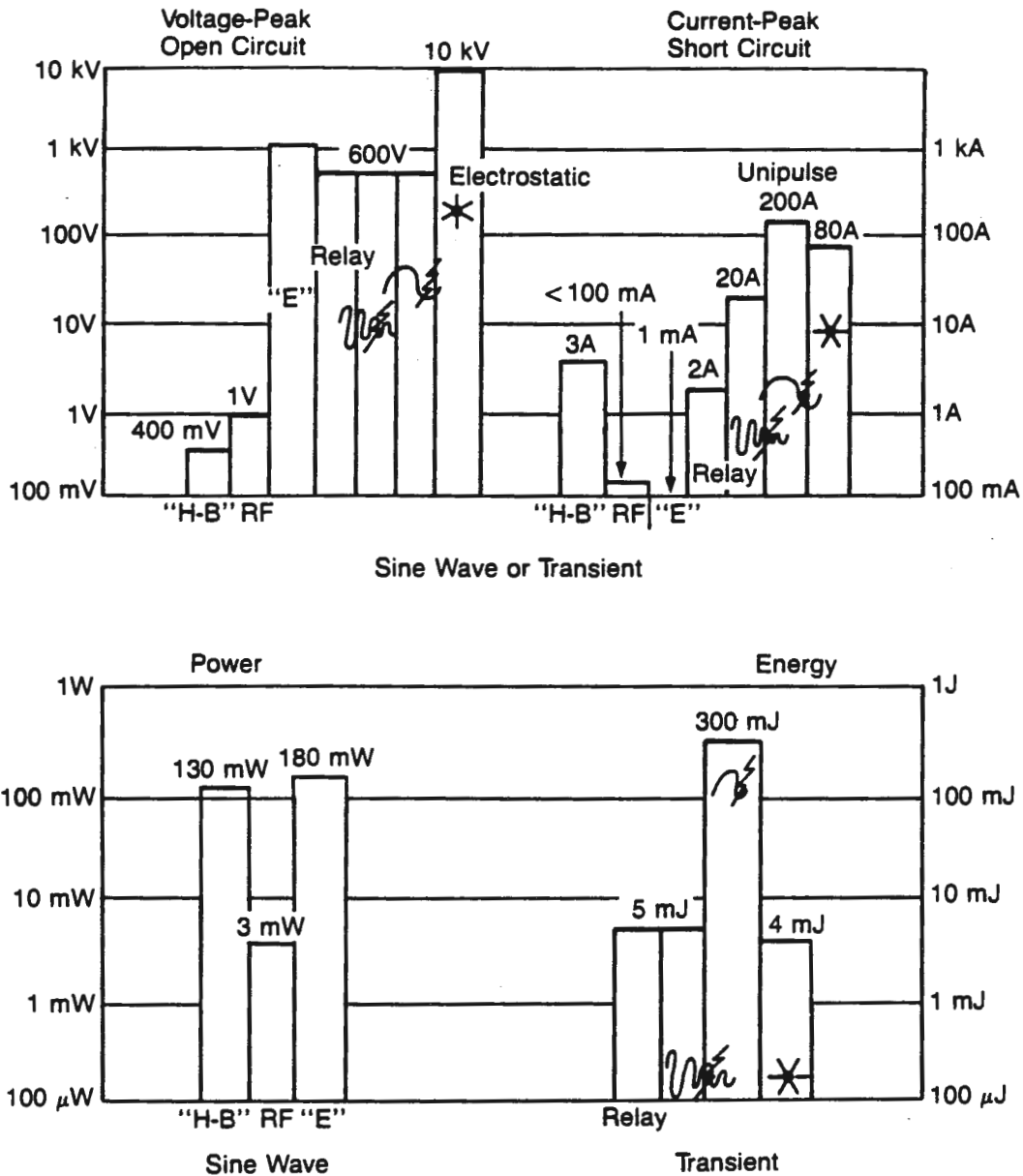


FIGURE 3.4-4. SELECTED EMI LEVELS

Seven electromagnetic interference types can be listed for the purposes of identifying, summarizing, and comparing their voltage-current-energy and power profiles. In these selected laboratory test setups, certain circuit parameters and conditions must be arbitrarily defined. Therefore, the levels shown in the figure are illustrative and helpful for making comparisons, but are not appropriate for detail design conditions and must not be used or adapted to specific designs. The electrostatic discharge test is not an industry standard

test level and is shown here for comparison to the other forms of noise. The amplitude is arbitrarily set at 10 kV, a nominal value out of an actual range of about 3 kV to 50 kV. Referring to figures 3.4-4 and 3.4-5:

- The radio frequency test induces a low voltage, 1V and a low current, approximately 100 mA, on a single wire that has a ground plane return. Observe on the bar chart the radio frequency voltage level of 1V and then the radio frequency current level of less than 100 mA. Now move down to the power bar chart and see the radio frequency power level of 3 mW. The power level is shown because it is a continuous signal. This test is performed in the laboratory using a current probe to induce the radio frequency voltage measured as an open circuit voltage (figure 3.4-5). The test level here is chosen as 1V (in the HF-VHF range) induced into a wire that is 5m (15 ft) long. The calculated current is the short circuit current and the load resistance is adjusted for maximum power.

These radio frequency signal products come from aircraft microprocessor clock and switching regulator stray noise signals as well as HF-VHF aircraft communication, HF-VHF television, and FM radio broadcast transmitters. Radio frequency carries with it the ability to sneak through avionics internal circuit capacitors or diodes connected to power supplies, and on a circuit card it will pass from circuit trace to trace. The radio frequencies are not damaging, but they can alter circuit operation and performance.

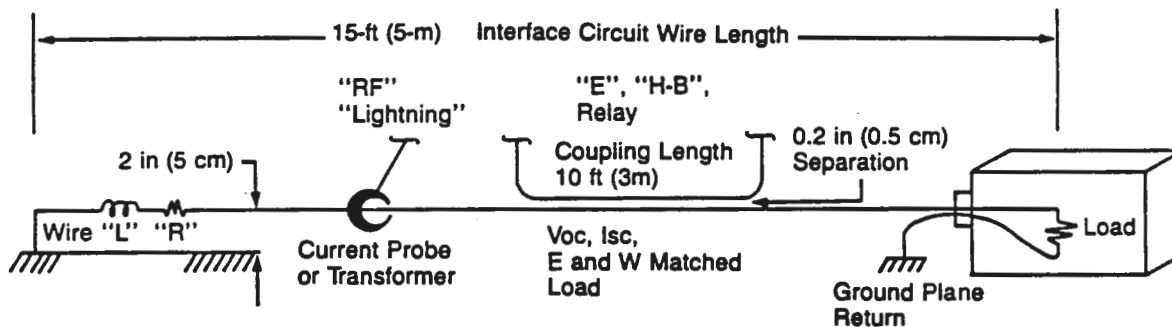


FIGURE 3.4-5. LABORATORY TEST SETUP

- The power line 400-Hz magnetic field, designated as "H-B" on the chart, delivers a low-frequency effect into analog circuits or operational amplifiers. Mark on the chart how the magnetic field might impose 400 mV with a power of 130 mW.

For this laboratory setup, a 40A current is coupled into a 3m (10 ft) section of wire, with a 0.5-cm (0.2-inch) separation, and positioned 5-cm (2-inch) above the ground plane. This simulates a 4A, 30-m (100A-ft) maximum coupling condition on an aircraft. The power is calculated for a load resistance equal to the 5-m wire resistance.

- The power line 400-Hz electric field ("E") also delivers a low-frequency effect into analog circuits or operational amplifiers. See how the electric field ("E") is about 1000V, but note the extremely low current, less than 1 mA (off the chart). The power level is low but easily large enough for upset or alteration of equipment performance. Notice here that the power line test is performed at 10 times the 115V line level, or 1150V, in order to use a 10-ft coupling length. This simulates a 115V 100-ft maximum coupling condition on an aircraft.
- The electrical switching transient test (relay induced transient) may impose voltages of 600V or higher with currents of just a few amps and energies in the range of millijoules - large enough to damage sensitive solid state devices. For this test, the electrical parameters were estimated with the following values chosen: 600V peak-to-peak voltage; 2A current; a 1-MHz damped sine wave, repeated 1000 times to simulate a transient of 1-ms total duration. This is an arbitrary electrical switching transient for illustration only.
- The 1-MHz lightning-induced transient (designated by the damped sine on the bar chart) is a damped cosinusoid, has a set 600V amplitude, is transformer-induced into the wire, has an initial fast risetime of about 100 ns (3 MHz), and is to simulate lightning resonance on the aircraft. The induced current is high but the energy is fairly low.
- The so-called "ground potential test" (designated by the pulse waveform on the bar chart) is a 600V 10 μ s unipulse or "double exponential" waveform to simulate a controlled lightning induced current transient in structure. Observe on the figure the dramatically higher current, 200A, and much higher energy of the unipulse. The unipulse is a very powerful noise source. It carries a considerable amount of energy and can easily damage and destroy electronic components. A balanced circuit design or substantial protection is needed to divert or block this transient. The test setup is different from that shown in the figure. The 600V waveform from the transient source is pre-established on a 5 Ω load, then the voltage is applied with a "source" transformer connected between ground and the avionics case, housing (and any signal return) where the resulting waveform will vary depending on avionic equipment circuit loading and wiring design. Any circuit using structure as return will receive the full impact of this transient.
- The profile of the electrostatic discharge transient (designated by the "star" on the bar chart) shows an exceedingly high-voltage pulse of 10 kV, which is capable of causing dielectric breakdown of insulation rather than the usual condition of thermal burnout of a semiconductor or microcircuit. The short circuit current can be high but the width or duration of the pulse is so narrow or short that it results in a low energy level. The application is different from that shown in the test setup figure. The transient is applied directly to the circuit wire or connector pin. These electrical parameters apply: 150-pF, 50 Ω source through a 1- μ H, 1-m wire to a simulated 50 Ω load for maximum power transfer.

3.4.3. EMC Quality in Maintenance

Structural shielding, wire shielding, and interface circuit protection must be maintained through the life of the aircraft. Here are some of the most important items:

- Keep the controlled wire routing and the wire separation design intact. The original design of wiring is formulated to ensure that faults will not propagate, that critical functions are redundant, and that electrically noisy circuits are separated from vulnerable circuits.
- Concentrate on maintaining short shield ties to the structure or to "line replaceable unit" (LRU) box if structure is graphite-epoxy. Shield ties or "pigtails" do not usually receive the attention they warrant. Shield tie length is extremely important in the effectiveness and quality of shielding. A shield tie that is short, less than 2 or 3 inches, is highly desirable; longer lengths, 6 inches or more, degrade the entire shield. The best termination is to the connector backshell. Future connectors will have backshells and filter pins that will need maintenance.
- Maintain electrical bonding and grounding quality. Careful surface preparation, proper joining techniques, care of bonding straps, and finally adequate conductive sealing of joints and seams ensure the continuation of the excellent shielding and electrical grounding provided by structure.
- Be aware of electrostatic discharge. Electrostatic discharge is a key intruder in handling and maintenance procedures. Microcircuits may be impaired or destroyed by a pulse from a hand or an item of clothing. The event can go unnoticed. Conductive materials and grounding procedures, along with a training program, will provide techniques for failure prevention.

3.4.4. Shielding and Shield Ties

Shields may be single braid; double braid; braid and foil; shields inside an overall bundle shield; solid conduit, tray, or cableway; and aircraft structure.

The shielding effectiveness (SE) of shields is based on materials, dimensions, circuit connections, impedances, and shield tie lengths and has been one of the most elusive electromagnetic compatibility protective defenses to pin down. (A significant issue for future aircraft needing to be addressed is that of the wiring lengths installed during laboratory tests versus the actual aircraft installed length.) Length influences SE and SE can dictate length or design. A single braided shield with a 2-inch shield tie may offer less than 25 dB of protection over the span of 10 kHz to 100 MHz. A long wire, 100 ft (30m), may only show 10 dB above 1 MHz at some resonant frequencies under certain conditions. A solid 360-deg connection to a backshell can improve protection. Conditions that establish the grounding of shields vary, but there are some that need emphasis:

- Ground audio or analog shields at receiver end only.

- Ground digital or wideband signal circuit shields at both ends.
- Ground shields subjected to high frequencies (greater than 50 kHz) at both ends.
- Ground shields that contain or are a barrier to transients at both ends.
- When audio and HF requirements conflict then the circuits and installation must be evaluated. (Solution may be rerouting of wires or double shielding.)

Carrying a shield tie through an avionics unit connector, into the internal wiring harness, and to a circuit card connector is poor wiring practice. A 360-deg, peripheral shield connection to the backshell is the best (figure 3.4-6).

3.5. Aircraft Protection Measures

3.5.1. Structure Conductivity

Conductivity is the predominant electromagnetic compatibility consideration of materials in the basic steps to a unified aircraft structure design (figure 3.5-1). What are the separate electrical functions so highly dependant on conductivity (figure 3.5-2).

- Electrical stability: A low noise ground reference plane - a stable zero reference foundation for electrical and electronic circuit and shield ties (may have less than 500-mV ground noise). This electrical "ground" embodies structure, shelving, skin, spars, equipment chassis, and possibly uniquely installed grids, sheets, and foil.
- Shielding: Aircraft structure, skin panels - foils, flame spray, plating, paint-shelves, equipment enclosures, and wire shields. Shielding affords a barrier to external and internal radio frequencies, 400-Hz electric fields, and 600V transients.
- Fault path: Structure, skin panels, cable shields, safety wiring (green wire). The engineered fault paths divert currents to assure safety of passengers and personnel, prevent hazardous voltages, avoid ignition of combustibles (fire prevention), and limit equipment failure and upset.
- "Diverter": Structure, skin panels. The aluminum aircraft inherently offers the current control paths and bypass to eliminate shock hazard and damage from electrostatic charge and lightning.
- Signal return and power return: Cost and weight savings accrue through the use of structure as a return instead of the installation of wire.
- Reliability and redundancy: Parts of the aircraft may be employed as a baffle or wall to provide separation of wiring or equipment.

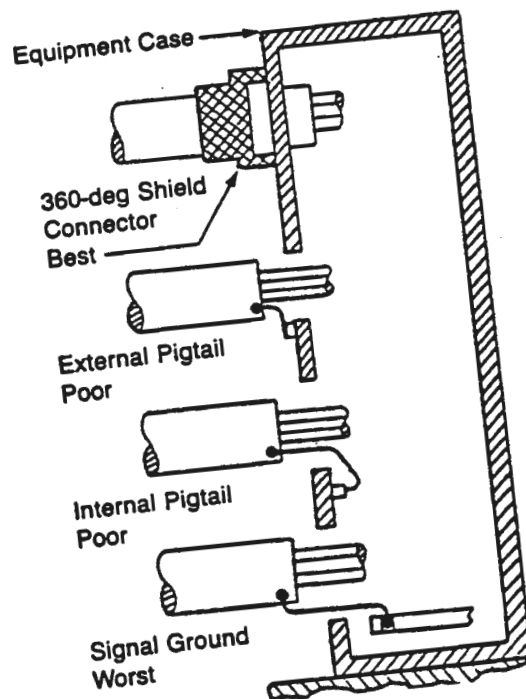


FIGURE 3.4-6. SHIELD TIES

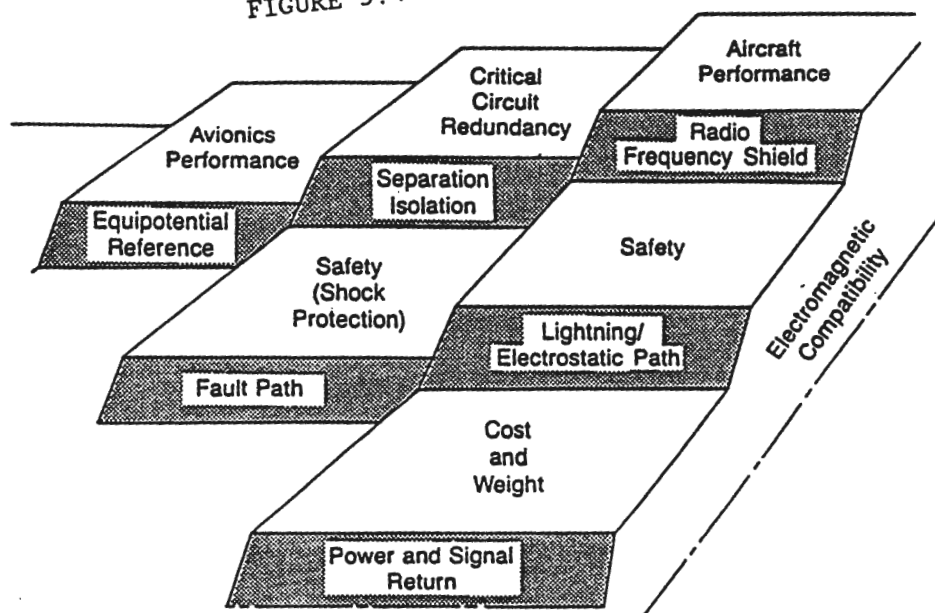
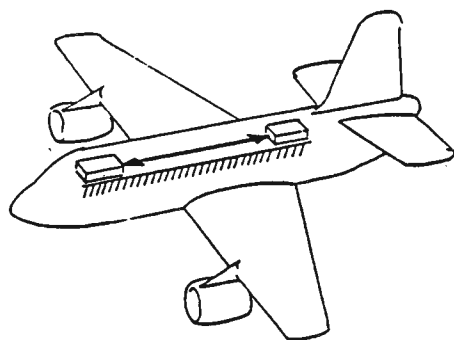
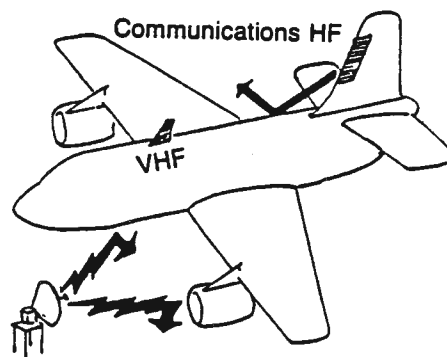


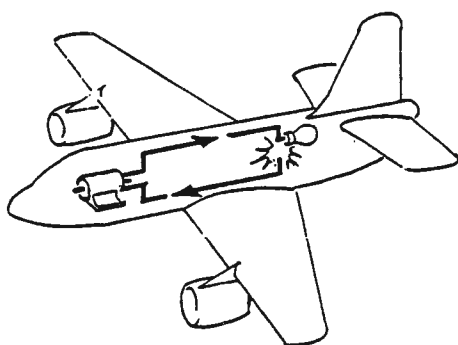
FIGURE 3.5-1. THE GROUNDING STEPS



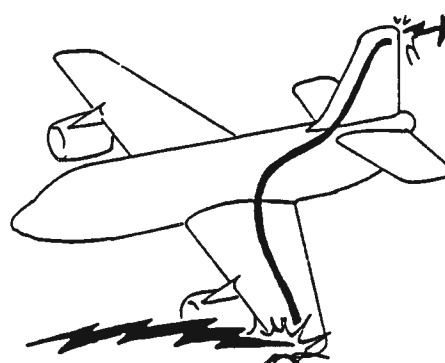
**Equipotential Ground Plane
Reference (Stability and
Performance)**



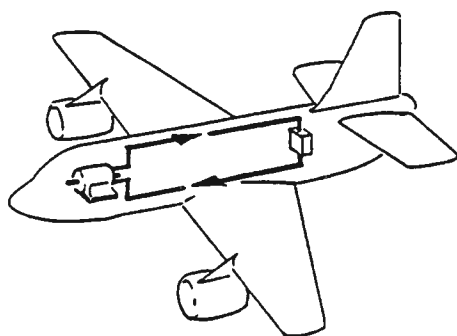
Radio Frequency Shield
• Radar (Critical
• Radio or Equipment
• Television Operation)



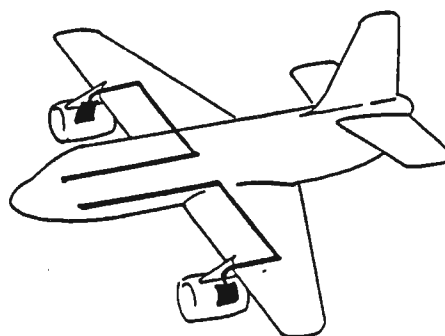
**Safety Fault Path
(Passenger and Personnel
Protection)**



**Lightning Diversion Path
Electrostatic Drain
(Passenger Safety and
Equipment Protection)**



**Power and Signal
Return Path
(Weight and Cost Reduction)**



**Separation-Isolation Redundancy
(Critical Equipment Operation)**

FIGURE 3.5-2. STRUCTURE EMC ROLES

The engineering of the structure, etc., to accommodate all of the electrical functions, entails detail design of electrical interfaces, bonding straps, foils, paints, etc. (figure 3.5-3). Bonding resistance tests can be made during the aircraft EMC test (see section 7).

It is illustrative and instructive to compare an aircraft with an electronics facility to help recognize the significance of the aluminum structure as a substantive electrical component (figure 3.5-4). Except for framing, many of the facility building materials are not conductive.

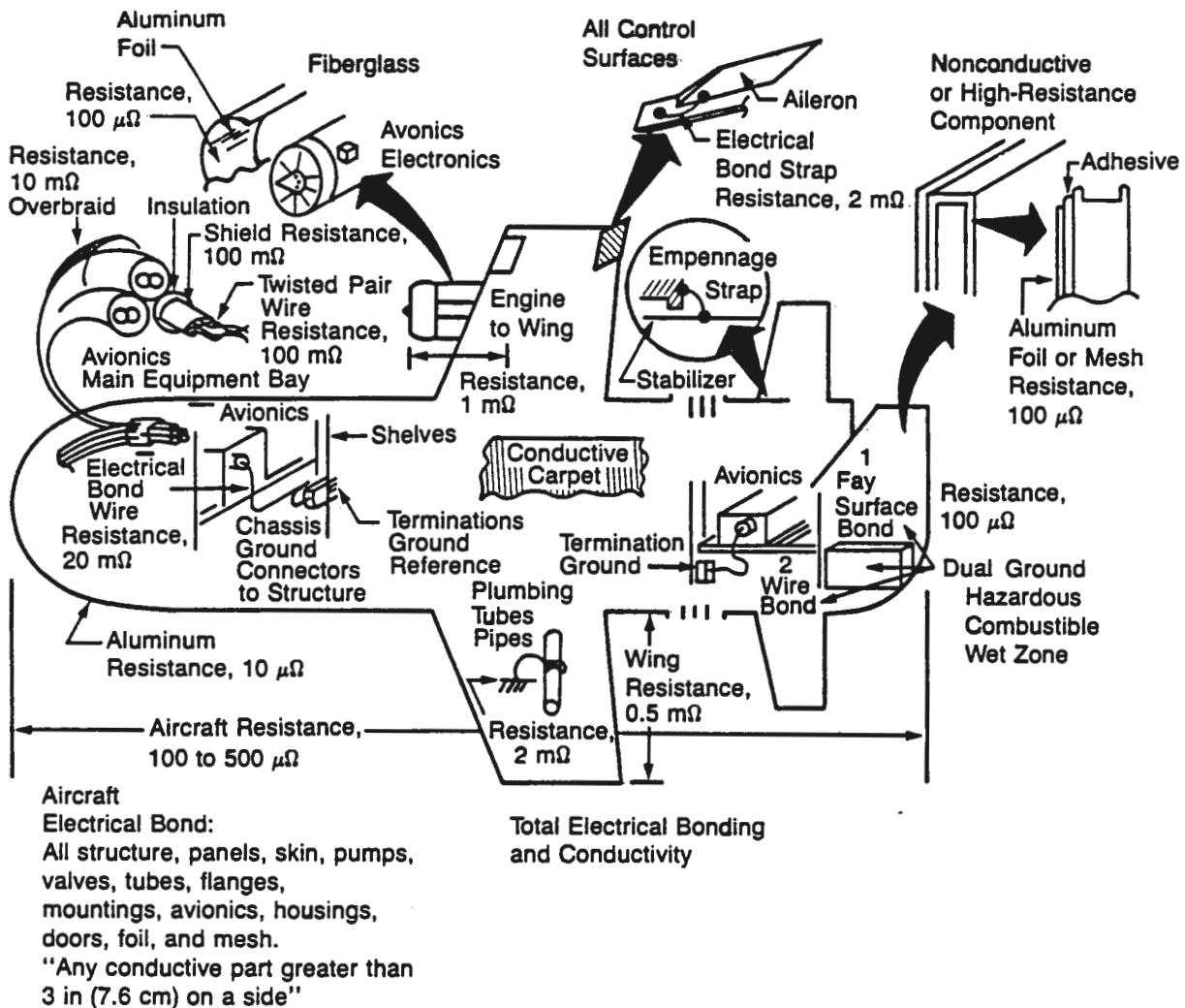


FIGURE 3.5-3. DETAILS AND INSTALLATION

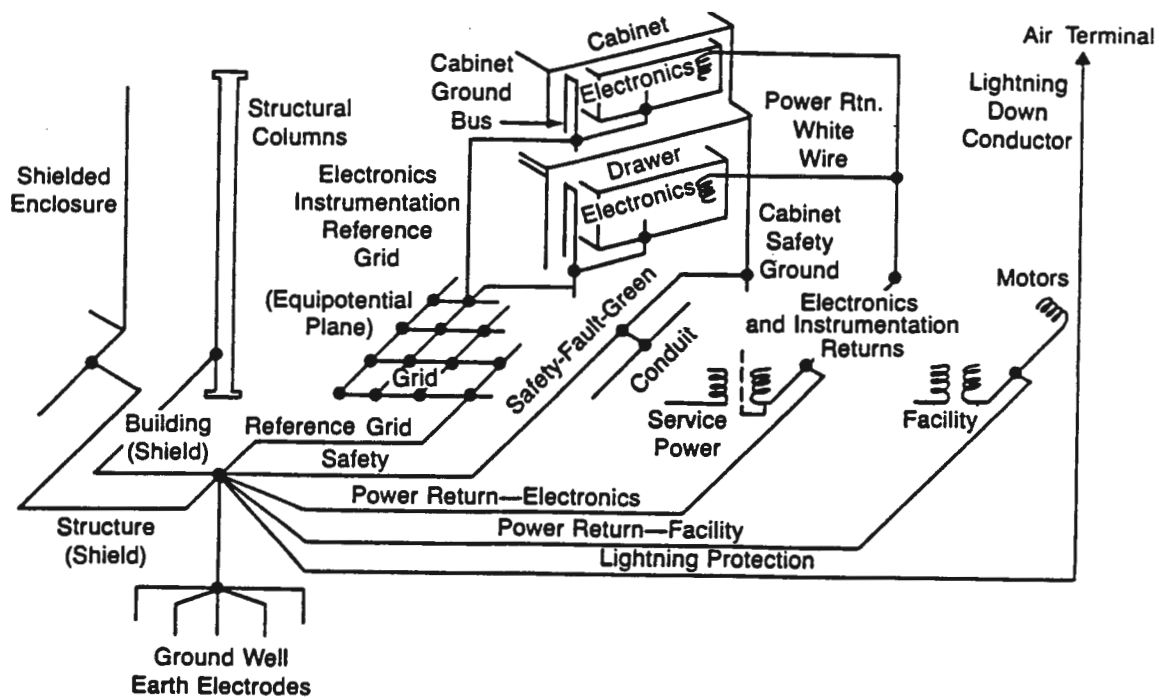


FIGURE 3.5-4. THE SEVEN EARTH GROUND CONNECTIONS

The electrical functions so freely supplied by the airplane structure must be built into a properly constructed facility using extra materials and supports. Whereas the airplane unifies the functions, in the facility they are separate (figure 3.5-5).

3.5.2. Shielding

Aluminum, after copper, is one of the best electromagnetic shields. Its effectiveness varies with thickness and frequency. One or two thousandths (1 or 2 mils) is good (figure 3.5-6).

Wherever shielding, grounding, or conductive properties are lost at joints and seams (figure 3.5-7) or are not available from structure (therefore compromising continuity), shielding must be added. Aluminum foil, flame spray, plating, and paint are candidate solutions (figure 3.5-8).

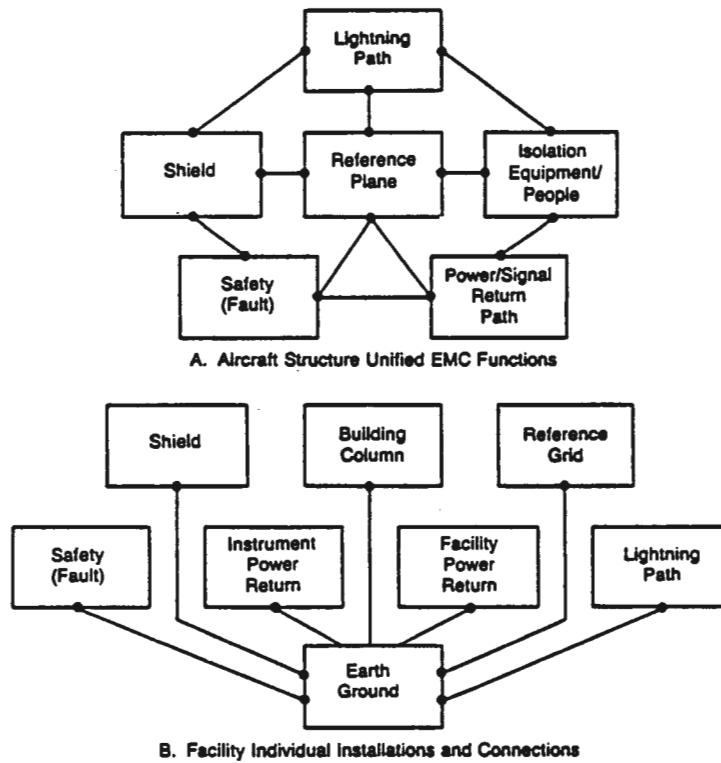


FIGURE 3.5-5. EMC ANATOMY BLOCK DIAGRAM

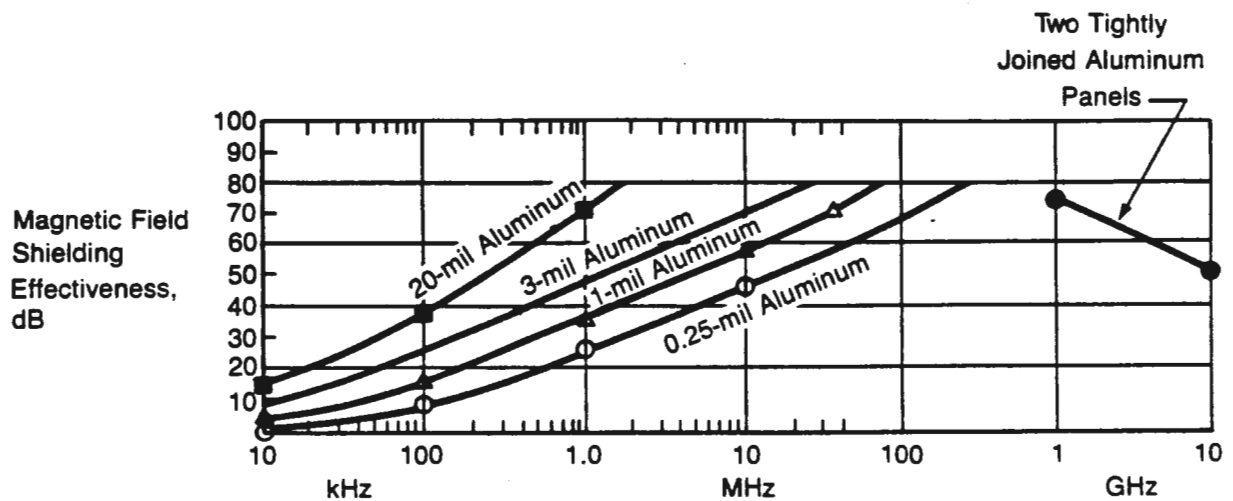


FIGURE 3.5-6. MAGNETIC FIELD SE OF ALUMINUM

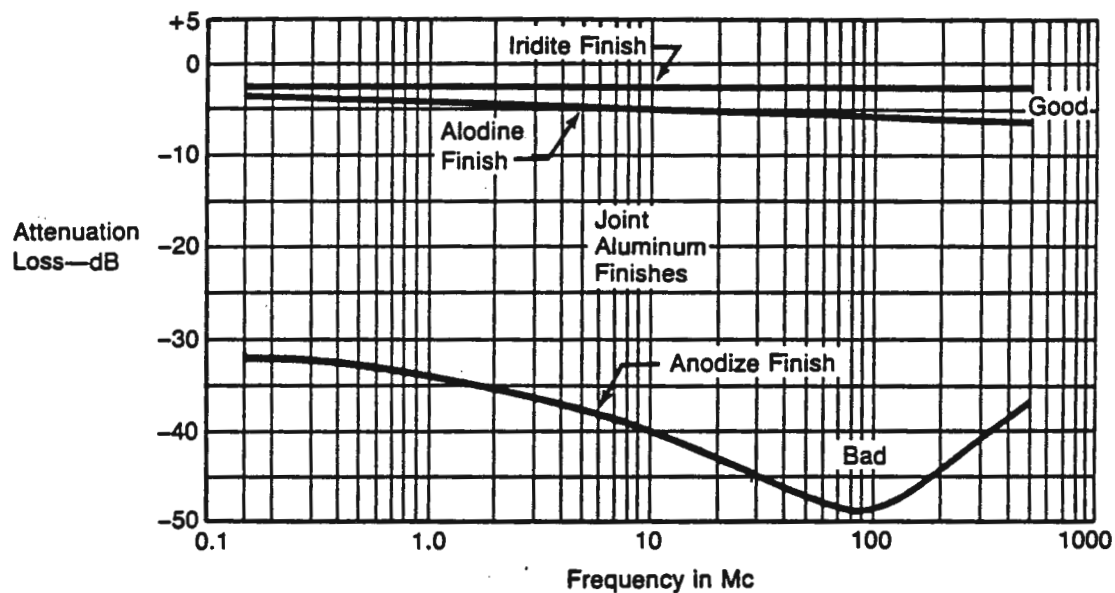


FIGURE 3.5-7. LOSS OF SE WITH JOINT FINISH

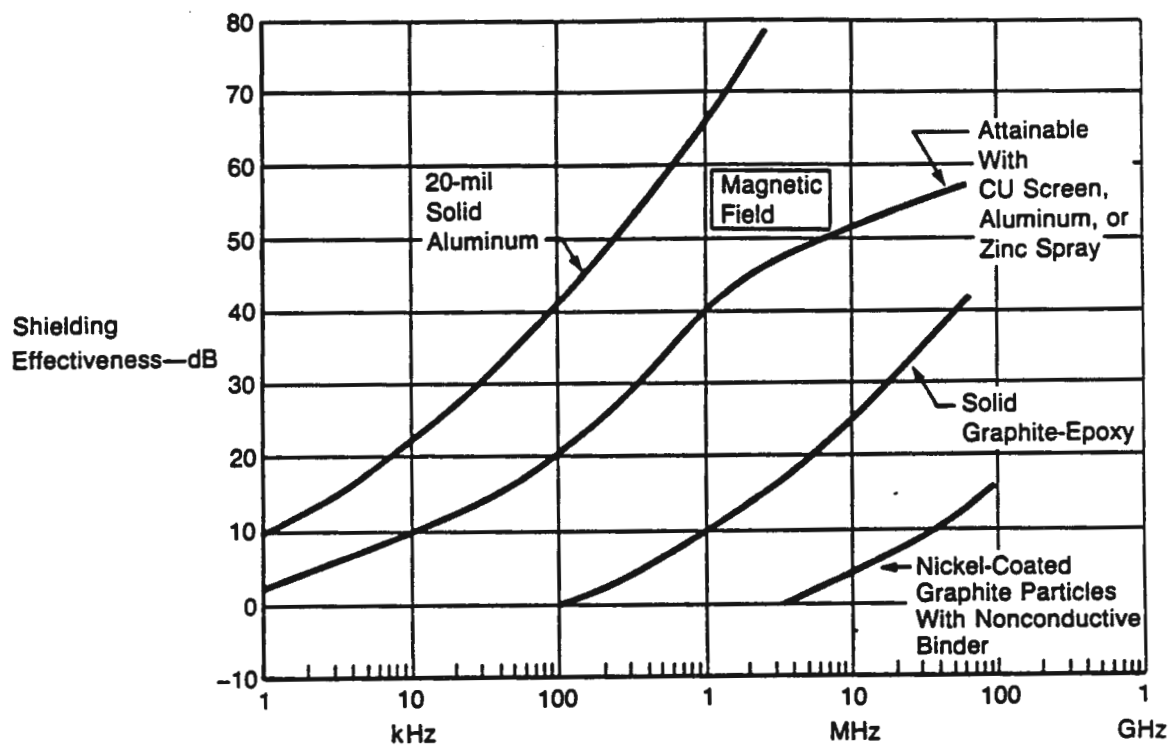


FIGURE 3.5-8. SE COMPARISON

Foil (aluminum or copper) is a good shield, conductor, and reference plane. Foil may bring with it an increased effort to establish quality of bonding to adjacent parts, reliability under environment and vibration, and integrity and durability. Moreover, the foil size, thickness, and geometric configuration dictate its economy of installation. Foil can make a good wire shield.

Flame spray applications (aluminum, copper, or zinc), even by skilled operators, can be difficult to apply in a controlled fashion. The resistance of flame spray coatings is higher and shielding effects lower even with greater thicknesses than foil. Its dense coat and coarse finish on a substrate can lead to flaking or cracking under vibration and moisture conditions with obvious loss of qualities.

Paints (metal or containing conductive agents or fillers) provide good shielding and are used successfully. Insulating or moderately resistive material (graphite-epoxy) may be painted to provide shielding or conductivity or they may be overlaid with aluminum mesh or foil.

Wire shielding provides another layer of protection (figure 3.5-9). Aircraft structural shielding is an important companion to wire shielding. The levels of shielding effectiveness are computed for each wiring run location and configuration to determine the accurate values and these can then be combined with modeled and calculated values of structural shielding. Figure 3.5-10 shows some SE levels for silver or gold film that might be deposited on glass and aluminum or copper screen that could be for shielding ventilation ports on equipment. The front and rear spar areas exhibit very poor shielding. The figures depict nominal or typical estimated levels. Shielding exhibits such wide spreads and variations around these levels (dependant on conditions), and especially as frequency varies that these summary representations are useful as a tutorial tool, but are not appropriate as a design tool.

3.5.3. Safety in Grounding and Returns

Structure is a ground and a return. Seven separate grounds or returns on electrical/electronic equipment can be identified. Some of these ground/returns are interconnected and perform common functions. Certain individual grounds are always separated to avert mixing noisy and sensitive circuits.

- Case enclosures or housings (electrical/electronic equipment) ground: May be a wire or the case surface. This case enclosure ground acts as a safety ground (fault return), a static ground, and a radio frequency ground. It sometimes is a signal return and possibly a 115V, 400-Hz return.
- 115V, 400-Hz return: Isolated and brought out of the equipment on a separate wire (figure 3.5-11). A twisted pair power supply circuit will improve electromagnetic compatibility.
- 28V, 400-Hz return.
- 28V, dc return.

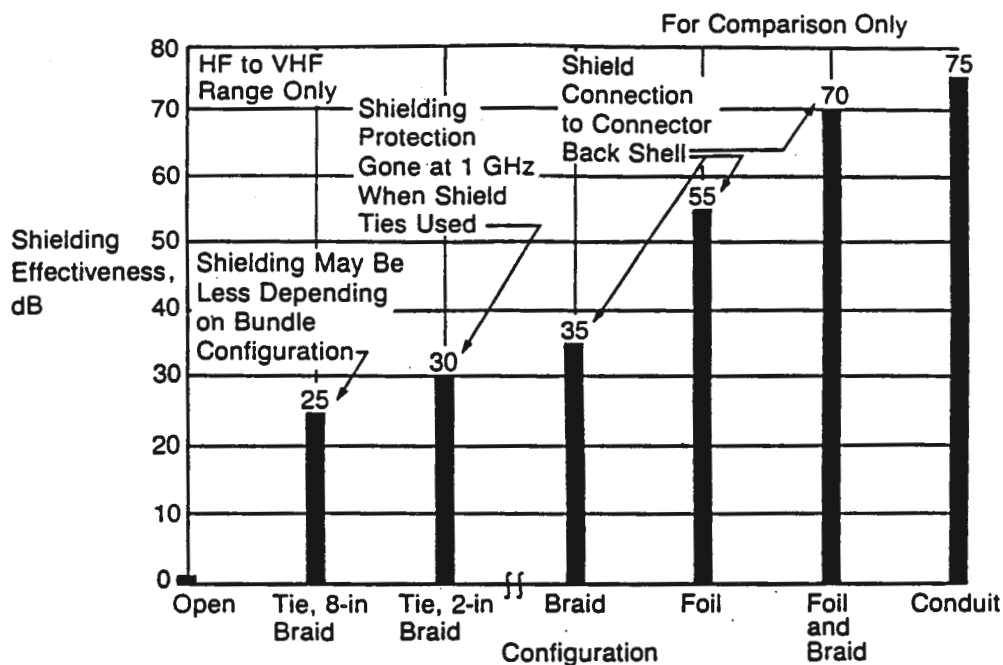


FIGURE 3.5-9. WIRE SE COMPARISON AT HF-VHF

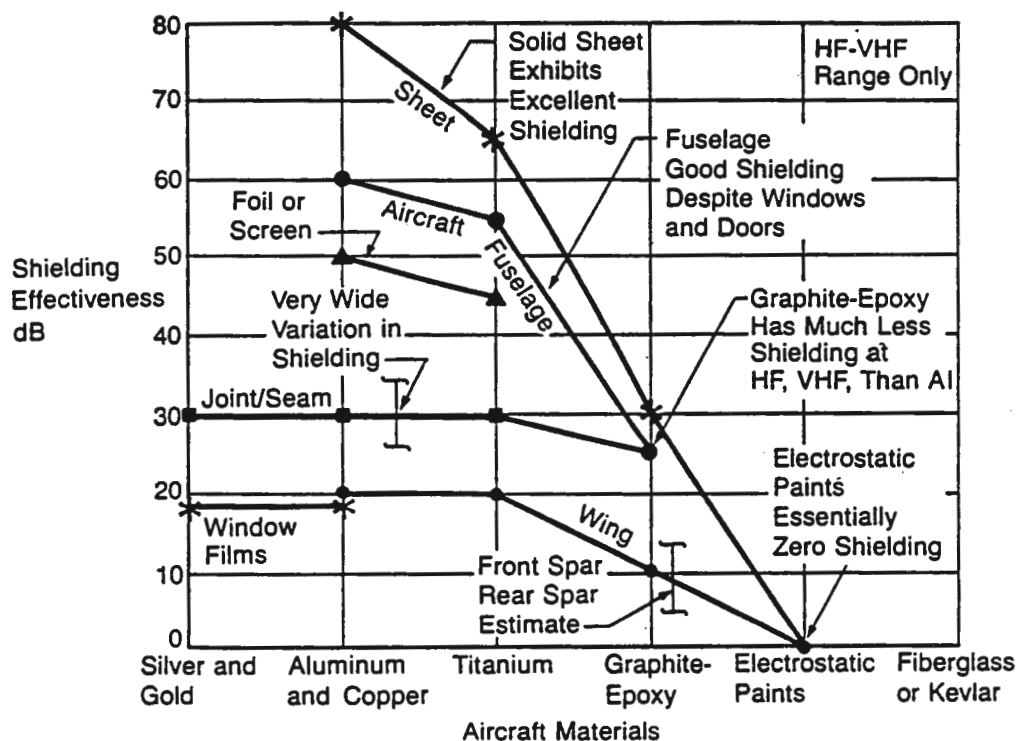


FIGURE 3.5-10. MATERIAL/CONFIGURATION SE OVERVIEW

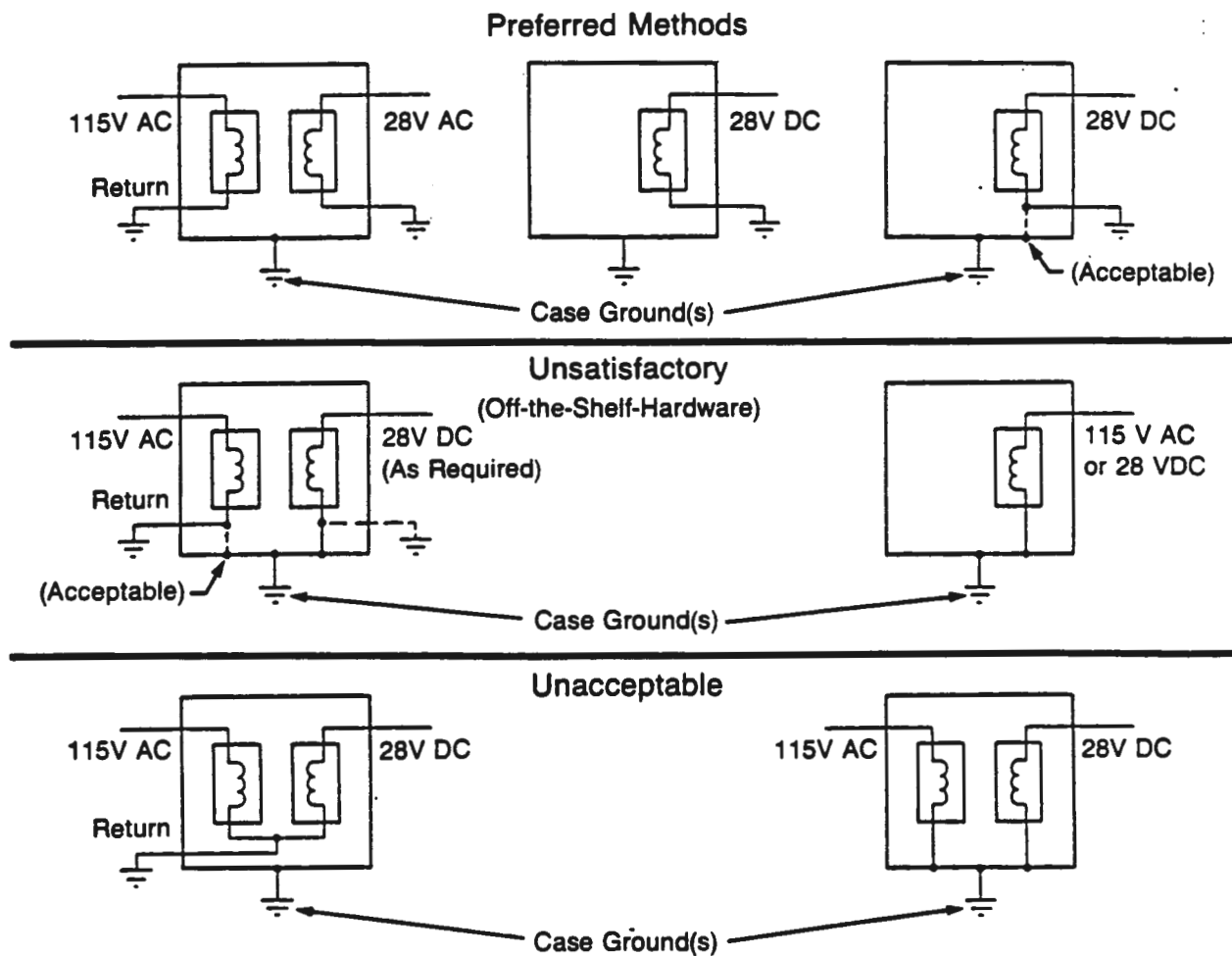


FIGURE 3.5-11. POWER RETURNS

- Audio or analog circuit return.
- Digital circuit return.
- Shield ties (pigtails).

Certain case enclosures require dual grounding in a flammable leakage zone or water exposure area, possibly around wing, cargo, and tail locations, to offer redundancy and safety in case of electrical fault. Power returns are brought out of these zones before being terminated to structure. Each circuit is exhaustively reviewed for voltage, current, resistance, and potential voltage drop at each connection. MIL-B-5087 gives resistance limits of $6\text{ m}\Omega$ to $11\text{ }\mu\Omega$ maximum when computed fault currents are 50A to 7 kA.

3.5.4. Resistance

Two and one-half milliohms (0.0025Ω) is the most often quoted resistance maximum for a termination or connection when designing for electromagnetic compatibility. It is applied to wire terminations, case enclosures, mounts, shelves, panels, doors, and radio frequency components. The $2.5\text{-m}\Omega$ limit is appropriate wherever any component or structure forms a part of the ground plane, shielding, fault path, or power or signal return.

Panels, rails, frames, access doors, and all conductive items in the flight deck are bonded especially to reduce electrostatic discharge.

Pumps, valves, flanges, lines, vents, and penetrations in the fuel tank are bonded especially to prevent arcing from lightning. Transport aircraft undergo thorough research and assessment of conductivity or resistance. It is sometimes difficult to attain $2.5\text{ m}\Omega$. The resistance of aluminum is a good basis for comparison of other materials (figure 3.5-12 and figure 3.5-13).

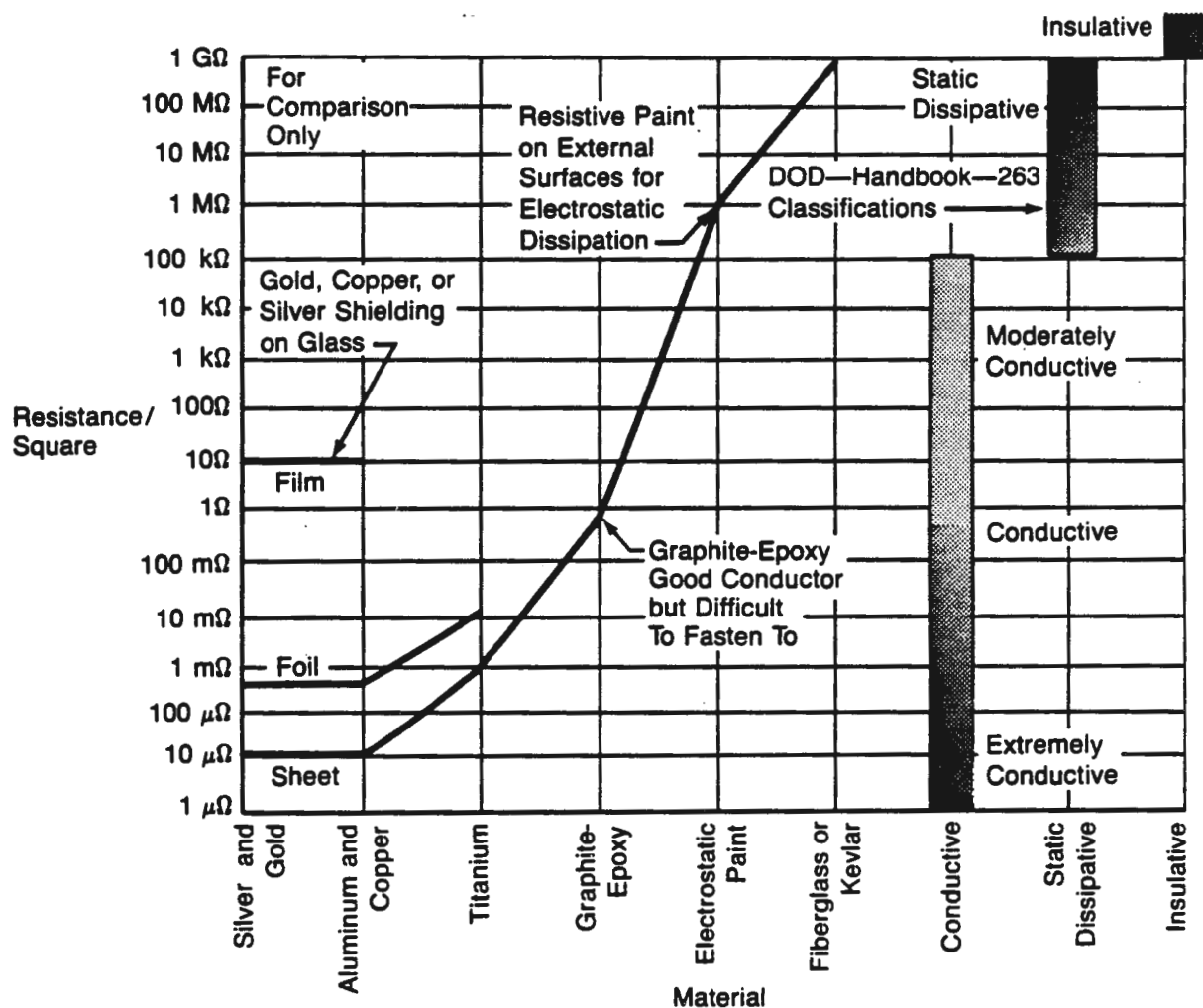


FIGURE 3.5-12. MATERIAL/RESISTANCE OVERVIEW

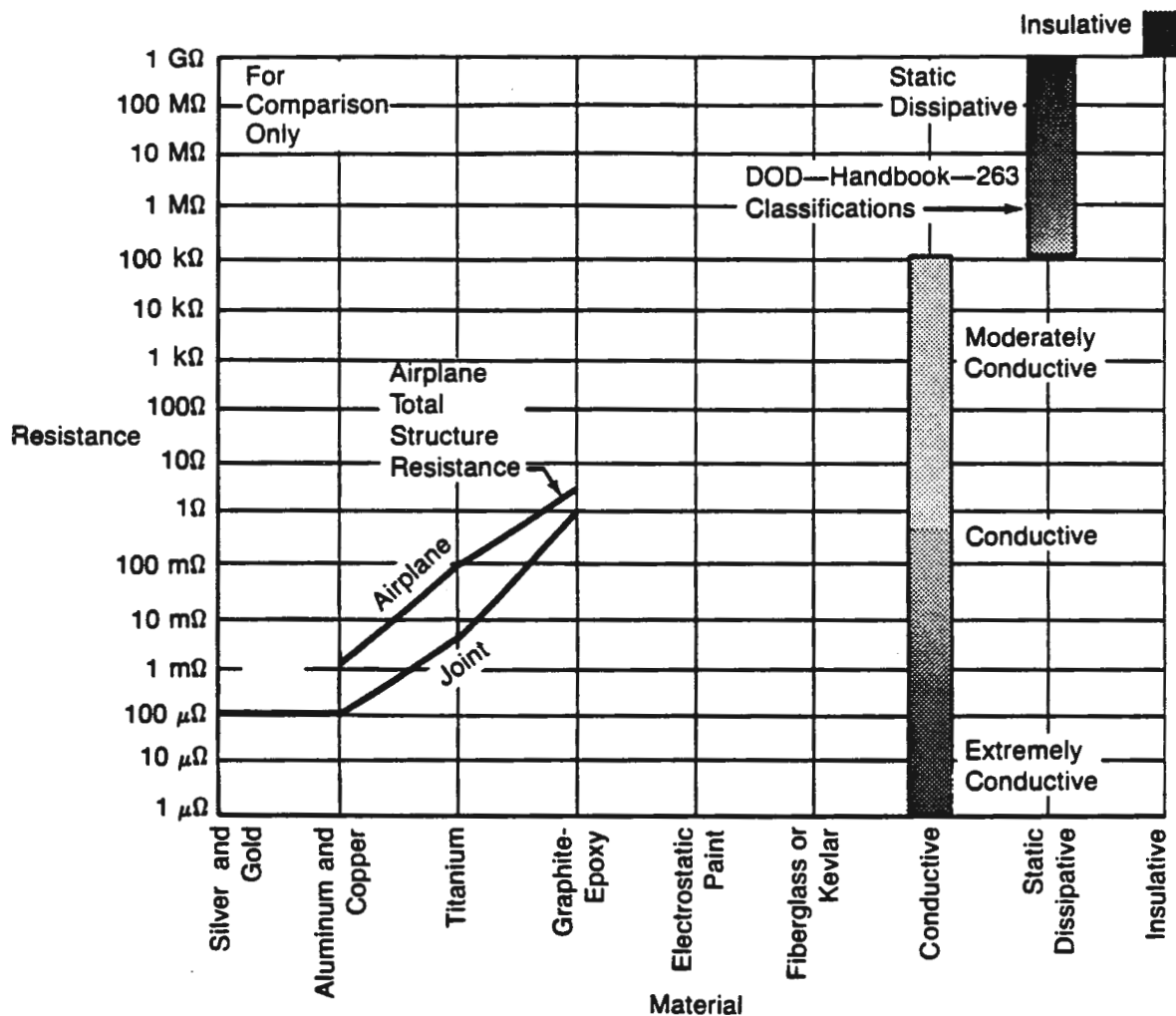


FIGURE 3.5-13. CONFIGURATION/RESISTANCE OVERVIEW

Ground planes establish the electrical foundation for any system. Copper is practical, affordable, and is usually employed in the electromagnetic compatibility laboratory (figure 3.5-14). Currents flowing in copper have a low IR drop and therefore provide a low noise system. Aluminum usually offers good conductivity through faying surface bonds or permanent joints. A number of factors affect resistance: pressure and surface finish are significant (figures 3.5-15 and 3.5-16).

When components do not form a part of the electrical design but must be grounded for safety, a 1Ω resistance is often adequate. Where electrostatic charge buildup is of concern on nonconductive materials such as on external dielectric surfaces, the resistance limit may be allowed to be much higher. Military Handbook 263 gives $500 \text{ k}\Omega$ to $100 \text{ M}\Omega$ as the range to adequately drain away electrostatic charge.

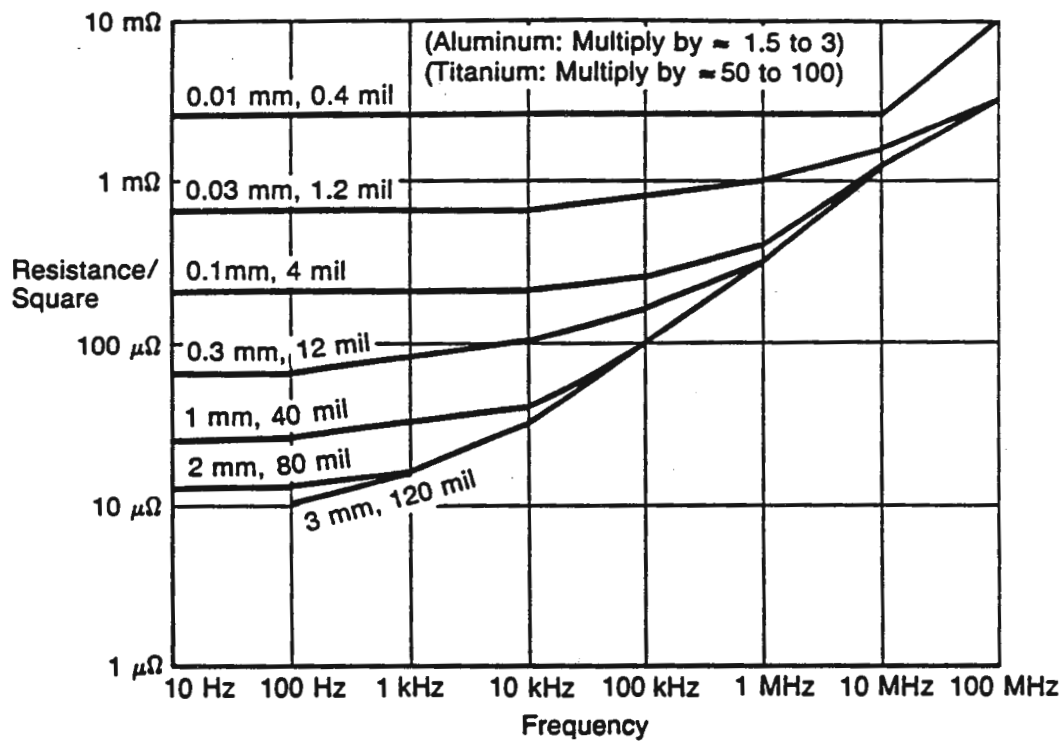


FIGURE 3.5-14. RESISTANCE PLOT - COPPER PLANE

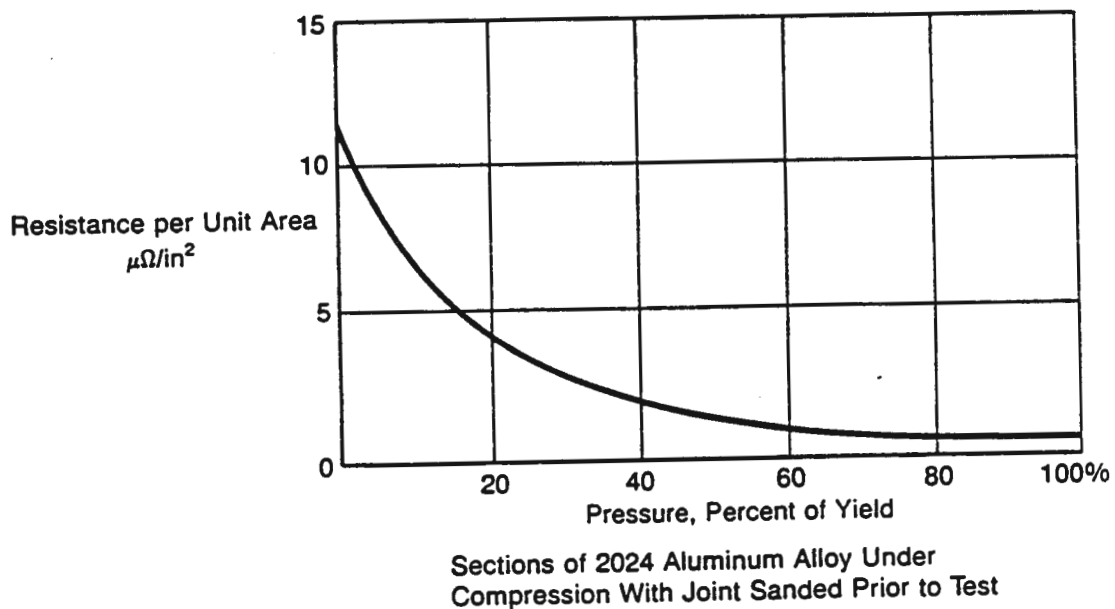


FIGURE 3.5-15. RESISTANCE VERSUS PRESSURE

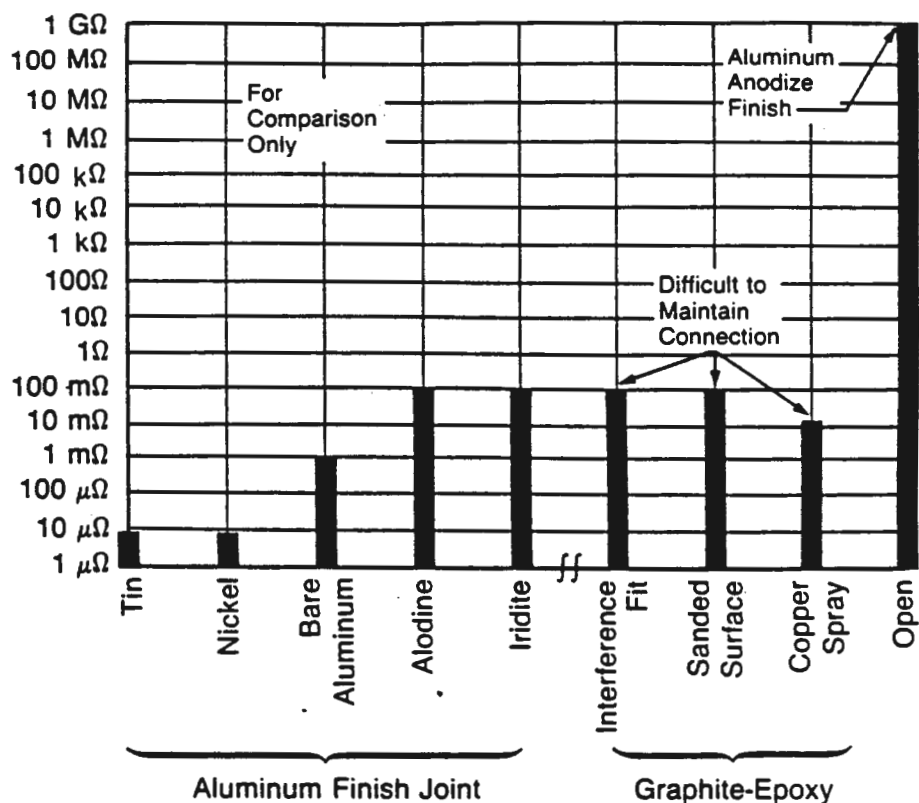


FIGURE 3.5-16. JOINT FINISH RESISTANCE

In summary, the complexity of the joining process and its maintenance is contrasted by the simplicity of the electrical bonding resistance limit: 2.5 mΩ. Aluminum is the grand conductive foundation. The part that it plays cannot be overstated. It stabilizes, shields, conducts, isolates, and protects.

3.5.5. Resonance

Aluminum structure and single wires as conductors have a drawback. They resonate (figure 3.5-17). Structure and single wires must be avoided in high-frequency circuit design.

3.6. Composites

Graphite-epoxy is a fair to good conductor and shield.

Kevlar and fiberglass are insulators and have no affect on radio frequency fields. Dielectric structural materials need to be modified to provide electromagnetic shielding.

Nonmetallic materials form many parts in today's aircraft (figure 3.6-1) and are expected to increase in the future. When varieties of materials are brought

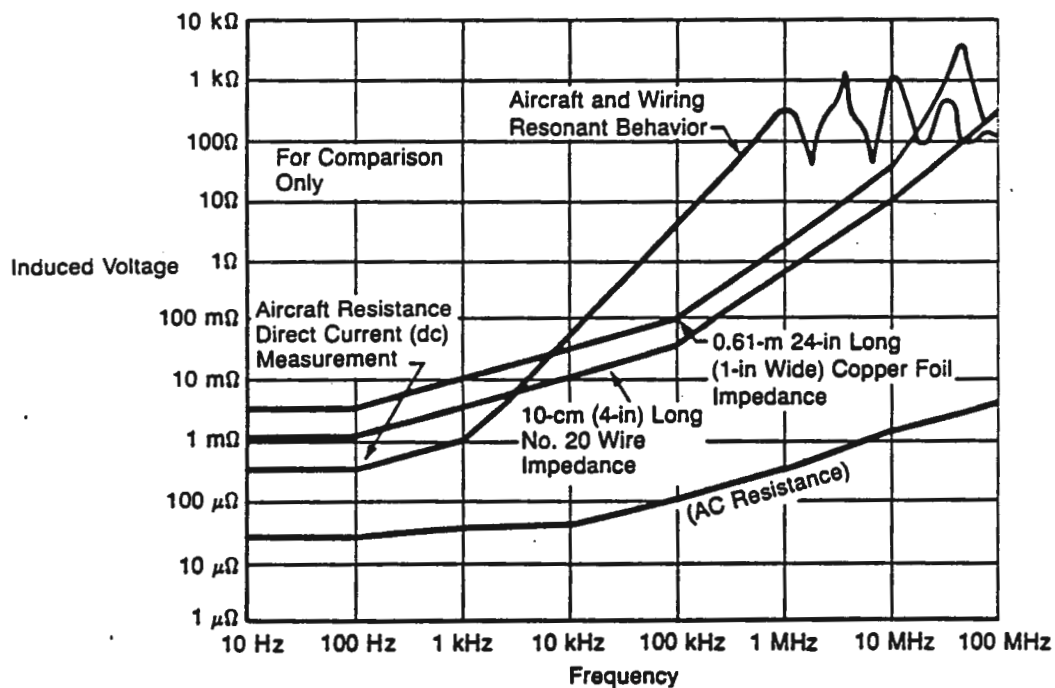


FIGURE 3.5-17. AIRCRAFT AND STRAP RESONANCE

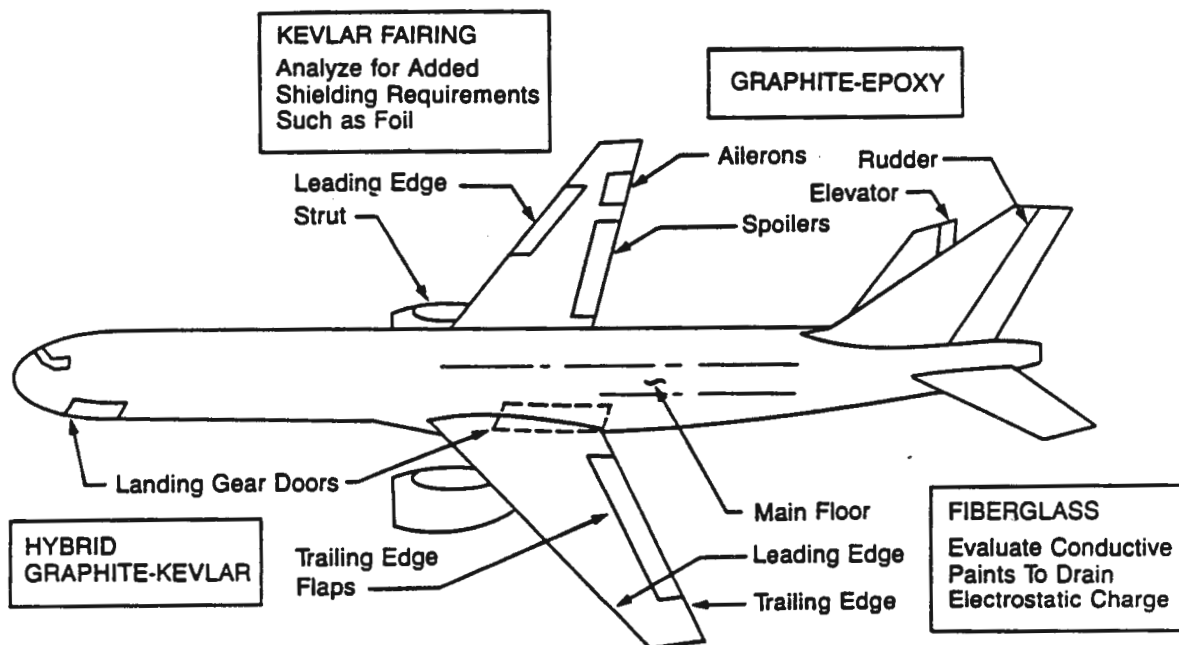


FIGURE 3.6-1. TYPICAL NONMETALLIC APPLICATIONS

together, the interfaces lead to escalating possibilities of material mismatch between finishes, fasteners, and adhesives with possible adverse implications for the quality of electromagnetic compatibility.

Graphite-epoxy will conduct and will shield. Electric field shielding is very good. Following is a comparison of appropriate electromagnetic compatibility-related characteristics:

- Dc resistivity: more than 1000 times greater than aluminum in longitudinal direction and 100 thousand to 1 million in the transverse direction.
- Joint resistance: 30 mΩ to 1Ω or higher. It is difficult to create an electrical connection to graphite-epoxy, and once the connection is made, it is difficult to maintain. New techniques and better procedures are being developed. Today, graphite-epoxy cannot be used as a power or signal return.
- Magnetic shielding starts at about 1 MHz, rises to 60 dB at 100 MHz with about 35 dB in the HF-VHF range.

If graphite-epoxy is not well bonded at seams and joints, it will not act as a shield.

If composites are used extensively for structure, the greatest impact is in providing for a ground plane, the signal return system, and the power return system, which will require their own installations of wire, conductive foils, strip, or cableways. Graphite-epoxy resistance is too high to provide an adequate ground plane (except for antennas). Figure 3.6-2 shows the general level of resistance of a large cylinder or tube, and indicates that its use as a power return path and signal return path is unsatisfactory.

If graphite-epoxy or insulating materials are built into the structure, loss of shielding may open up a path for noise from fluorescent lights, switching regulator and clock oscillator harmonics to reach the ADF, HF, or VHF receivers. Or, in turn, HF frequencies may enter wiring and the avionics. The effects of radio frequency electromagnetic interference are becoming more of a concern (figure 3.6-3). Figure 3.6-4 illustrates some of the resistance levels that might be encountered and shows the use of twisted pair or twisted pair shielded wire.

In the world of electromagnetic compatibility, our protection and safety lies in an integrated design: architectural or structural shielding, shielded wiring, interface circuit voltage limiters, and software correction techniques. The soul of electromagnetic compatibility is a balanced, isolated, shielded interface circuit: it is practically impervious to conductive, inductive, and capacitive attack of electrical noise.

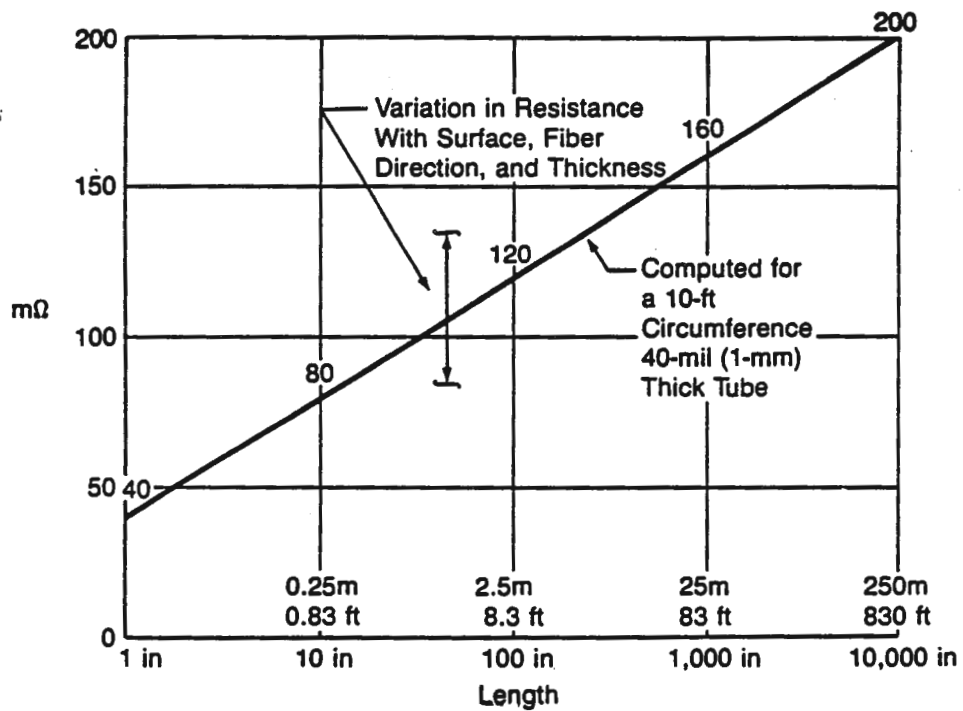


FIGURE 3.6-2. GRAPHITE-EPOXY RESISTANCE

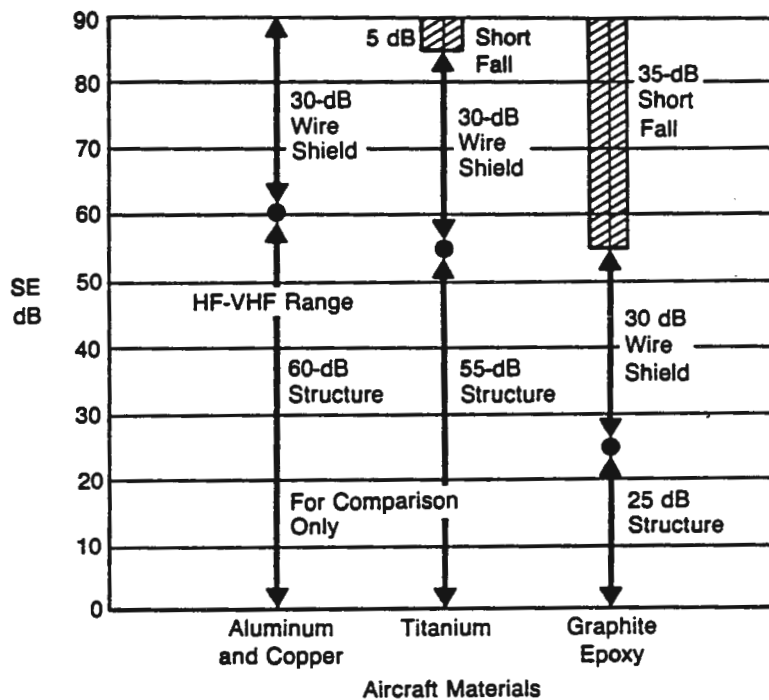


FIGURE 3.6-3. GRAPHITE-EPOXY SE SHORTFALL

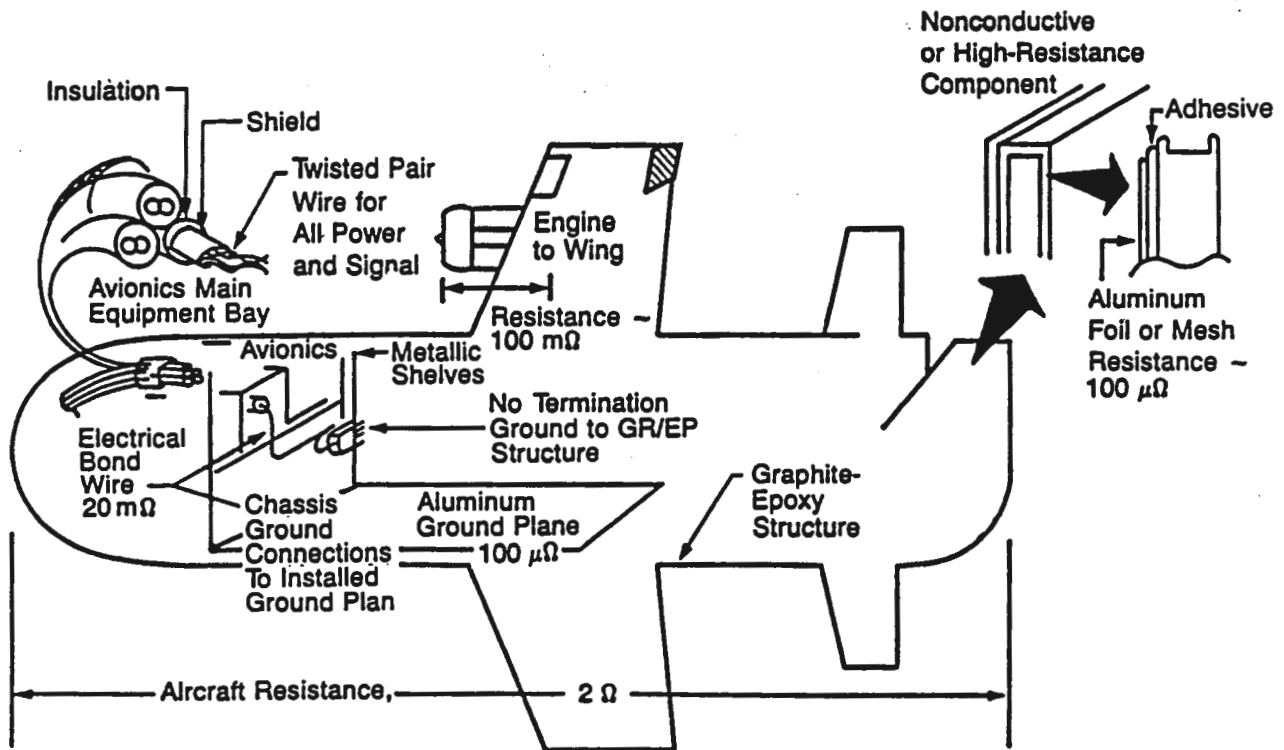


FIGURE 3.6-4. GRAPHITE-EPOXY DETAILS AND INSTALLATION

4. EMC ANALYSIS AND ENVIRONMENT

4.1. Radiated Environment

4.1.1. Radio Frequency Field Distribution

4.1.1.1. Environment

The electromagnetic radio frequency spectrum is vast. Electromagnetic radio frequency field strengths - measured in volts per meter - vary widely. But how do we bind them and treat them so we understand their significance? Well, for the purposes of electromagnetic interference simplicity, one might say we, as humans, and the aircraft, as an object, exist in a dimensional world of roughly 300m to 1m which, converted to radio waves, is 1 MHz, to 300 MHz.

That is very important for electromagnetic compatibility.

The American National Standards Institute (ANSI) radio frequency protection guide for personnel has the strongest field strength limitation over the span of 3 to 300 MHz. People, airplanes, wiring, and the avionics equipment itself are most susceptible to the frequency spectrum spanning the range of 1 to 300 MHz. This is the HF-VHF range (figure 4.1-1). The conditions and behavior of radio frequencies are sometimes so complex that radio frequency measurement and design efforts have often been given the title of "black art." Stray fields, cavities, diffraction, absorption, resonance, constructive interference, destructive interference: all of these interact to make the details of radio frequency studies conceptually difficult over some of the transitional frequency ranges.

The spectrum of radio frequency fields that penetrate aircraft wiring and enter into the avionic inputs to attack integrated circuits can be bounded and summarized over three basic regions:

- A low frequency below 1 MHz where radio frequency noise is normally less of a concern because fields are very inefficiently received on a wire.
- The HF and VHF region, 1 to 300 MHz, where radio frequency fields are very much of a concern and aircraft wiring acting as an antenna is efficient.
- An upper frequency range above 300 MHz where induced voltages in wiring drop off rapidly with increasing frequency and are easier to control.

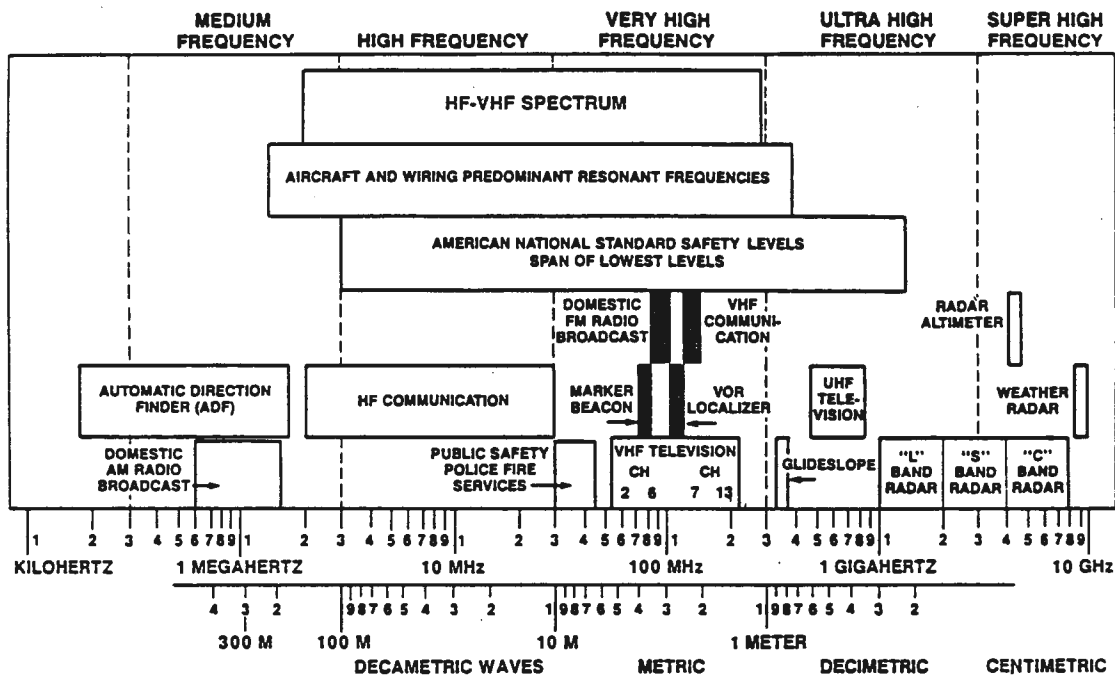


FIGURE 4.1-1. HF-VHF RANGE

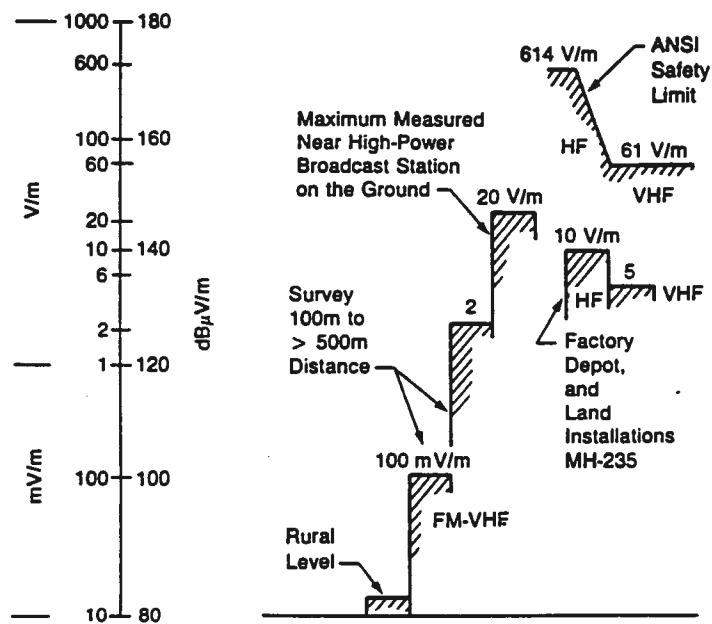


FIGURE 4.1-2. SELECTED RF FIELDS: HF-VHF RANGE

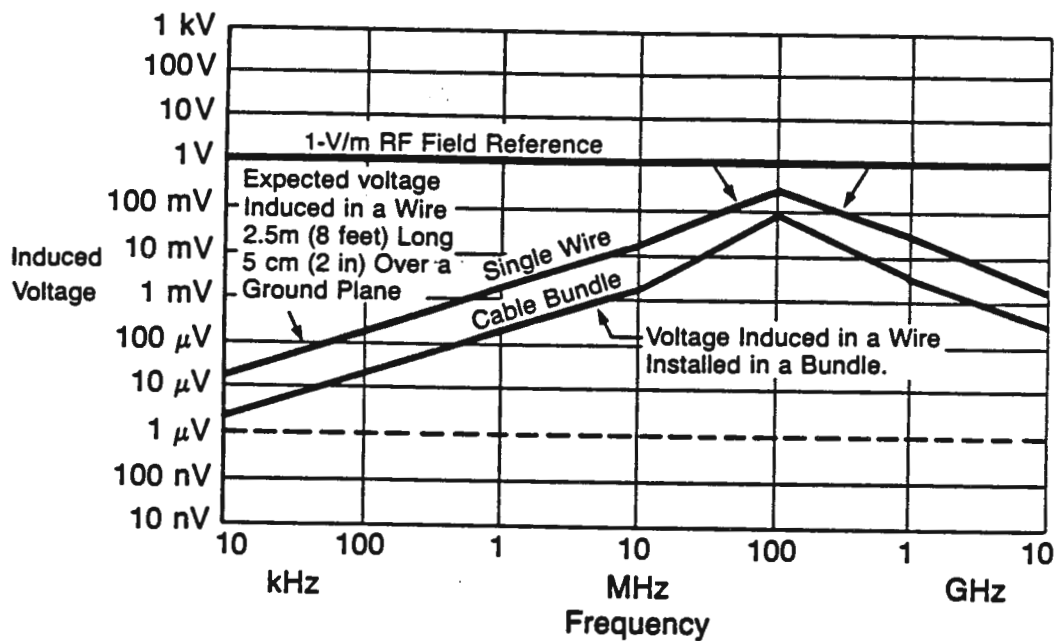


FIGURE 4.1-3. EXPECTED WIRE VOLTAGE - EIGHT FEET

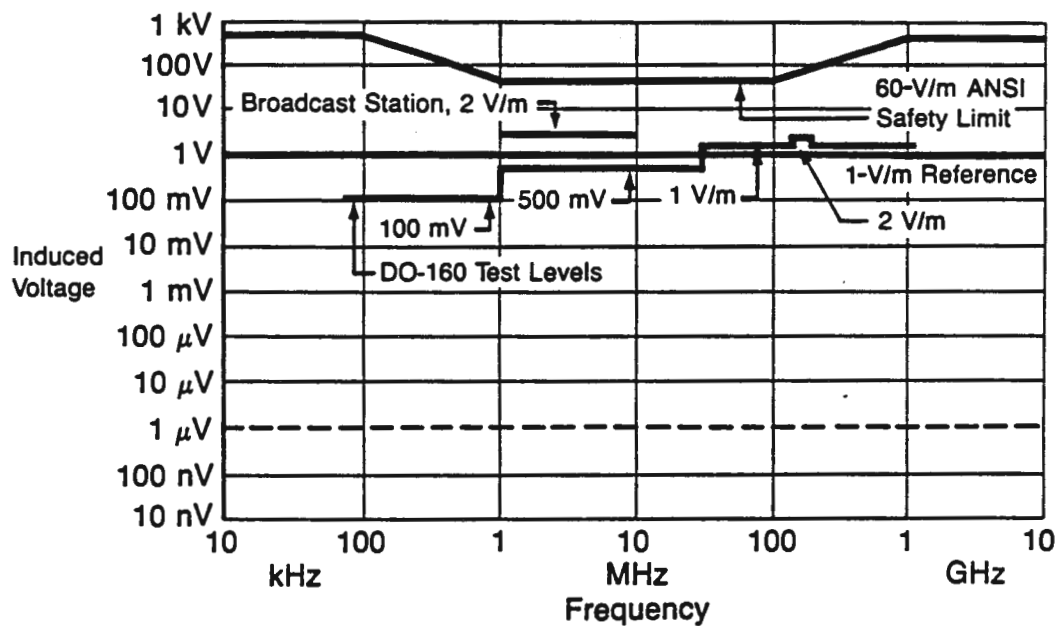


FIGURE 4.1-4. ENVIRONMENTS, SAFETY, AND TEST

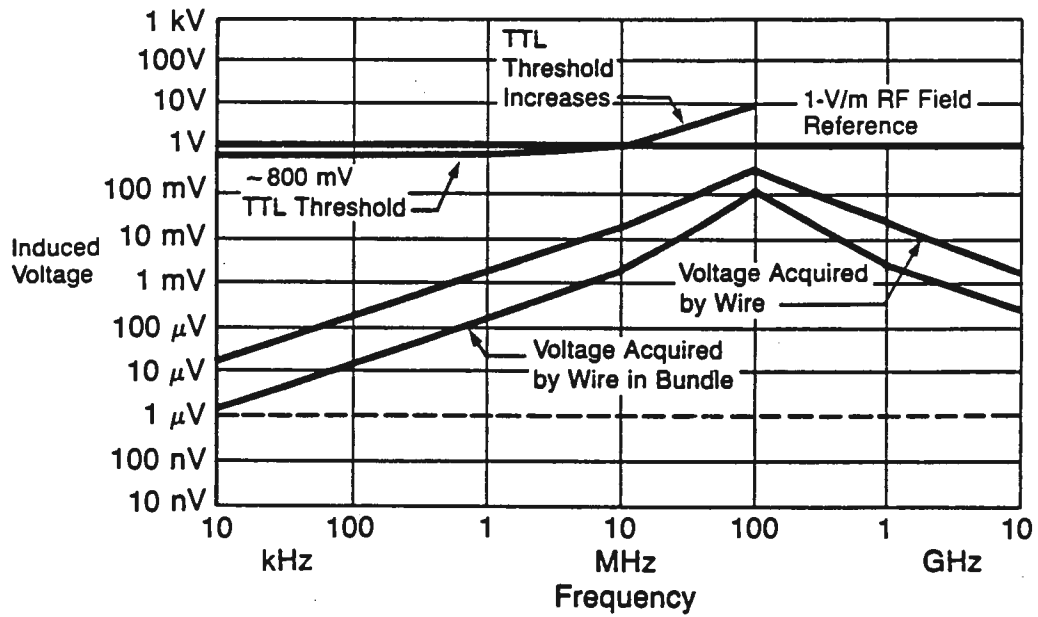


FIGURE 4.1-5. TRANSISTOR THRESHOLD

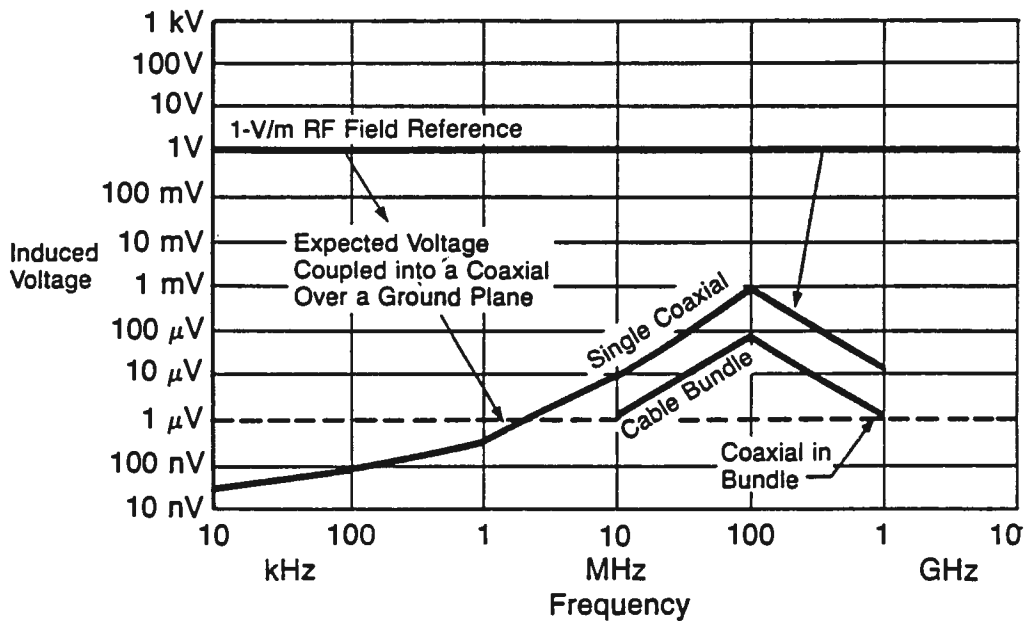


FIGURE 4.1-6. REPRESENTATIVE COAX VOLTAGE COUPLING

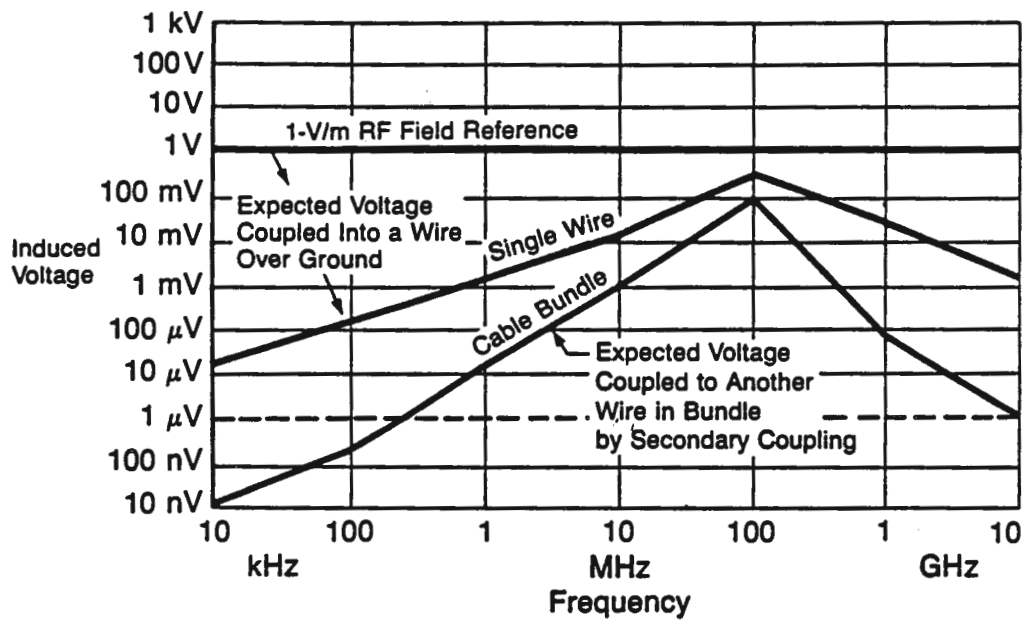


FIGURE 4.1-7. REPRESENTATIVE SECONDARY COUPLING

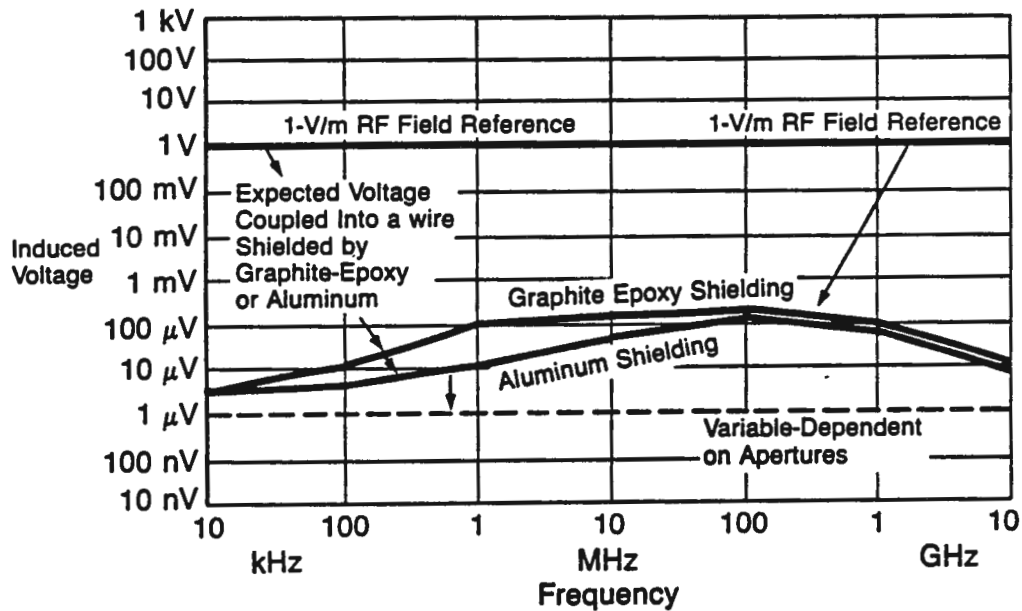


FIGURE 4.1-8. GRAPHITE-EPOXY OR AL SHIELDING ADDED

HF and VHF, ground-based FM radio and TV broadcast signals join with their airborne companions, the aircraft HF and VHF communication links, to induce significant voltages on aircraft wiring in the 1- to 300-MHz range. HF-VHF field strengths measured at ground level for selected urban, suburban, and rural areas are plotted in figure 4.1-2. At altitude, stronger fields can exist (see section 10, Bibliography, ECAC study). The ANSI personnel safety limit and the Military Handbook 235 Environmental Test Specification are plotted for comparison.

Figures 4.1-3 through 4.1-8 give a general view of the expected induced voltages (in some specific wires under selected conditions) and also a comparison of related environments. This information and data is illustrative and advisory to resolve significance and make comparisons of parametric properties and behavior and is not for use or adaptation to specific designs.

Aircraft structures and avionics are engineered and designed by the airframe manufacturer and subcontractors to limit and control these fields. Some lines are shielded, some filtered, and others incorporate a balanced circuit design that will very effectively reject most radio frequencies.

Another main defense against HF-VHF radio frequency signals is a comprehensive and all-inclusive electrical bonding system of all aircraft structural shielding materials to eliminate resistive seams and open seams or apertures. During a specific design program, it is becoming more and more necessary to develop a sophisticated set of models and computations to characterize and define the particular installation and uncover the behavior of the radio frequency fields and induced voltages.

So, in summary, the HF-VHF frequencies are important to aircraft designers because the wiring, equipment, structure, and the aircraft itself, having the same dimensions as the radio frequencies, become efficient antennas.

4.1.1.2. Resonant Behavior

Resonance is the ingredient that brings the arcane aspect to electromagnetic interference. Resonance, standing waves, and reflections share a commonality with black magic in their illusory and unseen character. Electrical parameters, voltage, and current, under the influence of the infinite variability of resistance, capacitance, and inductance, depending on the number of electrical poles, and when viewed over the frequency spectrum, may start out with high impedance and fall quickly to a short circuit minimum. Then they curve back to a maximum. On the other hand, the impedance may start low and rise to open circuit values. Voltage and current in RLC circuits may rise and fall in a periodic fashion or in one simple cycle. They may reach constant highs and lows, or may vary with inconsistency. Resistance, capacitance, and inductance vary endlessly with material electrical properties and length, size, and height above ground. But, they are bounded by maximum and minimum envelopes and the spreads, although large, are limited. In a shield room, standing waves on wiring may rise up at around 10 MHz.

Shield room resonance itself usually is around 50 MHz depending on size. On a large transport aircraft, resonance may occur around 1 MHz and continue into the higher frequencies. The fuselage may resonate at 1 MHz, the wing and other structural details at higher frequencies.

In a balanced transmission line circuit, resonance is kept to a minimum; signal stability and quality are maintained.

Single wires and braids (at their resonant frequencies in the 10s and 100s of MHz) offer very high impedances. A 40-inch, or 24-inch, or 9-inch wire or braid at 70 MHz (the VHF range) may have an impedance that has risen to 10 k Ω . A 4-inch long braid or a #20 wire exhibits about 500 Ω at 100 MHz, roughly the same frequency range. With this kind of impedance increase and resonant condition, adding a wire connection between a circuit and ground may alter the surrounding field strength levels by 10 or 20 dB. The significant consideration here is that they are not stable conductors. As soon as appreciable capacitance and inductance are added, conductor quality is lost in a single wire or braid.

In unencumbered space, radio frequencies behave linearly, constantly, and predictably. Introduce conductors, metallic planes, and a range of materials, resistance, and dielectrics; and vary the structure, cabling shape orientation, and location; then only modeling and measurement techniques can focus on and map the radio frequency fields with any accuracy and reliability.

Even though there is a wide range of variation in the HF-VHF radio frequency region, the testing and measuring is performed in an equivalent shield room, metallic environment, complementary to the aircraft fuselage metallic environment. Increased field strengths from resonant and reflection activity in the shield room have been a rough emulation of the final installation.

4.1.2. Magnetic and Electric Field Distribution

From the 400-Hz powerline, those ubiquitous, unidentical twins, electric field and magnetic field, although easy to contain or block, nevertheless crisscross the length and breadth of today's aircraft and are present in every bundle and on every circuit. The magnetic field permeates all materials. It may couple to other wires. A wire with 4A may cause noise voltages up to 400 mV (figure 4.1-9) depending on wire length. These noise voltages can mar the signal fidelity in the input circuitry of analog sensors, passenger entertainment, radio headsets or interphones. The electric field charges all materials. It can induce high voltages. A 100-ft wire with 160V peak, 115V rms, may induce over 100V peak (figure 4.1-10) onto high-resistance circuits. These noise voltages will pester operational amplifiers or comparator circuits, although they will ignore low-resistance analog or digital signal drivers.

These twin marauders, almost always together, are strengthened with increasing bundle length and encouraged by proximity of common wire routing. The magnetic field and electric field induced voltages of figures 4.1-9 and 4.1-10 have been plotted together on figure 4.1-11 and shown with the added variable of wire separation - the other dominant wire coupling parameter. See in figure 4.1-11 how the induced voltage drops with wire separation.

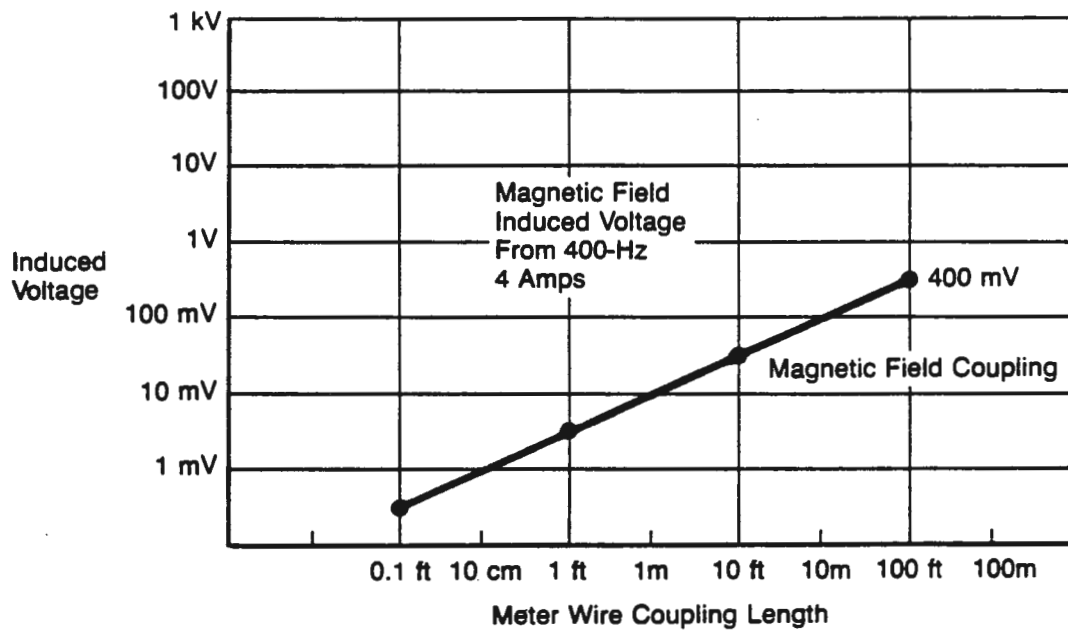


FIGURE 4.1-9. "H" FIELD COUPLING

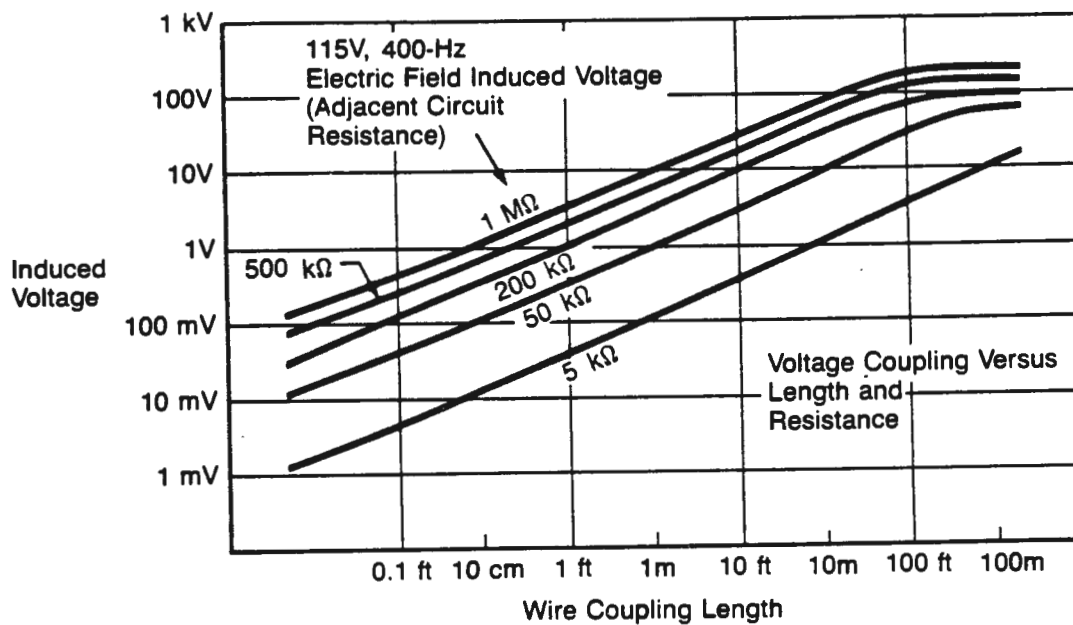


FIGURE 4.1-10. "E" FIELD COUPLING

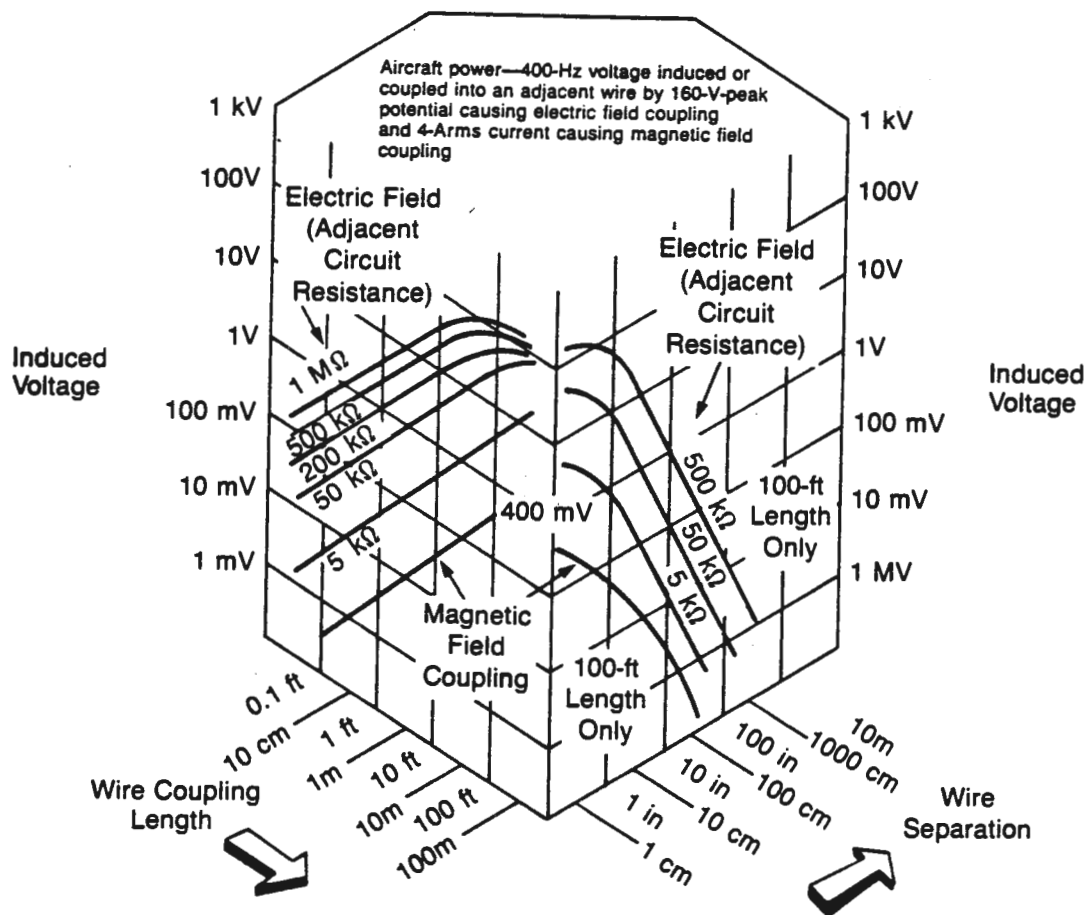


FIGURE 4.1-11. COUPLING VERSUS SEPARATION

Now, there is one exception to the design panacea of using a twisted pair, shielded circuit as the perfect defense against electromagnetic interference. On present day aircraft, a circuit, even though it is twisted pair, will be subject to magnetic field induced voltages if it has a shield and the shield and circuit are grounded at both ends. (Aluminum aircraft structure carries the 400-Hz power line return currents. The 400-Hz current will thus travel in shields grounded at both ends. The shield current then transfers a voltage to the internal circuit.) A balanced isolated circuit design or double shielding will compensate. The first twin, magnetic field, induces voltages on other wires in a bundle, but it also radiates out from the bundles into the aircraft interior space and into equipment (figure 4.1-12). The radiated magnetic field emanating from the power lines varies in field strength from microgauss to gauss (milliamps per meter to amps per meter). The fields are of interest when flight-deck CRT displays, hand-held transceivers, and microphones (whose operational functions rely on the use of magnetics for control or display processes) are present. The magnetic field will easily cause distortion or "banding" on display tubes or will develop the sound of a 400-Hz "hum" in a speaker. Magnetic fields drop off in magnitude very rapidly with distance.

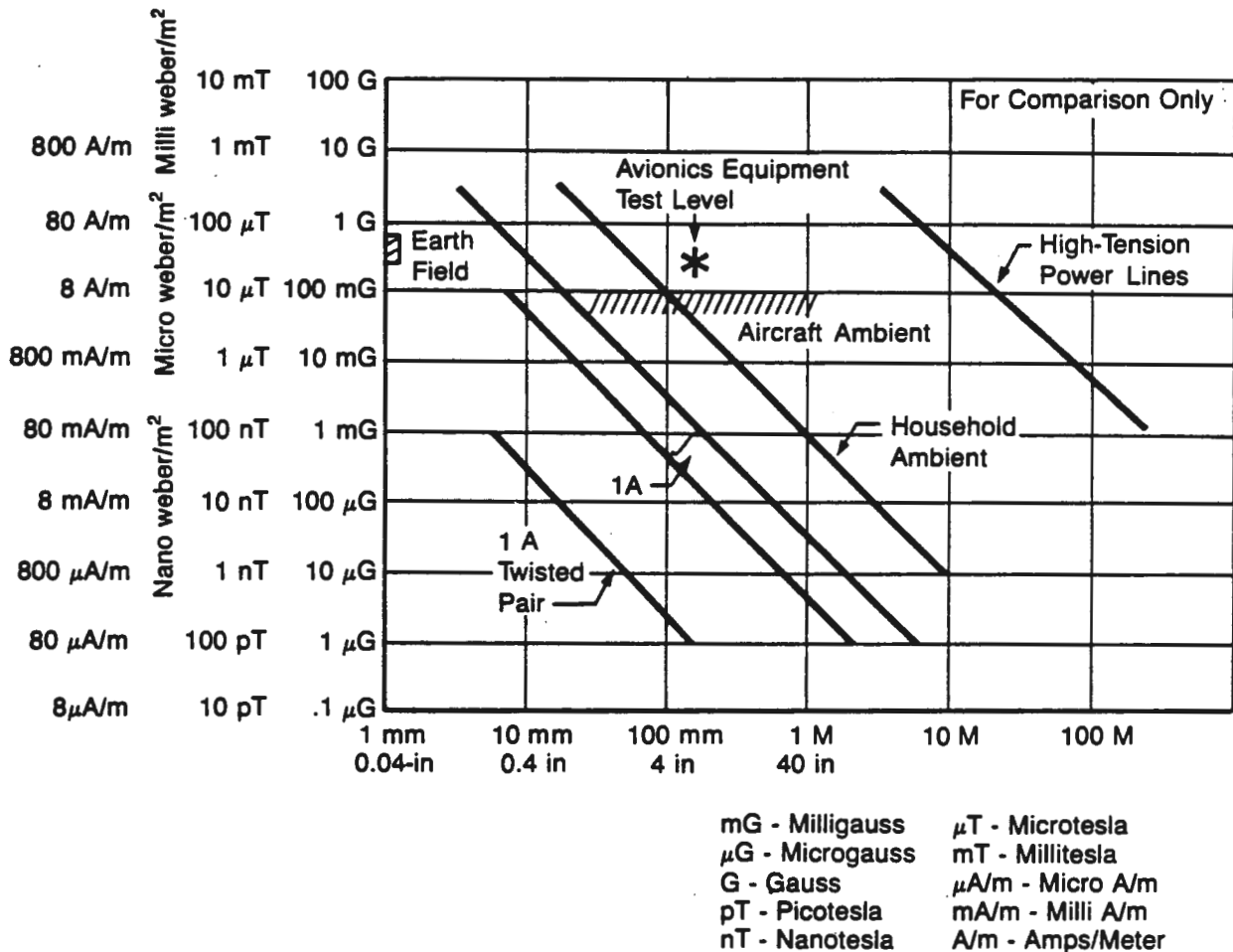


FIGURE 4.1-12. MAGNETIC FIELDS

The second twin, the 115V electric field from 400-Hz power lines, is in almost every aircraft cable bundle. The power line wire has a 115V rms (160V peak) potential. Another wire circuit brought into proximity of the 115V wire holds the 115V potential unless it is destroyed by the "electrical resistor divider" action of a low resistance on the adjacent or "victim" circuit. Figure 4.1-11 shows the reduction of voltage with reduced resistance on the adjacent circuit. The resistance is the parallel resistance of source and load. Either a low resistance or a shield will stop electric field coupling.

Careful attention to wiring details and design such as the use of twisted pair wire, shielding, balanced circuits, or fiber optics helps protect the aircraft against electric and magnetic fields.

4.1.3. Transients

A 115V power line supplying an inductive load, when interrupted by the opening of a switch or circuit breaker, will create an electrical switching transient

often called an "inductive switching transient" or a "relay-induced transient." Electrical switching transients from lights, fans, pumps, or control surface actuator operation reach levels of 600V. Their duration can be over 1 ms, and their rise times nanosecond to microseconds. They harbor repetitive ringing waveforms or pulses (figure 4.1-13). These recurring ringing frequencies span the 1 to 10-MHz range. The switch opens, an electrical arc is started, and an inductive coil unloads stored energy through the arc in repetitive pulses onto a wire - a wire that may be bundled with other circuits carrying digital data or analog signals. The high-frequency energy is transferred capacitively and inductively to those circuits and directly into the internal harnessing and etched circuit card traces and microcircuits of an avionic unit. The electrical transient distributes to microprocessor clock, timing, or data lines.

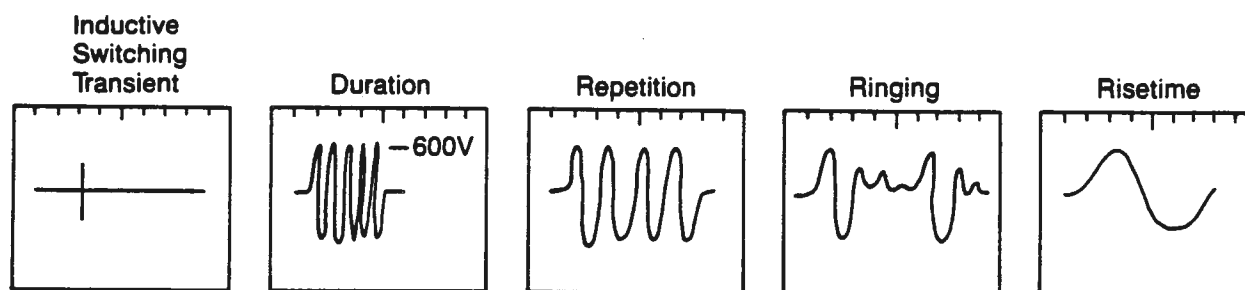


FIGURE 4.1-13. TRANSIENT RINGING

Very extensive measurements have been made of transients and their characteristics. L. Bachman states in his 1981 report: "This paper summarizes the results of the most comprehensive study ever conducted of U. S. Navy shipboard power line transients. Transient data was acquired on 13 ships - over 9400 hours of monitoring time - 2300 transients were encountered." Figure 4.1-14 is a summary. This study is remarkable in its extent and completeness. These transients show very similar, if not identical, characteristics of magnitude, risetime, and duration to those measured on commercial aircraft. In contrast to the high-level, inductive switching transient, the power bus may also experience momentary power interruptions, where the voltage drops to zero for as long as 50 ms.

Electrostatic discharge is another form of transient that can suddenly erupt under the right conditions of humidity, air or fluid flow, and juxtaposition of materials. Electrostatic discharges have much higher voltages - up to and over 10 kV, but much narrower pulse widths, possibly a 100-ns duration. A person on a dry Arizona-like day touching an avionic unit may discharge 10,000V onto the housing or wiring.

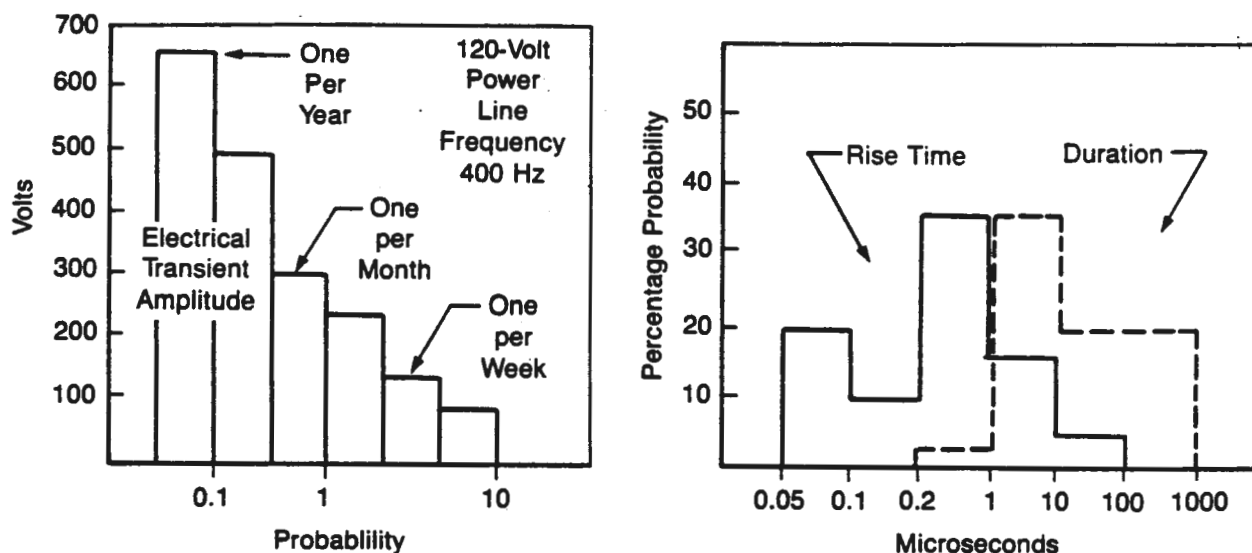


FIGURE 4.1-14. ELECTRICAL TRANSIENTS

One of the most evident electrical discharges, of course, is lightning. Large currents flow on the wing structure, fuselage, landing gear, or empennage leaving induced voltages in unprotected circuits. These induced voltages have resonant behavior often established by the characteristic of the size of the aircraft and length of wiring. Lightning protection is engineered to divert and contain voltages and currents in metallic structures to avoid damage to electrical wiring and parts. The transient voltage levels induced in wiring are required to be less than 600V. Induced voltages are usually less than 200V for the most severe strikes.

So you see, HF-VHF radio frequencies, 400-Hz power, and the various transients are electromagnetic interference or noise forms that are freely distributed and transferred into microprocessors to affect aircraft performance unless constrained and controlled by carefully integrated wiring and avionic and structure design.

4.2. Aircraft Protection

4.2.1. Shielding and Ground Reference

Aircraft and circuit protection is formed in a layered design: 1st layer, structural shielding, liners, trays, overbraid; 2nd layer, circuit shield; 3rd, balanced, isolated circuit; 4th, voltage, current limiting; and 5th, software.

Double shielding (two layers of shielding) fends off lightning and radio frequencies. For critical circuits having exposed wiring, double shielding is necessary.

Structure, skin, and panels of the aircraft form the first major level of shielding, and it is the system level barrier. Fuselage shielding is extended into open sections or unshielded areas, such as leading and trailing edge or landing gear, with a cableway, overbraid, or foil (figure 4.2-1). Shielding and wiring for radio frequencies, electrostatic charge, and lightning, safety fault returns and ground reference all call for a highly conductive material. Liners can be installed in nonmetallic electronic bay sections. Where spar, skin, or panel shielding is not inherent in structure, then foil, metal spray, or metal mesh (figure 4.2-1) can be designed into or onto nonconductive parts. Leading and trailing edges, engine bays, wheel wells, bulkheads, tail sections: all need evaluation to delineate the shielding and ground planes. Nonconductors, composites, and graphite-epoxy are not a ground or current return.

The second level or layer of shielding is individual circuit shielding continued (with back-shells) to the avionics enclosure shield. The second level offers protection against external threats and internal noise in wiring bundles too. The second level is connected, "grounded," to the first level.

4.2.2. Apertures and Electrical Bonding

Conductive panels, foil, or paint form and continue the shielding enclosure on skin panels and trays and, unless electrically bonded, will develop harmful voltages or electrostatic charge centers. (See excellent reports in bibliography by L. O. Hoeft on shielding and aperture losses.) Often good continuity is provided by fasteners. External items are electrically bonded and grounded to provide static discharge paths. Conductive paint is applied to external nonconductive surfaces.

Everything is searched out to check resistance.

- Materials: titanium, steel, stainless steel, aluminum alloys, fiber glass, graphite-epoxy, Kevlar, phosphor bronze, composites.
- Parts: instrument panels, keyboards, switching panels, seats, frames, window films or mesh, quick access doors, skin panels, cowls, fairings, trays, overbraid, backshells, plumbing, brackets, covers, control surfaces.
- Liners: foil, mesh, plating, depositions.
- Finishes: alodine, anodize, iridite, organic applications, copper plating, tin, silver, nickel.
- Sealants: paint, film, special substances.

Bonding is detail work, and often research and development is needed on new materials and techniques. Materials and procedures are recorded in a bonding and corrosion prevention document.

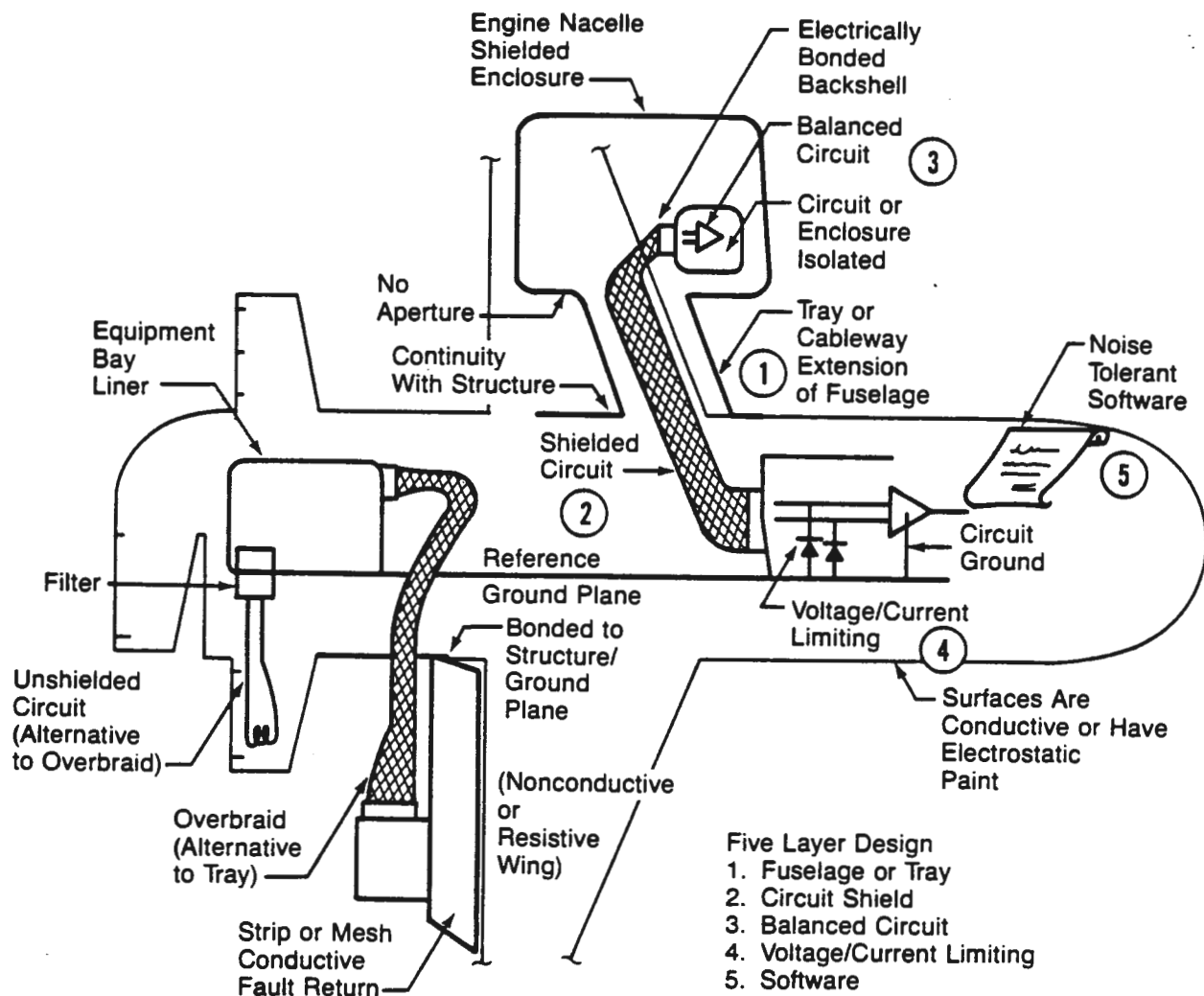


FIGURE 4.2-1. LAYERED DESIGN

4.2.3. External Wiring Interface Circuits

Long wiring runs extending out to the wing and to the engines are unintentional antennas that collect noise. Data, control, and sensor lines, outside of the shielding of the fuselage, demand protection against radio frequencies impinging on engine struts or mounts, leading and trailing edges, and landing gear.

Engine instruments, pressure, temperature, speed, thrust control, air data, fire, flight control computers (circuits for actuators and control surfaces), proximity switches, position indicators, temperatures, and braking circuits are candidates for analysis and protection. Circuits are categorized by criticality. Line replaceable units may number from 50 to 100 units and digital buses,

200 to 300, but the buses will reduce to 30 or so different types with only 5 or 10 types being external to the fuselage. Future systems may have less than 10 digital buses. Discrete circuits number 30 or 40 different types, with around 10 external. They can be thoroughly analyzed.

Good rules for external circuits (there are exceptions):

Grounding:

- Circuits isolated from structure at exposed end.
- EMI tests applied to returns/grounds of units not on a ground plane.
- Primary to secondary power isolated.

Bonding:

- Case bonded to ground plane (when circuits isolated from case).
- Case isolated from ground plane (when circuits grounded to case).
- Backshell/connector bonded to case.

Shielding:

- Double shielding installed on transmitter lines.
- Internal shield grounded internally.
- External shield grounded externally (to backshell).

Input/output:

- Interface circuits balanced; clock and data signals routed together.
- Interface circuits isolated: transformer, LEDs, fiber optics.
- High resistance (greater than 10 k Ω) designed into circuits.
- All circuits filtered.
- No shared power wire returns.

Wiring/packaging:

- Return wire twisted with signal wire.
- No wires installed across an aperture (apertures bonded).
- Line drivers/receivers packaged close to connector.
- Connectors on equipment case placed in one local area.

4.2.4. Circuit Protection

Protection depends upon electronics design, packaging and, especially for external wiring, the voltage stress conditions: stress on insulation, thin films, integrated circuits, and trace spacing. Solenoids and motors with heavy insulation and no electronics usually do not require protection. Thin film and integrated circuits do. (See comprehensive reports in bibliography by R. L. Carney, R. A. McConnell, and D. L. Sommer for excellent treatment of protection devices.)

Every interface circuit needs analysis and voltage/current limiting. Naturally, the transient or radio frequency threat must be known; first define the open circuit voltage; second the surge or characteristic impedance of the wiring and the input impedance of the input capacitor, resistor, or diode; then determine short circuit current; and finally the transient time and energy, or radio frequency power. The following are variable, but important, voltage withstand requirements or test levels that apply to insulations, parts, and electronics and are usable benchmarks.

No protection usually required:

- 1500V rms, 2100V peak, 400-Hz signal, impressed for one minute; this is the voltage withstand specification for insulation (solenoids, motors).
- 3000V or greater, one microsecond transient withstand for insulation, varies with humidity, configuration, altitude, time.
- 1500V one microsecond transient, discrete resistor (molded part).
- 1500V fifteen mil circuit card spacing (at sea level).

Protection required:

- 200V or less, thin film, transient withstand voltage.
- 200 to 800V receivers protected by integrated circuit diodes.
- 100V transistors.
- 30V or less, operational amplifier receiver.
- 360V or less, fifteen mil circuit card trace spacing, at 100,000 ft.
- 360V or less, corona initiation at 100,000 ft, varies with shape.

The five layers of protection: (1) structural shielding, overbraid, cableway; (2) circuit shields; (3) balanced, isolated circuit; (4) voltage/current limiting; and (5) software combined with a stable ground reference plane help to guarantee compatibility.

4.3. Trades

It is critical to know program design requirements-and the environment. Dissect the electromagnetic topology of the aircraft including digital transmission, power system, and antenna fields. Round up new and off-the shelf equipment and bay locations. Search out aircraft structural and skin panel materials. Define the exposed critical circuits and equipment.

1st Major Trade: Location of equipment, categories, and tailoring of electromagnetic compatibility requirements of each unit. The wiring and the equipment is kept shielded under aluminum or graphite-epoxy and away from radio frequency fields (figure 4.3-1). Units of a subsystem collocated in a protected environment may have the design/test levels eased: lowered for susceptibility, raised for emission. Units not on the wing will not experience resonant conditions. Off-the-shelf equipment is often unchangeable at the input/output interface, and if located internally may save on shielding or externally mounted filters.

2nd Major Trade: Wire length and separation. Shortened wiring or elimination of wiring. Equipment location, equipment combination and wiring deletion are waiting for evaluation, for example, on-engine electronics supplied with on-engine power; major interfacing units physically located close together. Noise amplitude is proportional to length. Metallic wiring may be replaced with fiber optics to save tens to hundreds of pounds and 25% to 50% reduction of wires (figure 4.3-2) and major reduction of EMI.

3rd Major Trade: Digital, balanced, isolated circuits versus analog circuits and single-ended or discrete circuits. Delete or reduce open wiring. Reduction of wiring, shielding, transformers, power, weight through use of multiplexed digital bus (figure 4.3-3).

4th Major Trade: The five levels of layered protection. If structure shielding is not available, the following alternatives may be investigated. Trade cableways, overbraid, and exposed wiring with filter/voltage/current limiting. For one or two wires on noncritical circuit, filter may trade off better than overbraid or cableway.

A recent large program relied on the fuselage for the major portion of circuit shielding. Wheel wells, engine bays, outer wing sections, and radome areas were shielded. Quick access is often necessary. Overbraid (20% of the bundles) was installed for critical flight control circuits to "extend" the airframe. It was indicated in that design that the use of metal overbraid on all bundles might be 10 times heavier than reliance on the fuselage. Also, complexity of the topology limited the use of overbraid, and in certain areas a combination of protection was needed. For circuits inside the fuselage, filter pins were used; for outside circuits, voltage limiters. Discrete filters for each circuit were not pursued because of excessive volume and weight.

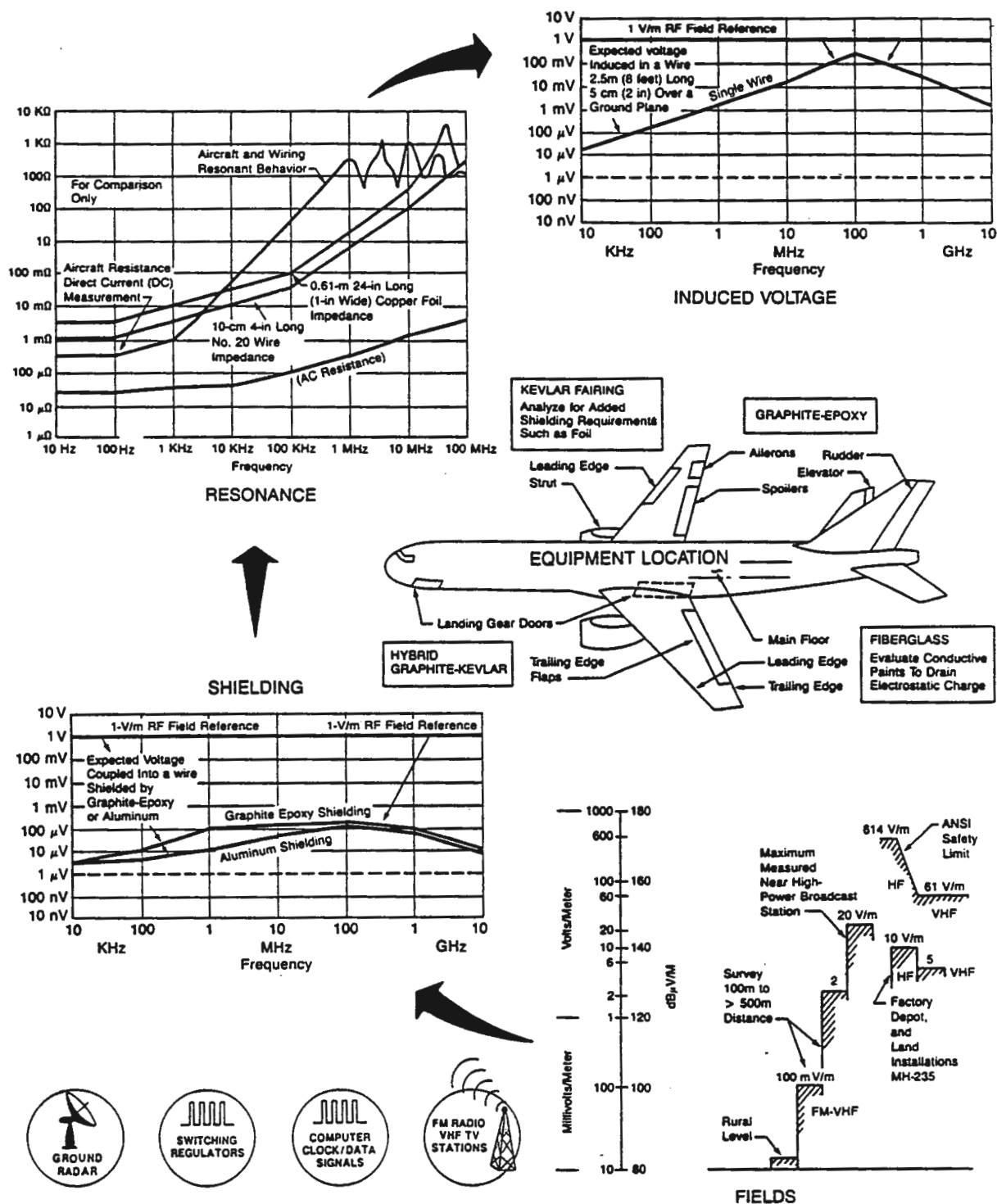


FIGURE 4.3-1. TRADE-EQUIPMENT LOCATION

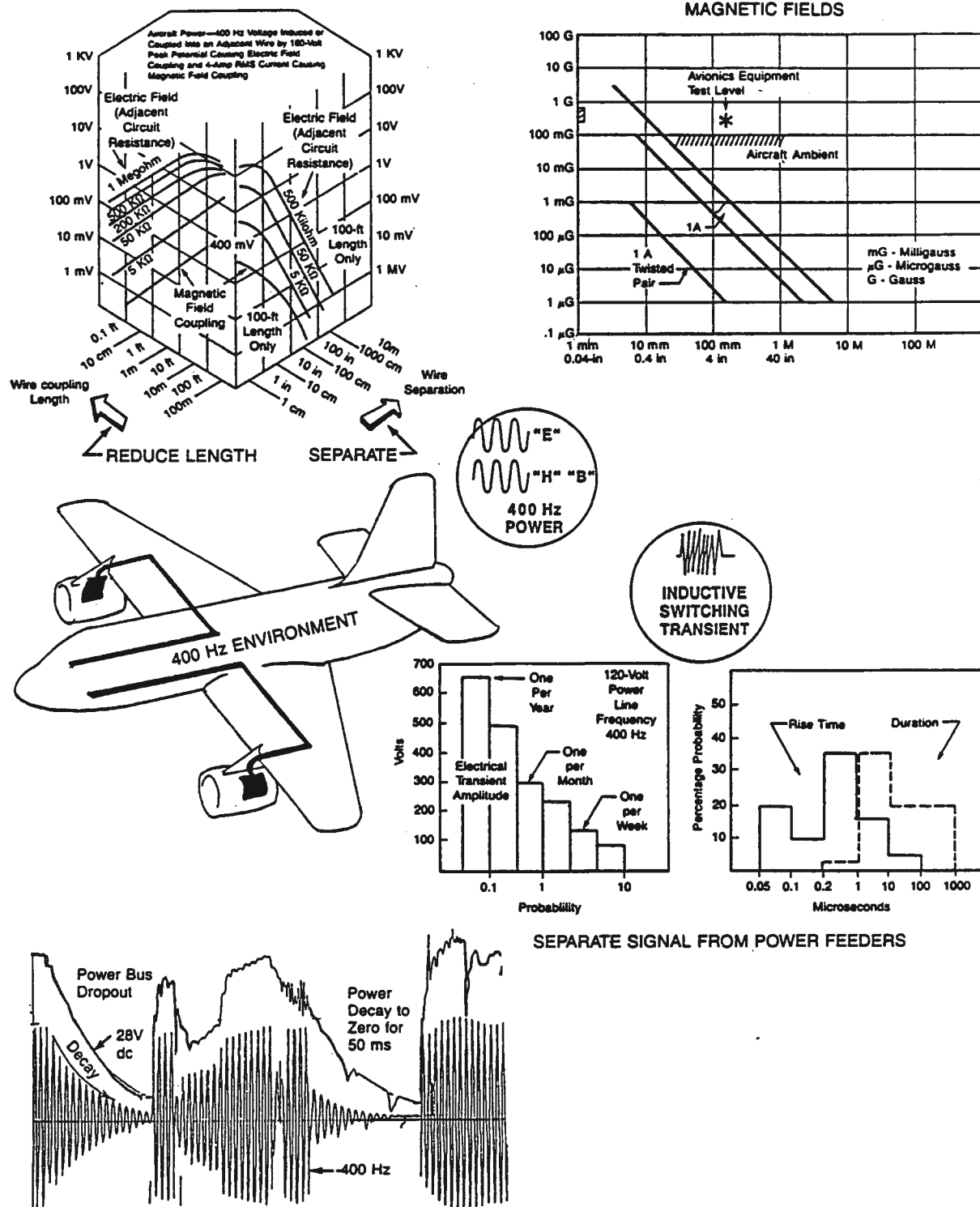


FIGURE 4.3-2. TRADE-WIRE LENGTH/SEPARATION

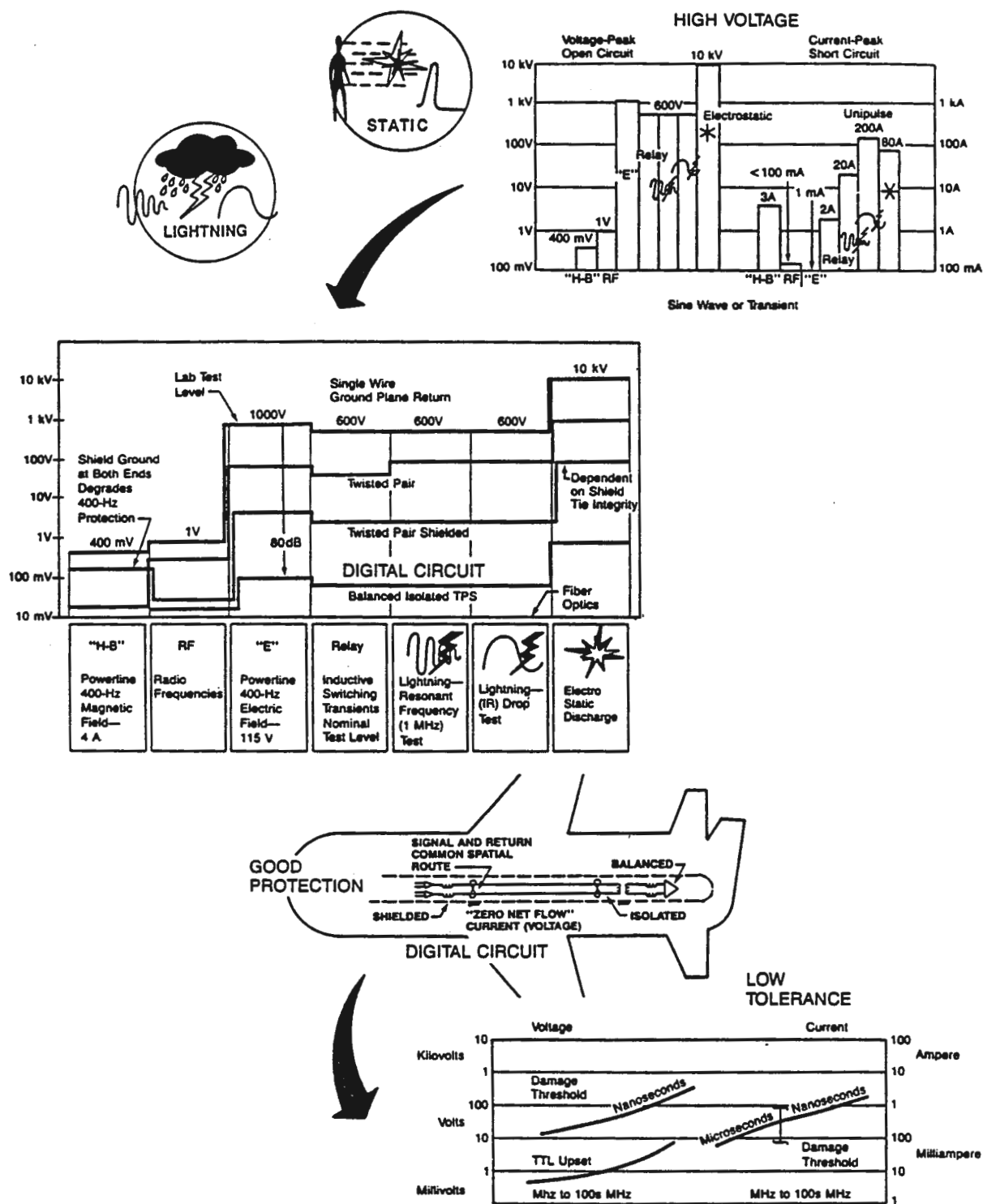


FIGURE 4.3-3. TRADE-DIGITAL CIRCUITS

If it is at all possible to install a cableway, tray or liner, it may be better in the long run. Consider these facts in favor of a tray:

Design:

- Proven designs/models and in-service experience.
- Easily formed, integrated with structure.
- Fewer electronics and "sneak paths".
- Freedom from reliability studies.

Protection:

- Line replaceable units installed with known protection.
- Freedom from frail electronics.
- Excludes high voltages and arcs.
- Rugged and durable.

Test:

- Lower test levels on electronics.
- Freedom from complex verifications.

Life cycle cost:

- Reduced engineering, part control, troubleshooting, maintenance.
- Easy in-service monitoring.
- Long life.

Computerized design procedures will speed the capability to make trades.



5. ACTIVITIES AND DOCUMENTS

5.1. Specifications/Documents

Some have postulated that most of the electromagnetic compatibility designs and trades at the airframe manufacturer's level must be quickly accomplished before go-ahead or at least at the very beginning of the program, that is: all of the major specifications must be set up - the equipment located and categorized - the environment documented - procurements contracted.

This may not, however, be as extensive an effort as it sounds because most specifications for environments and existing technologies are already available from other programs and need only to be reworked, shaped, and adapted.

Specifications set guidelines-standards-requirements. They are the foundation for any program, and a path that provides a reference and continuity. They help bypass unproductive squabbles and unnecessary changes. They sidestep duplication: "Reinventing the wheel?" They even help tie into the next program.

From the top documents, which outline the intent and scope, down to the lower tier designs, look for these specifications:

Industry:

- FAA Code of Federal Regulations, Aeronautics and Space #14, Parts 1-59; Part 25 Airworthiness Standards: Transport category Airplanes, reference 25.1309, 25.1353, and 25.1431; Part 23 Rotorcraft, 23.1309; Part 27 General Aviation, 27.1309.
- RTCA/DO-160B "Environmental Conditions and Test Procedures for Airborne Equipment," July 20, 1984, sections 1 to 3, and 15 to 22.
- (A generic industry system specification here would be appropriate.)

Program:

- "Program System Specification": Simple one or two paragraph delineation of top EMC references, generic system requirements, or scope.
- "System Electromagnetic Compatibility Design Requirements": Organized sections of system and aircraft program requirements for external environment, equipment categories, interface circuit policy, grounding, bonding, shielding, wiring, isolation, electrostatics, aircraft shield policy, special analyses, applicable EMC document list, and policy for critical equipment.

- "Equipment Electromagnetic Environmental Requirements Specification": Explicit equipment level design and test requirements including test setup and conditions-patterned after the industry standard DO-160.
- All Procurement Specifications: Reference in the design section to the "Equipment Electromagnetic Environmental Requirements Specification?" List of deviations. Requirements for power, dielectrics, circuit interfaces, grounding, bonding, shielding. Table of tests. Statement of Work: Developmental analysis and breadboard test report, analysis of off-the-shelf equipment, technical exchange meetings, qualification test plan and report.
- Design Handbook/Guidelines/Design Notes: Interpretation and conversion of program design and test requirements into protection techniques.
- System Test Plan: Test policy and outline. Subsystem tests. Aircraft tests: ground test, environment test, electrical bonding measurements, power switching test, flight test.

Related specifications:

- "Power and Electrical Requirements Specification": Aircraft and external ground power quality, returns and grounding design.
- "Wiring and Shielding Design and Manufacturing Procedure": Wire separation, shield construction.
- "Electrical Bonding and Corrosion Prevention": Materials, compatibility, fasteners, surface preparation, joining and sealing.

5.2. Tasks/Activities

At the beginning of the program, some participants think immediately of the analytical tasks and modeling. Some think of staffing, organization, and people. Others think of schedules. The EMC engineer must pursue the design; he thinks of what the new aircraft is: new materials, new equipment, old equipment, size, layout, wiring runs. Defining the aircraft for electromagnetic compatibility is a monumental effort. The effort cannot be delayed. It must be done quickly. Some details will not be available. There are roadblocks, but these are the engineering tasks.

- Identify in a table or spreadsheet all equipment: new and off the shelf; in-house and vendor; supplying agency; purchasing specification number. List the equipment versus applicable EMC requirements (including outmoded EMC limits), deviations, waivers, and approvals.
- Assess aircraft and equipment architecture: flight deck, electronic bays, externally mounted units, engine units, power and signal circuitry. Identify each equipment and installation and focus on grounding, bonding, shielding, materials and wiring.
- Set up electromagnetic compatibility design categories/groups.

- Track down procurement specifications: remove or add limits and EMC interfaces; establish the developmental, engineering model, qualification, and acceptance tests; communicate with Procurement, Project, and the subcontractor.
- Define topology, field patterns, power of transmitters, and dig out the receiver thresholds.
- Extract power system type, power quality, ground returns and ground points.
- Chase down aircraft advanced technologies and materials. Unearth the differences from past programs.
- Dissect the new environments, external and internal, that are different from DO-160.
- Document new test equipment needs.

With the aircraft definition in hand, the electromagnetic compatibility engineer can take each of the EMC program documents and cut and fit them together: no overlap, no deficiencies, just acceptable coverage. He tailors the requirements based on: (1) past experience, (2) new environments and technologies (3) equipment location, (4) critical circuits, (5) weight, volume, and (6) test. Important: aircraft specifications are finalized more easily before formal design release has been invoked.

Analysis cannot wait. Analysis and models feed on physical dimensions: fuselage-wing-landing gear, length, width, height; and electrical parameters: resistance, capacitance, inductance. Computer models are bringing new capabilities for quick evaluation of multiple design alternatives, and keyboard editing speeds the rapid reevaluation and turnaround when changes are proposed.

The apertures, seams, wire routing, overbraid, isolation, and the shielding effectiveness of aluminum, titanium, graphite-epoxy, and steel must be known along with interface protective designs that must be pulled out of the schematics: filter pins, discrete filters, balanced circuits, transformer isolators, fiber optics. Getting an early handle on these items expedites the many wire coupling and radiation analyses of common noise sources, for instance: power frequencies and ripple; pulse width modulated power; transients from solenoids, valves, motors; and clock oscillator harmonics.

Just a word about responsibility. The electromagnetic compatibility engineer is usually responsible for either a system and product or, conversely, for a technology or even a combination of both. Becoming knowledgeable in a technical area is often the best. For example, for technology:

- Computers/digital circuits software.
- Radio frequency transmitters, receivers, antennas.
- Power quality, generators, switching regulators.

- Analog instrumentation, transducers, sensors, fiber optics.
- Grounding/bonding/shielding/wiring/packageing.
- Dielectrics, corona, and materials.

And for example, product or system:

- Environmental control and cargo (power).
- Flight control/management (digital/software).
- Communication/navigation (radio frequencies).
- Power and lighting and fuel.
- Air data, flight instruments (video, analog, CRTs, displays).

The definition and specification of electromagnetic compatibility on an airplane program is a sporty task. It must be initiated early. Most airplane programs need at least three engineers. Future programs with advanced avionics and flight control systems may require four or five.

6. EQUIPMENT SPECIFICATION

6.1. New and Existing

The countdown is on until the specifications on every last unit of new and existing equipment are found, cut apart, organized, and set straight. Equipment "procurement specifications" are the bottom line. They define design and test requirements and waivers or deviations. If a unit meets its tailored, allocated electromagnetic compatibility specifications, then electromagnetic compatibility on the aircraft is almost always assured. Deliberate on and wring out every last detail:

- Tailored requirements for each unit (per RTCA/DO-160B).
- Each bonding/grounding/shielding interface (between the supplier and airframe manufacturer).
- Power quality specific requirements.
- Dielectric voltage withstand levels.
- Circuit interface wiring design.
- Detailed unit tests: developmental, engineering model, qualification, acceptance.

The specifications on new and existing units give exact requirements, item by item, on electrical and physical parameters and tolerances.

Existing equipment has already proven itself with its "in-service history," and treatment is different from new equipment. Existing equipment is addressed with attention to conformance to past specifications and any new program requirements. On a new program, old requirements may not be adequate: transients may need redefinition and reapplication depending on unit location; electric field protection may need to be increased; radio frequency susceptibility tests may need strengthening; and power bus momentary interruptions may be a problem. Development tests cannot be overemphasized for new and even existing units. The policy is "make it upset" in order to find margins. Development tests are important and are a major element of successful programs. EMI protection cannot be competently added after the finalized design.

The packaging engineer and circuit designer at the supplier have to know the overall design policy for new and old equipment. They must also know what their interface design is going to be. Key design requirements may be dispersed in various documents but should be either in a system specification or in the equipment procurement specification. Policies or requirements vary from program to program, product to product; these are some standards:

Grounding:

- Designed, controlled, equipotential ground plane system.
- Optional digital circuit ground to case.
- No analog (audio) circuit ground to case without approval.
- Primary to secondary power isolation (transformer electrostatic shield).
- Power single point ground system; "star" architecture.
- No power current in nonmetallic structure.
- EMI requirements apply to returns/grounds of units not on ground plane.

Bonding:

- Case bonded directly to ground plane or,
- Connector pin for case ground wire.
- Backshell/connector bonded to case.

Shielding:

- Internal shield grounded internally.
- External shield grounded externally (to backshell).
- Shields grounded at both ends (except analog).
- No circuit current on shield (except coax).
- Double shield on transmitter lines.

Input/output circuits:

- Interface circuits balanced; clock and data signals routed together.
- Interface circuits isolated; transformer, LED, fiber optics.
- Increased circuit power ratings at outputs.
- Return wires run twisted with the signal wires.
- Connector pin for circuit ground.
- Connector pin for case ground or fault wire.
- Voltage/current limit all interface circuits.

- No shared power returns.

Packaging:

- Separation/isolation of power from signal.
- Power, signal, and ground returns on separate pins.
- Line drivers/receivers close to the connector.
- High-frequency circuits close to the connector.
- Low-frequency circuits at back of the box.
- Connectors on equipment case placed in one local area.

6.2. Aircraft Equipment Categories.

Category definition needs immediate attention. Avionics must not be over or under designed. Equipment categories represent the modification and tailoring of DO-160 requirements to units of equipment having widely variant properties or environment:

- Computers, power generators, transducers, motors.
- Units with extra long interface wiring.
- Units placed on or off the aircraft ground plane.
- Equipment exposed or not exposed to high-energy radio frequency.

Here are some typical categories:

Category 1A:

Energy storage devices (having no electronics): inductors, valves, motors, solenoids, and relays switched continuously or automatically. The items in this category are designed and tested to conducted emission and radiated emission requirements only. They do not require susceptibility tests.

Category 1B:

Energy storage devices operating on an intermittent basis, less than once every three minutes. The conducted emission and radiated emission requirements are raised (relaxed) 20 dB.

Category 1C:

Energy storage devices having short duration transients and operating two times or less per flight. Emission and susceptibility requirements are waived.

Category 2A:

Electrical/electronic equipment: avionics, power equipment, any unit having electronics located within the fuselage and protected by the fuselage metallic structure. All of the standard equipment electromagnetic compatibility requirements, equivalent to DO-160, apply: conducted emission, radiated emission, conducted susceptibility, radiated susceptibility.

Category 2B:

Electrical/electronic equipment within the fuselage, but having long wiring runs, over 100 ft. All standard requirements apply along with a raised (possibly 3 dB) 400-Hz electric/magnetic field and radio frequency field tests.

Category 2C:

Transmitters/receivers: All standard requirements, and with antenna terminal conducted emission and susceptibility tests added.

Category 3A:

Electrical/electronic equipment outside the fuselage and well shielded, but exposed to higher lightning induced transient activity. All standard requirements plus lightning induced transient requirements.

Category 3B:

Electrical/electronic equipment unshielded under nonconductive material or on external mountings. All standard requirements apply and lightning plus increased radio frequency susceptibility test levels (possibly 100 V/m).

Category 3C:

Electrical/electronic equipment not on a ground plane. All standard requirements apply and susceptibility and emission requirements are also applied to the equipment circuit ground, power ground, and case ground interface wires.

Category 4:

Support equipment associated with the aircraft. Apply radiated emission and conducted emission requirements on power that connects to aircraft.

Category 5:

Flight-test equipment: Apply radiated emission requirements and radiated susceptibility requirements for external equipment. Flight equipment is isolated from aircraft circuits.

6.3. Suppliers

Early in the program the EMC engineer hastens to know the supplier (vendor or subcontractor). It is important to record and establish a file of equipment identification, electrical characteristics and the tailored requirements:

- Unit: name, acronym, model, unique identifications.
- Company: name, location, engineers, organization.
- Affiliated paper: procurement specification, interface control drawing, statement of work, interface schematics.
- Unit history: new, existing, modified, previous program usage, in-service experience, equipment similarities, deviations, waivers.
- Schedule: technical meetings, prototype/engineering models, test dates, qualification test procedure.
- Grounds/bonds/shields: internal ground, interface bonding.
- Test: development, engineering model, qualification, acceptance.
- Subsystem: project engineer, staff engineer, contracts engineer.

At program startup, there may be specific candidate proposals and subcontractor control plans or procedures to evaluate. These are important attributes to look for:

- Treatment and awareness of specification requirements.
- Unit in-service use on previous programs.
- Past programs experience and success.
- Evaluation and analysis capabilities.
- Implementation of design techniques.
- In-house or provisions for testing facilities.

Of course, the size of the program will dictate the extent of the subcontractor effort, but this might be a typical statement of work:

"The subcontractor shall conduct an early investigation and developmental EMC analysis and test on the breadboard or prototype hardware for new or modified equipment."

"A developmental test report and evaluation shall be forwarded to the prime contractor containing information on noise measurements of interface circuits and proposed wiring, protection, and emission control techniques."

"The subcontractor shall obtain available EMC data and in-service history on existing, off-the-shelf equipment; determine compliance with the program requirements; and, where requirements are not met, recommend options for equipment modifications, deviations, or waivers."

"The subcontractor shall conduct timely technical exchange meetings and present grounding, shielding, hardware, and software protection. A qualification test plan and report shall be submitted for comment and approval."

The statement of work for the subcontractor may be more or less detailed depending on complexity and size of the equipment or subsystem.

Subcontractor equipment design changes are made throughout the entire program. An EMC engineer cannot monitor or assess all engineering changes on all equipment, but a change involving electrical properties may require some reevaluation or retest. For example, anytime a change deals with new coupling characteristics, that is: wiring harness rerouting, new input/output circuits, different circuit board layout, extensive software modification that changes timing sequence or adds new loops, then an analysis or susceptibility retest is appropriate. The vendor maintains complete responsibility and accountability for design and test verification of his unit or subsystem to fully utilize his own facilities, experience, test equipment and software capabilities.

7. VERIFICATION AND VALIDATION

7.1. Key EMC Designs

Verification: Equipment qualification test at a supplier to prove that a design meets specification. The qualification test of equipment forms the very keystone of verification.

Verification is also accomplished in the airframe manufacturer's laboratories to prove performance of key EMC designs and properties, such as:

- Grounding of digital or analog circuits.
- Bonding resistance of aircraft part, wire, or joint.
- Shielding properties of new materials.
- Leakage of joints and apertures.
- Emissions of pulse width modulated power.
- Wiring design and coupling conditions.
- Aspects of aircraft mounted filters and limiters.
- Susceptibility thresholds of interface circuits.

Verification tests quickly lead to the ultimate proper operation on the aircraft. Measurement reveals thresholds and noise upset margins. Verification lays a groundwork before validation.

Validation: Demonstration of and confidence in proper equipment function along with acceptable control of noise during aircraft operation in its intended environment.

Demonstration - A two part measurement and test:

- Operation of an aircraft (or subsystem) through its normal modes while operating equipment and devices.
- Introduction of jeopardizing noise environments to stress equipment: all the while monitoring functioning flight deck instruments and aircraft circuit operation internally and externally: internally with built in test circuits/equipment (BITE) and externally with meters, transient recorders, oscilloscopes, spectrum analyzers, and digital bus analyzers.

An airplane program must be structured on a foundation of solid verification and validation (see section 9).

7.2. Validation Plans

First, hasten to collect, document, and rely on existing qualification data, existing validated technologies, and validation by similarity.

Otherwise, validation tests can be run at the subsystem level in a flight avionics laboratory or on the production aircraft.

When considering subsystem versus aircraft test, assess the following deficiencies and benefits in cost, effectiveness, flexibility, and timeliness:

TABLE 7.2-1. SUBSYSTEM VERSUS AIRCRAFT TEST

COST:	CAPITAL EQUIPMENT	ATTENDANT PERSONNEL	EQUIPMENT REWORK	TEST EQUIPMENT	LEARNING CURVE
Subsystem	Very Low	Low	Low	Low	Low
Airplane	High	High	High	High	High
ENVIRONMENT:	EQUIPMENT COLOCATION	RF SIM- ULATION	GROUND/SHIELDS	RESONANT CONDITIONS	
Subsystem	Poor	Good	Poor	Different	
Airplane	Actual	Good	Actual	Actual	
DESIGN:	FINAL HARDWARE	FINAL SOFTWARE	TEST SOFTWARE	FINAL WIRING	ENGINEER AVAILABLE
Subsystem	Good	Good	Excel	Fair	Good
Airplane	Excel	Excel	Excel	Actual	Fair
PROCEDURE:	APPROVAL SIGNATURES	FORMAL SEQUENCE	DECISIONS/ DIAGNOSTICS	CHANGE CONTROL	PROGRAM TIMING
Subsystem	Few	No	Best	Good	Fair
Airplane	Many	Yes	Fair	Good	Late

Aircraft offers fidelity, but demands a high price with inflexibility in a dedicated airplane outfitted with test equipment. The avionics laboratory is an active testing and simulation facility and testing can often be done speedily on a "noninterference" basis when test time is slack. Circuit access panels already exist. Testing can be easy, informal, flexible and promote diagnostics and real-time knowledge of noise behavior. A range of tests can be run at the subsystem level with selected tests on the aircraft.

7.3. Validation Procedure

The subsystem or aircraft configuration, setup, test equipment, and modes are identified and recorded paragraph by paragraph for each separate mode. The procedure then records step-by-step operation of equipment.

CONFIGURATION: Model, make, serial numbers, outstanding engineering changes or deviations, wiring or customer configuration, software, test software, test date, location.

TEST SETUP: Equipment bay lights, test stands, ground planes, shield ties, ground wires, unique simulations, monitors.

TEST EQUIPMENT: List test equipment and installation for the following systems by paragraph section:

- Flight control/flight management: computer breakout boxes, digital bus analyzers, oscilloscopes.
- Communication/navigation: interphone electronics breakout boxes, receiver breakout boxes, interphone headsets, RF voltmeters, spectrum analyzer.
- Power: power supply breakout boxes, chart recorders, transient analyzers, peak reading (transient) voltmeters.
- Flight indicating/recording: flight instrument breakout boxes, digital bus analyzers.
- Engine: monitors, indicators.
- Environment: magnetic field measuring instruments, spectrum analyzer and current probe (wiring) and antennas (radiated emission), signal generators.

PERSONNEL: Knowledgeable subsystem or equipment engineers, experienced EMC engineers and technicians. Monitor and record all initial settings, indicator light status, measurement instrumentation, BITE readouts. On functioning displays, observe and record status of flight instruments, CRT panels on flight deck, left, right, and center control panels, power panels. During step-by-step procedure, record upsets, circuit breaker disconnects, state changes/events, autopilot disconnect, warning signals/flags, annunciations, CRT distortion on all subsystems.

SOFTWARE: Identify software installation and initiation for each mode. Automated test software routines exercise processing functions and interfaces and monitor the data, status, error counters, or fault logging in real-time or on printout.

MODES OR PRESETS: Establish modes and settings for all systems:

- Environmental control: pressurization, temperature, air control.
- Flight control/management: mode control panel, autopilot engage at "left or right command"; test software installation, flight management computer; control display unit, flight path program, takeoff, cruise, landing, autoland; engine control operation.

- Communication/navigation: ADF, HF, VHF, ATC transmit/receive frequency modes and settings, IRS navigation modes, selector panel interphone left and right settings.
- Power: circuit breakers engaged/disengaged.
- Fuel and hydraulics: pumps, valves.
- Flight instruments and recorders: flight instrument, caution/advisory readouts, recorders.
- Lighting: light settings, dimmers, strobes.
- Engine: electronic engine control settings, indicators.
- Degraded modes: loss of redundant unit, low battery.

FUNCTIONAL SWITCHING TEST PROCEDURE: Establish a step-by-step procedure paragraph for each mode and possibly for each major subsystem. For chosen mode perform "functional switching test?" That is: operate, switch on/off, cycle, or exercise all units:

- Environmental control: pressurization, temperature, fans.
- Flight control/flight management: motors, actuators, flaps, slats.
- Communication/navigation: HF-VHF transmit.
- Power: all circuit breakers.
- Fuel and hydraulics: pumps, valves, solenoids.
- Flight indicating/recording: flight instruments, CRTs, recorders
- Lighting: cabin, flight deck, strobe, and landing lights.
- Engine: electronic controls, indicators, actuators, indicators.

A subsystem or airplane EMC validation test procedure demonstrates the absence of malfunction or undesirable noise and authenticates that the repeatability, accuracy, and reality of operation in normal modes of an airplane baseline production configuration meets the Federal Aviation Regulations and the Aircraft System Specification.

ENVIRONMENTAL STRESS TEST PROCEDURE: Establish separate paragraph for each mode or subsystem as above and operate equipment while introducing transients or radio frequency energy. The policy is "make it upset" not "show success?" If noise is transient, vary the pulse repetition rates relative to computer processing control cycles. Equipment should operate within performance specification and also demonstrate:

- Maintenance of proper internal configuration.

- Successful automatic restart.
- Continued operation.
- No permanent faults.
- No adverse control surface motion.
- Automatic transfer to secondary systems.

NOISE MEASUREMENT: Measure and characterize the signature of noise on the digital data bus, communication circuits or cables, interphone circuits, powerline bus, analog instrument circuits, and in the aircraft ambient to assist in validating operation and establishing a data base for diagnostics, not only for the existing program, but for future programs. Establish a separate paragraph for each mode. The following are noise types or thresholds:

- Clock oscillator harmonics or HF-VHF transmitter signals on circuits and cables.
- ADF-HF-VHF threshold sensitivity; passenger address circuits, voice recorder, crew interphones noise and thresholds.
- Powerline switching transients at circuit breakers and equipment terminals, switching regulator harmonics on buses and on cables.
- 400-Hz electric and magnetic field strengths, HF-VHF field strengths.

BONDING RESISTANCE TEST: Measure resistance of structure joints, skin panels, electrostatic paints, liners, foils, doors, flight deck panels, strut fairings and wing.

Testing and test assessment rests so heavily on a product's history and the intent of the product's function and future use, that it is, of course, an individual or program responsibility to define test concepts, requirements, and procedures. This validation test information, therefore, represents possible guides or techniques, but must be considered as not being appropriate for a specific test.

7.4. Program Design Reviews

During a program there is no quicker way to help improve the validity and timeliness of the design and specifications than by periodic reviews to establish agreement on perceived requirements and outcome. The following items are for consideration and might be selected and presented in a program preliminary design review or critical design review to help lay groundwork for verification, validation and certification:

- Program documentation, organization.
- Aircraft system and RTCA/DO-160 design/test requirements.

- Environment assessment, equipment EMC categories.
- Subsystem, equipment architecture/topology.
- Grounding system plan.
- Bonding: radio frequency/static/safety.
- Shielding: aircraft and wiring
- Wiring design, critical circuits.
- New technologies analysis/models.
- Equipment/subsystem/aircraft verification/validation plan.

The prompt review of key documentation and requirements helps to start and keep the program on a successful path.

APPENDIX A - TEST AND TEST LIMITS

Early EMC history has seen the radio receiver as the centerpiece of electromagnetic field testing. It was necessary to find the source of noise entering aircraft audio circuits and radios. The keys today, in measuring emission and susceptibility parameters, are the oscilloscope and spectrum analyzer; the need for noise control spread to instrumentation and automatic pilot circuits.

We are now seeing the need for digital logic analyzers, bus controllers, and automatic, interactive testing capability to measure performance in fly-bywire, negative stability, aircraft systems.

Spectrum analyzers and storage oscilloscopes are proven and powerful troubleshooting tools in the measurement of noise. The scope displays the analog time domain waveform. The spectrum analyzer lays out the frequency components of that waveform. The scope shows peak amplitude. The spectrum analyzer reveals amplitudes versus frequency. With a scope, the pulse repetition rate is measurable. And with the spectrum analyzer, harmonics are caught in any frequency band. But, although spectrum analyzers and digitizing oscilloscopes are useful in finding and characterizing noise, it is ineffective to apply them to the task of detection of errors or noise margins in computers. The EMC community now needs these units to be paired with high-speed data bus analyzers and logic analyzers in order to provide even simple measurements of computer processor operations, such as: timing sequence, state activity, and bus status.

Recently, the processing functions of an avionic computer were upset by noise causing the unit to suspend operation, to "lock up?" It then required recycling of power to restart operation. Personnel from the electromagnetic compatibility group were asked to help diagnose the unpredictable upsets. An oscilloscope (analog waveforms) and the spectrum analyzer (frequency components) displayed noise occurring on the wire returns, grounds, and logic power supply lines; the noise levels were high enough to interfere with clock and data signals; wavering clock timing pulses destabilized the processing sequences; noise and unstable signals appeared everywhere on the circuit boards traces.

Many days were spent on this problem, but with "analog" instrumentation a solution could not be found until logic analyzers and bus controllers were brought in. Without the capability to control data entry formats, observe and evaluate output activity, timing sequences, and to correlate state changes with noise events, timely solutions to complex problems become impossible.

Future validation testing on automated aircraft systems will require laboratory personnel to monitor simultaneously occurring events, to visually correlate timing, logic state changes, and to automatically record data, decoded and converted, in real-time under a number of aircraft modes. Digital interface bus

analyzers, logic analyzers, and interactive computer controllers will offer fast solutions and bring about professional insights to noise problems on new digital avionics.

Laboratory personnel are called upon to execute a variety of tasks:

- Circuit research.
- Test equipment construction.
- Diagnostics and troubleshooting.
- Evaluation of avionics performance.
- Avionics and airplane qualification testing.

And laboratory personnel diagnose problems in a variety of aircraft units: computers, power controllers, transmitter/receivers amplifiers, motors/generators, analog sensors, all of which encompass a wide range of characteristics. Functions of the aircraft, in a flight context, span the systems of environmental control, flight control, flight management, fuel, communication, navigation, power, and engine controls, but to the EMC engineer and laboratory technician these functions bring to mind susceptibilities and emissions:

- Power: 400-Hz, dc buses, motors, relays, switching events.
- Dc-to-dc switching regulators, pulse width modulation controllers.
- Radio frequency receiver thresholds and transmitter antennas fields.
- Data transmission and clock oscillators.
- Sensitive analog (audio) sensors and circuits.

Data, information, and histories already exist on environmental levels and are documented in the RTCA/DO-160B and MIL STD 461 specification (see table A-1).

But, for noise effects on data, transmission and timing sequences relative to conditions of circuit stability and upset margins, the EMC community today does not have an adequate data base.

Speeds of future systems will increase, voltage levels will rise, and sensor and receiver thresholds will be lower and more sensitive. Emerging flat panel displays, microprocessor controls, voice controlled systems, dc power systems, and pulse width modulated, electric actuators will be available soon for full authority flight controls and will be beyond the reach of engineers in the EMC community for test and analysis.

In the shield room, the controlled environment and standard electrical references (for instance, voltage, current, impedance, frequency meters, and ground planes) offer established laboratory conditions for analysis and tradeoffs of digital flight control designs. But, most of the time during

today's EMI tests, current probe factors, antenna factors, line losses, attenuation factors, and bandwidth conversions are now essentially hand calculated and joined with raw data almost on a point-by-point basis in an anachronistic, time-consuming process. These computations are a hindrance, but the more important loss is inability to assess trends and compare, in real-time, circuit operational changes in a controlled and repeatable manner under various noise levels.

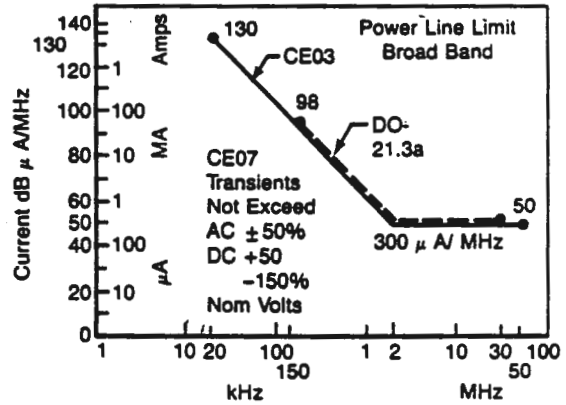
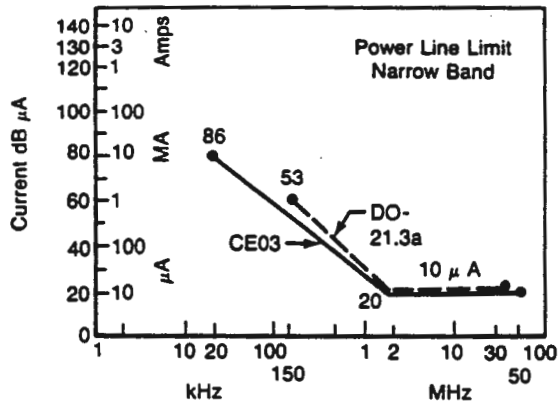
Controlling, probing, comparing, and recording status and data transmissions (being digital in nature) against the itinerant noise occurrences (having analog waveforms) stands or falls on the test and simulation capability.

Effectiveness resides in the technician's skill and his experience coupled with the tools and equipment with which that skill and experience is implemented. New, modern equipment having automatic microprocessor controls and automatic data readout is becoming available to provide interactive test decisions. Equipment and tools of the "analog" 1960s and 1970s cannot carry the load or be compatible with the flight controls of the "full-authority," "fly-by-wire" 1990s.

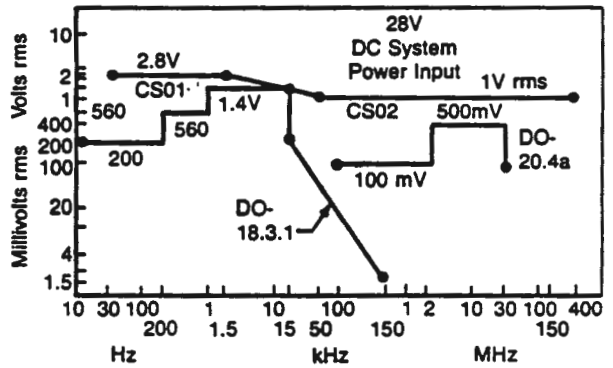
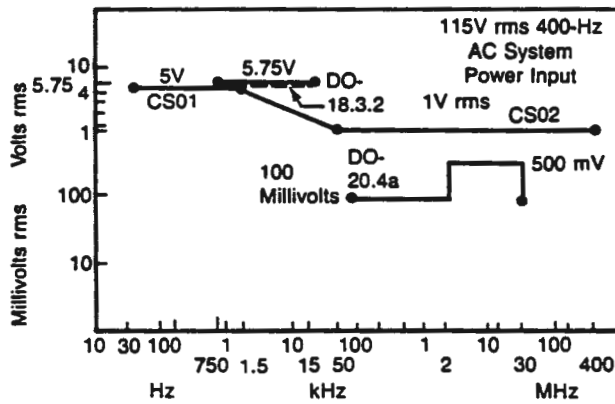
TABLE A-1. DO-160B AND MIL STD 461B. TEST CROSS REFERENCE
(see figure A-1)

TEST	160B	461B
POWER BUS:		
Conducted Emission	21.3a	CE03
Switching Transients CE	None	CE07
Conducted Susceptibility (CS):		
Audio Frequency CS	18.3	CS01
Radio Frequency CS	20.4a	CS02
Power Line Spike	17.3	CS06
Bus Momentary Interruption	16.5.1.4	None
EQUIPMENT AND SIGNAL CABLE:		
Conducted Emission Cable	21.3b	None
Induced into equipment and cable:		
Magnetic "H" Field Equipment	19.3.1	RS02
Magnetic "H" Field Cable	19.3.2	RS02
Transient Spike: 200V	None	RS02
Electric "E" Field Cable	19.3.3	None
Inductive Switching Transient	19.3.4	None
Radio Frequency CS Cable	20.4b	None
EQUIPMENT & INTERCONNECTING CABLE:		
Radiated Emission	21.4	RE02
Radiated Susceptibility	20.5	RS03

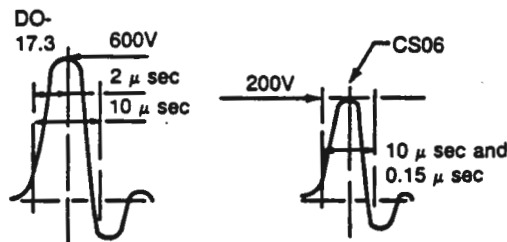
POWER LINE CONDUCTED EMISSION



CONDUCTED SUSCEPTIBILITY



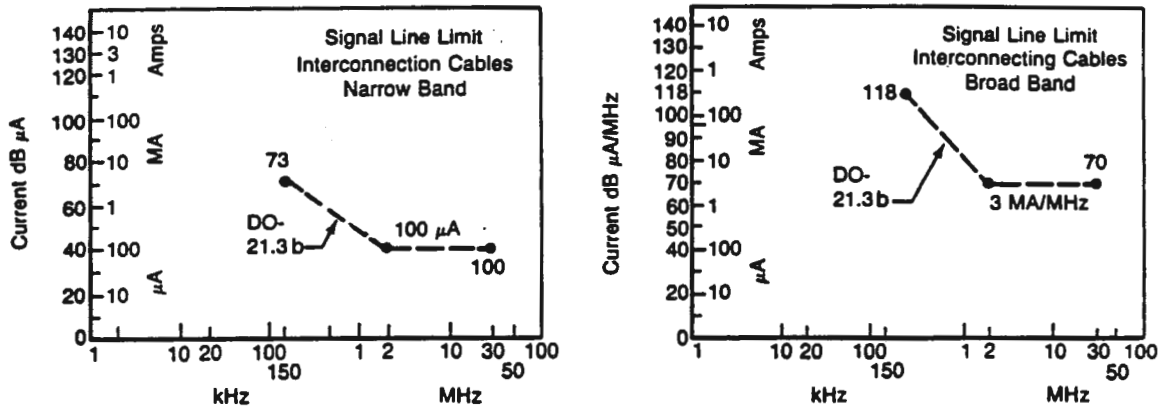
Power Input Transient



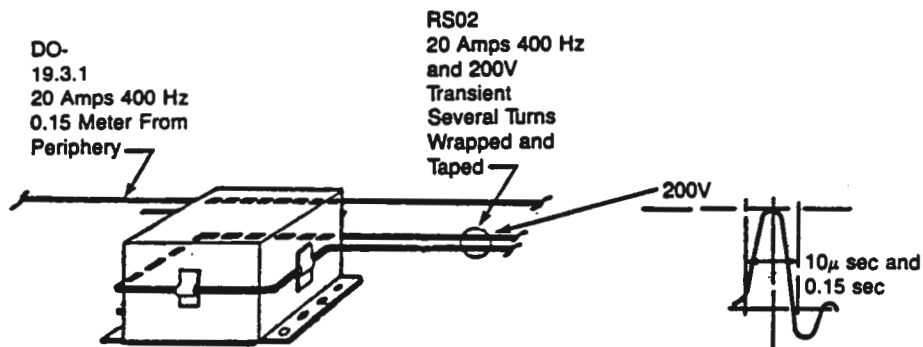
Note: The "DO-" indicates the RTCA/DO-160B paragraph.

FIGURE A-1. EMI TEST LIMITS (1 of 4)

SIGNAL INTERCONNECTING CABLES CONDUCTED EMISSION



INDUCED FIELDS—EQUIPMENT-CABLE



SIGNAL INTERCONNECTING CABLE MAGNETIC ("H") FIELD

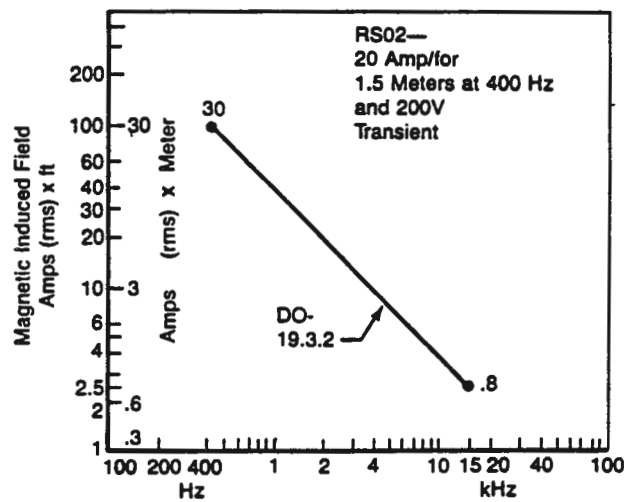


FIGURE A-1. EMI TEST LIMITS (2 of 4)

INDUCED FIELD—EQUIPMENT-CABLE (CONTINUED)

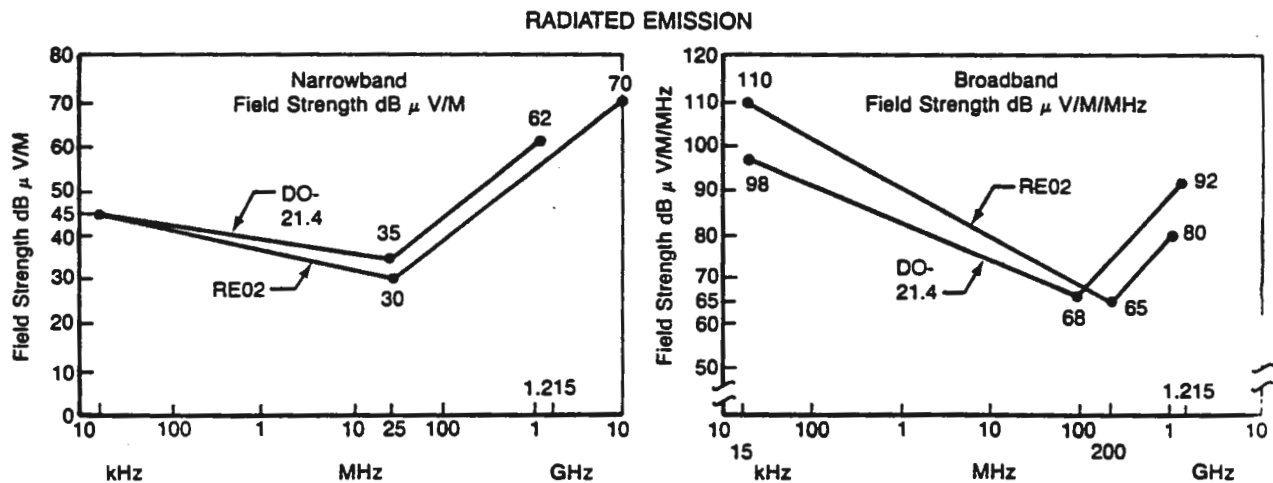
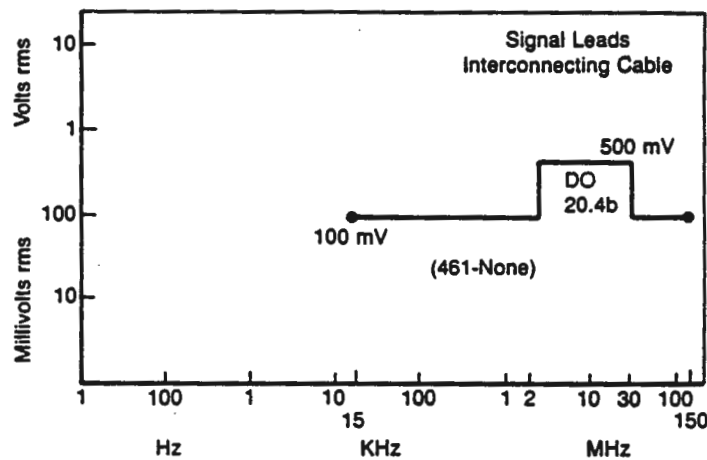
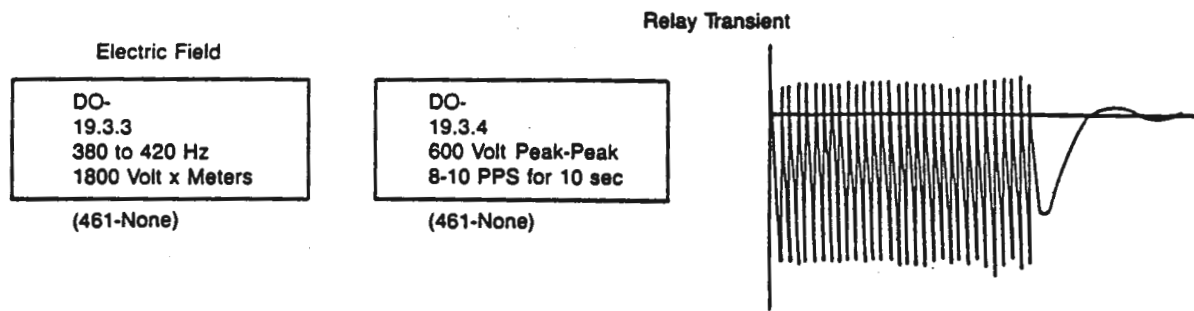


FIGURE A-1. EMI TEST LIMITS (3 of 4)

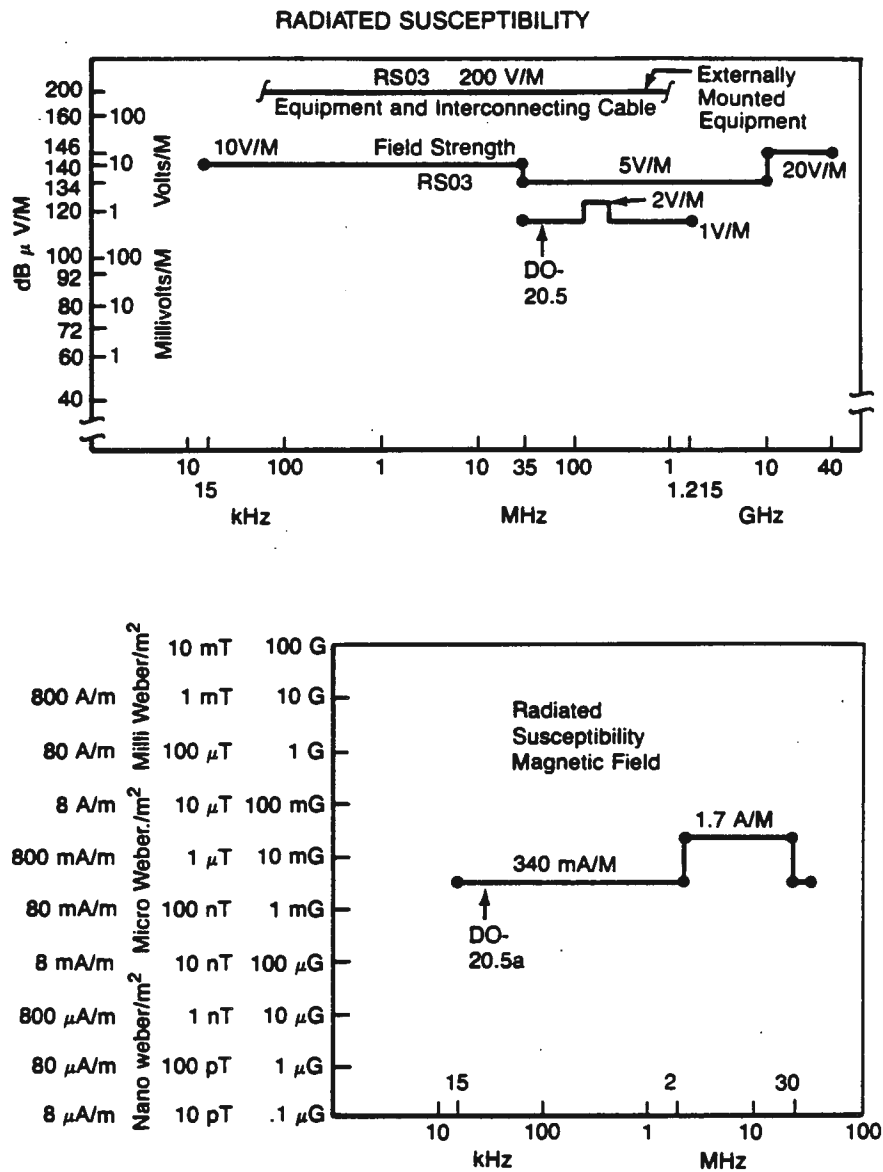


FIGURE A-1. EMI TEST LIMITS (4 of 4)



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25.1351, General

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GLOSSARY OF TERMS

ABSORPTION LOSS. Attenuation or retention of electromagnetic energy passing through a material, a shield. Absorption loss and reflection loss contribute to total shielding effectiveness (SE).

ANODIZE. A preparation by electrolytic process that deposits a protective oxide, insulating film on a metallic surface (aluminum). The oxide defeats electrical bonding. Alodine and iridite finishes on aluminum are conductive.

APERTURE. An opening, such as a nonconductive panel joint, slot or crack, allowing electromagnetic energy to pass through a shield.

AUDIO FREQUENCY (AF). The spectrum (20 to 20,000 Hz) of human hearing, often defined as extending from approximately 20 Hz to 50 kHz and sometimes to 150 kHz. Audio noise is nuisance hum, static, or tones from power line 400 Hz, switching regulator and digital clock harmonics, or HF,VHF transmitter frequencies.

BACKSHELL. Metal shell connecting circuit shields or overbraid to an electrical connector.

BALANCED CIRCUIT. A signal, acting line-to-line, between two conductors having symmetrical voltages identical and equal in relation to other circuits and to ground. "Differential mode" is line-to-line; "common mode" is line to ground.

BANDWIDTH (BW). Frequencies bounded by an upper and lower limit in a given band associated with electronic devices, filters, and receivers.

BOND, ELECTRICAL. Electrical connection at two metallic surfaces securely joined to assure good conductivity often 2.5-m maximum for electrical/electronic units and 1 Ω for electrostatic dissipation or safety. A "faying surface" bond maintains contact between relatively large or long surfaces. Inherently bonded parts are permanently assembled and conductivity exists without special preparation: such as with welding, brazing.

BRAID, OVERBRAID. Fine metallic conductors woven to form a flexible conduit or cableway and installed around insulated wires to provide protection against electric fields and radio frequencies. Best when peripherally connected to backshells. A grounding strap/jumper may be made of braid.

CABLE OR HARNESS. A bundle of separate, insulated, electrical circuits, shielded or unshielded, usually long and flexible and having breakouts, terminations, overbraid, and mounting provisions completely assembled.

CABLEWAY. A solid metallic housing (liner, foil, coating) surrounding and shielding insulated electrical conductors. Also called conduit, tray, or

raceway. Crosswise or transverse openings or breaks in the metallic cableway cause noise voltages to be transferred to internal wire circuits.

COMMON MODE (CM) IMPEDANCE. Impedance or resistance shared by two or more circuits so that noise voltages/currents generated by one are impressed on the others.

COMMON MODE REJECTION. The ability of wiring or an electronic device to reject common mode (line-to-ground) signals and maintain fidelity of differential mode (line-to-line) signals.

COMMON MODE SIGNAL. Identical and equal signals on input conductors or at the terminals of a device relative to ground.

CONDUCTED EMISSION (CE) OR INTERFERENCE. Voltage/current noise signals entering or leaving a unit on interface conductors-emission is the general term, interference is undesired noise.

COUPLING. The transfer of energy between wires or components of a circuit electrostatically, electromagnetically, or directly.

CROSS COUPLING (CROSSTALK). Transfer of signals from one channel, circuit, or conductor to another as an undesired or nuisance signal or the resulting noise.

DAMAGE. The irreversible failure of a component.

DECIBEL (dB). Decibel expresses the ratio between two amounts of power, P_1 and P_2 , at two separate points in a circuit. By definition, the number of dB = $10 \log$ to the base 10 of (P_1/P_2) . For special cases, when a standard power level $P_2 = 1 \text{ mW}$ or 1 W or 1 kW , then the ratio is defined as "dBm," "dBw," or "dBKW." Because $P = V^2/R$ and also I^2/R , decibels express voltage and current ratios. Ideally, the voltages and currents are measured at two points having identical impedances. By definition, $\text{dB} = 20 \log V_1/V_2$ and $\text{dB} = 20 \log I_1/I_2$. For convenience, V_2 or I_2 are often chosen as $1 \mu\text{V}$ or $1 \mu\text{A}$ and the ratio is defined as dB above a μV or dB above a μA when graphing emission or susceptibility limits.

DIELECTRIC STRENGTH. Voltage withstand capability that an insulating material sustains before destructive arcing and current flow, usually expressed in volts per mil thickness. Dielectric withstand voltage is the voltage level at which insulation breakdown occurs.

DIFFERENTIAL MODE (DM) SIGNAL. The signal in a two-wire circuit measured from line-to-line.

DUAL GROUND. Equipment case ground/return through two independent circuit paths to structure implemented in flammable zones and water leakage areas-each path meeting electrical conductivity (resistance) requirements.

ELECTRIC FIELD. High-impedance, radiated voltage field, positive or negative, from a voltage source as contrasted to a low-impedance magnetic field from a current source.

ELECTROMAGNETIC COMPATIBILITY (EMC). Operation within performance specification in the intended electromagnetic interference environment.

ELECTROMAGNETIC INTERFERENCE (EMI). Conducted and radiated voltage/current noise signals, broadband (BB) or narrow band (NB), that degrade the specified performance of equipment.

ELECTROSTATIC CHARGE. Electric potential energy with a surrounding electric field, uniform or nonuniform, moving or at rest, on a material.

EMISSION. Voltage/current noise on a wire or in space. Broadband emission has uniform spectral energy over a wide frequency range and can be identified by the response of a measuring receiver not varying when tuned over several receiver "bandwidths." Or, energy present over a bandwidth greater than the resolution bandwidth where individual spectral components cannot be resolved. Broadband (BB) may be of two types: (1) impulse and coherent varies 20 dB per decade of bandwidth and (2) random or statistical, varies 10 dB per decade. A narrow band (NB) emission or signal, sometimes called continuous wave, occurs at a discrete frequency and does not vary with bandwidth.

FAULT CURRENT. The maximum current (magnitude and duration) flowing through a fault point-equal to the supply voltage divided by the dc resistance of power line leads, circuit breakers, and the current return in wire or structure.

FILTERING. Device or unit that passes or rejects a frequency band and designed to block noise from entering or leaving a circuit or unit.

GROUND. A generic term having multiple meanings and indicating a circuit return path or a voltage reference: not "zero" voltage reference. Four hundred millivolts of noise voltage is common on "quiet" grounds. There are several types of returns and references.

Return:

- Structure, for power, fault, and "discrete" circuits.
- A grid of wires, solid sheet, or foil.
- A wire from circuit load back to source or to case.
- Circuit card "ground plane," also a reference and shield.

Reference:

- Structure, for electronics, shields, power.
- A grid of wires, solid sheet, or foil.
- A wire from circuit to grounding block or case.
- A wire from circuit to structure.

- Shield tie.
- Earth.

IMMUNITY. Capability of a circuit or unit to operate within performance specification in a specified electromagnetic interference environment.

ISOLATION. Electrical separation and insulation of circuits from ground and other circuits or arrangement of parts to provide protection and prevention of uncontrolled electrical contact.

JUMPER/STRAP. A short wire, strip, strap, or braid conductor installed to make a safety ground connection, to dissipate electrostatic charge, or establish continuity around a break in a circuit.

LIMITING. VOLTAGE/CURRENT. Semiconductor components, diodes, Transorb, or filter designed to clip and shunt to ground an applied transient or steadystate voltage. Used to protect against noise frequencies, faults, lightning, and inductive switching transients.

MAGNETIC FIELD. A radiated, low-impedance field having lines of "flux" or magnetomotive force associated with an electrical current.

MALFUNCTION. Failure or degradation in performance that compromises flight safety.

NOISE. Conducted or radiated emission causing circuit upset, performance disorder, or undesired sound.

PRECIPITATION STATIC (P-static). Electrostatic discharge, corona, arcing, and streamering, steady state or impulsive, causing circuit upset, receiver noise or component damage.

RADIATED EMISSION (RE). Electromagnetic energy transmitted and propagated in space usually considered as audio frequency or radio frequency noise.

RADIO FREQUENCY (RF). Frequencies in the electromagnetic spectrum used for radio communications extending from kilohertz to gigahertz.

RADIO FREQUENCY INTERFERENCE (RFI). Electromagnetic interference in the radio frequency range.

SEALANT. An applied substance enclosing and protecting the integrity of a joint, fastener, or electrical bond from moisture, contaminants, oxidation, and acid or alkaline corrosion.

SHIELD. A conductive material, opaque to electromagnetic energy, for confining or repelling electromagnetic fields A structure, skin panel, case, cover, liner, foil, coating, braid, or cable-way that reduces electric and magnetic fields into or out of circuits or prevents accidental contact with hazardous voltages.

SHIELD EFFECTIVENESS (SE). The ability of a shield to reject electromagnetic fields. A measure of attenuation in field strength at a point in space caused by the insertion of a shield between the source and the point.

SIGNAL RETURN. A wire conductor between a load and the signal or driving source. Structure can be a signal and power return. Commonly, it is the low voltage side of the closed loop energy transfer circuit.

SINGLE-ENDED CIRCUIT. A circuit with source and load ends grounded to case and structure and using structure as return.

STRUCTURE. Basic members, supports, spars, stanchions, housing, skin panels, or coverings that may or may not provide conductive return paths and shields for electrical/electronic circuits.

SUSCEPTIBILITY. Upset behavior or characteristic response of an equipment when subjected to specified electromagnetic energy-identified with the point, threshold, or onset of operation outside of performance limits. Conducted Susceptibility (CS) applies to energy on interface conductors; Radiated Susceptibility (RS) to radiated fields.

THRESHOLD, NOISE. The lowest electromagnetic interference signal level that produces onset of susceptibility.

UPSET. Temporary interruption of performance that is self-correcting or reversible by manual or automatic process.

UNACCEPTABLE RESPONSE. Upset, degradation of performance, or failure, not designated a malfunction, but is detrimental or compromising to cost, schedule, comfort, or workload.

UNDESIRABLE RESPONSE. Change of performance and output, not designated a malfunction or safety hazard, that is evaluated as acceptable as is because of minimum nuisance effects and excessive cost burdens to correct.

VALIDATION. Demonstration and authentication that a final product operates in all modes and performs consistently and successfully under all actual operational and environmental conditions founded upon conformance to the applicable specifications.

VERIFICATION. Demonstration by similarity, previous in-service experience, analysis, measurement, or operation that the performance, characteristics, or parameters of equipment and parts demonstrate accuracy, show the quality of being repeatable, and meet or are acceptable under applicable specifications.

ACRONYMS AND ABBREVIATIONS

A/C	Aircraft
ACARS	ARC Communications Addressing and Reporting System
ACT	Active Controls Technology
ADC	Air Data Computer
AF	Audio Frequency
ADF	Automatic Direction Finder
AFCS	Automatic Flight Control System
ANSI	American National Standards Institute
APU	Auxiliary Power Unit
ARC	Aeronautical Radio, Inc.
ASDS	Airport Surface Detection System
ATCRBS	Air Traffic Control Radar Beacon System
BB	Broadband
BITE	Built-In Test Equipment
BW	Bandwidth
CAS	Criticality Advisory System
CDU	Control Display Unit
CE	Conducted Emission
CM	Common Mode
CRT	Cathode Ray Tube
CS	Conducted Susceptibility
DFDAU	Digital Flight Data Acquisition Unit
DFDR	Digital Flight Data Recorder
DITS	Digital Information Transfer System
DM	Differential Mode
DME	Distance Measuring Equipment
E	Electromagnetic Environmental Effects
EADI	Electronic Attitude Director Indicator
ECAC	Electromagnetic Compatibility Analysis Center
ECS	Environmental Control System
E/E	Electrical/Electronic
EEC	Electronic Engine Control
EED	Electro-Explosive Device
E-FIELD	Electric Field
EFIS	Electronic Flight Instrument System
EGT	Exhaust Gas Temperature
EHSI	Electronic Horizontal Situation Indicator
EICAS	Engine Indication and Crew Alerting System
EM	Electromagnetic
EMC	Electromagnetic Compatibility
EME	Electromagnetic Effects
EMI	Electromagnetic Interference
EMIC	Electromagnetic Interference/Compatibility
EMP	Electromagnetic Pulse
EPR	Engine Pressure Ratio
ESD	Electrostatic Discharge

ESE	Electric (field) Shield Effectiveness
FCC	Flight Control Computer
FDEP	Flight Data Entry Panel
FMC	Flight Management Computer
Gr/Ep	Graphite/Epoxy
GPS	Global-Positioning-System
GPWS	Ground Proximity Warning System
HF	High-Frequency
H-FIELD	Magnetic Field
IAAC	Integrated Application of Active Controls Technology (to an Advanced Subsonic Transport Project)
IDG	Integrated Drive Generator
ILS	Instrument Landing System
INS	Inertial Navigation System
IRS	Inertial Reference System
LCC	Life Cycle Cost
LOC	Localizer
LRRA	Low Range Radio Altimeter
LRU	Line Replaceable Unit
MCDP	Maintenance Control and Display Panel
MCP	Mode Control Panel
mil	One thousandths of an inch (0.001)
MLS	Microwave Landing System
MSE	Magnetic (Field) Shielding Effectiveness
NB	Narrow Band Signal
N1	Fan Speed
N2	Core Engine Speed
OMEGA	Very Low Frequency Navigation
PCU	Power Control Unit
PRF	Pulse Repetition Frequency
P-static	Precipitation Static
PWM	Pulse Width Modulation
RDMI	Radio Distance Magnetic Indicator
RE	Radiated Emission
RF	Radio Frequency
RFI	Radio Frequency Interference
RS	Radiated Susceptibility
RTCA	Radio Technical Commission for Aeronautics
S/A	Spectrum Analyzer
SE	Shielding Effectiveness
SHF	Super High-Frequency
TCAS	Traffic Alert and Collision Avoidance System
TLA	Thrust Lever Angle
TMC	Thrust Management Computer
TTL	Transistor-Transistor-Logic
UHF	Ultra High-Frequency
VHF	Very High-Frequency
VLF	Very Low-Frequency
VOR	VHF Omnidirectional Range
VORTA/VHF	Omnirange/Tactical Air Navigation
VSI	Vertical Speed Indicator
WRU	Weapons Replaceable Unit





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HANDBOOK-VOLUME II DIGITAL SYSTEMS VALIDATION

CHAPTER 12 FAST RISE-TIME ELECTRICAL TRANSIENTS IN AIRCRAFT



PREPARED BY:

**COMPUTER RESOURCE MANAGEMENT, INC.
950 HERNDON PARKWAY, SUITE 360
HERNDON, VIRGINIA 22070**

PREPARED FOR:

**FEDERAL AVIATION ADMINISTRATION
TECHNICAL CENTER
ATLANTIC CITY INTERNATIONAL AIRPORT, NEW JERSEY 08405**

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1. INTRODUCTION

1.1. Topic History

Fast rise-time transients have always had potentially serious effects upon aircraft structures, wiring, and systems. With the application of fly-by-wire control systems to engines and control surfaces, an understanding of the nature and effects of fast transients is now even more essential for certification authorities.

The measures, which can be taken to protect aircraft and their systems against fast transients, must also be understood for certification authorities to decide whether an aircraft is adequately protected.

Solid state devices and digital systems have improved reliability by orders of magnitude compared to the vacuum tubes, analog systems, and electromechanical devices which they replaced. Unfortunately, solid state devices and digital systems are far more susceptible to electromagnetic interference than were the older devices. Therefore, much care must be given early in the design phases of new systems to ensure against upset or damage from a wide variety of electromagnetic fields generated both by the aircraft systems themselves and by sources external to the aircraft.

This chapter describes and analyses some of the fast rise-time transients to which modern aircraft are exposed. Estimates are made of the energy levels and field strengths which these transients are capable of generating. These energy levels and field strengths are compared to the typical upset and damage thresholds which exist in aircraft systems and devices. Various methods of testing systems for susceptibility to fast transients are described. Finally, the measures which can be taken to protect systems and devices from these transients are described.

1.2. Topic Overview

Fast rise-time transients include the Nuclear Electromagnetic Pulse (NEMP), lightning, and Electrostatic Discharge (ESD). The primary power lines of aircraft are also subject to fast transient disturbances. These types of transients will also be described.

Historically, there has been a large difference in the levels of electromagnetic protection applied to civil and military aircraft. Civil aircraft are subject to the rather modest requirements of Radio Technical Commission for Aeronautics (RTCA) DO-160B. Military aircraft are subject to the rigorous requirements of MIL-STD-461C and MIL-STD-462.

Recently there has been widespread recognition that the electromagnetic environment is similar for many military and civil aircraft. (One notable

exception is the very severe environment to which Navy carrier aircraft are exposed.) Ground-based radar systems to which aircraft are exposed are much the same at military and civilian airports. Many airports are jointly used by both civil and military aircraft. Air traffic control routes and procedures expose civil and military aircraft to a similar electromagnetic environment. In the future, the Electromagnetic Compatibility (EMC) requirements for civil aircraft will be close to the requirements for military aircraft.

The potential for exposure of civil aircraft to the NEMP is a debatable point, but the civil air fleet constitutes a logistical reserve for the military. More importantly, the electrical characteristics of NEMP, lightning, and ESD are so similar that a description of all three and a comparison of their characteristics is useful as a tutorial method. Furthermore, the methods used to protect aircraft against all these transients have much in common.

In this chapter, military specifications and test methods are referenced. It is recognized that these specifications and methods do not apply directly to civil aircraft, but a thorough knowledge of the military approach is most important for understanding the evolving field of aircraft EMC.

The effects of lightning are addressed in chapter 13 of this handbook, and therefore will not be covered in depth in this chapter. The characteristics which the lightning transient has in common with other fast transients will, however, be explored.

1.3. Function Criticality

The fast transients described have the capability of upsetting or damaging a variety of systems which are critical to the continued safe flight and landing of the aircraft.

For junction type devices (which include transistors, integrated circuits, and diodes) it is usual to assign a burnout level of 100 microjoules. This energy level can easily be exceeded when an aircraft is subjected to NEMP, lightning, or ESD.

The consequences of circuit damage are obvious: the circuit function will be lost or at least impaired. Backup systems may have to be called into service.

Circuit upset, short of damage, may also occur. Transistor-Transistor Logic (TTL) circuits can be upset by spurious voltages of 0.4 volts. The electromagnetic fields generated within the aircraft by fast transients can generate voltages which readily exceed this value.

There are several possible consequences of circuit upset. An automatic computer controlled reset may occur. A manual reset by a flight crew member may be necessary. Or it may be necessary to completely power down a system and restart it.

The consequences of upset or damage depend, of course, on what the affected system is controlling and how long it takes to reset the system or bring backup controls into operation. Obviously, loss of engine or flight surface controls

may be extremely critical depending upon the phase of flight. Loss of navigation or communications systems is less critical. The presence of backup systems, such as on present day fly-by-wire engine controls, greatly affects the criticality.



2. TRANSIENT ANALYSIS: THE DOUBLE EXPONENTIAL PULSE

Lightning, NEMP, and ESD can all be described in terms of the double exponential pulse. The common characteristics of all three phenomena are that they display a fast rise-time followed by a much slower fall-time. Such a pulse is shown in figure 2-1.

For each type of pulse, the rise- and fall-times are different, and the maximum voltages or currents are different, but the pulse shapes are similar.

The NEMP has a rise-time of about 5 nanoseconds and a fall-time of perhaps 600 nanoseconds. Lightning has traditionally been assigned a rise-time of 1 to 2 microseconds and a fall-time of 50 to 100 microseconds. However, recent investigations have shown that a significant proportion of lightning discharges have rise-times in the range of 200 nanoseconds or less. The ESD pulse has a typical rise-time (10 percent to 90 percent) of 2 nanoseconds, and a fall-time to the 50 percent level of 200 nanoseconds.

The ESD pulse has an amplitude of from 1 to 30 kilovolts. The NEMP pulse has an amplitude of about 50 kilovolts, and the voltage induced by lightning on an aircraft structure can range up to about 60 kilovolts, although the peak voltage of the lightning strike itself may be much higher. The voltages induced on aircraft wiring by lightning and NEMP are much smaller, on the order of 600 to 1500 volts, and voltages of this magnitude are often specified in pin level tests of system cabling.

Mathematically, the voltage of the double exponential pulse can be represented as:

$$V = V_0 (e^{-at} - e^{-bt}) \quad (1)$$

where the fast rise is governed by the "b" coefficient, and the slow fall is governed by the "a" coefficient.

The fast transient pulse itself can be compared to a hammer blow. The response of the aircraft structure, cabling, and systems can be compared to the ringing of a bell when struck by the hammer.

2.1. A Distinction between Lightning, NEMP, and ESD

The following analysis applies principally to the NEMP and to the indirect effects of lightning. Because the source impedance of an electrostatic discharge is a function of frequency and is also much higher than that of NEMP and lightning, the system response to ESD is somewhat different, and its analysis will be addressed separately.

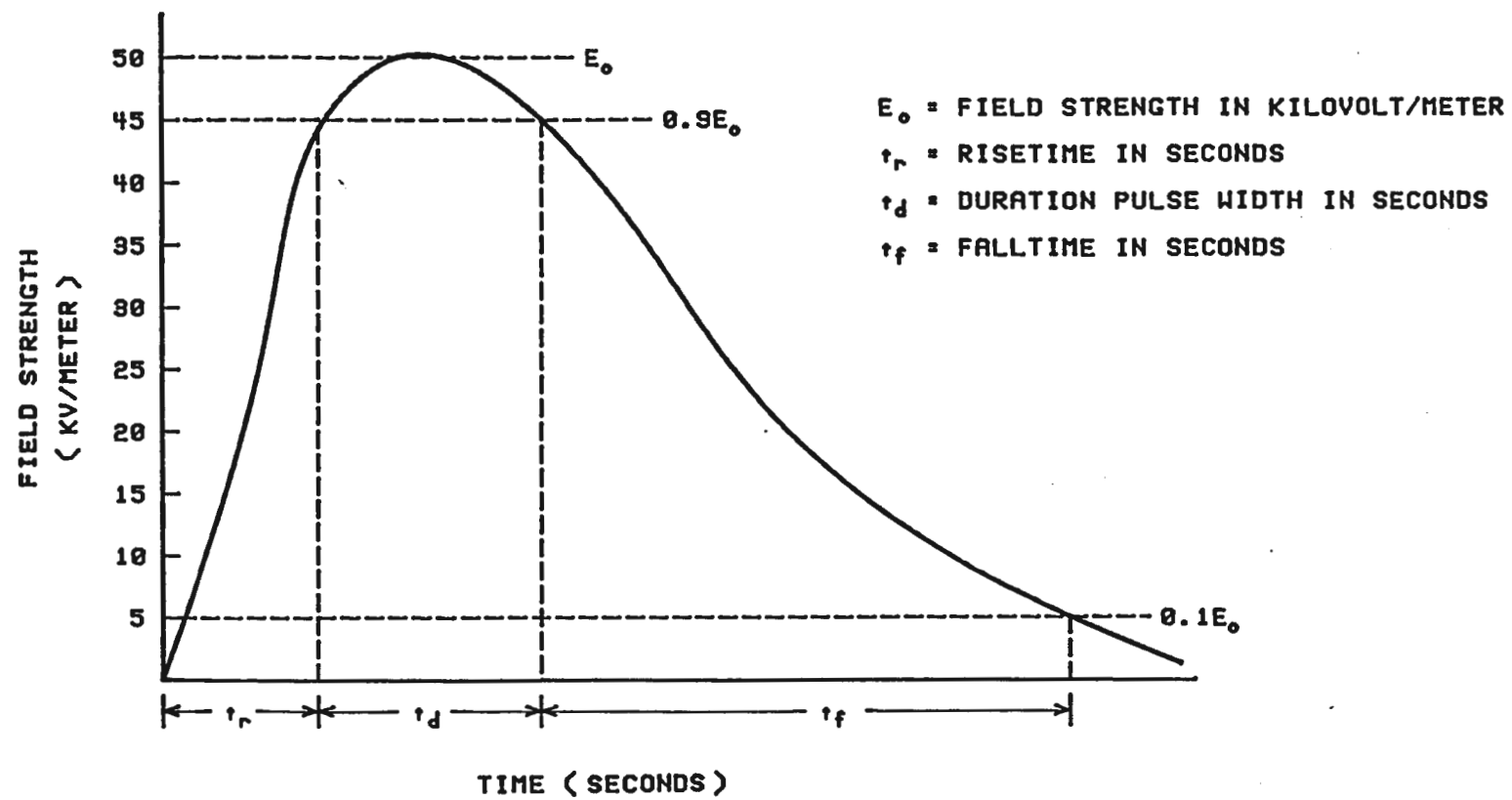


FIGURE 2-1. DOUBLE EXPONENTIAL PULSE

2.2. Response of the Aircraft to Lightning and NEMP

The size of the aircraft, the length of the cables within it, and the exact method of electrical termination of those cables determine the response of the aircraft to the double exponential pulse.

For example, in a large commercial transport, if a long cable in the wings or fuselage is terminated at one end with a low impedance, and at the other end in a high impedance, the cable can resonate in a quarter wavelength mode. The resulting resonant frequency might be on the order of 1 Megahertz (MHz).

When the double exponential pulse excites a resonant system in the aircraft, the response is an exponentially damped sinusoidal wave form. The frequency of this sinusoidal response is determined by the dimensions of the structure or cabling. The damping is determined by the energy storage and losses of the aircraft system, or quality factor, Q .

The resulting waveform can be represented mathematically as:

$$V = V_0 (e^{-at} - e^{-bt}) \cos(2\pi f_0 t) \quad (2)$$

where f_0 is the resonant frequency assigned to the aircraft structure or cabling. The rise- and fall-times are not those of the excitation pulse, but are determined by the physical properties of the resonant system. "Physical properties of the resonant system" refer to the energy storage capabilities of the aircraft wiring (determined by the inductance and capacitance of the wiring) and to the resistive losses (determined by the loads connected to the cables).

The voltage V_0 in Equation 2 is a function of the exciting pulse amplitude, the aircraft fuselage shielding, and the cable shielding. V_0 is usually much smaller than the voltage amplitude of the double exponential pulse itself.

A simpler version of Equation 2 is often used to represent the aircraft system response:

$$V = 1.05 V_0 (e^{-at}) \sin(2\pi f_0 t) \quad (3)$$

In this representation the rise-time is governed by the choice of the system resonant frequency. The rise-time is on the order of the time required for the sine wave to rise from zero to its first peak, at one-quarter of one period of the resonant frequency. The "a" coefficient, which determines the rate of decay, is defined as $\pi f_0/Q$. The value of Q is usually between 15 and 25 for aircraft cabling systems.

This form of the equation is somewhat more easily analyzed than that of Equation 2 and will be used throughout this chapter. Although Equation 2 is a more exact

representation than is Equation 3, the differences between them in terms of maximum voltage, maximum current, frequency spectrum, and energy transfer are negligible.

It should be noted that the V_0 and "a" terms in Equation 1 and Equation 3 are not the same. The only connection is that the peak amplitude of the system response is proportional to the peak amplitude of the NEMP or lightning pulse.

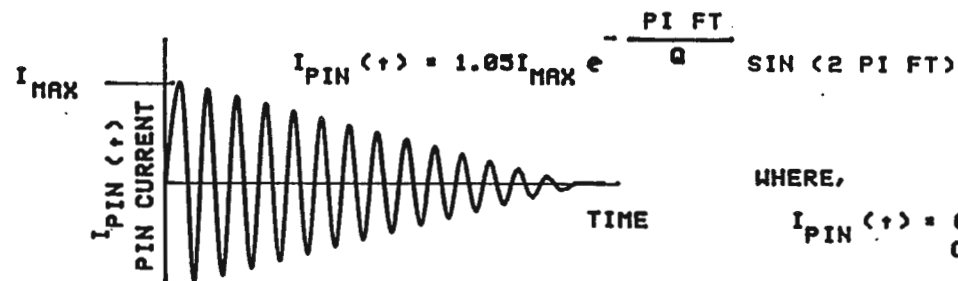
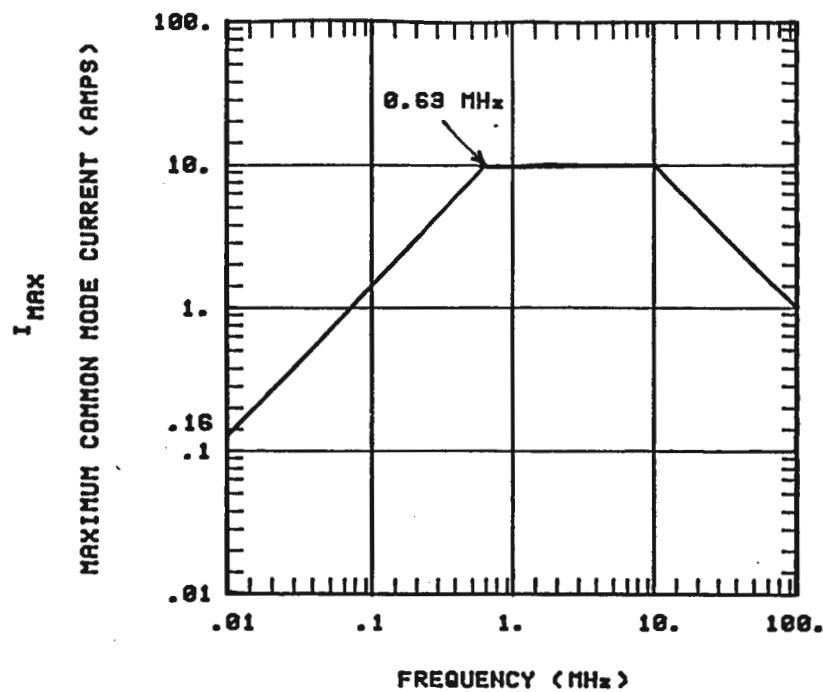
In Equation 3 the coefficient 1.05 is a function of the Q of the system, and can vary from about 1.03 to 1.05 as the Q varies from 15 to 25 in typical resonant structures. The exact value of this coefficient is:

$$k = e^{\frac{\pi}{4Q}} \quad (4)$$

This coefficient, multiplied by the maximum voltage induced on the cabling (V_0), yields an initial over-voltage such that when the falling exponential curve meets the rising sinusoidal curve, the resulting voltage is the maximum voltage of the transient response. The exact value of k is not particularly critical to the energy analysis, but the value assigned to Q is, since the energy of the system response is closely proportional to Q.

For NEMP analysis, the voltage V_0 is replaced with a current I_0 . MIL-STD-461 CS10 and CS11 define I_0 as in figure 2.2-1. The current shown in figure 2.2-1 represents the current that could be induced on aircraft wiring when the aircraft is subjected to a NEMP or a NEMP simulator during testing. Figure 2.2-1 also shows the exponentially damped sinusoidal waveform.

Lightning and NEMP test requirements may specify either the voltage or the current of the test waveform. Through a knowledge of system impedances and with the use of Ohm's Law, voltages and currents can be rendered equivalent to one another.



WHERE,

$I_{PIN}(t)$ = COMMON MODE PIN
CURRENT IN AMPS

f = FREQUENCY, HERTZ

t = TIME, SECONDS

Q = DECAY FACTOR

FIGURE 2.2-1. NEMP CURRENT



3. ENERGY TRANSFER FROM THE PULSE TO THE SYSTEM

When the double exponential pulse is applied to the system, and the system responds in its resonant fashion, energy is transferred from the pulse into the system. It is this energy which can cause upset or damage.

In a pin-level test the voltage of Equation 3 is applied directly to one of the conductors in a multiconductor cable. If the source resistance of the testing device and the load resistances connected to each end of the cable are known, the power developed in the system is then found by squaring Equation 3 and dividing the result by the sum of the source and load resistances.

The total power is:

$$P = \frac{V^2}{(R_S + R_L)} \quad (5)$$

where V is obtained from Equation 3. R_S is the source resistance of the testing device and R_L is the load resistance of the cable under test.

To find the energy in the system, it is necessary to integrate Equation 5 with respect to time. The total energy delivered to both source and load is:

$$E_t = \frac{k^2 V_o^2}{(R_S + R_L)} \int_0^{\infty} e^{-2\alpha t} \sin^2(\omega_o t) dt \quad (6)$$

The total energy is divided between the source and load resistances as follows:

$$E_S = \frac{E_t R_S}{(R_S + R_L)} \quad (7)$$

$$E_L = \frac{E_t R_L}{(R_S + R_L)} \quad (8)$$

where E_S and E_L are the energy into the source and load resistances, respectively.

Integrating Equation 6 and allocating the energy to source and load as in Equations 7 and 8 produces the following equations:

$$E_S = \frac{k^2 V_o^2 R_S Q}{(R_S + R_L)^2 \left(1 + \frac{1}{4Q^2} \right) (4\pi f_o)} \quad (9)$$

$$E_L = \frac{k^2 V_o^2 R_L Q}{(R_S + R_L)^2 \left(1 + \frac{1}{4Q^2} \right) (4\pi f_o)} \quad (10)$$

3.1. An Energy Transfer Example

As an example of the energy that can be transferred from a NEMP or from the indirect effects of lightning, Equations 9 and 10 can be solved by assigning typical values to the variables:

$V_o = 600$ volts
 $Q = 20$
 $k = 1.04$
 $R_S = 20$ ohms
 $R_L = 20$ ohms
 $f_o = 1$ MHz

The voltage (600 volts) is typical of the voltage induced on aircraft wiring by fast transients.

The Q value of 20 is representative of aircraft systems. The source and load impedances can have many values. The values chosen are low and will result in high energy transfer.

The resonant frequency of 1 MHz is suggestive of a very large aircraft. It also implies a large energy transfer.

When Equations 9 and 10 are solved using the assigned values, 7.74 millijoules of energy are delivered to the source and to the load. The burnout level for solid state devices is often taken to be 100 microjoules, where 1 joule is equal to 1 watt-second. The calculated value of energy is 77.4 times the burnout level, so obviously there is a problem. The system affected by the transient pulse must be protected in some way.

The energy distribution of the system response in the frequency domain will now be examined. This approach will lead immediately to a low-pass filter specification which will limit the amount of transferred energy to below the burnout level.

4. FREQUENCY SPECTRUM OF THE SYSTEM RESPONSE

The transient response of the system as a function of time has been depicted, as shown in the lower part of figure 2.2-1. There is also a frequency spectrum representation from which the energy distribution as a function of frequency may be found.

Mathematically, the Fourier transform is used to derive the voltage as a function of frequency from its time domain representation. An explanation of the Fourier transform is beyond the scope of this chapter. However, the frequency spectrum derived through the Fourier transform is shown in figure 4-1. The spectrum is that of a system having a resonance at 1 MHz and a Q of 20. The amplitude scale at the left is in decibels (dB), with the zero dB level corresponding to the voltage in the system at zero frequency. The voltage at resonance is Q times the voltage at zero frequency. From this consideration it may be observed that the voltage peak at resonance is $20 \log Q$ dB above the zero frequency reference level. The bandwidth between the points 3 dB below the peak response also yields the value of Q through the relation:

$$Q = \frac{f_0}{\text{BandWidth}(3\text{dB})} \quad (11)$$

At about two times the resonant frequency the slope of the voltage characteristic begins to approach a 40 dB per decade slope. As a consequence, the energy contribution at frequencies more than about 4 times higher than the resonance is negligible.

If the voltage versus frequency characteristic of figure 4-1 is squared, divided by the sum of the source and load resistances, and integrated with respect to frequency, the energy as a function of frequency can be found. If the integration limits are zero and infinity, the total energy over the entire frequency spectrum is found and is the same as that previously determined in the time domain by Equations 9 and 10. If the integration limits are zero and 500 kilohertz (kHz), for example, then the energy between those limits can be determined.

A program in Electromagnetic Computer aided design (EMCadtm) usable on IBM compatible PCs has been written to perform this integration and has generated some useful guidelines for the energy distribution:

- One-half of the energy lies between the half-power (3 dB) points of the resonance.
- Approximately one-fourth of the energy lies below the lower half-power (3 dB) point of the resonance.

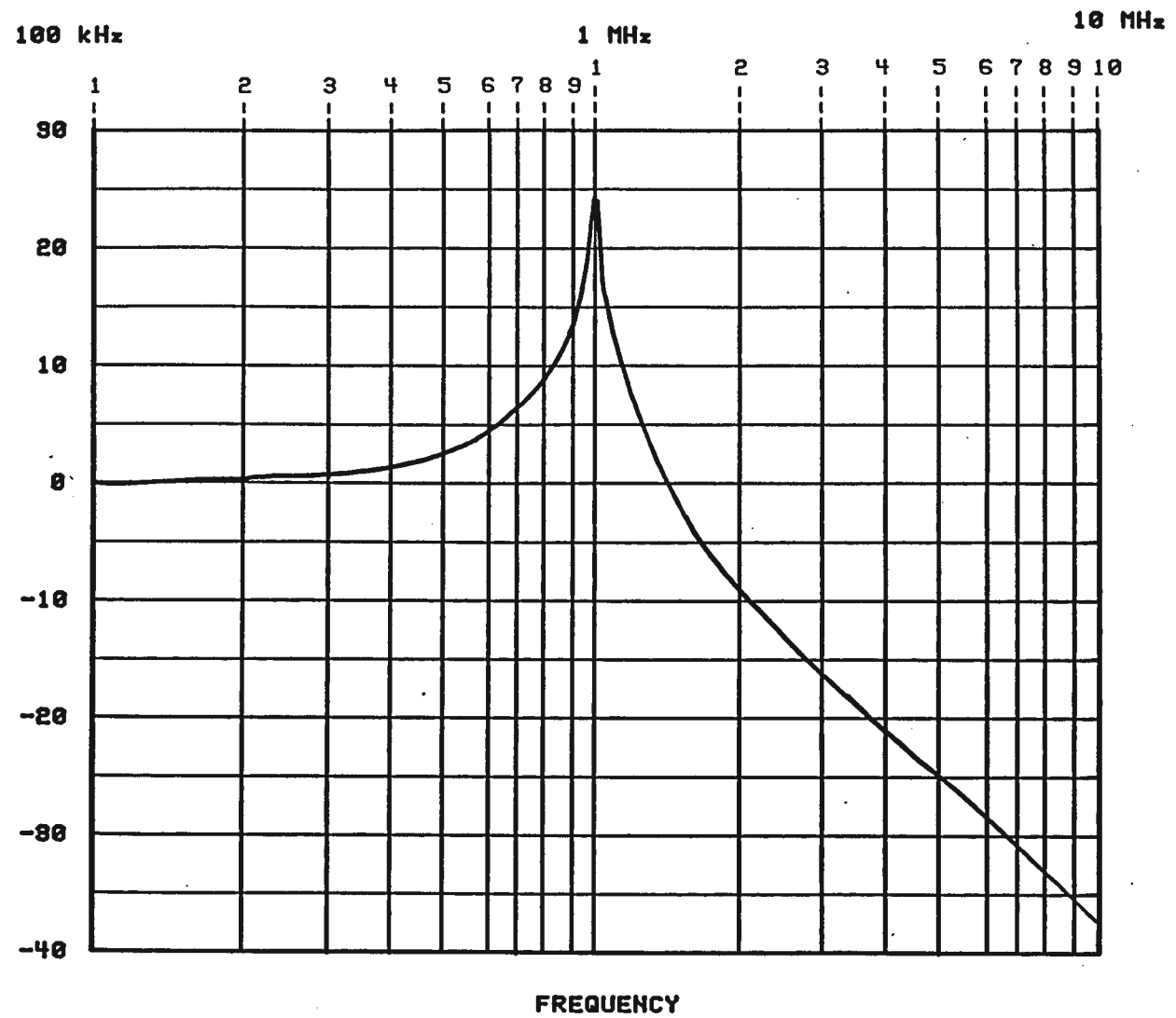


FIGURE 4-1. FREQUENCY SPECTRUM OF A 1 MHz DAMPED SINUSOID

- Approximately one-fourth of the energy lies above the upper half-power (3 dB) point of the resonance.
- Approximately 2 percent of the energy lies below 10 half-power bandwidths below the resonant frequency.

Using these guidelines, or (for more precision) the EMCadtm program, the frequency at which the burnout level is reached can be determined. From the examples in Equations 9 and 10, the energy delivered to the load is 7.74 millijoules. The half-power bandwidth of the 1 MHz resonance is 50 kHz (from Equation 11). The 10 half-power bandwidths below resonance then yield a frequency of 500 kHz, at which 2 percent of 7.74 millijoules will have accumulated. That is, over the frequency range from Direct Current (DC) to 500 kHz, the transient pulse will deliver 155 microjoules into the load impedance. For a more precise answer, the computer program shows that 100 microjoules will be delivered between DC and 368 kHz.

An approximate, though accurate, formula has been developed for the energy as a function of frequency. By setting the frequency equal to zero in the Fourier transform, the voltage at zero frequency can be found:

$$V(0) = \frac{V_0 k}{2\pi f_0 \left(1 + \frac{1}{4Q^2} \right)} \quad (12)$$

Assuming that the system losses (represented by the "a" in e^{-at} of Equation 6) are zero, then Q is infinite. For infinite Q, the voltage as a function of frequency is given by:

$$V = \frac{V(0) f_0^2}{(f_0^2 - f^2)} \quad (13)$$

This expression is extremely accurate for Q's as low as 15 up to frequencies within a few half-power bandwidths of the resonant frequency.

The total energy as a function of frequency is given by:

$$E_t = \frac{2 V(0)^2 f_0^4}{(R_S + R_L)} \int_0^{f_1} \frac{df}{(f_0^2 - f^2)^2} \quad (14)$$

The factor of 2 is required because the energy integral should be evaluated between limits of $-f_1$ and $+f_1$ rather than 0 and f_1 . The integral of the time

function, Equation 6, is still integrated starting from time $(t) = 0$ and is always zero for time before $t = 0$.

After integrating the expression above, the energy as a function of frequency is given by:

$$E(f) = \frac{V(0)^2 f_0^2}{(R_s + R_l)} \left[\frac{f_1}{(f_0^2 - f_1^2)} + \frac{1}{2f_0} \log_e \frac{f_0 + f_1}{f_0 - f_1} \right] \quad (15)$$

The energy calculated here divides resistively between source and load as shown in Equations 7 and 8.

If Equation 15 is evaluated at f_1 and f_2 instead of 0 and f_1 , the energy in any specified portion of the spectrum can be found.

Criteria for a filter design are now established. To protect the circuit with the values given earlier from energy levels above 100 microjoules, the circuit must have low-pass filtering which cuts off all energy above 368 kHz.

A NEMP or lightning specification sometimes identifies several frequencies of resonance in an aircraft, such as 1, 5, 10, and 45 MHz. Each frequency corresponds to some structural or cable dimension which resonates at a wavelength usually related to that dimension in a quarter or half-wave mode. An aircraft typically displays a multitude of resonances in the 1 to 50 MHz range.

Computer analysis has shown that if the aircraft resonances are separated in frequency by just a few half-power bandwidths, each resonance and its energy contribution may be treated as independent. A separate analysis may be made for each resonant frequency, and the energy levels from each may be added to find the total energy transfer. In a multiple-resonant system the energy contribution of each resonance scales inversely with frequency, so in many cases only the two lowest resonances make a significant energy contribution.

Multiple resonances can readily occur if a cable is resonant at 1 MHz in a quarter wave-length mode. The cable will then also be resonant at odd multiples of 1 MHz, such as 3, 5, and 7 MHz. In the numerical example worked above, if a 3 MHz resonance is assumed in addition to the 1 MHz resonance, the total energy will increase from 7.74 millijoules to 10.33 millijoules. Resonances at even higher frequencies will make only a small additional energy contribution. The voltage spectrum of a system with resonances at 1 and 3 MHz is shown in figure 4-2.

A low-pass filter, which takes care of limiting the energy transfer at the lowest frequency resonances, will also eliminate the energy transfer at the higher frequency resonances, if the filter retains its attenuation characteristics at the higher frequencies. For this reason the choice of filters or filter components, which have good high frequency characteristics, is important.

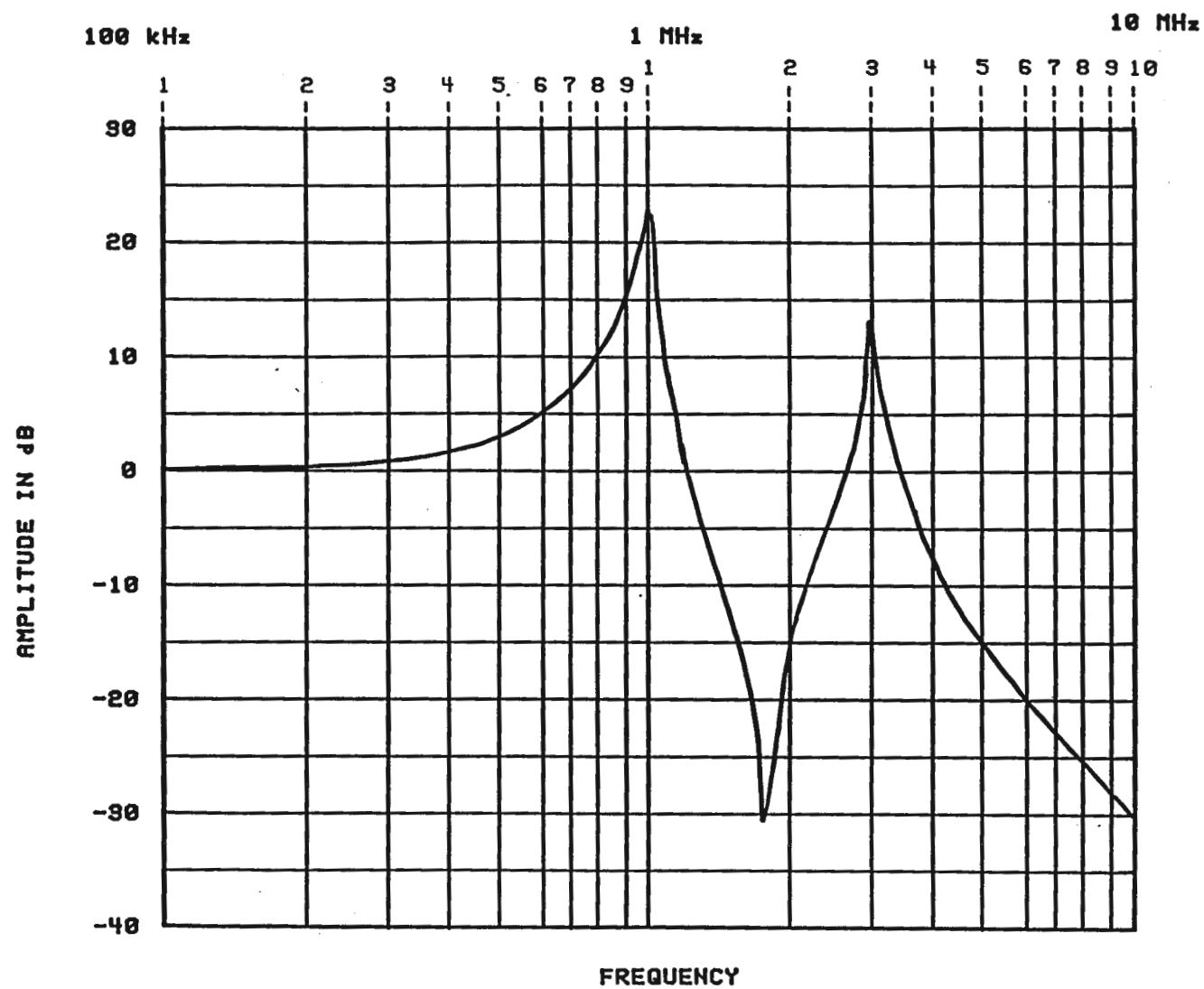


FIGURE 4-2. FREQUENCY SPECTRUM OF A DAMPED SINUSOID WITH TWO RESONANCES

If a lightning or NEMP requirement specifies only one resonant frequency, good practice dictates that a safety factor of 2 should be applied to the cut-off frequency determined by the energy analysis to eliminate the multiple resonance condition.

Since hundreds or thousands of individual circuits may have to be analyzed, a computerized approach to an evaluation of the system is essential. Spread sheets identifying conductors, load resistances, and energy levels are useful in this type of analysis. Computerized filter design programs are also helpful. The usual filter is a low-pass type, and usually only a series inductance and a shunt capacitance are required. Sometimes the necessary characteristics can be provided by a filter pin connector, in which the filter elements are incorporated into the body of the connector.

5. ELECTROSTATIC DISCHARGE

ESD is the familiar phenomenon experienced in stroking the back of a cat or touching a metal door knob after crossing a carpeted room. Some typical methods of generating electrostatic voltages are shown in table 5-1.

The accumulation of charge on the human body, or upon objects, can generate voltages up to 30 kilovolts. When the accumulated charge is eventually redistributed via a discharge arc, the fast rise-time pulse that results can have adverse effects on susceptible electronics equipment, ranging from upset to damage.

Aircraft are well suited for the generation of ESD. Relative humidity is often low, so that accumulated charge cannot slowly and harmlessly leak off without an arc. The decrease of atmospheric pressure with altitude somewhat decreases the insulating properties of air, and the motion of the aircraft through the air can cause an accumulation of charge. Finally, the movements of passengers and crew can cause charge accumulation in the same way as on the ground. Discharges from passengers or crew to metallic objects in the cabin can generate a field which may couple to cables routed behind decorative panels.

5.1. Energy Transfer in ESD

Although the waveform of the transient pulse for ESD is similar in shape to that of lightning and NEMP, the frequency spectrum of the energy transferred to the victim circuit from an ESD pulse is rather different. The source impedance of lightning and NEMP test devices, and the source impedance of the physical phenomena are usually quite low and mainly resistive. The ESD source, as shown in table 5-1, is characterized as a resistance in series with a capacitance. The impedance of this source is quite high at low frequencies and decreases with increasing frequency as the impedance of the capacitor decreases. As a result, the spectral content of an ESD has less energy at low frequencies and more at high frequencies than lightning and NEMP. Also, much higher voltages are required in ESD to generate the same total energy as for lightning and NEMP. However, the voltages available from ESD generating mechanisms are in fact very high, and the total transferred energy can be comparable to that of lightning and NEMP.

The energy of an ESD pulse may be delivered directly to a sensitive conductor or component, but more often the discharge is from the human body to a grounded object. Most of the energy is dissipated harmlessly to ground, but the flow of current to ground generates an electric field. In the analysis of ESD effects, it is useful to consider the effect of this electric field on surrounding objects in addition to the energy transfer considerations.

TABLE 5-1. TYPICAL ELECTROSTATIC VOLTAGES

Means of Static Generation	Electrostatic Voltages	
	10 to 20 Percent Relative Humidity	65 to 90 Percent Relative Humidity
Walking across carpet	35,000	1,500
Walking over vinyl floor	12,000	250
Worker at bench	6,000	100
Vinyl envelopes for work instructions	7,000	600
Common poly bag picked up from bench	20,000	1,200
Work chair padded with polyurethane foam	18,000	1,500

5.2. Description of the ESD Pulse

The typical ESD pulse is a double exponential and has been described earlier in this chapter. For easier analysis the pulse can be approximated by a triangular wave form as shown in figure 5.2-1. The frequency domain characterization of the pulse is also shown in figure 5.2-1. Significant amounts of energy appear well into the VHF region. The ESD pulse is inherently broadband, and susceptible equipment must be protected against it from very low frequencies to several hundred MHz.

This broadband pulse is capable of exciting resonances in nearby structures. The resulting response is a damped sinusoid, as shown earlier in figure 2.2-1.

5.3. Direct Discharge ESD

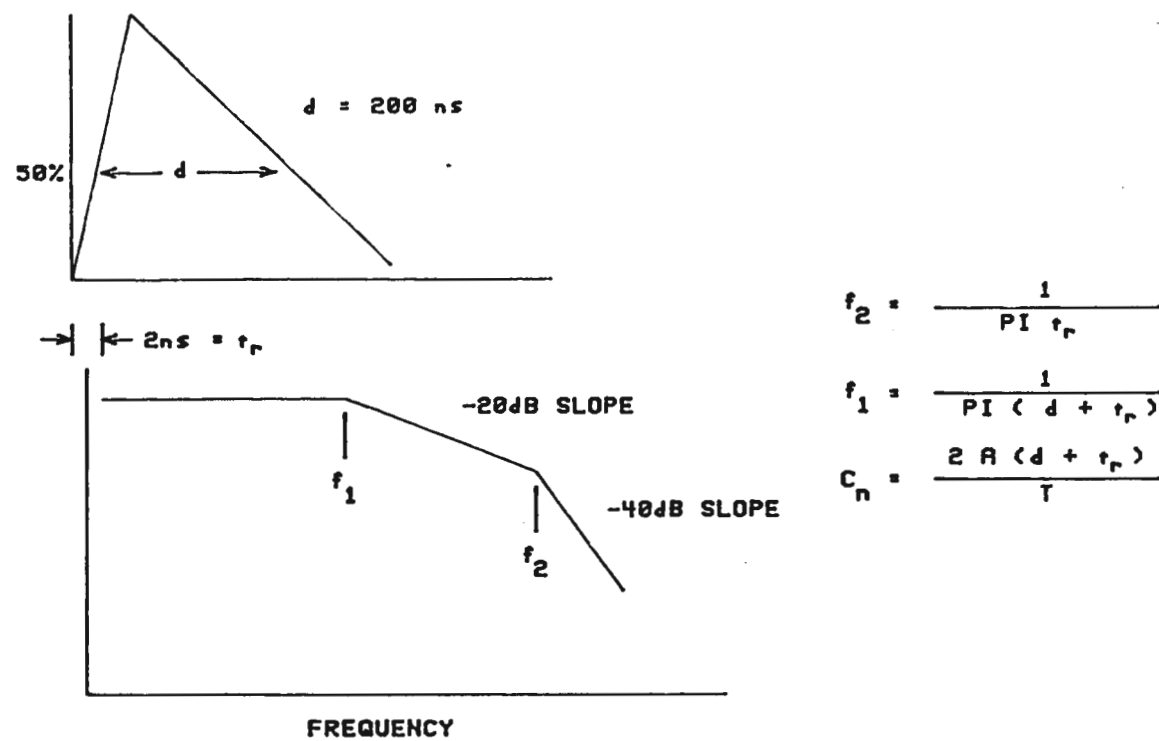
An arc from the hand to a susceptible device is called direct discharge. The human body, for ESD purposes, can be modeled as a simple series RC circuit (shown in figure 5.3-1).

For the simulation of an ESD, an ESD simulator is used (shown in figure 5.3-2). The energy stored in the human body or in the capacitor of the ESD simulator is given by $1/2 CV^2$, where C is the capacitance and V is the voltage. This energy is in joules or watt-seconds. For a 100 pf capacitor charged to 12 kV, the total energy delivered to the series resistor and the device under test is 7.2 millijoules. The total energy divides between the series resistance and the resistance of the test device. The energy delivered to the test device can be compared to its energy specification, if given, to see if there is danger of damage. The energy division occurs as shown in Equations 7 and 8.

The time required for the voltage to decay to $1/e$ (37 percent) of its initial value is given by $t = RC$. For a series resistance of 1500 ohms and a 100 pf capacitor, t is 0.15 microseconds.

In many cases upset rather than damage is the result of this kind of discharge. This condition is best checked during actual operation of the system under test. Repeated discharges will sometimes cause upset when a single discharge does not. A single discharge may occur in the time between data bit transmissions, whereas a repeated discharge has a good chance of occurring at the time of a data bit transmission and can cause a loss or alteration of data.

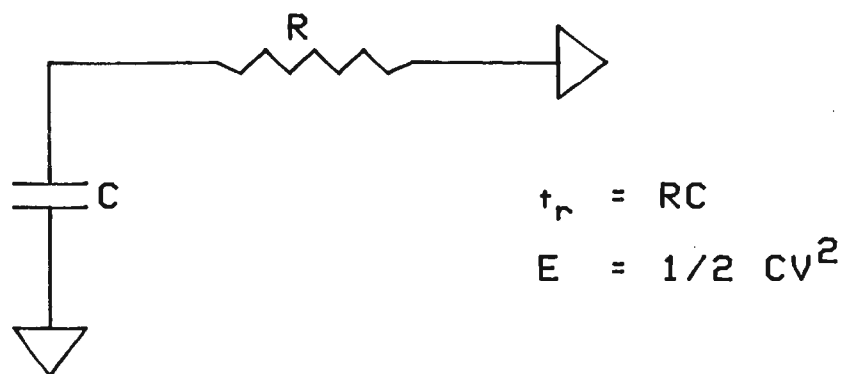
Many solid state devices are very susceptible to permanent damage from ESD incidental to handling. Careful training of installation and maintenance personnel, and adherence to handling procedures is required to avoid damage. Complimentary Metal-Oxide Semiconductor (CMOS) integrated circuits are known to be particularly subject to damage by mishandling, both as isolated components and when installed on circuit boards. For the proper handling procedures for ESD sensitive components, much useful information can be obtained from ESD Control in the Manufacturing Environment, published by the Department of Defense (DOD) Reliability Analysis Center. Military Handbook DOD-HDBK-263, Electrostatic Discharge Control Handbook for Protection of Electrical and Electronic Parts, Assemblies and Equipment is also of use.



LET $f_0 = 1 \text{ SEC. } T = 1$
 $R = 12 \text{ KV}$

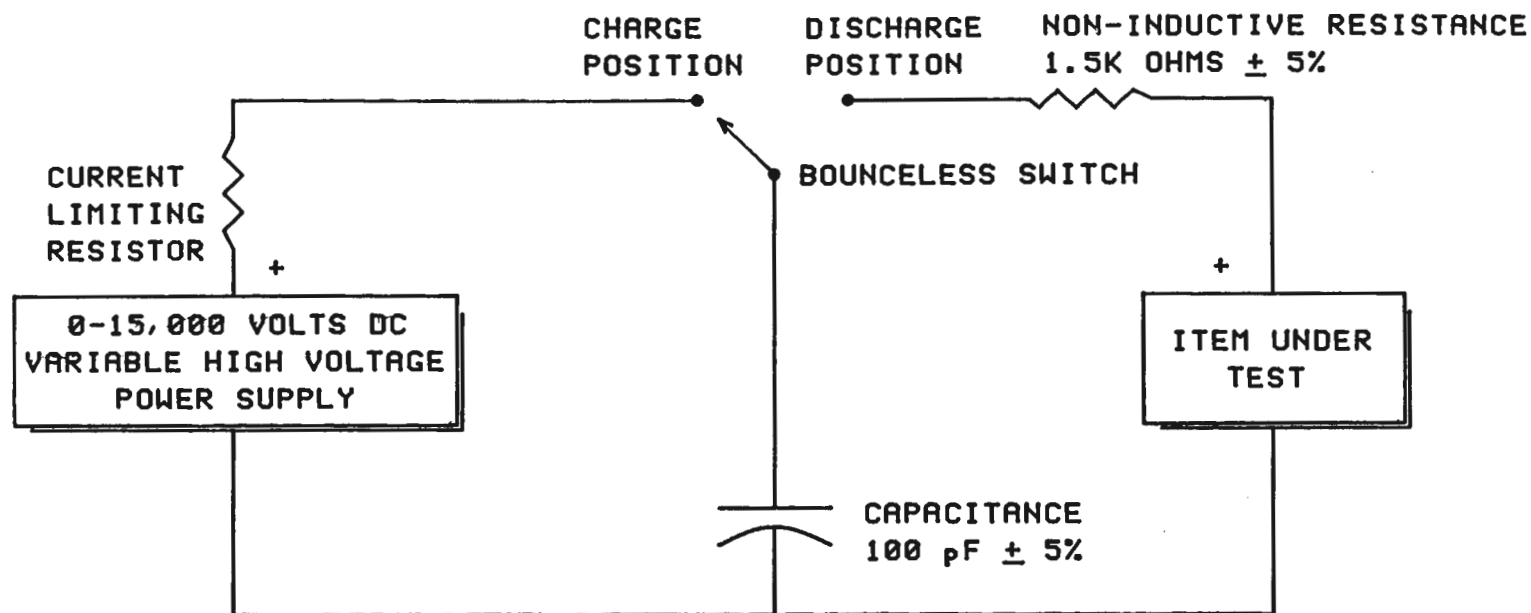
<u>FREQUENCY</u>	<u>Cn mV</u>	<u>Cn dBuV</u>	<u>Cn dBuV/MHz</u>
1 Hz	4.8	78.7	199.7
$f_1 = 1.58 \text{ MHz}$	4.8	78.7	199.7
$f_2 = 159 \text{ MHz}$	4.8	33.7	159.7

FIGURE 5.2-1. ESD VOLTAGE PREDICTION



<u>PARAMETER</u>	<u>LOW</u>	<u>HIGH</u>	<u>TYPICAL</u>
R	5	100 K OHMS	1-4 K OHMS
C	25 p f	.003 u f	100-150 p f
V	1 k v	30 k v	8-12 k v
t_r	125 P SEC	300 U SEC	1-6 N SEC
E	100 u J	1.3 J	3-10 m J

FIGURE 5.3-1. HUMAN ESD MODELING



NOTE: TEST VOLTAGES ARE MEASURED ACROSS THE CAPACITANCE. THE CAPACITOR SHALL BE DISCHARGED THROUGH THE SERIES RESISTOR INTO THE ITEM UNDER TEST BY MAINTAINING THE BOUNCELESS SWITCH TO THE DISCHARGE POSITION FOR A TIME NO SHORTER THAN REQUIRED TO DECAY THE CAPACITOR VOLTAGE TO LESS THAN 1 PERCENT OF THE TEST VOLTAGE OR 5 SECONDS, WHICHEVER IS LESS. POWER SUPPLY VOLTAGE SHALL BE WITHIN A TOLERANCE OF $\pm$ 5 PERCENT OF TEST VOLTAGE.

FIGURE 5.3-2. ESD TEST CIRCUIT

Table 5.3-1 lists a wide variety of electronic devices, grouped by their sensitivity to the voltage of an ESD. Table 5.3-2 shows the failure mechanisms that occur in various devices from ESD exposure.

Testing for direct discharge effects is performed as shown in figure 5.3-3. In the laboratory, the Alternating Current (AC) power would be provided as shown. On board an aircraft, it may be convenient to power the ESD simulator from batteries. Systems are best checked by operating them and watching for upset or damage when the ESD pulse is applied.

The effects of ESD on a digital system can range from missing data, to a condition requiring reset of the system, to damage requiring the replacement of parts.

5.4. Prediction of ESD Radiated Electric Fields

In the following example, a prediction is made of the effects of an ESD from the human hand to a metallic object in the cockpit of a commercial transport. A cabin attendant comes forward, touches a metallic object located one meter away from a susceptible item of equipment, and causes a discharge to occur. This analysis starts with the triangular pulse approximation of figure 5.2-1, and uses the chart in table 5.4-1 to guide the calculations.

In table 5.4-1, the amplitude of the pulse is 12 kV, which is quite typical of the ESD associated with the human body. The Fourier coefficient, which gives the amplitude of the pulse in the frequency domain, is C_n measured in dB microvolts per MHz. Note in the legend at the bottom of figure 5.2-1 that the amplitude as a function of frequency is constant from essentially DC to 1.58 MHz and then decreases at a rate of 20 dB per decade to 159 MHz. Above 159 MHz the amplitude decreases at a rate of 40 dB per decade. The electric field strength that is generated at a distance of one meter from this 12 kV discharge can now be calculated.

Referring to the frequency column of table 5.4-1, 1.58 MHz was entered as the frequency at which the radiated field of a 12 kV discharge was first calculated. Enter in column 1, 193.7 dB μ V/MHz, the Fourier coefficient obtained in figure 5.2-1. In column 2 enter -34 dB, which is a standard transfer function value to convert from a voltage on a conductor to the electric field generated one meter away. In column 3 enter a figure in dB (for the antenna factor) to determine how efficient the source is as a radiator of energy. This calculation compares the level of radiation from a short antenna to that of a quarter wave-length antenna at the same frequency. In this case, the entire human body is the major part of the radiating circuit and can be determined to be 2 meters. Using Note 3 in table 5.4-1, the antenna factor is calculated as 13.8 dB. Finally, in columns 4, 5, and 6 enter dB values for the number of conductors carrying the signal for the distance from the discharge to the susceptible nearby equipment and for the height above ground of the radiating object. Values of zero dB have been entered in each of these columns, indicating one conductor, a distance of 1 meter, and a height above ground of 0.5 meters.

TABLE 5.3-1. ESD PARTS BY PART TYPE

CLASS 1: SENSITIVITY RANGE 0 TO $\leq$ 1000VOLTS
• Metal Oxide Semiconductor (MOS) devices including C, D, N, P, V, and other MOS technology without protective circuitry, or protective circuitry having Class 1 sensitivity • Surface Acoustic Wave (SAW) devices • Operational Amplifiers (OP AMP) with unprotected MOS capacitors • Junction Field Effect Transistors (JFETs) (Ref.: Similarity to MIL-STD-701: Junction field effect transistors and junction field effect transistors, dual unitized) • Silicon Controlled Rectifiers (SCRs) with $I_o < 0.175$ amperes at 100° Celsius ($^{\circ}$ C) ambient temperature (Ref.: Similarity to MIL-STD-701: Thyristors (silicon controlled rectifiers)) • Precision Voltage Regulator Microcircuits: Line or Load Voltage Regulation < 0.5 percent • Microwave and Ultra-High Frequency Semiconductors and Microcircuits: Frequency > 1 gigahertz • Thin Film Resistors (Type RN) with tolerance of ≤ 0.1 percent; power > 0.05 watt • Thin Film Resistors (Type RN) with tolerance of > 0.1 percent; power ≤ 0.05 watt • Large Scale Integrated (LSI) Microcircuits including microprocessors and memories without protective circuitry, or protective circuitry having Class 1 sensitivity (Note: LSI devices usually have two or three layers of circuitry with metallization crossovers and small geometry active elements) • Hybrids Utilizing Class 1 parts

TABLE 5.3-1. ESD PARTS BY PART TYPE (Continued)

CLASS 2: SENSITIVITY RANGE > 1000 TO $\leq$ 4000 VOLTS
• MOS devices or devices containing MOS constituents including C, D, N, P, V, or other MOS technology with protective circuitry having Class 2 sensitivity • Schottky diodes (Ref.: Similarity to MIL-STD-701: Silicon switching diodes (listed in order of increasing tr)) • Precision Resistor Networks (Type RZ) • High Speed Emitter Coupled Logic (ECL) Microcircuits with propagation delay $\leq$ 1 nanosecond • Transistor-Transistor Logic (TTL) Microcircuits (Schottky, low power, high speed, and standard) • Operational Amplifiers (OP AMP) with MOS capacitors with protective circuitry having Class 2 sensitivity • LSI with input protection having Class 2 sensitivity • Hybrids utilizing Class 2 parts

TABLE 5.3-1. ESD PARTS BY PART TYPE (Continued)

CLASS 3: SENSITIVITY RANGE >4000 TO ≤15,000 VOLTS
● Lower Power Chopper Resistors (Ref.: Similarity to MIL-STD-701: Silicon Low Power Chopper Transistors) ● Resistor Chips ● Small Signal Diodes with power ≤1watt excluding Zeners (Ref.: Similarity to MIL-STD-701: Silicon Switching Diodes (listed in order of increasing tr)) ● General Purpose Silicon Rectifier Diodes and Fast Recovery Diodes (Ref.: Similarity to MIL-STD-701: Silicon Axial Lead Power Rectifiers, Silicon Power Diodes (listed in order of maximum DC output current), Fast Recovery Diodes (listed in order of tr)) ● Low Power Silicon Transistors with power ≤5watts at 25° C (Ref.: Similarity to MIL-STD-701: Silicon Switching Diodes (listed in order of increasing tr), Thyristors (bi-directional triodes), Silicon PNP Low-Power Transistors ($P_c \leq 5$ watts $T_A = 25^\circ$), Silicon RF Transistors) ● All other Microcircuits not included in Class 1 or Class 2 ● Piezoelectric Crystals ● Hybrids utilizing Class 3 parts

TABLE 5.3-2. PART CONSTITUENTS SUSCEPTIBLE TO ESD

Part Constituent	Part Type	Failure Mechanism	Failure Indicator
Film Resistors	Hybrid ICs: Thick Film Resistors Thin Film Resistors Monolithic IC-Thin Film Resistors Encapsulated Film Resistors	Dielectric breakdown voltage dependent-creation of new current paths Joule heating-energy dependent-destruction of minute current paths	Resistance shift
Metalization Strips	Hybrid ICs Monolithic ICs Multiple Finger Overlay Transistors	Joule heating-energy dependent metalization burnout	Open
Field Effect Structures and Nonconductive Lids	LSI and Memory ICs employing nonconductive quartz or ceramic package lids especially ultraviolet EPROMS	Surface inversion or gate threshold voltage shifts from ions deposited on surface from ESD	Operational degradation
Piezoelectric Crystals	Crystal Oscillators	Crystal fracture from mechanical forces when excessive voltage is applied.	Operational degradation
Closely Spaced Electrodes	Surface Acoustic Wave Devices Thin metal unpassivated, unprotected semiconductors and microcircuits	Arc Discharge melting and fusing of electrode metal	Operational degradation

TABLE 5.3-2. PART CONSTITUENTS SUSCEPTIBLE TO ESD (Continued)

Part Constituent	Part Type	Failure Mechanism	Failure Indicator
MOS Structures	MOS FET (Discretes) MOS ICs Semiconductors with metalization crossovers Digital ICs (Bipolar and MOS) Linear ICs (Bipolar and MOS) MOS Capacitors Hybrids	Dielectric breakdown from excess voltage and subsequent high current	Short (high leakage)
Semiconductor Junctions	Diodes (PN, PIN, Schottky) Transistors, Bipolar Junction Field Effect Transistors Thyristors Bipolar ICs, Digital and Linear Input Protection Circuits on: Discrete MOS FETs MOS ICs	Microdiffusion from micro-plasma secondary breakdown from excess energy or heat Current filament growth by silicon and aluminum diffusion (electromigration)	Operational degradation

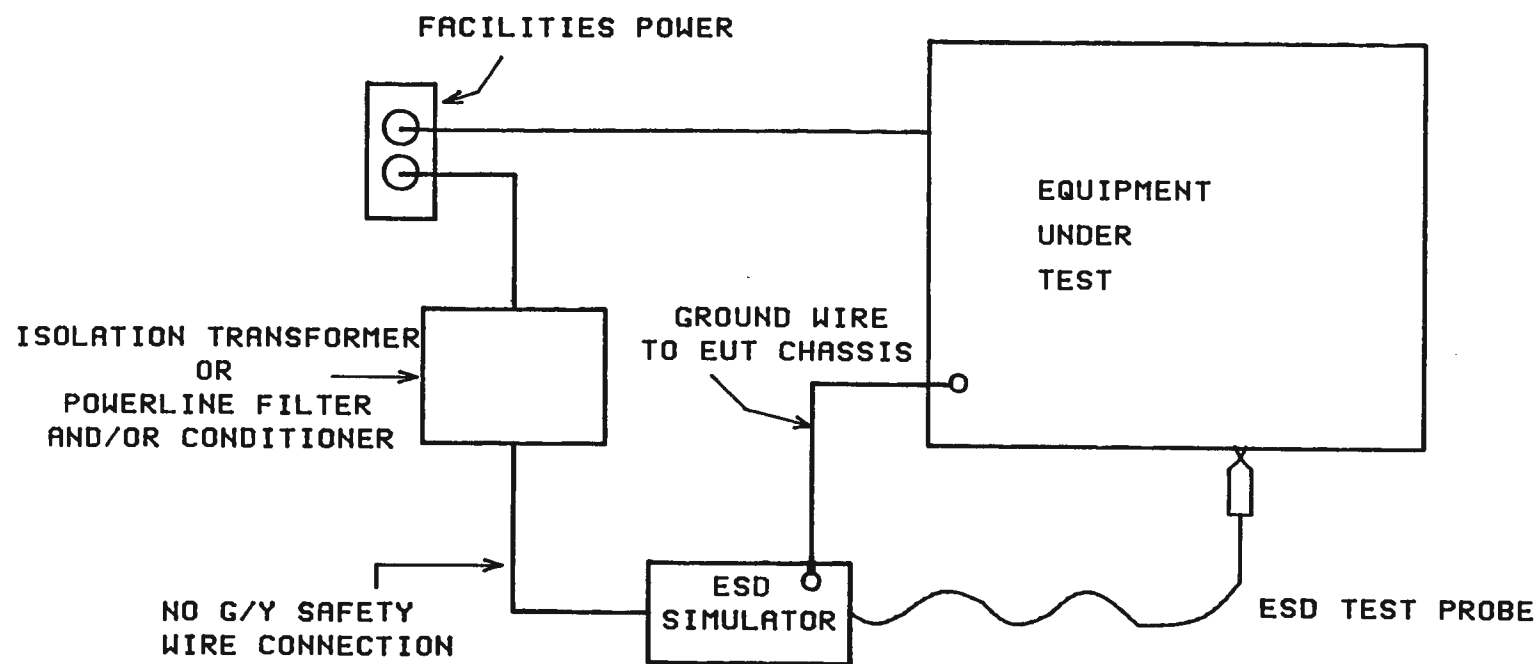


FIGURE 5.3-3. ESD DIRECT DISCHARGE EQUIPMENT CONFIGURATION

TABLE 5.4-1. PREDICTION OF RADIATED EMISSIONS

<u>FREQ</u>	1. <u>C_n dBuV/MHz</u>	2. <u>T_F</u>	3. <u>ANT. F.</u>	4. <u>N</u>	5. <u>D</u>	6. <u>HEIGHT</u>	7. <u>RESULT</u>
1.58 MHz	193.7	-34	-13.8	0	0	0	145.9

The dB values in columns 1 through 6 are now added to obtain the result in column 7, 145.9 dB μ V/meter/MHz. This value can be converted to a radiated field strength:

$$E = 10^{(145.9/20)} \mu\text{V/meter/MHz}$$

or

$$E = 19.7 \text{ V/meter/MHz}$$

To arrive at the magnitude of the field threatening a broadband device, the frequency domain plot of figure 5.2-1 can be integrated over the frequency range of interest. Piecewise integration from DC to 1.58 MHz yields a value of 31.2 volts per meter. This value is the field value that would be seen by a device which cuts off sharply at 1.58 MHz. Continuing the integration from 1.58 MHz to 159 MHz gives an additional field value of 1,550 volts per meter as the contribution from this portion of the spectrum. The total field is the sum of the two piecewise integrations and is then $31.2 + 1,550 = 1,581$ volts per meter per 159 MHz. This is the field which would be seen by a broadband device having response from very low frequencies up to 159 MHz. Above 159 MHz, the field strength decreases at a rate of 40 dB per decade, and the contribution from this part of the spectrum is negligible.

5.5. Prediction of Voltage Induced on a Susceptible Circuit

The next part of the problem is to calculate how much voltage might be induced on a susceptible circuit by the field calculated above. For the susceptible circuit a printed circuit board will be used into which a loop area has been inadvertently designed. The loop acts as an antenna which responds to the field calculated above. Assumed are loop dimensions of 4 inches by 6 inches, which could be created by a trace around the periphery of the board. The induced voltage as a function of field strength is given by:

$$V(\text{volts}) = \frac{2\pi}{\lambda} E N A \cos\theta \quad (16)$$

E is the field strength; N is the number of turns in the receiving loop; A is the area of the loop in square meters; λ is the wavelength; and θ is the angle of the loop with respect to the radiating field. (θ is usually taken to be zero for a worst case estimate.)

An effective way of combining the broadband field generated by the pulse with the response of the loop from DC to 159 MHz is to plot the field and the response in dB on semilog paper, as in figure 5.5-1. No values are shown above 159 MHz because the induced voltage above that point is negligible. The generated E field is shown in the uppermost curve and has frequency distribution as in figure 5.2-1. The loop response with an applied field of 1 volt per meter is shown in the lower curve and shows a linear response with frequency with a

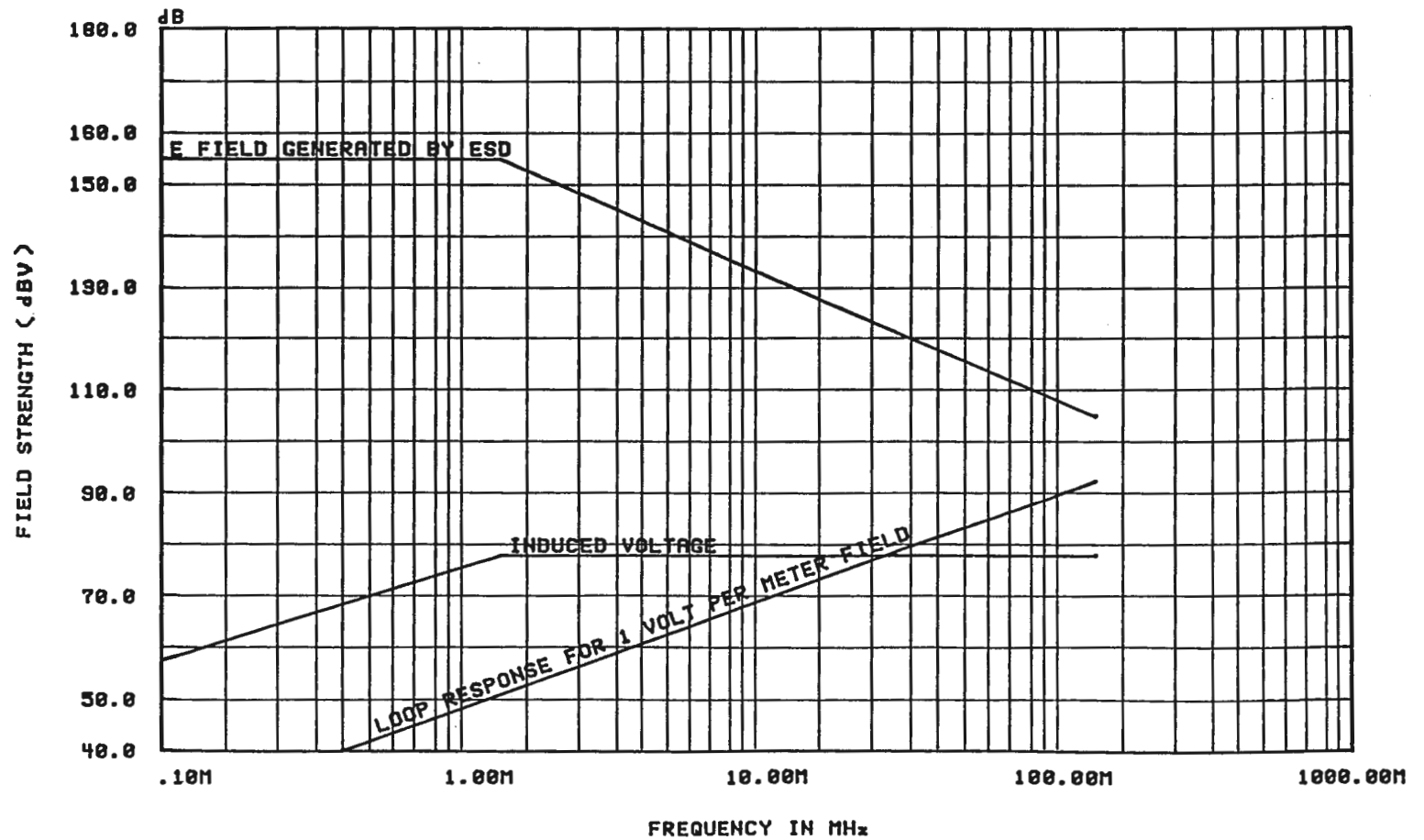


FIGURE 5.5-1. INDUCED VOLTAGE ON A LOOP FROM ESD

slope of 20 db per decade. The curve for the induced voltage in the loop is found by combining the applied field curve and the response curve. From 1.58 MHz up, the curve shows a steady value of 7.9 millivolts per MHz.

When the curve in figure 5.5-1 is integrated over the bandwidth to 159 MHz, the total voltage to which a broadband device would be exposed is about 1.25 volts, which is in the upset range for TTL devices.

In the analysis it was assumed that the receiving loop was unshielded. Shielding of the susceptible circuit will normally provide enough attenuation to reduce the induced voltage to a non-threatening level. However, an increase in the original 12 kV discharge voltage, a failure of shielding, a resonance condition in the susceptible circuit, or a decrease in the distance from the discharge to the susceptible equipment, can increase the induced voltage to the upset or damage levels.

Testing for radiated induced ESD is performed on installed systems using an ESD simulator while operating the systems under test and looking for upset conditions or damage conditions.

5.6. ESD Induced Voltages on Cables

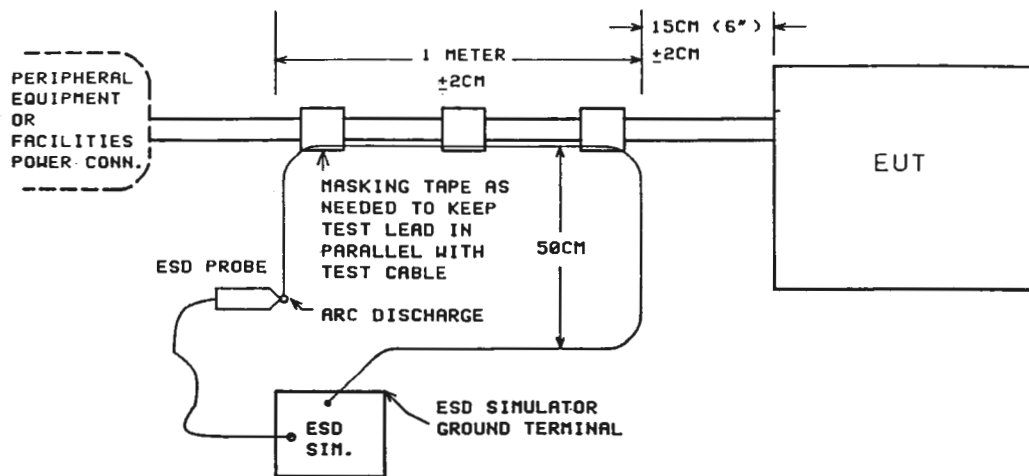
When a discharge occurs to a cabinet, which is connected to a shielded cable, some of the discharge current flows to ground via the cable shield. The noise of the discharge can be coupled to the cable center conductor through the transfer impedance of the cable, or by direct coupling to an exposed center conductor (in the case of improperly terminated shields), or directly through the shield itself. Figure 5.6-1 shows a test arrangement used for the ESD testing of cables. In an installed system, the ESD simulator voltage would be applied near the connectors at each end of a cable and to as much of the shielding between as is accessible.

In tests by the 3M Company, voltages larger than 500 volts could be induced on a center conductor by ESD if one end of the shield was unterminated. When the shield was terminated with a pigtail connection, the induced voltage was reduced to 16 volts. When the proper 360° backshell termination was made, a further reduction to just over a volt occurred. These results point out both the threatening nature of ESD and the necessity of proper cable shield termination.

5.7. Summary of ESD Prediction Technique

The procedure just given falls into four parts. First, the field at a distance of one meter from the discharge is calculated. Later, if desired, the field strength can be adjusted for distances other than one meter. A starting point for this field calculation is the information contained in figure 5.2-1.

Next, the susceptibility of the threatened circuit is calculated. Then the field and the susceptibility curves are combined to produce a resultant curve of induced voltage versus frequency. Graphical combining of the curves is most readily performed on a semilog scale as shown in figure 5.5-1.



- NOTES:
1. TEST LEAD PLUS LOOP SHOULD BE 1 METER AWAY FROM ANY CONDUCTING SURFACE INCLUDING CEMENT FLOOR. A WOOD TABLE IS AN IDEAL TEST PLATFORM.
 2. WHEN TEST CABLE IS LESS THAN 1.15 METERS IN LENGTH, THEN TEST PARALLEL SHALL BE SHORTER AND STARTED A MINIMUM OF 5CM (2") AWAY FROM EUT CONNECTOR.

FIGURE 5.6-1. ESD TEST CONFIGURATION (CABLE INDUCED)

The voltage induced in the susceptible circuit is found by converting from dB microvolts per meter per MHz to volts per meter per MHz.

Finally, the average value of the induced voltage over the frequency range is found by integrating over the entire spectrum of interest.

After appropriate adjustments are made for shielding or other attenuation factors, the induced voltage can be compared to the upset and damage levels of the threatened circuit or system.

5.8. ESD Testing Recommendations

It is recommended that ESD tests be performed because of the flight critical nature of new systems and the high susceptibility of broadband logic devices to ESD phenomena. The specific ESD test recommendations for flight critical systems follows.

Equipment, systems, and installed systems should be ESD tested according to table 5.8-1.

A soft failure is one which causes an alteration of data or missing data. A hard failure is one which requires a reset of equipment. Damage requires repair or replacement of a system, subsystem, or component.

All tests should start at the 2 kV level and should be advanced in 1 kV increments until a failure occurs (or the specified level is reached without failure).

TABLE 5.8-1. ESD TESTS

Failure Mode	ESD Test Level	Simulator	Circuit Values
Soft	8 kV	150 pf	1200 ohms
Hard	12 kV	150 pf	1200 ohms
Damage	25 kV	150 pf	1200 ohms



6. CONDUCTED FAST TRANSIENTS ON POWER LINES

When an aircraft is on the ground and being serviced, it is possible that fast transients will be introduced on the primary power lines. The connection and disconnection of Auxiliary Power Units (APU) and switch-over from the APU to aircraft power are likely sources of these transients. In flight the operation of other on-board equipment may cause similar disturbances.

Typical power line transients are described in two comprehensive military standards. MIL-STD-461C defines pulse shapes, amplitudes, and time characteristics. MIL-STD-462 gives detailed test procedures. The applicable test is Conducted Susceptibility, CS06. A review of this specification is recommended.

Two types of transients are described. One is a heavily damped spike which starts out with a positive polarity, goes negative, and then rapidly decays to zero. The second transient is a trapezoidal pulse. Tests are conducted with pulse times of both 0.15 microseconds and 10 microseconds. The pulse amplitude is specified as 200 volts. These transients are shown in figure 6-1.

Although the military specification states that these pulses are to be applied to equipment and subsystem AC and DC power lines, such testing may be inappropriate in some cases. For example, low voltage DC power supplies may be unable to tolerate 200 volt spikes applied to their input leads. In recognition of these difficulties, MIL-STD-462, Notice 2, states "A detailed test procedure shall be included in the test plan to provide a practical measurement on the item to be tested, as the specified procedure often cannot be accomplished."

The CS06 tests are most appropriate for 28 Volt, Direct Current and 115 Volt, Alternating Current, 400 Hz primary power lines.

RTCA/DO-160B describes rather similar power line transients and test procedures. A thorough review of Section 17 of that document is recommended.

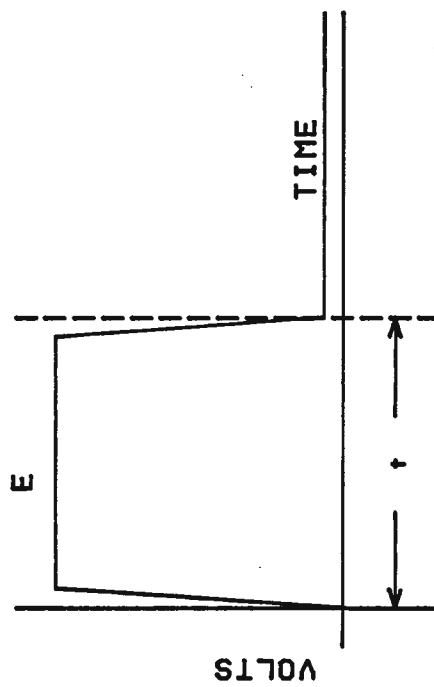
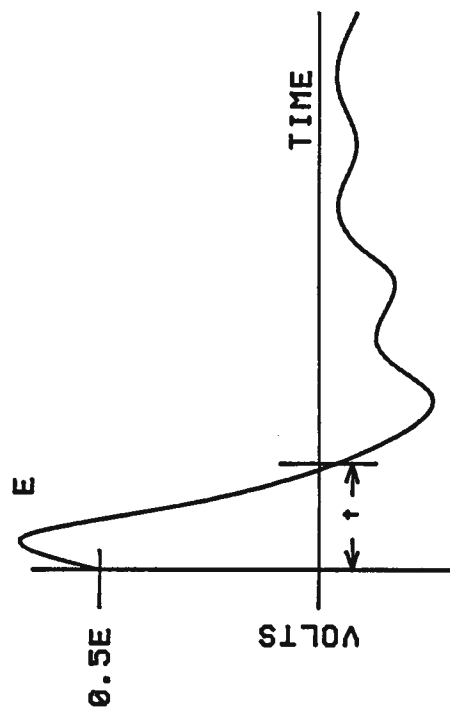
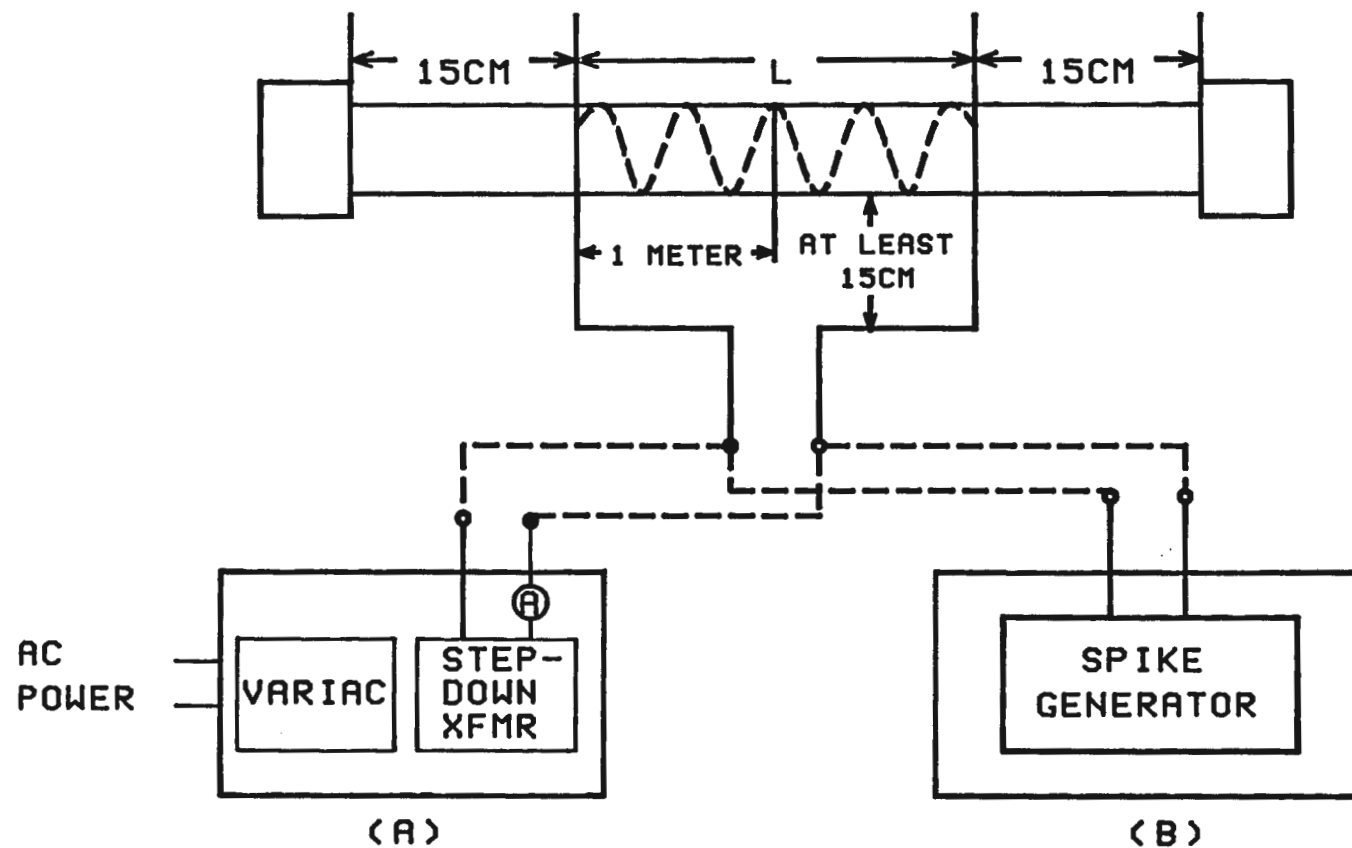


FIGURE 6-1. POWER LINE TRANSIENT TEST WAVESHAPES

7. RADIATED SUSCEPTIBILITY TO MAGNETIC INDUCTION FIELDS

The same transient spikes described in the previous section are also applied, under MIL-STD-461C, RS02, as magnetic induction fields to cables and equipment. The transient spikes must not cause "malfunction, degradation or deviation from specified indications..."

For these tests, a wire is wrapped around either cables or equipment, as shown in figures 7-1 and 7-2. The specified test spikes are applied from the spike generator. For complete details, see MIL-STD-461C and MIL-STD-462.



NOTE:

- (1) L SHALL BE THE LENGTH OF THE CABLE IN THE ACTUAL INSTALLATION OR 1.5 METERS, WHICHEVER IS LESS.

FIGURE 7-1. RADIATED SUSCEPTIBILITY, MAGNETIC INDUCTION FIELD, CABLE TEST

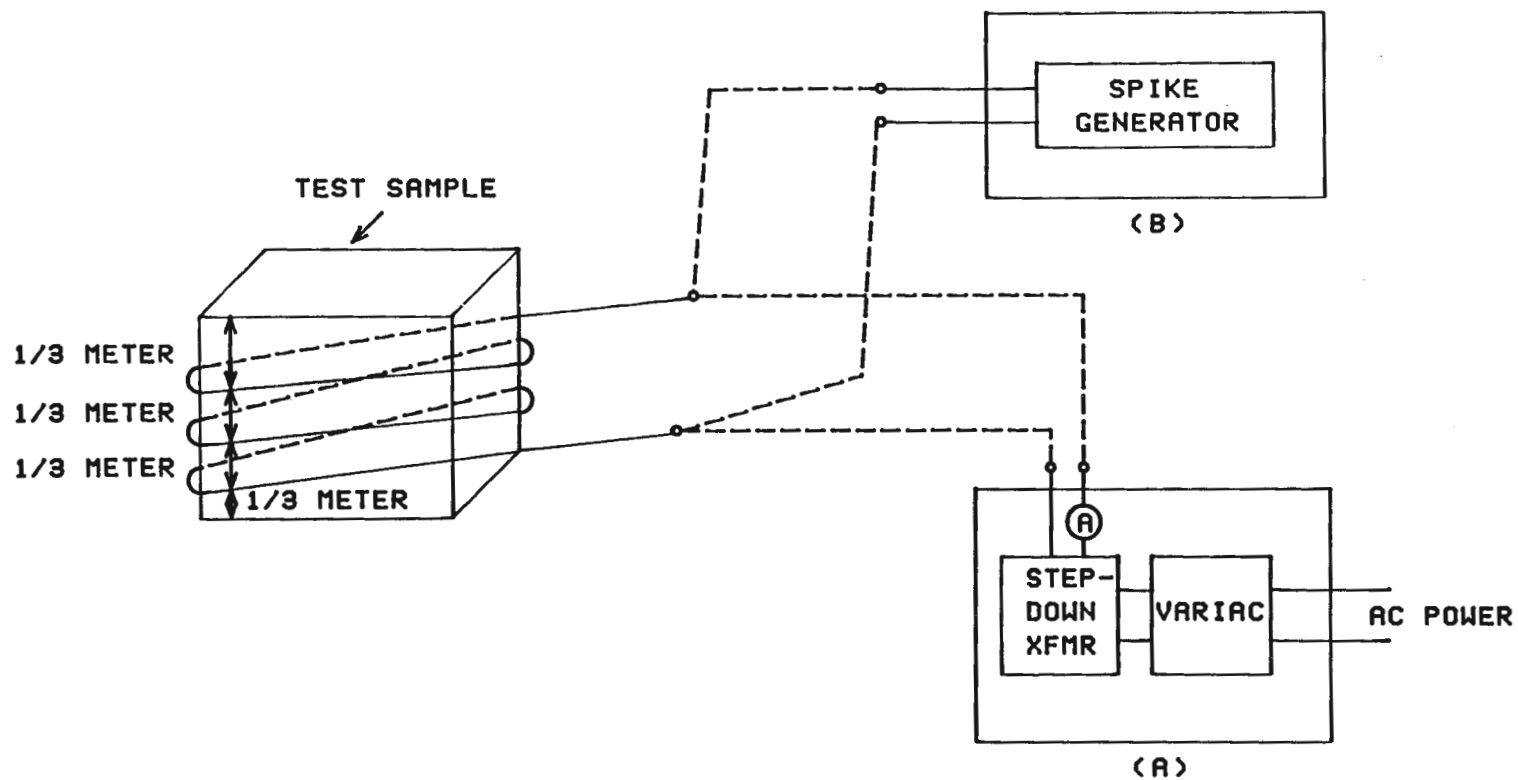


FIGURE 7-2. RADIATED SUSCEPTIBILITY, MAGNETIC INDUCTION FIELD, CASE TEST



8. MIL-STD-461C EMP TESTS

MIL-STD-461C defines Electromagnetic Pulse (EMP) pin level test limits under Conducted Susceptibility CS-10 and EMP cable test limits under Conducted Susceptibility CS-11. MIL-STD-462 gives detailed test procedures. A review of these references is recommended for a thorough understanding of these tests.

Briefly, a pin level test involves the direct connection of a test signal generator to a pin of a cable connector, followed by application of the test signal, and observation of the equipment under test for malfunction or degradation.

The cable test, Bulk Cable Injection (BCI), calls for inducing a specified current on a cable bundle or shield and again observing the equipment under test.



9. PROTECTIVE MEASURES

This section briefly describes various ways in which equipment and systems can be protected from the effects of fast transients.

Shielding is the most fundamental way of protecting both equipment and cables. All multiconductor cables, which carry signals that could be affected by transients, should be shielded. Other circuits, which can be regarded as generators of transient energy, should also be shielded to reduce their radiated emissions.

Particularly critical data can be transmitted on shielded twisted pairs. The twisting provides a measure of shielding effect by itself, and a single or even double shield provides further protection.

Coaxial lines with a single center conductor are frequently used. The braid-over-foil type is most effective. The foil provides nearly the effect of a solid outer shield, and the braid provides both physical protection and additional shielding. The type of fold is critical to the shielding effectiveness: a longitudinal "z-fold" or cigarette type fold is best. Spirally wrapped foil shields are less effective. The most effective coaxial cable shield is a solid metal outer conductor, but because of its lack of flexibility, it is ordinarily only used in short lengths within the avionic systems.

The method of shield termination is critical to maximizing the effectiveness of the shielding. All shields should be terminated with a 360° connection of the shield to the back shell of the cable connector. Experiments have shown a 30 dB or more degradation of shielding when a "pigtail" connection as little as 1 inch in length is used instead of the 360° connection.

Equipment case shielding is another line of defense against fast transients. Metal boxes provide effective shielding except against low frequency magnetic fields; however, holes, slots, seams, and apertures of any kind can allow entry of energy. In the case of ESD, some of the energy is in the 10 to 1000 MHz region and even fairly small apertures can degrade the shielding effectiveness.

A one-inch diameter hole has an attenuation of 40 dB at 60 MHz. This value scales downward (less attenuation) with increasing frequency at a rate of 20 dB per decade. Seams and slots can actually act as antennas. If the length of a slot or seam is one-half wavelength for an interfering signal, the slot or seam can resonate (like a magnetic dipole) and transfer signals with no attenuation at the frequency of resonance.

Filtering is another form of protection from fast transients. The energy in fast transients is primarily at higher frequencies. Low-pass filters which pass low frequencies and reject high frequencies can be effectively used on power lines and data lines to limit both radiation and reception of transient energy.

The cutoff frequency of the filter must be high enough to allow passage of the desired power or data frequency and low enough to prevent the flow of the transient energy. An example was given earlier in this chapter of the selection of a cutoff frequency in a low-pass filter. Filters can be designed in the form of discrete components. Filters may also be designed into the cable connectors. For many applications, a series inductance and a shunt capacitor provide adequate attenuation.

Ferrite beads placed over the end of a wire or cable can also provide a type of filtering. The common mode (line to ground) impedance is raised by the bead. For initially low impedance circuits, 6 to 10 dB of common mode attenuation results at frequencies from 10 to 100 MHz.

Various solid state protection devices exist which generally shunt transient energy to ground, while interrupting the normal flow of power or data for only brief periods. Variously known as transzorbs, surge suppressors, and varistors, these devices can all be considered as switches which are automatically operated by excessive voltage. They are connected in shunt-to-ground, and normally present a high impedance to the power or data line. When operated by the presence of the transient, they change state, go to a very low impedance, and short-circuit the line to ground. Designs are available in both unidirectional and bidirectional versions.

ESD protection requires attention to equipment case shielding. Control knobs and shafts can be grounded to the equipment case. This grounding will prevent entry of transient energy inside the case and will force the energy to flow on the outside to ground.

Attention should also be given to materials selection both in the aircraft structure and the equipment. Some materials, particularly plastics, are more apt to promote charge separation and accumulation, leading to ESD.

High voltage DC power supplies (found in video display units) are sources of charge and should be completely shielded to prevent charge transfer to nearby objects.

Keyboards on data entry units are often points of entry for ESD. Membrane-type keyboards are particularly vulnerable, but shielding can be incorporated between the keyboard and the sensitive components beneath.

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GLOSSARY

BROADBAND. A frequency spectrum which is wide compared to the bandwidth of the device used to detect it.

DECIBEL. A dimensionless unit for representing the ratio of values of power or voltage.

dB μ V. Decibels referred to one microvolt. Zero db represents one microvolt.

FALL-TIME. The time required for pulse amplitude to go from a predefined magnitude to a given level.

FOURIER TRANSFORM. A mathematical method for deriving the frequency spectrum from a time dependent function.

HARD FAILURE. A failure that requires a reset of the equipment.

JOULE. A unit of energy equal to one watt-second.

LOW-PASS FILTER. An electrical circuit which allows the passage of low frequencies and prevents the passage of high frequencies.

PIN LEVEL TEST. An EMC test in which voltage or current is applied directly to a conductor at a connector pin.

Q. The quality factor of a resonant circuit which is the ratio of the energy stored to the power dissipated per cycle.

RESONANCE. Resonance occurs in an electrical circuit when the energy stored in the inductance is equal to the energy stored in the capacitance.

RISE-TIME. The time required for a voltage pulse to reach a predefined percentage of the amplitude from a given level.

SHIELDING. Any metallic structure such as the aircraft fuselage or the woven braid on a cable that provides protection against electromagnetic fields.

SINUSOID. A wave form that follows the mathematical values of a sine function.

SOFT FAILURE. A failure which causes an alteration of data or missing data.

UPSET. A condition in which the state of a digital device is unintentionally altered, but may be restored by automatic means or by operator intervention.

ACRONYMS AND ABBREVIATIONS

AC	Alternating Current
APU	Auxiliary Power Units
BCI	Bulk Cable Injection
CMOS	Complimentary Metal-Oxide Semiconductor
dB	decibels
DC	Direct Current
DOD	Department of Defense
EMC	Electromagnetic Compatibility
EMCad tm	Electromagnetic Computer aided design
EMP	Electromagnetic Pulse
ESD	Electrostatic Discharge
kHz	Kilohertz
MHz	Megahertz
NEMP	Nuclear Electromagnetic Pulse
pf	picofarad
R-C	Resistor-Capacitor
RTCA	Radio Technical Commission for Aeronautics
TTL	Transistor-Transistor Logic





HANDBOOK-VOLUME II DIGITAL SYSTEMS VALIDATION

CHAPTER 13 LIGHTNING STUDIES



PREPARED BY:

COMPUTER RESOURCE MANAGEMENT, INC.
950 HERNDON PARKWAY, SUITE 360
HERNDON, VIRGINIA 22070

PREPARED FOR:

FEDERAL AVIATION ADMINISTRATION
TECHNICAL CENTER
ATLANTIC CITY INTERNATIONAL AIRPORT, NEW JERSEY 08405

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1. INTRODUCTION

While the effects of lightning strikes on metal aircraft are considered minimal, there is a greater threat to advanced technology aircraft. There are two major reasons for this increased threat. Digital electronic components are more susceptible to electrical transients than analog systems in older aircraft. Also, advanced composite structural materials offer less electrical shielding than metallic structures. Both of these factors reduce the inherent protection of electronic/electrical systems found on previous technology, all-metal aircraft.

Competition in the marketplace for aircraft sales and the increasing cost of fuel is developing pressure on manufacturers to employ advanced technology electronic equipment and materials, such as bonded honeycomb, kevlar, fiberglass, or Graphite Epoxy (G/E), in the next generation of aircraft. This is evident both in large transport and in general aviation aircraft currently under development. Several general aviation aircraft that employ advanced technology are nearing or are in the certification process. These include the Beech "Starship," Lear Fan, and AVTEK-400. In addition to the all electric engine control (Pratt & Whitney 2037) for the Boeing 757, transport aircraft manufacturers are researching the use of advanced composite structures, digital data busses (beyond ARINC 429), and all-electric systems.

Advanced technology materials and new structural fabrication methods, such as using adhesives in place of fasteners to minimize drag, reduce manufacturing costs as well as corrosion and fatigue. Several potential problems have prevented widespread use of the new structural technology. These include the variability of: lightning protection, static electrification, Electromagnetic Compatibility (EMC), mechanical and electrical parameters, impact resistance, effect of environmental factors, and production controls.

This chapter addresses the topic of lightning studies and the impact of lightning strikes to aircraft and digital systems.

Section 2 outlines some of the problems associated with the use of digital systems and composite materials in advanced technology aircraft. Currently accepted information about lightning current waveforms is reviewed. The lightning environment, which an aircraft might be expected to encounter, is discussed. A few examples of lightning effects on aircraft systems are given.

Section 3 covers the design approach to lightning indirect effects protection. A design margin must be established. Requirements for grounding and bonding must be taken into account. Preferred methods for equipment installation and cable routing to provide optimum protection from Electromagnetic (EM) hazards are reviewed.

Test methods for verification of protection design are reviewed in section 4. These test methods include a variety of lightning simulation techniques and equipment bench test techniques. Several simulation techniques are discussed and a comparison of test results for four methods is given. Verification of data quality is an important aspect of these test methods.

Analysis techniques are used to validate experimental results. Computer modeling methods are covered in section 5. Several techniques must be modified to be valid for composite structures where the dominant coupling mechanism is dependent on the resistance of the structure. The capabilities of several computer codes useful for predicting EM coupling due to lightning are summarized.

Section 6 contains worked examples to illustrate prediction of EM coupling to wiring. The examples cover very simple techniques as well as the procedure for a more detailed vehicle analysis.

2. BACKGROUND

2.1. History

Aircraft are exposed to a wide variety of EM environments from onboard and external sources including the following major items:

- The electrical power and electronics equipment often cause electric currents to flow in the structure, if not by design, perhaps through equipment faults or short circuits.
- Lightning strikes to an aircraft result in large currents of short duration. These currents may flow in the structure and in any of the metallic plumbing, control cables, or wiring for the electrical/electronic systems.
- Antennas on the aircraft and nearby high energy Radio Frequency (RF) sources may cause large RF currents in the structure.

Lightning is the most severe of these EM environments. Direct strike lightning currents may be as high as 200 kA and the rate of rise may exceed 100 kA per microsecond. The effects of such currents will be subsequently discussed. Power system and electronic fault currents may reach a few thousand amperes. RF current density may reach a few tens of amperes per meter. Lightning and electric circuit faults act upon a small point of contact; therefore, the structure could be damaged at entry points. The RF currents are not likely to result in structural damage, but could cause a functional disruption of digital electronic circuits and possibly of analog circuits as well.

Aircraft lightning strikes produce two types of effects.

- Direct effects: Physical damage from arcing and sparking.
- Indirect effects: Disruption of the electronic/electrical systems from electrical transients in the wiring and structural elements.

The trend in avionics/electrical equipment toward digital circuits having lower operating voltage and power levels adds to the concerns regarding protection against the indirect effects of lightning strikes and static electrification. The poor (lower) conductivity of composite materials and the bonds between structural members make it difficult to obtain the 2.5 milliohm bonding and grounding required by current military specifications (MIL-B-5087B(ASG), 1970). If the 2.5 milliohm specification were applied, the assessment indicates that all electrical systems would be safe.

Structures composed of many of the newer materials will be far more resistive than the present 2.5 milliohm requirement. Kevlar is an insulator and G/E

structures may have resistances of a few ohms. Designs using these materials should allow the addition of enough metal so that protection against lightning strikes may be achieved. These designs should also allow the addition of metallic pathways in the grounding and bonding systems so that onboard systems will function properly in the presence of currents anticipated from normal or fault conditions in onboard equipment.

There are also concerns about potential interference between different onboard digital systems and sources of Electromagnetic Interference (EMI). The EM shielding effectiveness of a composite fuselage without seams or joints is one to two orders of magnitude less than that of an aluminum fuselage. Seams and joints reduce the shielding to a practical upper limit of 25 to 40 dB depending on size and number of seams and joints. The EM Compatibility (EMC) between digital and RF circuits using the structure as a return path is more critical for new aircraft than for older aircraft.

Since advanced aircraft digital electronics operate at a few volts compared with a few tens of volts for analog systems, the design margin in future advanced aircraft electronic systems may be reduced by two to three orders of magnitude. (See section 3.1.) Considerable design efforts may be required to accomplish the protection for these advanced technology aircraft because of the reduced margins of safety between the EM induced transients (stress) and the ability of future technology equipment to withstand these transients (strain). Figure 2.1-1 shows the problems associated with EM protection of digital systems and changing electrical parameters when replacing metal structure with composite materials. Figure 2.1-2 indicates the variability in transfer impedance of various materials.

The characteristics of electrical transients induced in aircraft wiring and avionics systems are affected by the system response of the entire aircraft to the lightning stimulus. Induced coupling and susceptibility tests may be conducted on electrical hardware and associated wiring at the subsystem level. Determination of the voltage and current levels induced into the equipment by a lightning strike must take into consideration the structural interaction with the arc, the subsequent coupling of the magnetic and electric fields generated by lightning currents on the external surfaces, and coupling to internal wiring and equipment. The presence of an aircraft in the lightning channel may modify the natural lightning current waveform as a result of the resonant responses of the aircraft structure.

The demonstration and validation of the lightning/static electrification protection design is more critical for the new materials and electronics technology than for older aircraft technology. Extrapolation of prior protection designs on metal aircraft to those on advanced technology aircraft is not valid because the protection technology is completely different. Very few protection measures were required to protect older metal aircraft from indirect effects.

IMPACT ON ELECTRICAL PARAMETERS
FROM REPLACING METAL STRUCTURE
WITH COMPOSITES

IMPACT ON EM PROTECTION
FROM REPLACING METAL STRUCTURE
WITH COMPOSITES

MINOR IMPACT ON PROTECTION METHODS		SUBSTANTIAL IMPACT ON PROTECTION METHODS					
	LIGHTNING DIRECT EFFECTS	CONDUCTIVITY - STRUCTURAL END-TO-END RESISTANCE MUST BE SMALL TO PROVIDE A PATH FOR LIGHTNING CURRENTS.		TRANSFER IMPEDANCE - SHIELDING EFFECTIVENESS PROVIDED BY STRUCTURE WILL DEPEND ON MATERIAL TRANSFER IMPEDANCE.		JOINT IMPEDANCE - GOOD ELECTRICAL CONTACT AT JOINTS IS NECESSARY TO PREVENT ARCS AND SPARKS.	
						GROUNDING - GROUND CONNECTIONS MUST ACCOUNT FOR IR DROP IN STRUCTURAL MATERIALS AND BETWEEN EQUIPMENT.	
	STRUCTURAL EFFECTS						
	FUEL SYSTEMS						
	LIGHTNING INDIRECT EFFECTS						
	ANTENNA PERFORMANCE						
	HF AND LF						
	VHF - L BAND						
	ABOVE L BAND						
	STATIC ELECTRICITY						
	EMI/EMC/EMP						
	POWER SUBSYSTEM						

NOTE:  Shaded area indicates impact.

FIGURE 2.1-1. PROBLEMS ASSOCIATED WITH PROTECTION OF DIGITAL SYSTEMS IN ADVANCED TECHNOLOGY STRUCTURES

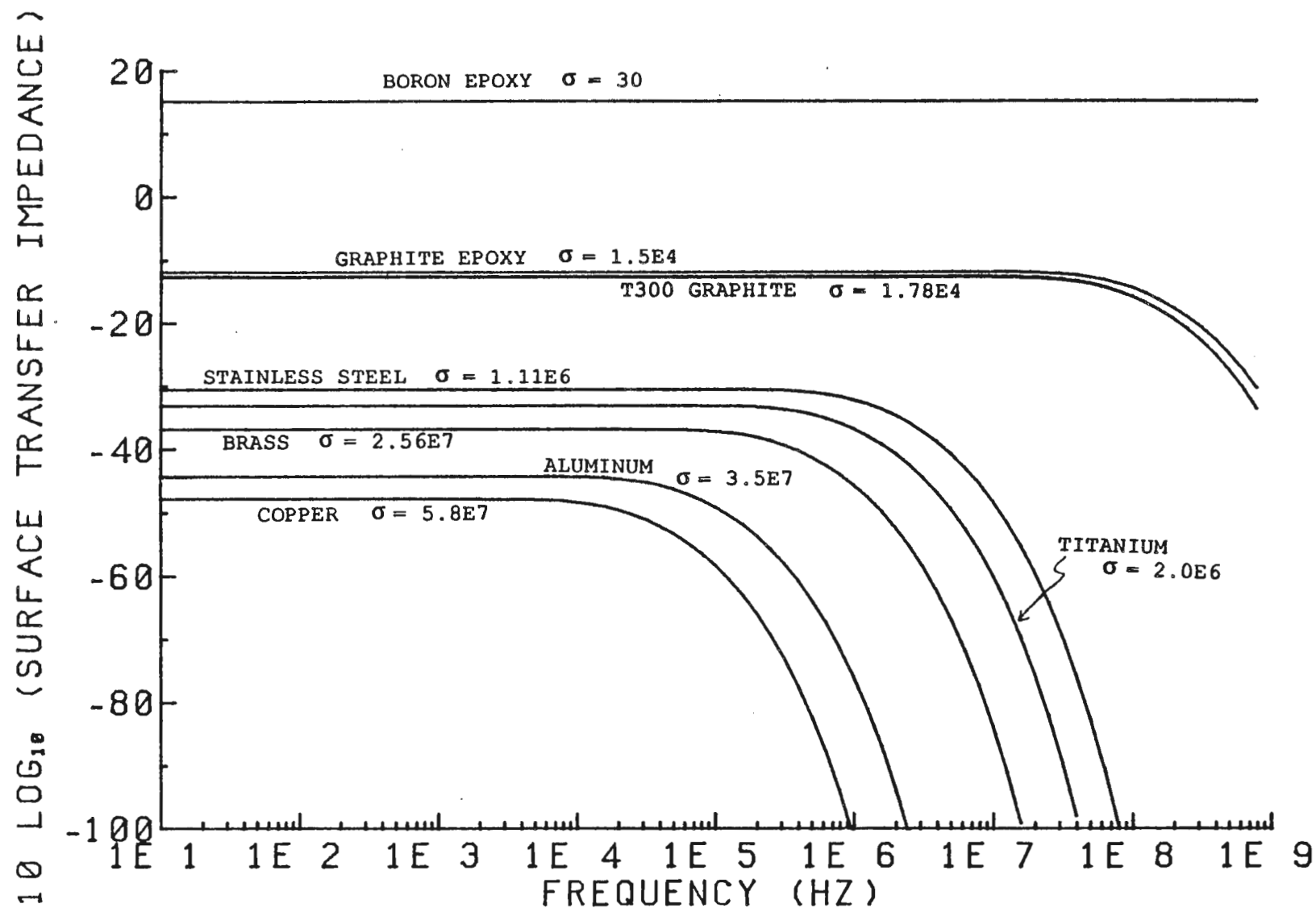


FIGURE 2.1-2. SURFACE TRANSFER IMPEDANCE FOR VARIOUS MATERIALS

The design margins did not exist for the older designs because of prior experience with the inherent insensitivity of metal aircraft to lightning strikes. Advanced technology aircraft are judged to have three to four orders of magnitude less design margin based upon the reduced shielding effectiveness of the fuselage and the sensitivity of the digital electronics (Sommer, 1981). It is natural to question the protection designs with such a great change in design margins.

Because of the increased potential for damage to electronic systems in advanced technology aircraft, simulated lightning tests must be performed to verify the adequacy of the design measures utilized to prevent serious loss or damage resulting from a lightning strike. It is recognized that the necessary protection verification test cannot be performed as a go/no-go test at full lightning threat levels, discussed in section 2.2, for several reasons as follows:

- The required electrical storage capacity in the lightning simulator would be excessive for a 200,000 amp discharge through an aircraft.
- The required fast rise-time (2.0×10^{11} amps/second) for a 200,000 amp discharge could not be attained.
- The discharge currents and charge densities around the aircraft would be different in flight.
- There is no predictable method to update test results for airplane modifications.
- The perceived lightning threat may be changed in the future as more data become available through the efforts of the National Aeronautics and Space Administration (NASA), Federal Aviation Administration (FAA), and others investigating the effects of lightning on aircraft in flight.

Lightning and static electrification protection design and demonstration of the design for aircraft and ground based systems are current topics for considerable technical research and development. The electrical impedance of a full-scale aircraft precludes the use of severe lightning current pulses such as those used for component indirect effects tests. Consequently, lower level current pulses have been used to evaluate the transients produced by a lightning strike. Lightning simulation testing is a very challenging technical area because it is very difficult, if not impractical, to conduct a full-scale simulated lightning test on large aircraft (MIL-STD-1757 and report of SAE Committee AE4L, 1978). The energy storage and electrical circuits capable of delivering a full-threat 200 kA peak current require simulator voltages greater than can be achieved in air (in excess of 2 million volts for a 25-foot long aircraft). Facilities to generate these conditions presently cost over a million dollars. Even with costly tests, there are no methods to update the results for airplane modifications, or methods to incorporate changes in the lightning threat. For the last several years there have been regular conferences dedicated to these topics. (See International Aerospace and Ground Conference on Lightning and Static Electricity in the Bibliography.)

Because of the practical considerations regarding simulator limitations, several different simulation test techniques are used for lightning and static electricity design demonstrations on full vehicles. The simulation and analysis methods are specialized to the unique character of lightning attachment, vehicle charging, high arc current flow, and vehicle discharging.

There are several variants on the high current pulse simulation techniques that are used for full vehicle lightning tests. The four principal lightning simulation test techniques are:

- Low-level swept Continuous Wave (CW).
- Low-level fast rise pulse.
- Full threat fast rise pulse.
- Shock-excitation.

Included in the accuracy of technique are the analytical models and computations used to verify and validate the test data. For some methods, analytical techniques provide extrapolation of the test data to other environments and guidance for modifications of the aircraft design.

2.2. Lightning Environment

Lightning environments have varying intensities and durations because of several different physical processes involved. The variation in parameters makes it impossible to specify the environment during any particular lightning flash. Lightning occurs most frequently due to charge separation in thunderstorm clouds. During a storm, air convection currents cause charge centers to build up within clouds. Because of mutual capacitance, the potential between these charge centers increases and will eventually break down a portion of the air path between them, redistributing the charges. The number of strokes within each flash and the peak currents depend on many factors including the storm intensity, the charge levels in the cloud before the leader has completed a path to the opposite charge, and the cloud base height (Melander and Cooley, 1984). These factors are variable, and more than an order of magnitude difference may occur between the currents of successive strikes.

Lightning flashes are of two fundamentally different forms, cloud-to-ground and inter/intracloud. Discharges may originate from either a positive or a negative charge center in a cloud and terminate at the ground or in an opposite charge center in the cloud. A negative discharge is characterized by several intermittent strokes and continuing currents. A positive discharge occurs only a small percentage of the time. It is characterized by both higher average current and longer duration within a single stroke. (See figure 2.2-1.)

Once preliminary breakdown within a cloud has begun, the breakdown process proceeds by a stepped leader which propagates from the cloud toward the ground or towards another charge center. As a stepped leader approaches the source of opposite charge, high electrical fields are produced. These fields give rise to streamers initiated at the ground or charge center until one of these

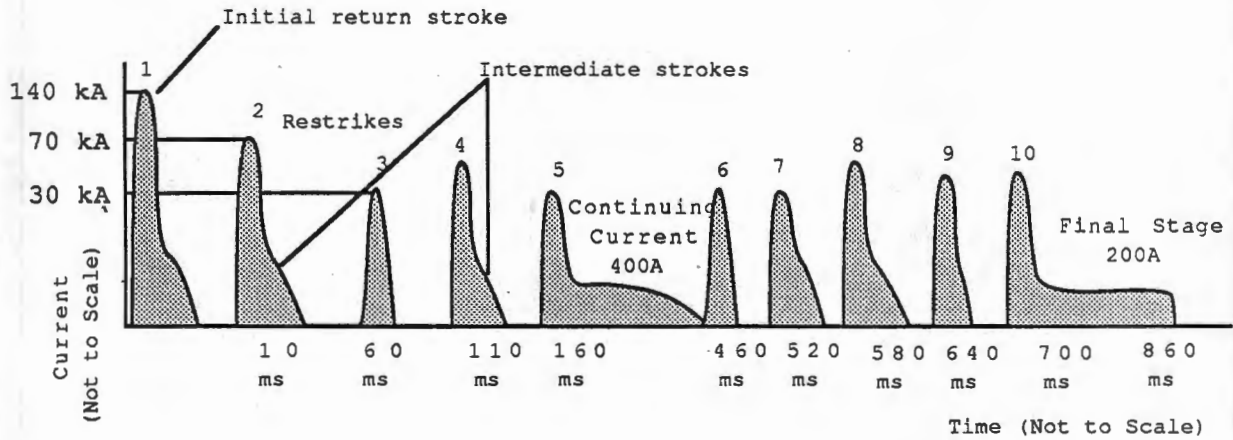
For Each Stroke:

Time to peak current = 1.5 μ s

Time to half value = 40 μ s

For the complete flash:

$$\int i^2 dt = 1.9E6 A^2 s$$

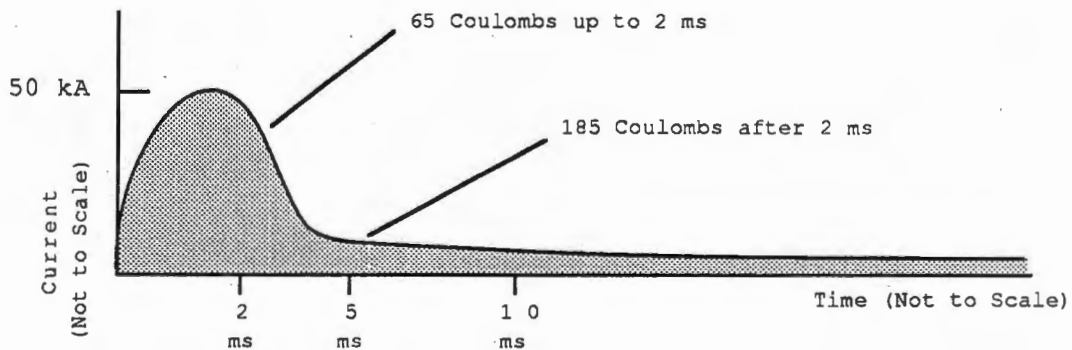


(A) Severe negative lightning flash current waveform.

(Courtesy of Cianos/Pierce)

For the complete flash:

$$\int i^2 dt = 2.5E6 A^2 s$$



(B) Moderate Positive Lightning Flash Current Waveform

FIGURE 2.2-1. LIGHTNING FLASH CURRENT WAVEFORM

streamers contacts the approaching stepped leader. The average velocity of propagation of the stepped leader is about one meter per microsecond for each step tens of meters in length and separated in time by approximately 50 μ s. The typical current is 1000 amperes and the average charge in the entire stepped leader channel is about 5 coulombs.

The high peak current associated with lightning occurs after the stepped leader reaches the ground and forms the return stroke of the lightning flash. This return stroke occurs when the charge in the leader channel is suddenly able to flow into the low impedance ground. The return stroke current expected value is much larger than the leader current and ranges between 10 and 30 kA. Although in rare cases a larger current can occur, a peak current of 200 kA represents a severe stroke and is considered a practical maximum value of lightning current. In rare cases, a larger current can occur. The current in the return stroke has a fast rate of change, ranging between 10 and 20 kA/ μ s. The maximum practical value is considered to be 200 kA/ μ s. No correlation has been shown to exist between peak current and peak rate of rise. The return stroke may be followed by a dart leader which initiates one or more subsequent return strokes or restrikes. Restrikes will continue until the charge is neutralized. Subsequent strokes typically have smaller magnitudes and faster rise times but comparable rates of change to the first return stroke. A lightning flash may contain as many as 24 restrikes following the first return stroke. Restrikes occur at intervals of several tens of milliseconds as different charge centers in the cloud are tapped.

The total charge transported by the lightning return stroke is relatively small, a few coulombs. Most of the charge is transported as a continuing current. Currents on the order of a few hundred amperes may flow in the ionized channel for periods up to one second. The maximum charge transferred is 200 coulombs. Continuing currents may link successive return strokes.

A discharge between clouds or charge centers within a cloud is characterized by a multitude of relatively low level intermittent strokes and continuing currents. The average velocity of propagation of the streamers is 10^5 meters per second. The peak current associated with an inter/intracloud discharge is in the range of 3 to 10 kA and occurs when opposite charge centers have been connected by a streamer or stepped leader. Each restrike within an inter/intracloud lightning flash may contain 20 separate pulses, known as a multiple burst. The number of restrikes is comparable to a cloud-to-ground flash.

Important parameters of a lightning flash are the peak current amplitude, maximum current rate-of-rise, energy, duration, rise-time, and the number of strokes within each flash. Statistical representations of these measured parameters are found in Melander and Cooley (1984). The great bulk of this statistical data is based on measurements of cloud-to-ground flashes. Several flight test programs within the past few years have gathered data on inter/intracloud lightning. These data are used to define parameters for lightning threat levels.

Representations of the various portions of lightning flashes have been developed by the Society of Automotive Engineers (SAE) committee AE4L as shown in figure 2.2-2. Multiple stroke and multiple burst waveforms are characterized

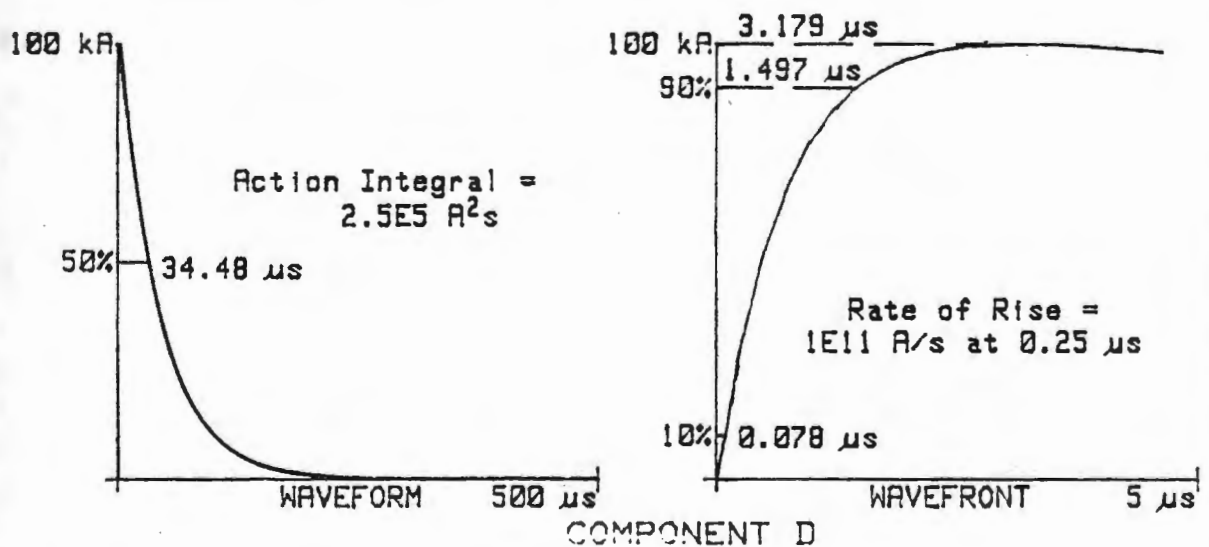
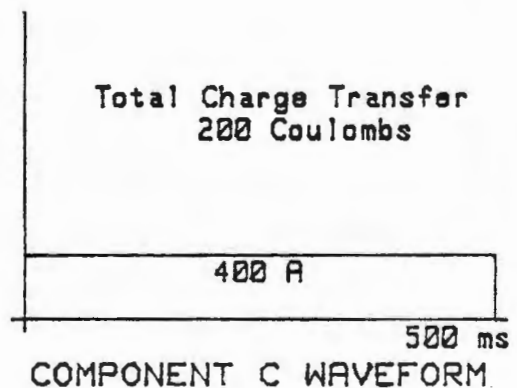
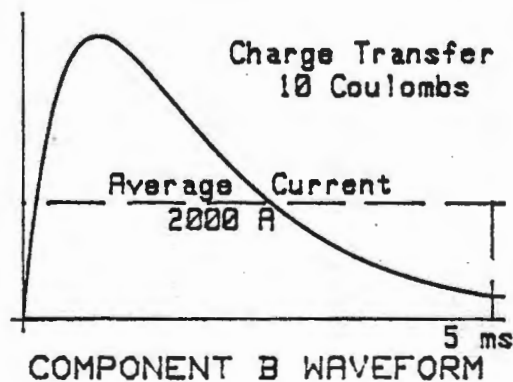
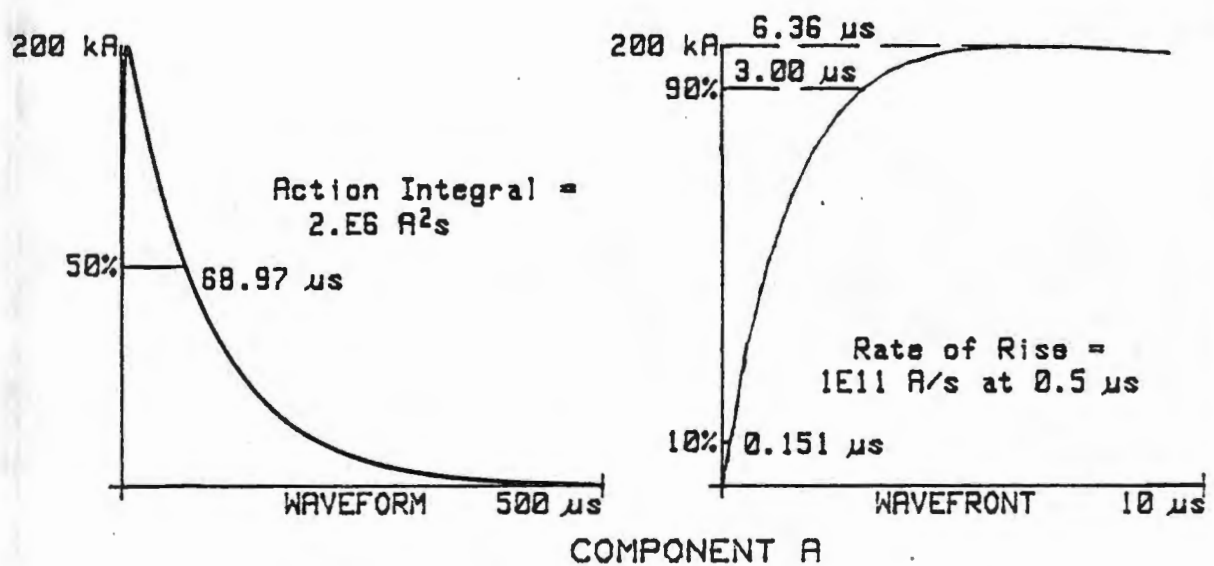


FIGURE 2.2-2. IDEALIZED LIGHTNING CURRENT WAVEFORMS

in figure 2.2-3. Table 2.2-1 summarizes the parameters for each waveform. A double exponential of the form shown below defines each component waveform.

$$I(t) = I_0 \times (e^{-\alpha t} - e^{-\beta t})$$

2.3. Interaction and Coupling to Aircraft

It is not possible to precisely quantify the lightning environment during any particular strike to an aircraft in flight. At low altitudes, cloud-to-ground strikes are more likely. Inter/intracloud strikes are more common at higher altitudes. Statistics on the number of strikes to aircraft as a function of altitude are given in figure 2.3-1.

It is generally accepted that the presence of an aircraft may trigger the leader discharge process between charge centers by disrupting the leader path. Recent experience by the FAA and the Air Force Wright Aeronautical Laboratory (AFWAL) in-flight program using a CV-580 to intercept lightning strikes indicates leader growth out of the wing of the test aircraft. Figure 2.3-2 illustrates the interaction of the aircraft with a charge center. When the aircraft is part of a leader not much current flows. The major effect occurs when the path is complete to the opposite charge center, and the return current passes through the aircraft.

A typical lightning flash consists of 20-200 current pulses over a one to two second interval. Because the aircraft is moving, the flash will sweep aft from an initial attachment point, all the while reattaching to spots along the line of flight aft from the original point of attachment. A portion of the current will enter the aircraft at several points along the path from the initial and swept attachment points. Figure 2.3-3 shows a typical path of attachment points for a swept flash. If the attachment is a trailing edge surface, then the current will hang on at this initial point and flow through there for the entire flash.

2.3.1. Standard Aircraft Lightning Environment

Since different parts of an aircraft must tolerate different levels of current, a standard definition has evolved defining the levels of current applicable to different zones of the aircraft. The SAE-AE4L committee has established generally accepted definitions of the possible strike zones as follows:

- Zone 1A: Initial Attachment Point with low probability of flash hang on, such as a leading edge.
- Zone 1B: Initial attachment point with high probability of flash hang-on, such as a trailing edge.
- Zone 2A: A swept stroke zone with a low probability of flash hang-on, such as a wing midspan.
- Zone 2B: A swept stroke zone with high probability of flash hang-on, such as the wing inboard trailing edge.

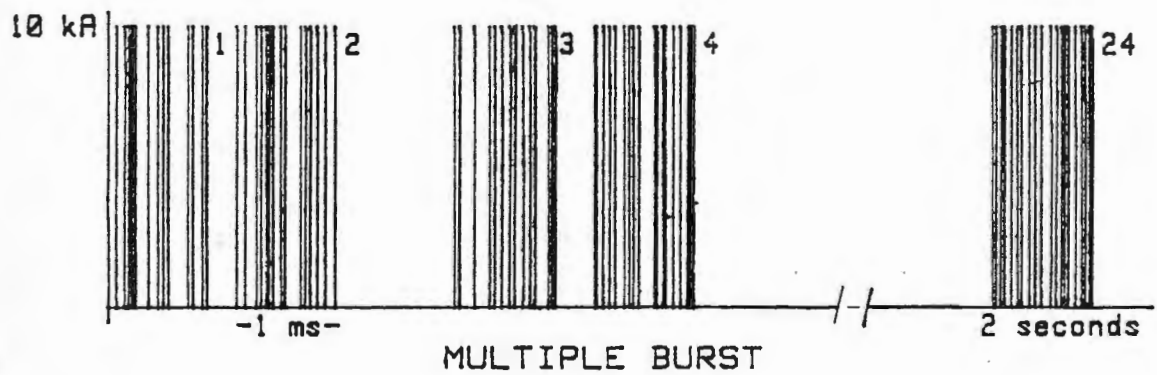
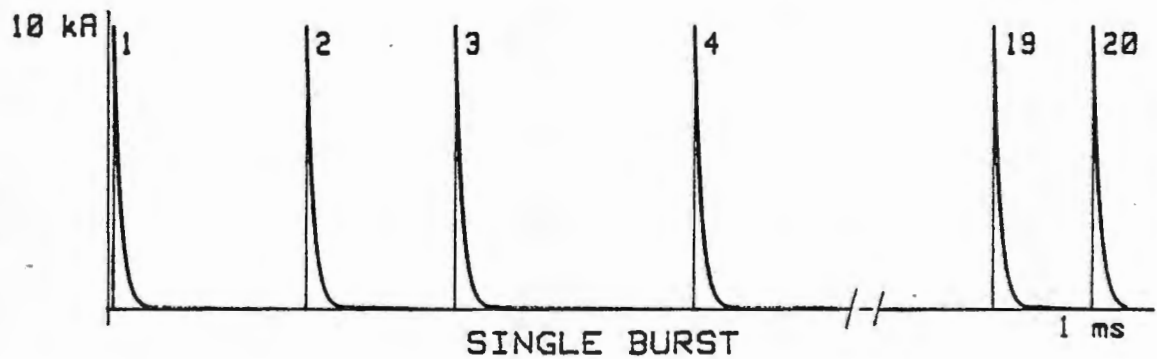
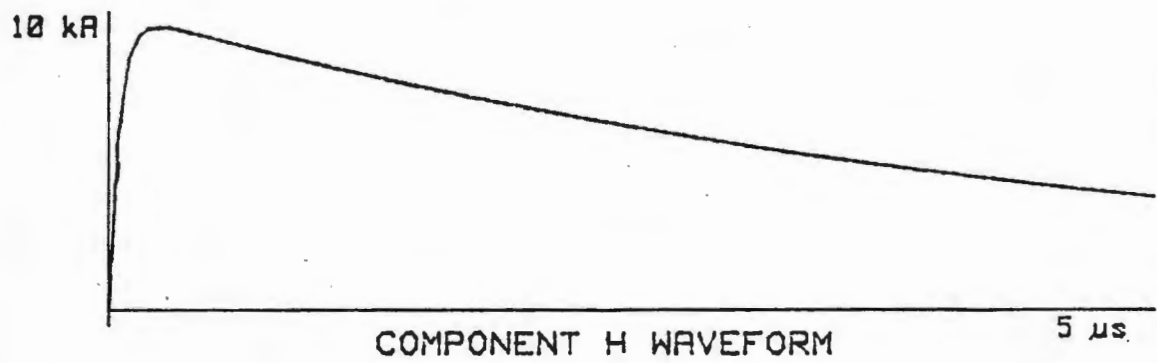
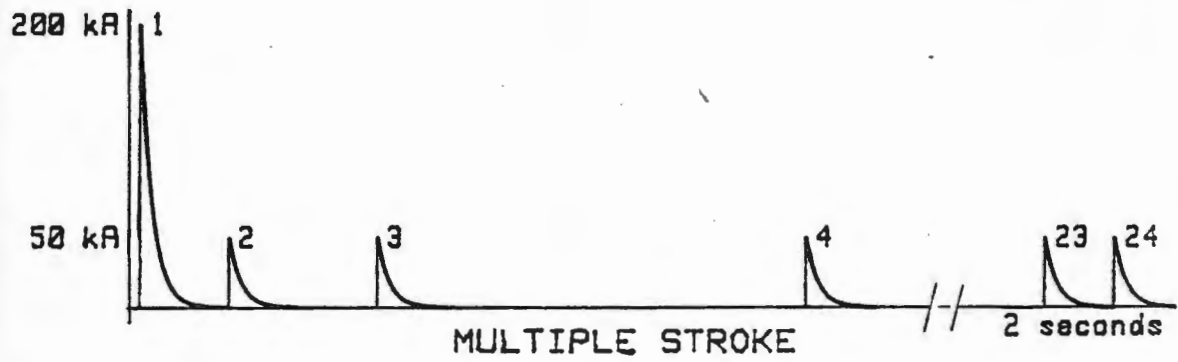


FIGURE 2.2-3. MULTIPLE STROKE CHARACTERIZATION

TABLE 2.2-1. IDEALIZED LIGHTNING WAVEFORM PARAMETERS

Threat	I_0 (A)	Alpha (1/s)	Beta (1/s)	Peak Current (kA)	Maximum Rate-of-Rise (kA/ μ s)	Action Integral (A ² s)
Severe Component A ^{1,2,3}	218810	11354	647265	200	140	2.0x10 ⁶
Intermediate Current Component B ²	11300	700	2000	4.2	NA	NA
Continuing Current Component C	400			0.4		
Restrike Component D ^{2,4}	109405	22708	1294530	100	140	2.5x10 ⁵
Multiple Stroke Component D/2 ²	54703	22708	1294530	50	70	6.2x10 ⁴
Multiple Burst Component H ²	10572	187191	19105100	10	200	NA

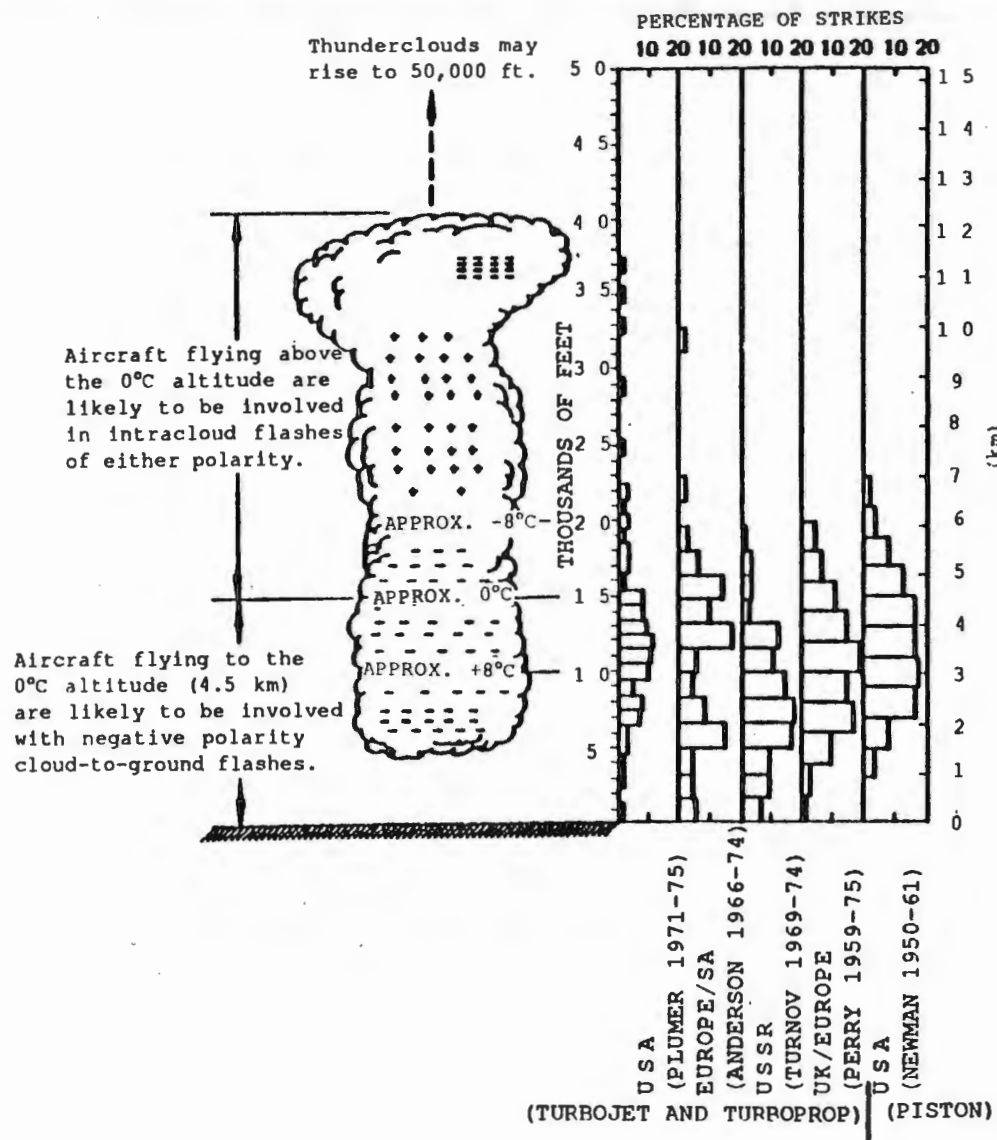
¹This component represents a very severe stroke, exceeding approximately 99.5 percent of all recorded strokes.

²The double exponential form is convenient for low level tests and analytical procedures.

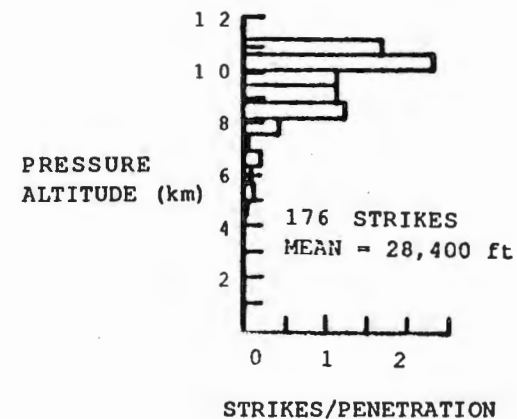
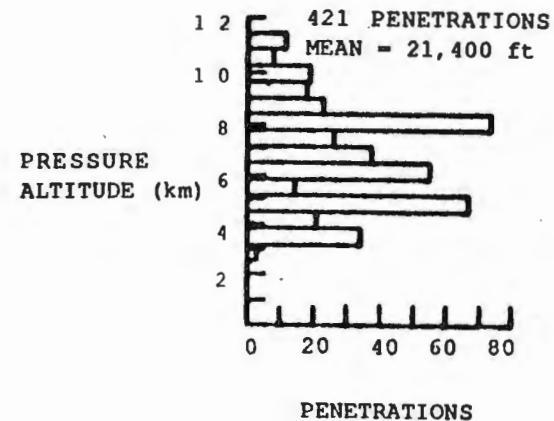
³Rate of rise represents 90 to 95 percent severity level.

⁴Rate of rise is the same as that of Component A to recognize the fact that recorded rise times of subsequent strokes are often as short or shorter than those of the first return stroke.

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Ref: Fischer & Plumer, 1977.



Ref: Fischer, et al., 1984.

FIGURE 2.3-1. AIRCRAFT LIGHTNING STRIKE INCIDENTS AS A FUNCTION OF ALTITUDE

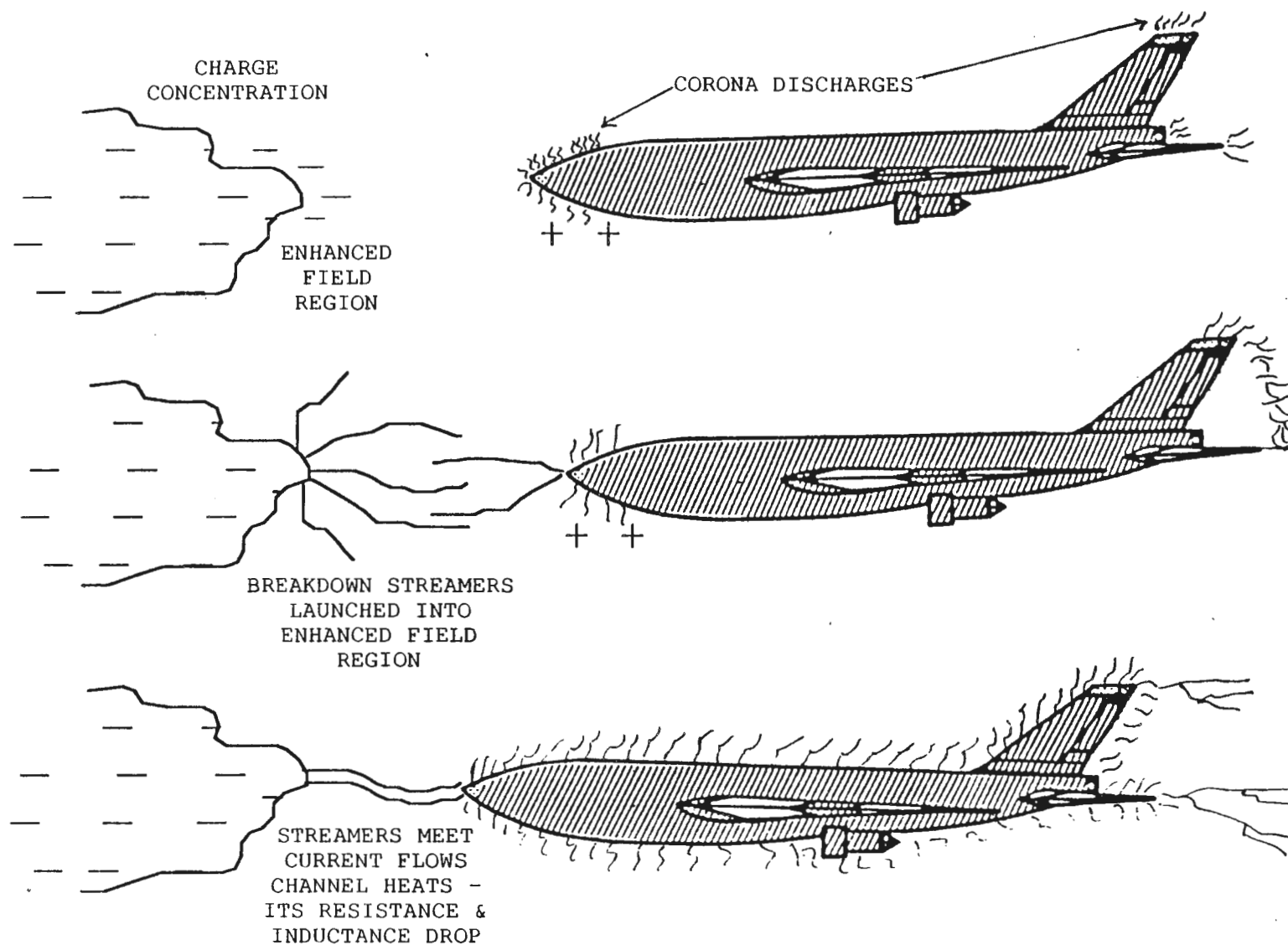


FIGURE 2.3-2. INTERACTION OF AIRCRAFT WITH CHARGE CENTERS

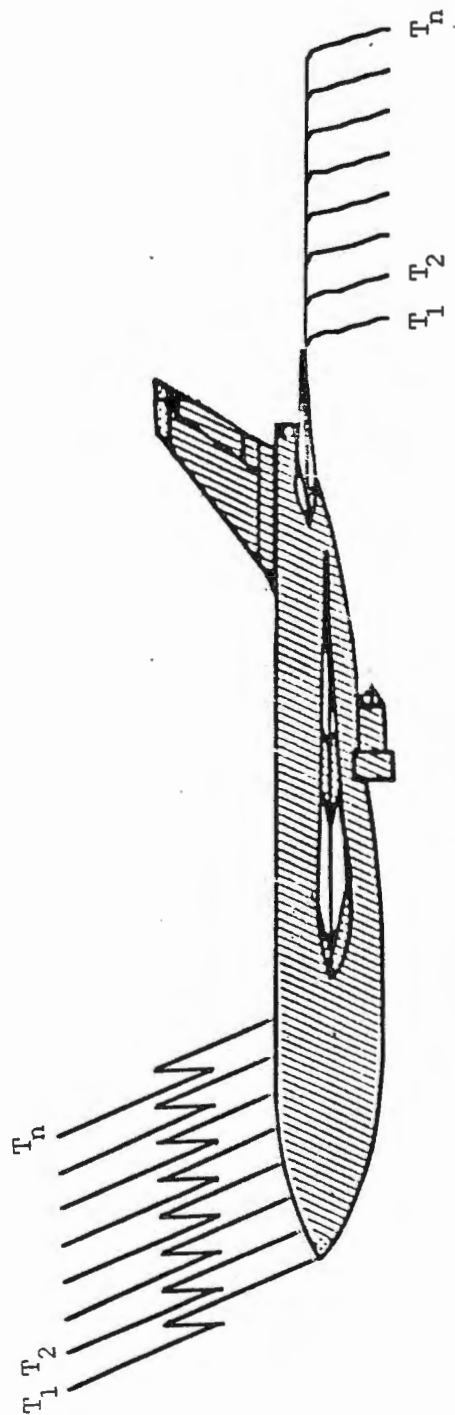


FIGURE 2 }-3. TYPICAL PATH OF SWEEP FLASH ATTACHMENT POINTS

- Zone 3: The remainder of the vehicle that is not covered by Zone 1 and Zone 2. There is a low probability of any attachment of the direct lightning flash arc. Zone 3 areas may carry large amounts of electric current, but only as a result of conduction between pairs of direct or swept stroke attachment points.

The SAE-AE4L committee has established the current levels to be expected in each of the strike zones to be used for test or certification purposes. An idealized representation of these defined current components for a complete flash is shown in figures 2.3-4 and 2.3-5.

The location of strike zones is well established for current aircraft configurations, see figures 2.3-6 and 2.3-7, but if configurations differ much from today's aircraft, locating the lightning strike zones may require careful interpretation of the particular geometry to determine the location and extent of the areas where lightning may attach, sweep, and hang-on.

A tabulation of lightning strike statistics is given in table 2.3-1 for major sections of commercial aircraft. In this grouping, the four sections most often struck by lightning are the nose, fuselage, wing, and horizontal stabilizer. The wide body jets also had high strike rates to antennas.

2.3.2. Structural Materials Effect on Strike Zones

A study by Grumman Aircraft Company investigated the strike attachment zones for an advanced fighter configuration (Craft, 1981). A scale model of the aircraft with replaceable panels for portions of the skin was used. The panels were made from aluminum, G/E, and kevlar. The study concluded that there is no essential difference between the strike zones for panels of G/E and aluminum. However, there is considerable difference for kevlar panels. Kevlar is a non-conductor; therefore, a lightning arc will jump over or along the surface rather than attach to kevlar.

Other studies have shown that the lightning arc will punch through kevlar to reach metal portions under the kevlar skin panels. This is also reported to be the case for radomes used to cover nose-mounted radar equipment. Lightning will readily puncture a radome to reach metallic portions of an all-metal aircraft. These items are described in a report by Fisher and Plumer.

2.4. Lightning Effects on Aircraft Systems

2.4.1. Range of Lightning Effects

Direct effects result from direct action of the lightning arc in the form of arcing and sparking on the aircraft; therefore, the term "direct effects." Evidence of this is seen as gross damage from the action of the arc on the materials. Protection against direct effects must come from the structure or protective diverters which prevent the currents from flowing into the sensitive areas. Important lightning parameters, which determine damage due to direct effects, are the dwell time on the aircraft, the peak current, the duration of

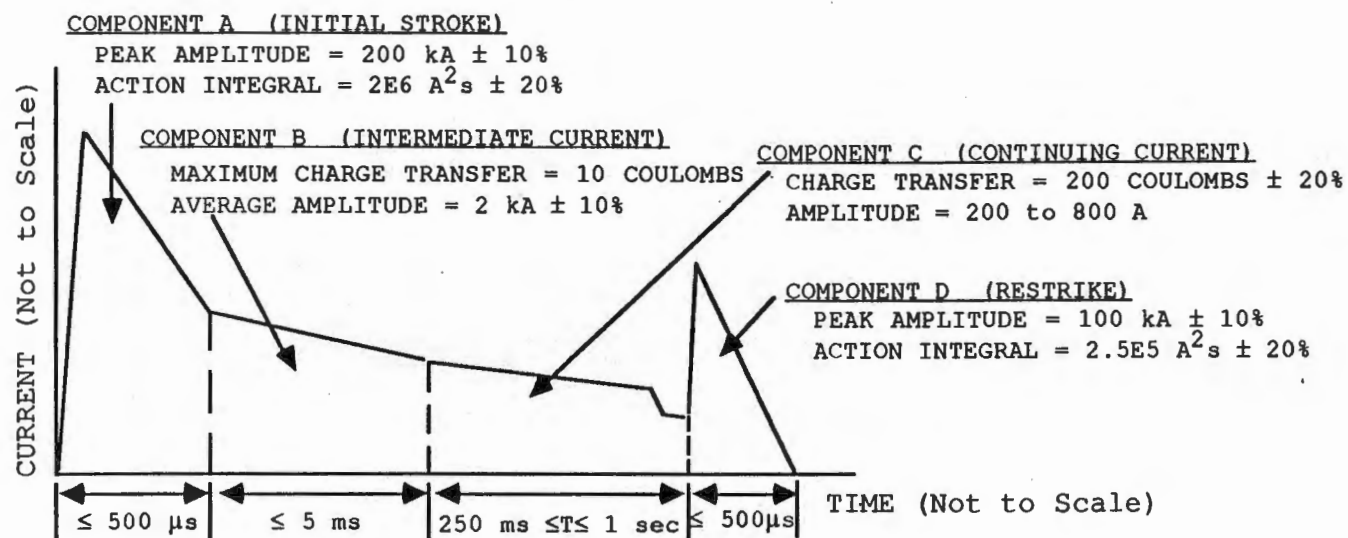


FIGURE 2 3-4. LIGHTNING CURRENT COMPONENTS EXPECTED IN EACH ZONE

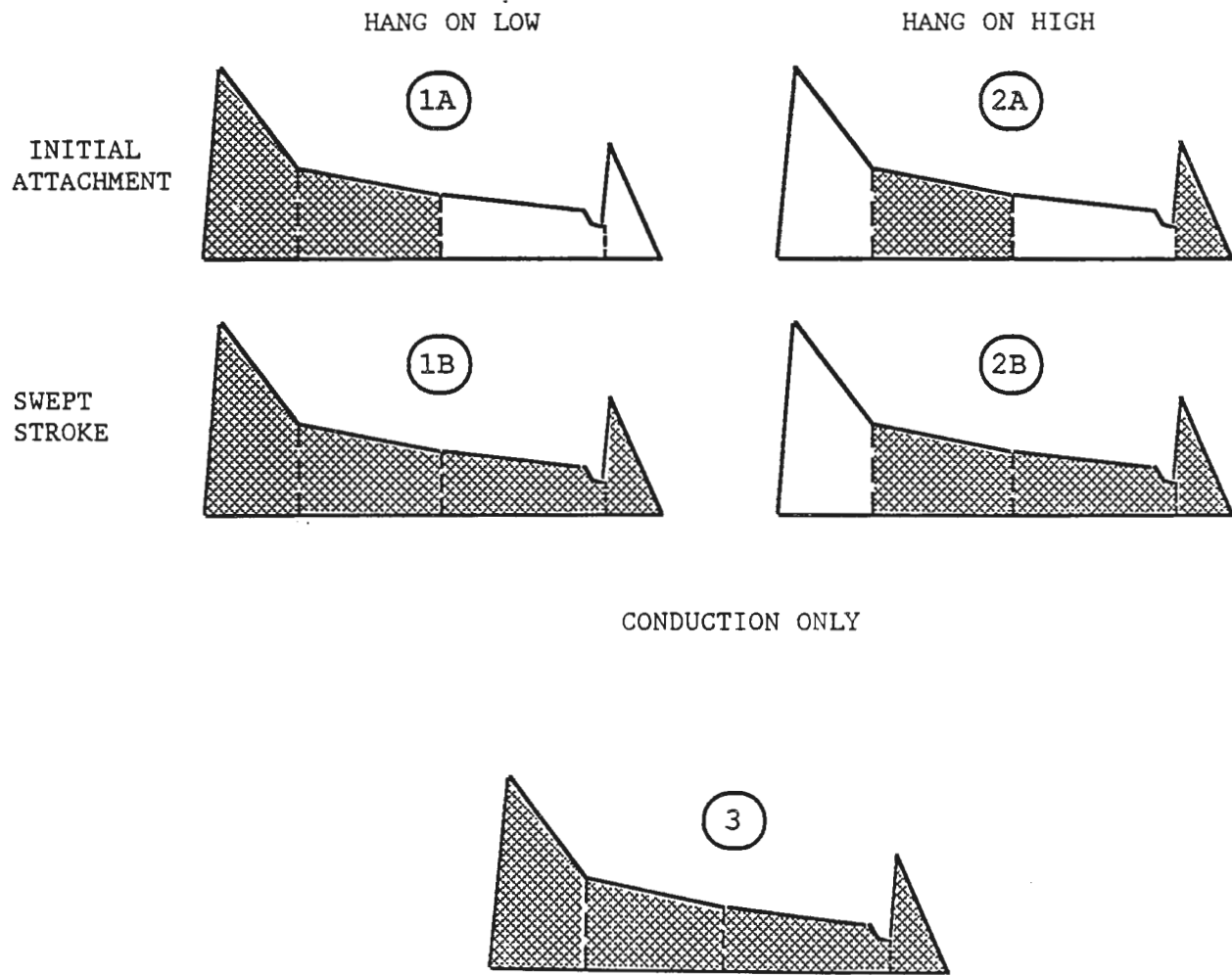


FIGURE 2.3-5. LIGHTNING CURRENT COMPONENTS EXPECTED IN EACH ZONE

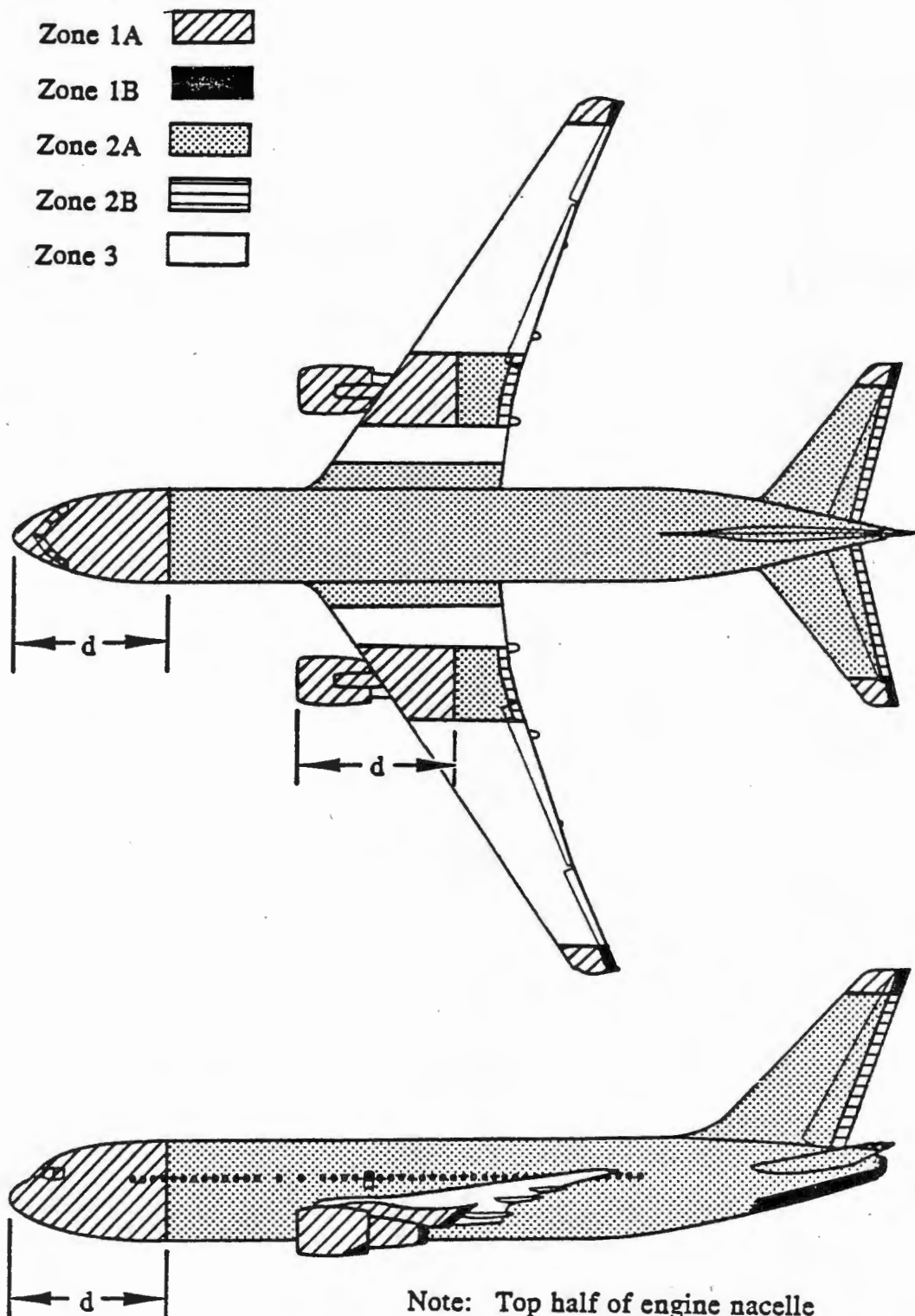


FIGURE 2.3-6. TYPICAL LIGHTNING STRIKE ZONES

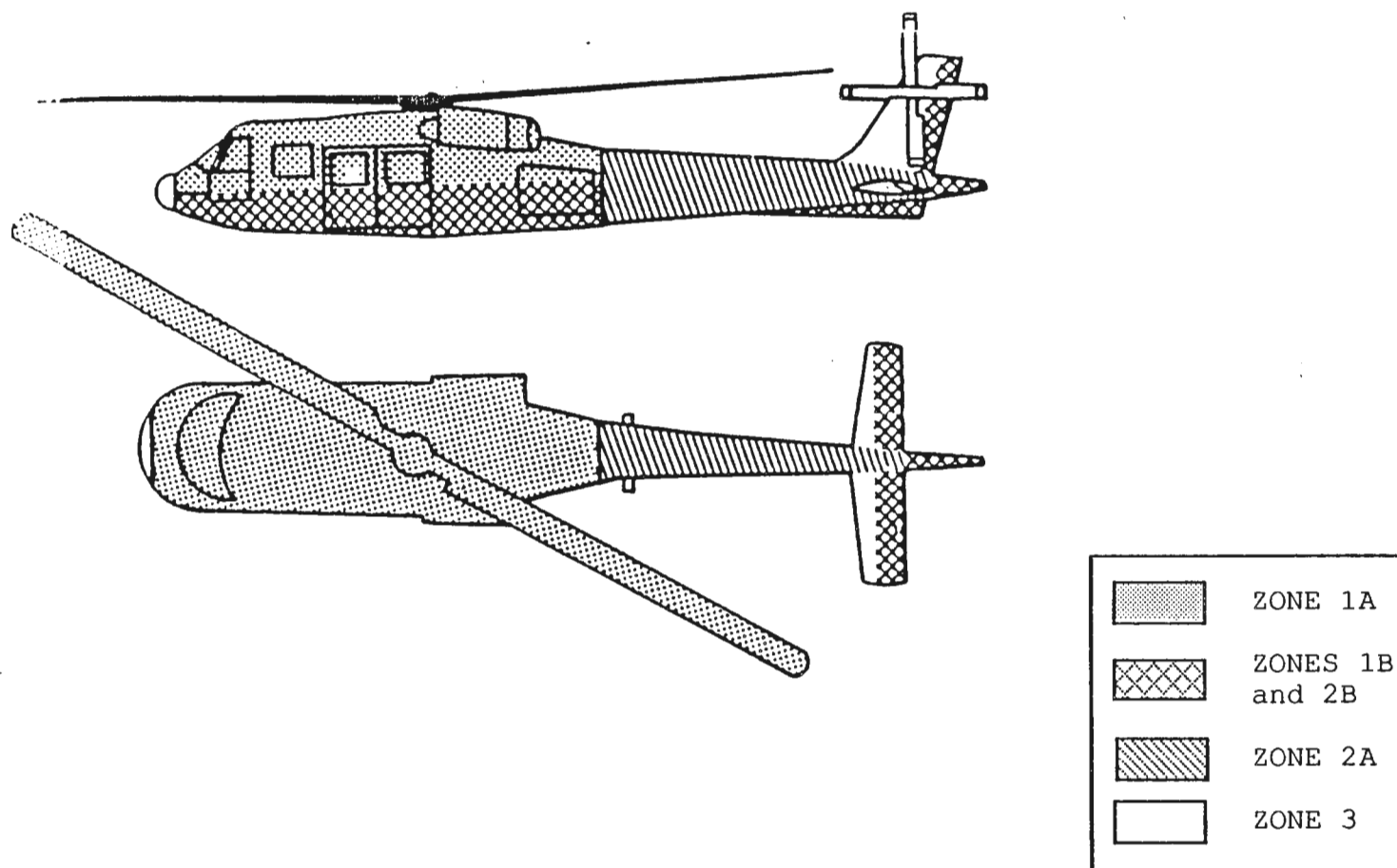


FIGURE 2.3-7. TYPICAL LIGHTNING STRIKE ZONES FOR HELICOPTERS
(User's Manual for AC-20-53, 1987)

TABLE 2.3-1. COMMERCIAL AIRPLANE LIGHTNING STRIKE STATISTICS

	JET 2 ENGINE AFT MOUNTED T-TAIL		JET 3 ENGINE AFT MOUNTED		JET 4 ENGINE WING MOUNTED		JET TOTAL NARROW BODY		JET WIDE BODY	
	NO.	%	NO.	%	NO.	%	NO.	%	NO.	%
NOSE	3	19	29	30	83	43	115	38	220	34
WING	-	-	24	24	33	17	57	19	99	15
HORIZONTAL STABILIZER	6	38	6	6	24	13	36	12	59*	9*
VERTICAL STABILIZER	-	-	14	14	12	6	26	8		
FUSELAGE	5	31	20	20	38	20	63	21	112	17
ANTENNAS	2	13	1	1	2	1	5	2	100	15
ENGINE	-	-	1	1	-	-	1	< 1	24	4
TAIL	-	-	3	3	-	-	3	1	35	5
TOTAL	16		98		192		306		649	

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Ref: Cooley, Geren, and Melander; Sept. 1982

* HORIZONTAL/VERTICAL COMBINED

current, the maximum current rate-of-rise, and the number of restrikes. Aircraft parameters, which determine damage due to direct effects, are skin type, skin thickness, and paint.

Damage may also occur from indirect effects that damage components or disrupt the software operations of electrical and electronic systems. Indirect effects are generally more subtle than direct effects, because the physical damage from indirect effects is not easily seen. Indirect effects are caused by the transient voltages and currents that are induced into the wiring as a result of the lightning arc and the associated EM fields on the aircraft structure. Important parameters are the peak current, maximum current rate-of-rise, maximum EM fields rate-of-rise, number of restrikes, and for non-metal aircraft the duration of current flow.

Protection design for lightning indirect effects has many factors. Parameters to be considered when protection beyond the baseline configuration is needed include protection provided by the structure, cost, weight, reliability, impact on vehicle performance, expected life cycle, and protection necessary against other EM threats.

Table 2.4-1 lists possible effects from direct or nearby lightning strikes to aircraft in flight.

2.4.2. Direct Effects on Non-Metallic Structures

The most important direct effect of lightning strikes is the possible arcing and sparking in fuel areas. Other important direct effects include pitting and burning in structural materials.

For reasons discussed in Cooley (1985), direct effects protection design requires more test and analysis for composite materials than for metallic materials. For example, it is usually assumed, in the design of lightning protection, that if an electrically conductive fastener is located in an attachment zone having a high probability of hang-on, lightning will attach to the fastener. Because of the currents in the strike zone definitions, the direct attachment to a fastener in zone 1A, 1B, or 2B represents a severe worst case for any lightning strike. Although direct attachment is not a severe problem in aluminum structures, it is presently not possible to develop a fastener capable of joining advanced composite materials and carrying the full lightning current without seriously weakening the fastener. Current composite aircraft designs depend upon not having fasteners in critical parts of zone 1 and 2. Where fasteners are used, a repair procedure may be necessary for lightning strike damage.

Painted metal surfaces have longer hang-on times for swept strokes than do unpainted metal surfaces. These longer dwell times will cause deeper burning, and additional metal is required for painted metal skins on wet-wing aircraft.

Composite materials have various resin binders that burn and affect the dwell time for swept strokes. Therefore, the thickness of composite materials required for structural integrity protection against lightning strikes will

TABLE 2.4-1. RANGE OF EFFECTS FROM ATMOSPHERIC ELECTRICITY INTERACTIONS WITH AIRCRAFT

EFFECT	CAUSE	CRITICALITY
Flight Control Disruption	Flight control systems have low tolerance to electrical transients caused by indirect lightning or static electrification effects. Transients may simultaneously affect parallel redundant systems.	Minor to Catastrophic
Fuel Tank Fire or Failure	Fuel vapor ignition may be caused by static electricity or lightning direct effects on structure. Fuel gauging and flow management electrical/electronics may spark from indirect effects.	Minor to Catastrophic
Loss of Engine Power	Power loss can result from possible direct effects which cause thermal or acoustic shock at engine inlet, or indirect effects of electrical transients on engine controls.	Minor to Catastrophic
Radome, Canopy, & Windshield Damage	Damage can occur due to direct effects of lightning strikes and arc discharge may be caused by static electricity buildup.	Minor to Catastrophic
Instrumentation Problems - Communications, Navigation & Landing System Interference	Instrumentation systems are disrupted by indirect transient effects caused by static electricity buildup and nearby or attached lightning strikes.	Minor to Catastrophic
Structural Damage	Structure damage is a result of burns or arcing and sparking from direct effects of lightning attachment to aircraft.	Minor to Catastrophic
Physiological Effects on Crew	Flash blindness & distracting electrical shock may be caused by the direct effects of nearby or attached lightning strikes.	Minor to Catastrophic
Unscheduled Deployment of Landing Gear or Control Surfaces	Premature activation of actuators is caused by indirect effects of lightning or static electricity buildup in electrical/electronic systems.	Serious to Catastrophic

depend upon the material's makeup as well as the fibers. It is expected that different materials of the same physical strength will behave differently when struck by lightning.

2.4.3. Indirect Effects on Non-Metallic Structures

The most important adverse feature of graphite and metal matrix materials is their lower conductivity. The conductivity factor, compared with aluminum, is 10 times less for metal matrix and 1000 times less for G/E. Lower conductivity means higher voltage drops in these structures than for a metal structure. Thus the voltage drop in a length of structure will be increased for the same level of lightning current. Because of the higher voltage drop, higher levels of currents will flow in any metallic paths such as plumbing and wiring inside an aircraft structure made from non-metallic materials. Additionally, any external EM fields will be attenuated less by a composite structure. For direct and nearby strikes, the internal EM fields will be at higher levels than for an aluminum aircraft. The exact difference in internal EM fields depends on the number of joints and openings, thickness of material, frequency range of the external EM fields, and the shape of the structure. The Atmospheric Electricity Hazards Protection (AEHP) program studies on an advanced composite (G/E) structure F-16 fuselage showed that the shielding was 15 dB for lightning pulses. It is estimated that a similar aluminum structure would have provided 30 to 40 dB of shielding, the limitation being primarily due to the number and size of openings.

In structures made from non-conductors such as kevlar and fiberglass, there is no EM shielding. This is also true for materials having very few conducting wires embedded into a non-conducting material such as Thorstrand kevlar. These materials can provide some protection against direct effects. There will be little or no improvement against indirect effects, because the magnetic shielding effectiveness of low loss factor metals, such as aluminum, is primarily due to currents flowing in the material. These currents generate EM fields which tend to cancel the incident fields. Electric field shielding is primarily due to an equipotential plane established by high conductivity metals. There are too few wires and too little metal in a material like Thorstrand to provide an effective shield against EM fields from nearby lightning sources or RF sources, such as airport radio and radar transmitters.

2.4.4. Effects on Aircraft Systems

Six examples of onboard system functions affected as a result of lightning and static electrification of aircraft are described below.

- The failure of engine instruments was experienced, as a result of lightning strikes, on an equipment version installed on a jet transport aircraft early in the development phase. The N1 tachometer and N2 were occasionally affected during lightning strikes. Electronic components in the engine tachometer transmitter were damaged. These components were connected to unshielded twisted pairs in a bundle of engine related wiring. The situation was repeated in laboratory tests and traced to the need for 10 dB additional shielding for the wiring. The solution to this problem was to add the necessary shielding by applying aluminum foil to the upper half

of the inside surface of the kevlar leading edge. This solution resulted in lighter weight than shielding the wiring.

- Possible sparks between fuel tank wiring and mechanical structures are a serious concern. For example, fuel plumbing can carry substantial currents in the event of a strike to the overflow vent tube. The fittings and plumbing must be capable of carrying these currents without sparking. Fuel gauge electrical wiring must be protected from currents on the structure path between lightning attachment and exit points along the leading and trailing edge. Currents from the cable shield to structure must not spark within the fuel tank or the fuel gauge.
- The pitot boom is often the point of attachment for lightning because of its shape and location near the front of an aircraft. The very high currents near the initial attachment can burn out the pitot heater and possibly damage electronic equipment inside the aircraft. This is particularly severe for pitot booms on aircraft with nose radomes.
- Hydraulic plumbing for flight control actuators needs to be grounded and bonded to the structure to remove high currents from the tubing fittings. These fittings have been damaged by the currents flowing in the structure and tubing from lightning strikes to the empennage. Damage was evidenced by loss of fluid in hydraulic systems.
- Static electricity results from triboelectric charging under normal flight conditions, from vehicle charging caused by nearby leaders, or as part of the initial attachment phase. High electric field breakdown discharges the accumulated charge in a repeated cycle that results in a series of sparkovers and therefore a repeating transient pulse. The electrical waveforms have wide band energy up to and including the Very High Frequency (VHF) band. Effects of static discharges appear as excessive radio noise and loss of sensitivity in the navigation and radio systems. The high electric fields have the potential to damage windscreens and damage electronic components in the window heater control.
- Lightning induced transient currents in electrical cables can cause errors in the digital data. Errors in digital bit streams are characterized by single bits or a burst of errors depending upon the baud rates. These errors are no different than those that should be expected due to other common EMI sources such as power transients, power faults, and RF and radar transmitter modulation frequencies. Digital systems designers need to consider possible disruptions in data streams similar to those that occur during long distance transmission over noisy lines and to include error detection and correction for flight critical data transmissions.

In addition to bit stream errors, errors in digital data can occur in the discrete data signals. Discrete data input signals can be modified from true, "1", to false, "0", by transient currents, as for example, in the sensing of flaps up, wheels down, or other aircraft configuration sensors which operate as a high/low switch position. Discrete signals should not be directly used by onboard digital systems without some checks for reasonableness or low pass filtering or both. Mode change commands should

be verified over five to ten samples. Discrete data outputs to indicators and actuators may also be subject to upset or damage. Faulty indicators increase crew workload and may contribute wrong information leading to hazardous flight conditions. Errors in these discrete signals are serious because they lead to wrong information regarding the aircraft configuration. Early raising of the flaps on takeoff or lowering the landing gear at cruise altitude are two obvious examples.

3. INDIRECT EFFECTS PROTECTION - DESIGN APPROACH

Lightning protection for aircraft should be addressed during the design phase of the vehicle. This allows optimization of the lightning protection design and possible integration with protection designs for other EM threats. A design margin provides the definition of protection needed. This section discusses design considerations for grounding and bonding, installation of equipment, and routing of cables.

3.1. Design Margin

Adequate protection of equipment or systems is defined by a design margin. The design margin represents the difference between the transients induced at the interface level (by a worst case lightning strike) and the transients that the equipment is designed to withstand. This should be a positive number. A negative design margin indicates that some redesign is necessary. The size of the design margin is inversely proportional to the confidence which is given to the tests and/or analysis results used for verification.

The first step in defining a margin is to determine a threshold where the equipment becomes damaged so that it can no longer perform its intended function. Both analytical and experimental techniques are available for assessing the damage threshold. The techniques might be used during equipment design where the thresholds are specified, or during overall vehicle assessment where thresholds are used to establish Equipment Transient Design Levels (ETDL).

The equipment transient susceptibility level is another name for the damage threshold. This level will normally be somewhat higher than the ETDL, which represents the amplitude of voltage and/or current that the equipment is required to withstand or tolerate and remain operational. The ETDL, in turn, will be set higher than the maximum amplitude of transients that are allowed to occur on interconnecting wiring, the Transient Control Level (TCL). The relationship between transient control, equipment transient design, and susceptibility levels is illustrated in figure 3.1-1.

Standardized definitions of induced voltage and current waveforms and amplitudes that are representative of transients which appear in interconnecting wiring are found in AC 20-53. ETDLs may be selected from among the amplitude levels presented; TCLs can be derived by providing an adequate margin. Normally the transient control and design levels will be established by the airframe manufacturer or system integrator, who will compare the penalties of vehicle or interconnecting wiring protection with those of equipment hardening to establish the most efficient levels.

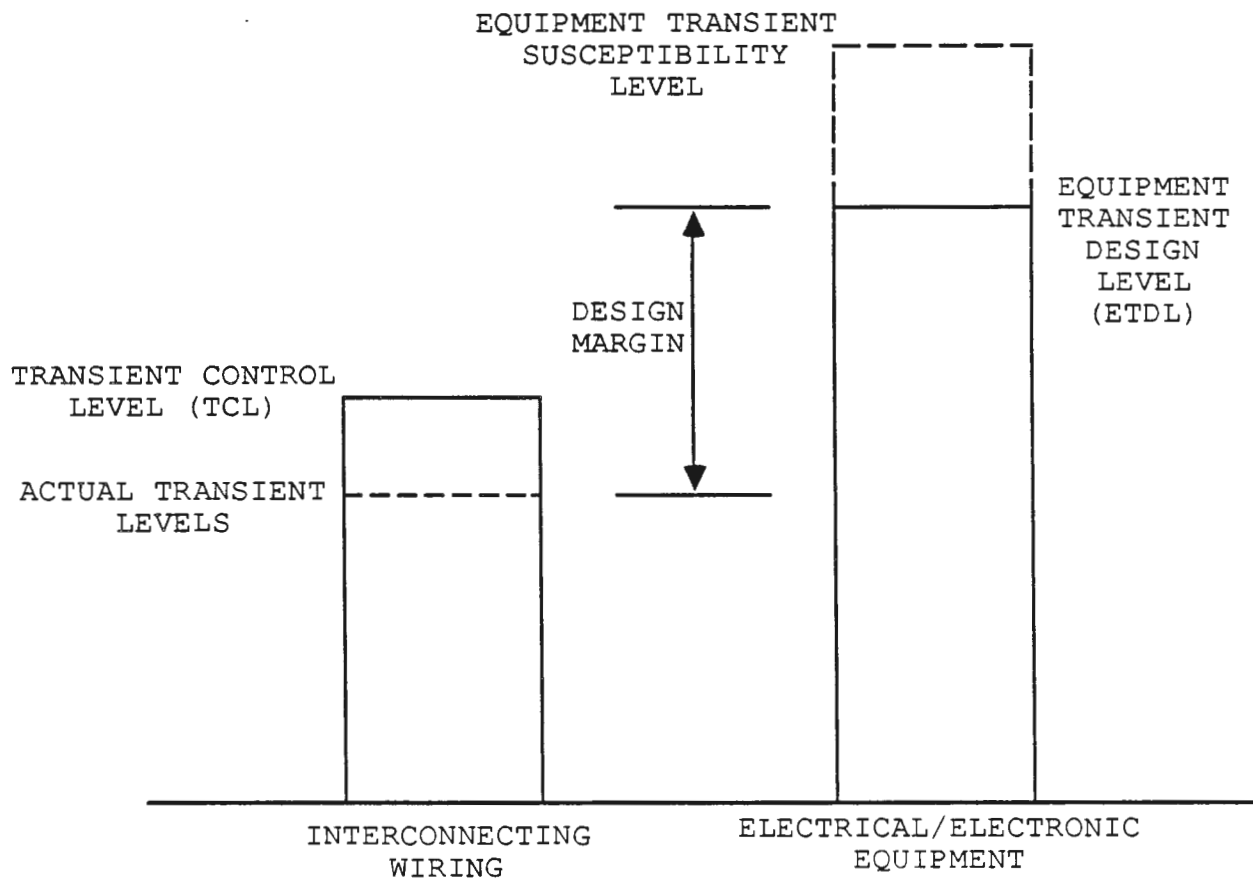


FIGURE 3.1-1. DESIGN MARGIN DEFINITION

3.2. Requirements for Grounding and Bonding

Grounding and bonding of the electronic equipment in an aircraft presently has a requirement for 2.5 milliohms from chassis to chassis (MIL-B-5087B, 1970). The resistance requirement was established early to assure that potential interference sources in different equipment would be isolated from sensitive electronics. The basis for the number chosen was one of engineering judgement. If the resistance was zero, there would be no interference or hazard to personnel due to power system faults or lightning. It is particularly convenient to use the 2.5 milliohm value for assessment of system protection against lightning. If 200,000 amps flows in a 2.5 milliohm resistor, the maximum voltage drop will be 500 volts. If every bond has less than 2.5 milliohms resistance, then no end-to-end validation testing is necessary, providing each unit of equipment will withstand 500 volts.

For the most part, aluminum and titanium alloys present no major difficulties in meeting the 2.5 milliohm specification. Maintaining adequate conductivity throughout an airframe has been a problem only when good contact between mating parts could not be maintained. Breaks in conduction paths are necessary at doors, hinges, and control surfaces. Corrosion control often requires a nonconductive coating on parts. Where interruptions in the grounding path occur, bonding requirements are met by reestablishing the conduction path using a metal jumper or strap. Utilizing present metal aircraft design approaches for composite craft will not yield the 2.5 milliohm resistance required.

Requirements for direct effects protection require that the grounding and bonding system be spark free in fuel areas, and that load bearing elements are not seriously degraded upon experiencing a lightning strike. Meeting these requirements requires an experimental approach.

Requirements for indirect effects protection of electrical/electronic systems against lightning strikes depend upon meeting the operational requirements of the equipment. The need for adequate grounding and bonding is well established by operational experience. For example, aircraft having poor grounding and bonding (i.e., poor or no electrical conduction throughout the structure and wiring) may be subject to operational hazards varying from major to minor. When different parts of the aircraft and wiring are not electrically connected, differences in potential can build up between them. Since the ultimate breakdown potential of air is lowered with altitude, corona or sparks may occur. Furthermore, fuses or circuit breakers to protect against shorts in equipment and wiring may not open properly. These factors lead to the following:

- Fuel explosion hazards from sparking in fuel gauges and wiring.
- Static discharges that may rupture windows (dielectric).
- Personnel electric shock hazards.
- Electronic equipment damage - burn out.
- Electrical/electronic systems malfunctions.

- Excessive radio noise when communicating with airport or other aircraft.

The electrical function of a bonding and grounding system for protection of electronic and electrical equipment is to provide a low impedance ground path for current flow. Table 3.2-1 indicates the current levels that may be expected from various sources.

The path impedance should be low enough that these interference sources will not produce a high voltage value that can disrupt equipment functions. Electrical/electronic equipment can be made to tolerate voltage differences of a few volts between units. Typical values for equipment tolerance against transients are shown in table 3.2-2.

Tolerable levels of ground impedance, based upon values in tables 3.2-1 and 3.2-2, are shown in table 3.2-3. This data indicates that the range of acceptable ground impedance values depends upon the source of interference and the degree of protection built into the installation and equipment. Equipment containing large scale integrated circuits can be designed and built to withstand 600 volt potential differences. This level of protection requires balanced circuits, high impedance input networks, and twisted pair wiring. More typical circuitry can withstand 10 volts between units by using balanced circuits. Single ended digital circuits can only tolerate a few volts between units for TransistorTransistor Logic (TTL) circuits that operate with 5 volt logic levels.

3.3. Guidelines for Cable Routing and Equipment Installation

The aircraft structure has a significant influence on the lightning protection measures needed for a specific design. The structure provides the first layer of protection from lightning. If the structure were a solid shell and a perfect conductor, no other protection measures would be necessary. In reality, there are apertures, joints, and imperfect conductors. These lead to leakage of fields and currents penetrating to the interior of the aircraft. It becomes necessary to select ways to minimize coupling to internal wiring.

Protection may be provided by increased shielding on exterior cables, good bonding at apertures, selective cable routing, surge arrestors, and wiring configuration between circuits.

Proper ways of penetrating the structure or equipment are illustrated in figure 3.3-1. These techniques maintain the protection integrity of the outer structure rather than allowing currents to penetrate the vehicle interior.

As a general rule, wiring should be routed near a good ground or reference plane and away from joints and apertures. A ground plane provides a common low impedance reference for equipment and also provides a fault return for power systems. Multiple connections between the ground plane and the structure allow division of the lightning current between the metal ground and structure. This is essential for composite structures.

TABLE 3.2-1. GROUND SYSTEM CURRENTS FROM A VARIETY OF EM SOURCES

CURRENT SOURCE	RANGE OF CURRENT
1. Lightning currents, broad band pulses.	20 KA to 200 kA
2. Power system fault currents, continuous wave 400 Hz.	100 A to 1000 A
3. RF current returns from on board transmitters and transponders, continuous wave HF, VHF, UHF, and microwave.	1 A to 10 A
4. Currents from nearby sources of RF energy such as radar and radio transmitters, continuous wave HF, VHF, UHF, and microwave.	1 A to 10 A
5. Power currents resulting from relays, strobe lights, and other high pulse current devices.	5 A to 50 A
6. Digital and analog signals between units of equipment, CW and Pulses 100 Hz to 10 MHz.	10 ma to 5 A

TABLE 3.2-2. TYPICAL INTERFERENCE TOLERANCE LEVELS BETWEEN UNITS OF ELECTRONIC EQUIPMENT

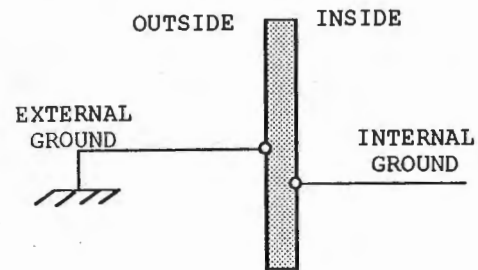
1. Lightning Pulses (a)	600 volts
2. Power System Faults (b)	10 volts
3. Continuous Waves HF Frequencies	50 volts
4. Continuous Waves VHF Frequencies	100 volts
5. Continuous Waves UHF Frequencies	100 volts
6. Tracking Radar (Sweep Frequency)	10 volts
NOTES: (a) Specially protected against lightning by balanced signals, high impedance, and twisted pair circuits. (b) Circuits unprotected except balanced signals, twisted pair wiring.	

TABLE 3.2-3. TOLERABLE LEVELS FOR GROUND IMPEDANCE IMPLIED BY TABLE 3.2-1 AND TABLE 3.2-2 VALUES

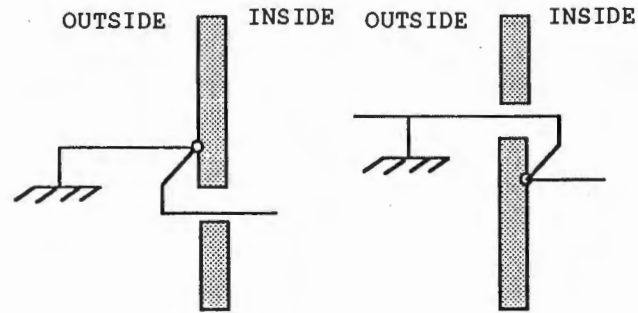
1. Lightning Pulses	3 to 30 milliohms
2. Power System Faults 400 Hz	10 to 100 milliohms
3. Continuous Waves HF	5 to 50 ohms
4. Continuous Waves VHF	10 to 100 ohms
5. Continuous Waves UHF	10 to 100 ohms
6. Tracking Radar (Sweep Frequency)	0.1 to 1 ohm
7. Power Currents From Onboard Pulsed Current Devices	20 to 200 milliohms
8. Digital and Analog Signals	0.2 to 100 ohms

13-35

CORRECT



COMPROMISING



UNACCEPTABLE

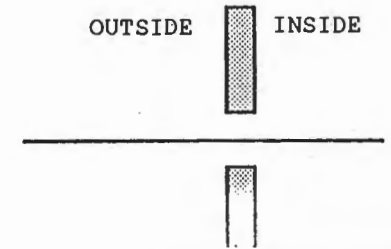
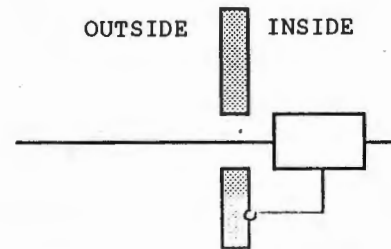
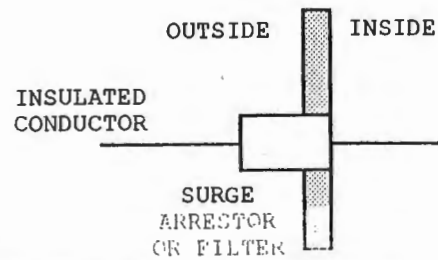
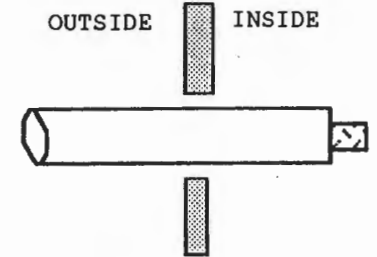
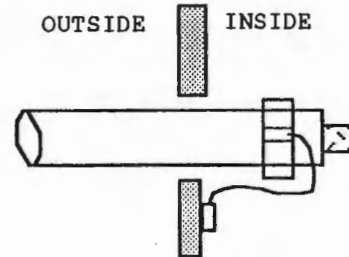
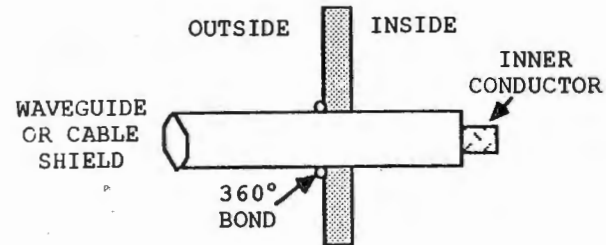
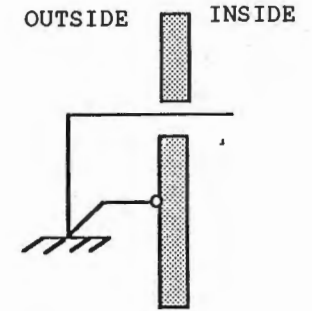


FIGURE 3.3-1. METHODS OF PENETRATING THE STRUCTURE WITH VARIOUS CONDUCTORS

Wiring is subject to differential mode and common mode coupling. Usage of twisted pair reduces the differential mode coupling. Shields tied to ground at each end of the cable significantly reduce the common mode coupling.

Figure 3.3-2 shows the coupling characteristics associated with various wire types and configurations. The wiring is assumed to be internal but routed near an aperture or non-metal structure.

Cable routing may not reduce transients from lightning indirect effects sufficiently to protect circuitry. Terminal protection devices to provide additional shielding can be placed at a vehicle penetration or within a circuit. Devices are typically in-line modules and may include limiters, filters, attenuators, buffers, and RF control devices. Each has different characteristics and must be evaluated for inclusion in specific designs. Important parameters, which should be evaluated for each device, are low shunt impedance, noise floor, insertion loss, insertion phase, and clamping capabilities at low and high levels.

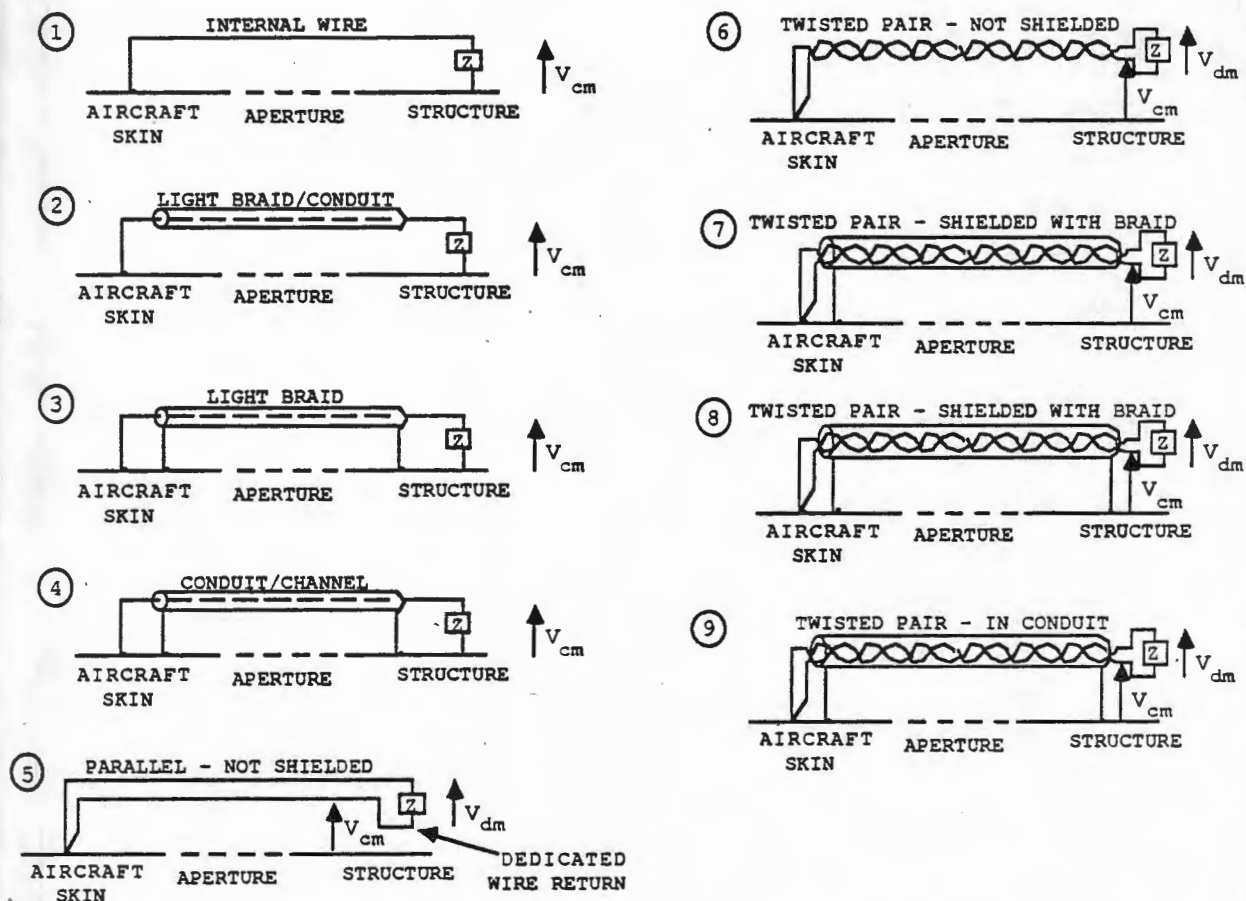
AEHP has defined the following guidelines for equipment installation to maximize the inherent shielding provided by the aircraft structure (Orange Book, 1987).

- Orient equipment where coupling with the EM fields is minimized. Avoid locations where fields are the highest (e.g., Zone 1 locations).
- Locate equipment as far from apertures and joints as possible.
- Cluster equipment in areas where the lowest flux density is expected, i.e., away from locations with small radii of curvature.
- Locate equipment in compartments with shielding, if possible.
- Where equipment must be located near apertures such as equipment bay hatches, the bay should be shielded and EMI gaskets used to improve bonding of the hatch to the compartment.
- Equipment in the canopy area, as well as the wiring leading to and from it, should be shielded. Equipment should be kept below the canopy sill, if possible.

A typical requirement for equipment protection is that both upset and damage protection be provided. This protection can be provided by either hardware or software.

The following AEHP guidelines apply to wire placement:

- Route the wiring in areas where lowest flux intensity is expected:
 - Within conductive structural enclosures.
 - Where minimum aperture coupling is predicted.
 - As close as possible to highly conductive structural members.



MAGNETIC FIELD INDUCTIVE/RESISTIVE VOLTAGES		ELECTRIC FIELD CAPACITIVE VOLTAGES	
①	FULL COMMON MODE VOLTAGE	FULL COMMON MODE VOLTAGE	
②	FULL COMMON MODE VOLTAGE	> 40 dB ATTENUATION	
③	20 TO 35 dB ATTENUATION	> 40 dB ATTENUATION	
④	> 60 dB ATTENUATION	> 60 dB ATTENUATION	
COMMON MODE	DIFFERENTIAL MODE	COMMON MODE	DIFFERENTIAL MODE
⑤ FULL VOLTAGE	20-30 dB LESS THAN V_{cm}	FULL VOLTAGE	} NOT EASILY SPECIFIED, BUT 30-40 dB LESS THAN V_{cm}
⑥ FULL VOLTAGE	50-55 dB LESS THAN V_{cm}	FULL VOLTAGE	
⑦ FULL VOLTAGE	55-60 dB LESS THAN V_{cm}	> 40 dB ATTENUATION	} NEGLIGIBLY SMALL
⑧ 20 TO 30 dB ATTENUATION	55-60 dB LESS THAN V_{cm}	> 40 dB ATTENUATION	
⑨ > 60 dB ATTENUATION	> 60 dB LESS THAN V_{cm}	> 60 dB ATTENUATION	

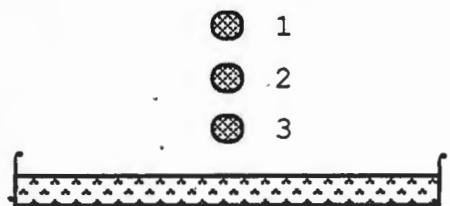
FIGURE 3.3-2. COMMON AND DIFFERENTIAL-MODE COUPLING CHARACTERISTICS FOR STRUCTURE RETURN AND TWO-WIRE CIRCUITS

- Close to a skin that can be considered a good shield, such as aluminum with controlled joints.
- Away from conducting structures with small radii of curvature..
- Orient cables perpendicular to primary current path and joints to minimize coupling.
- Separate wiring according to degrees of hardness and threat exposure (i.e., isolate wiring to equipment located in a lightning zone).

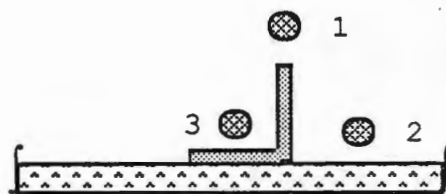
Figure 3.3-3 shows various locations for routing cables near structure. For all-metal structures, the numbers 1 through 4 indicate worst to best locations for cables. This is not the case with composite structures.

Coupling to these configurations was quantified using a two-dimensional method-of-moments code (Geren, Melander, and Hall, 1987) using typical thicknesses and conductivities for aircraft structures. Predictions tabulated in table 3.3-1 were made based on three structural materials. First, each configuration was assumed to be entirely made up of aluminum. Second, the skin was assumed to be G/E and primary structure was aluminum. The third configuration was identical to the second, except that kevlar, a non-conductor, replaced the G/E skin. These predictions were made with only the structure shown and assumed a 200,000 amp double exponential waveform input to the model. The values illustrate the differences in cable placement for metal and composite structures. See section 6.1 for an additional example of coupling to wiring in various locations.

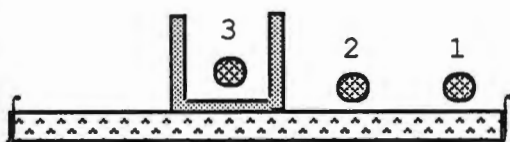
Peak current and voltage transient responses on wiring between aircraft systems, such as autopilot, engine control, fuel pump, fuel gauge, and electric power, due to a 200 kA direct lightning strike are tabulated in table 3.3-2. These values were obtained from results measured during a low-level swept CW lightning simulation test on a composite test bed representative of single engine general aviation aircraft (Cooley and Shortess, 1987). Wiring interconnections were representative of digital and analog signals, power, and grounds.



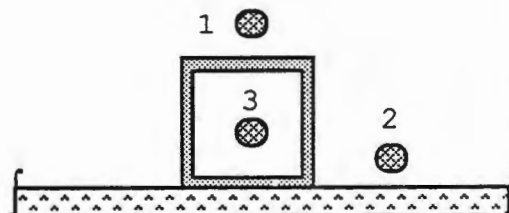
CONDUCTORS OVER A PLANE



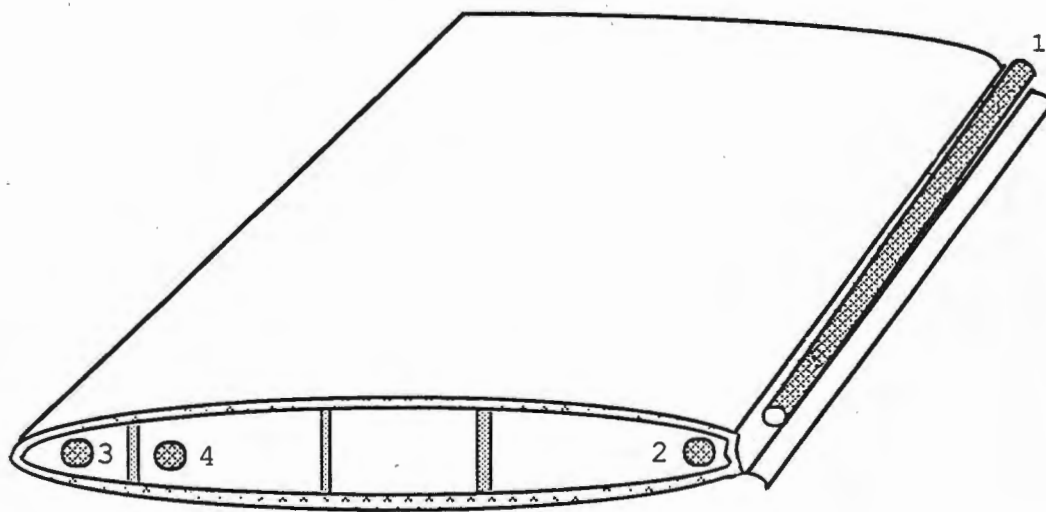
CONDUCTORS NEAR AN ANGLE



CONDUCTORS NEAR A CHANNEL



CONDUCTORS NEAR A BOX



CONDUCTIVE WING STRUCTURE

NOTE: The numbers 1 to 4 indicate worst to best cable locations.

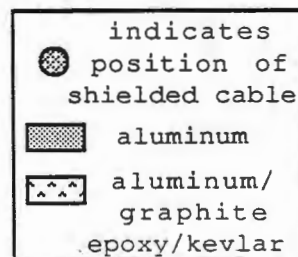


FIGURE 3.3-3. CABLE ROUTING NEAR CONDUCTIVE STRUCTURE

TABLE 3.3-1. ESTIMATED SHIELD CURRENTS DUE TO A DIRECT LIGHTNING STRIKE

Aluminum Skin			Graphite Epoxy Skin			Kevlar Skin		
Shield Location	Peak Current (kA)	Peak Time (μ s)	Shield Location	Peak Current (kA)	Peak Time (μ s)	Shield Location	Peak Current (kA)	Peak Time (μ s)
Shields over a plane								
1	4.76	6	1	110	50	1	200	6
2	3.67	6	2	110	50	2	200	6
3	2.48	6	3	110	50	3	200	6
Shields near an angle								
1	5.91	6	1	33.8	30	1	50.3	6
2	2.51	6	2	39.4	30	2	57	6
3	2.46	6	3	13.6	32	3	21.7	6
Shields near a channel								
1	3.05	6	1	46.7	20	1	55.4	6
2	1.00	6	2	20.6	30	2	30.5	6
3	.77	6	3	4.62	32	3	6.7	6
Shields near a box								
1	4.32	6	1	17.9	26	1	26.2	6
2	2.14	6	2	35.6	24	2	46.4	6
3	.04	104	3	.88	124	3	.88	108
Shields in a wing								
1	10.5	6	1	35.9	30	Not Calculated		
2	.38	16	2	35.9	32			
3	.02	112	3	3.4	28			
4	.46	12	5	28.4	20			

TABLE 3.3-2. COUPLING RESPONSES TO SUBSYSTEM WIRING DUE TO A DIRECT LIGHTNING STRIKE

System	Coaxial (RG-58)	Twisted Pair (no shield)	Twisted Pair (shielded)	Twisted Triple (no shield)	Single Wire
Nose-to-Tail Attachment					
Fuel	100-300 V				
Electrical	35 A				
Fuel			30-210 V		
Mechanical			10 A		
Engine	10-50 V		55 V		
Control	800 A				
Lights				40-1500 V	
Power				15-50 A	
Autopilot		50-1070 V	50-230 V 1800-7000 A		540 V
Autopilot shield disconnected		550-2400 V	250-1600 V		1300 V
Wing-to-Tail Attachment					
Fuel	800-2600 V				
Electrical	20 A				
Fuel			90-1200 V		
Mechanical			60 A		
Engine	< 10 V		150-400 V		
Control	140 A				
Lights				10-32,000 V	
Power				15-210 A	
Autopilot		1100-1070 V	250-2600 V 15,000-60,000 A		1850 V
Autopilot shield disconnected		1500-20,000 V	700-11,000 V		2500 V



4. INDIRECT EFFECTS PROTECTION - TEST METHODS

4.1. Review of Lightning Simulation Methods

This section presents a review of the lightning simulation technology used for evaluating the effects of lightning and static electrification on aircraft. The evolution of the simulation technology and considerations in selecting a simulation technique are discussed.

4.1.1. Lightning Simulation Technology

Several lightning simulation test techniques are currently being utilized for verification and validation of aircraft lightning protection. These different methods are the outgrowth of research using ground based simulations of lightning interactions with aircraft in flight. Under these test methods several aspects of natural lightning are simulated by pulse generators having various waveshapes and energy content as well as swept CW generators. The methods are often utilized, singly and in combinations, throughout an aircraft development cycle. These test methods, together with analysis procedures, characterize the indirect effects of lightning on aircraft electrical/electronic equipment.

The main reason for the different test methods is the difficulty of accurately generating natural lightning in the laboratory for large sized objects. It is very difficult (if not impossible) to accurately simulate severe lightning environments on an aircraft. This is because of the high levels of energy, current, and electric and magnetic fields associated with naturally occurring lightning. Simulation of these high energy levels requires very large and expensive facilities for an object the size of an airplane. Tests on aircraft and electrical/electronic equipment more commonly attempt to simulate portions of the lightning environment.

High level simulations of lightning appropriate for protection against direct effects, such as structural damage, are easier to implement because only small portions of the aircraft need to be exposed during a test. These simulations for direct effects of lightning on aircraft have been codified and are generally accepted as published in the report of the SAE Committee AE4L (1978).

Simulations for indirect effects, such as for the potential damage to flight control electronics from induced electrical transients, could require the use of the entire aircraft and several attachment locations for full evaluation. Thus, simulations for the indirect effects of lightning on aircraft and hardware have evolved along with the research efforts leading to a better understanding of these effects. It was recognized early on that electronic equipment would be affected by induced transients from lightning and other EM energy. However, the aircraft developments were allowed to go forward while answers were sought for protection against these effects. Research efforts were initiated in the

mid 1970's to develop the means of testing, assessing, and protecting the advanced electronics equipment. Since indirect effects testing necessarily involves the full vehicle and simulation of severe lightning levels on the full vehicle is impractical, approximate test methods were sought.

Several different investigators arrived at a variety of simulation techniques and test equipment because of different starting points and available resources. The data in table 4.1-1 summarizes the prior research and development tests on full scale aircraft. Major research and development efforts were concentrated in the AEHP program sponsored by the Air Force Flight Dynamics Laboratory (AFFDL) and supported by Tri-Services, FAA, NASA, and the Defense Nuclear Agency (DNA).

4.1.2. Considerations in Selecting Simulation Techniques

Many techniques and facilities are currently available for simulation of atmospheric electricity effects on aircraft. The selection of an appropriate method is challenging; it depends upon the ultimate use of the data and state of development of an aircraft.

Existing techniques for simulating static electrification, such as precipitation static, provide adequate data for aircraft design and protection. Nearby lightning strikes may be simulated using simulators developed for Electromagnetic Pulse (EMP) tests. The real challenge is to adequately simulate effects of lightning directly attached to an aircraft. A simulation technique that imposes all features of the lightning environment in a proper time sequence is desirable for full aircraft level tests. However, full vehicle tests are not time- and cost-effective for subsystem tests or to provide design data. It is especially important that the simulation technique provide data on the system, subsystem, component equipment, or Line Replaceable Unit (LRU). This data can be extrapolated to the values that occur when the aircraft is exposed to the real lightning environment.

Lightning simulation techniques must account for the significant features of natural lightning and the natural EM environments. These environments ultimately determine the indirect effects on advanced aircraft electronic equipment. Features necessary in the simulation setup include:

- The number of strokes in a lightning flash and the time between them.
- The peak current amplitude, rise time, and decay time reflecting parameters recently measured on natural lightning strikes.
- The proper EM environment geometric configuration around the test bed and the proper time phasing.
- The high voltage and electric fields associated with approaching leaders and static charging, preceding the high current and magnetic fields associated with the lightning return strokes.
- Lightning channel impedance and attachment.

TABLE 4.1-1. PRIOR RESEARCH AND DEVELOPMENT TESTS ON FULL-SCALE AIR VEHICLES

TEST	DESCRIPTION	RESULTS
AFFDL F-16 Mockup; G/E Composite 1983	Direct Attached - Coaxial - CW Tests - Moderate Threat 20 kA; 50 kA/ μ s	- V, I, B-Dot, D-Dot Data - Detailed Model - CW response within 4 dB of measurements at 28 kA
AFFDL ALCM 1983	Direct Attached - Coaxial - Full Threat 200 kA; 200 kA/ μ s - Moderate-Level 50 kA	- V, I Data - EM Model - Extrapolated values 7.7 dB greater than measurements at 200 kA
NASC F/A-18 1982	Direct Attached - Coaxial - High-Level 100 kA; 100 kA/ μ s - Low-level CW 760 A	- Functions - V, I Data - EM Model - No Upset/Damage - Predicted CW results 7.2 dB greater than measure- ments at 173 kA
NASC F-14 1982	Direct Attached - Coaxial - High-Level 180 kA; 180 kA/ μ s	- Functions - V, I Data - EM Model - Predictions
NASA F-106 Calibration 1982	Direct Attached - Shock-Excited - Radiated	- V, I, B-dot, D-dot Data - Spectral Analysis - Simple Model
RAE UK JAGUAR 1982	Direct Attached - Damped Sine Wave - Several Levels 100 kA; 20 kA/ μ s	- V, I, E, H Data - Spectral Analysis - External Currents/Fields
NSWC F-16 Mini-Test 1982	Direct Attached - Low-level 30 kA; 30 kA/ μ s	- Currents - Spectral Analysis

TABLE 4.1-1. PRIOR RESEARCH AND DEVELOPMENT TESTS ON FULL-SCALE AIR VEHICLES (continued)

TEST	DESCRIPTION	RESULTS
AFFDL C-130 1981	Direct Attached - Radiated - Low-level	- B-dot, D-dot Data - Simple Models
AFFDL F-111 1978	Direct Attached - Low-level; 2.5 kA	- Voltages - Simple Model
AFFDL YG-16 Composite; Fwd. Fuselage; 1978	Direct Attached - Moderate to High-Level 30-100 kA; 100 kA/ μ s	- Voltages and Fields - External EM Models
AFFDL A-7 1977	Direct Attached - Low-level 2 kA; 1.25 kA/ μ s	- Voltages - Simple Model
GENERAL DYNAMICS F-16 1975 & 1976	YF-16 #1,2 Lightning Tests 2-30 kA; 50 kA/ μ s	- Voltages - Simple Model
NASA F-8 1975	Direct Attached 5 kA; 5 kA/ μ s	- Voltages - Data Analysis
UH-60 Helicopter	Direct Attached - Low-level; 5 kA	- Unknown
757 Transport	Direct Attached - Low-level CW Test 250 A	- Spectral Analysis
ACAP Helicopter	Direct Attached - Low-level CW Test < 20 A - Moderate-level Pulse 24 kA; 18 kA/ μ s - High-level Pulse 200 kA; 146 kA/ μ s	- Spectral Analysis - Voltages, Currents, and Fields - EM Models - Extrapolated CW results 6 dB greater than meas- urements at 200 kA

- Voltage and current levels high enough to excite arcs and nonlinear effects, if any.

It is not possible to meet all these requirements with any one of the simulation techniques currently in use. The best present techniques only satisfy a few of the above requirements in any one setup.

The desired output from the lightning simulation technique determines which of these features are implemented and the degree of analysis required to demonstrate lightning protection. For example:

- Full vehicle, severe lightning protection demonstration tests with a minimum of analysis require simulation of the extreme values for the lightning parameters.
- Protection design demonstration tests can be performed with lower levels of pulse or with frequency domain measurements but require the support of analysis.
- Avionics and equipment susceptibility can be determined using bench tests on equipment outside the aircraft under test but require analysis or additional testing to relate to specific vulnerabilities of any aircraft.

The most widely applicable simulation methods are the pulse testing at various levels and rates of rise and the swept CW testing. The CW method differs from the pulse methods in the type of generator used and in the amount of analysis needed for interpretation of the data.

4.2. Lightning Pulse Generator Techniques

Many pulse techniques exist for generating simulations of lightning waveforms to investigate the induced effects of lightning on aircraft and avionic equipment. These techniques are based on discharging a high voltage source into the test aircraft, using switches and wave shaping elements to produce the appropriate current waveform. Most lightning simulators use variations of an RLC circuit to produce some of the relevant lightning characteristics. The circuit may be configured in either the underdamped, critically damped, or overdamped configuration. Four variants of the RLC circuit are in use today: the linear damped sine wave, the critically damped, the double exponential, and a nonlinear generator having sine wave rise and an exponential tail.

The underdamped RLC circuit generates the fast rise times and a moderate-level action integral desired with a practical circuit, but the waveform is oscillatory rather than unipolar as in natural lightning. J. Robb of Lightning and Transients Research Institute (LTRI) and B. Burrows of Culham Laboratory (England) have used damped sine wave pulsers for achieving high currents and fast rise-times (Riley et al., 1984).

The critically damped circuit gives a unipolar waveform, but the decay-time to rise-time ratio is smaller than that of natural lightning. However, the simplicity of constructing this type of pulser makes it attractive to generate a moderate-level pulse. With a low inductance aircraft and return circuit this

type of pulser can provide moderate rise-times and peak currents. The AFFDL has built moderate-level pulse generators using the standard RLC circuit arrangement. Significant improvements to the pulse generator switch and transient measurement system have been incorporated into this system (Walko, 1974).

An overdamped circuit requires a large amount of stored energy to produce the desired waveform. Therefore, this configuration is generally not used to provide both fast rise times and high current levels on a single pulse. B. Burrows utilized a pulse generator with an overdamped RLC circuit. This arrangement was combined with an approximately coaxial return conductor arrangement to produce moderate-level current pulses of 20 kA with a peak rate-of-rise 170 kA/ μ s (Butters, et al., 1982). Field distribution calculations were used to determine the placement for the return conductors.

The nonlinear "crow-bar" generator provides an efficient means for generating high peak currents with high energy in the decay. The sine wave efficiently transfers energy from the main capacitor bank into the test aircraft inductance. This energy is then short circuited by a spark gap and discharges with a Resistive/Inductive (RL) decay, producing the long decay tail on the test airplane's current pulse. Sandia National Laboratories has built a high-level lightning simulator for the Department of Energy (DOE) using a crowbar circuit design. This produces an essentially nonlinear circuit, because before the crowbar switch is closed it is an RLC circuit and after the switch is closed, an RL circuit.

An evaluation of four simulation methods for testing for indirect effects of lightning on full-size aircraft, having advanced technology structural materials and electronics equipment, is given in Cooley and Shortess (1987). The capabilities and limitations of these methods are summarized in table 4.2-1. A detailed description of each test method is given in the following sections.

4.2.1. Low-Level Swept Continuous Wave

This lightning simulation technique utilizes a CW rather than a pulse to excite the transients. Because the levels of testing are generally at low levels of current relative to the severe threat lightning current, there is more uncertainty. Consequently, more analysis is required to extrapolate to severe current levels.

Basically, this method utilizes a network analyzer to measure transfer functions (amplitude and phase) from a lightning attachment point to test points (voltage or current) within the aircraft. Figure 4.2-1 shows an example test setup.

The transfer functions, measured over a wide frequency range, are subsequently processed by multiplying by the severe lightning current spectrum to develop test point spectral responses. The test point responses are then numerically Fourier transformed to generate the pulse response expected from severe lightning. The method is generally applicable to other EM threats as well. Standard swept CW network analyzers provide coupling transfer function data as amplitude and phase versus frequency.

TABLE 4.2-1. SUMMARY OF SIMULATION TECHNIQUE CAPABILITIES

REQUIRED FEATURES FOR DIRECT STRIKE SIMULATION ON FULL VEHICLES	LOW LEVEL			HIGH LEVEL
	SWEPT CW BENCH TEST	FAST-RISE PULSE TEST	SHOCK EXCITATION	FAST-RISE PULSE TEST
1. High Voltage - E Field Phase Simulating a Leader Stroke	Linear Only	Limited By Pulse	Limited By Pulse	OK
2. High Current - H Field Simulating a Return Stroke Represented by Component A	Yes	Yes	Yes	OK
3. Continuing Currents Which Link Successive Return Strokes Represented by Component C	Yes	Yes	Yes	OK
4. Fast Rise Times (30 - 100 ns) To Match Recent Measurements	Yes	Yes	Yes	Limited By Pulse
5. Coaxial Transmission Line Test Setup of Test Object/Return Circuit				
• Simulation of In-Flight Conditions	Yes	Yes	?	Yes
• Simulation of Uniform Currents and Fields	Yes	Yes	?	Yes
6. Lightning Channel Attachment	Yes	Limited By Pulse	Limited By Pulse	Limited By Pulse
7. Test for Equipment Functional Responses				
• Upset	Yes	No	?	Yes
• Damage	Yes	No	?	Yes
8. Adapt Data to New Threat Parameters				
• EM Coupling	Yes	Limited By Pulse	Limited By Pulse	Limited By Pulse
• Functional Responses	Yes	Limited to Observed	Limited to Observed	Limited to Observed

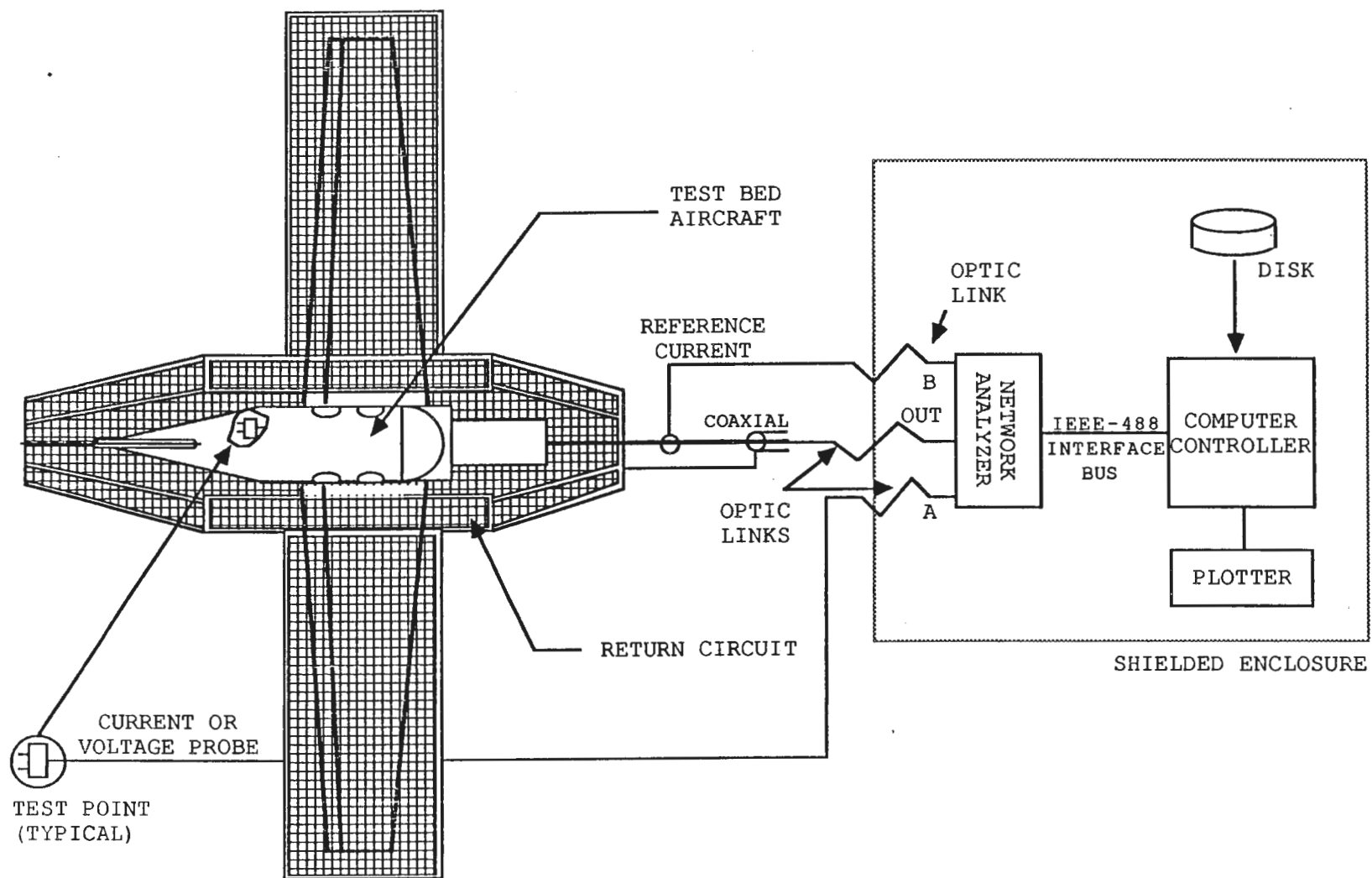


FIGURE 4.2-1. TYPICAL LOW-LEVEL SWEEPED CONTINUOUS WAVE TEST SETUP

Figure 4.2-2 provides an example of measured transfer function data, (a), and the pulse response, (d), obtained by Fourier transform processing of the CW measurement. This analysis requires a significant amount of numerical processing to extrapolate values to in-flight results. The intermediate data processing steps are illustrated in (b) and (c). Between 10 and 100 MHz in (a), the noise becomes more dominant than the signal. Here a judgement must be made as to the validity of the data. If the data are not valid, an estimate of the signal response must be made over the frequency range in question. The plot in (b) shows the expected signal with the noise removed. The 200 kA double exponential threat convolved with data in (b) is shown in (c). Fourier analysis of this spectrum yields the pulse response in (d).

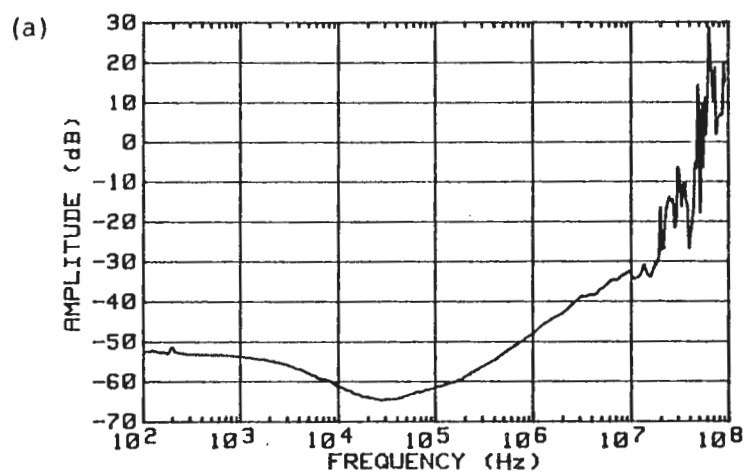
A principal advantage of the CW test method is the low level of injected current that can be applied (a few amperes) while attaining a high level of signal-to-noise ratios. Low levels of current can be used because of the wide frequency range contained in the lightning induced transient's interaction and coupling within the aircraft. Therefore, the energy delivered to the test object can be averaged over a relatively long time compared to pulse methods. This averaging provides an improved signal-to-noise ratio particularly when a narrow band receiver is used in the network analyzer.

Due to the transfer function complexity for most coupling responses, data are much easier to understand when obtained with CW measurements versus pulse techniques. The transfer function complexity contains clearly displayed and quantified resonances, superior signal-to-noise ratios, and lower required generator levels than the pulse methods. Under the CW method, an attached current of a few amperes can provide as high a signal-to-noise ratio as a severe lightning pulse. With these low levels of testing the CW measurement method offers several advantages for obtaining parametric sensitivity data, for example, the quick evaluation of a variety of mock-up protection options.

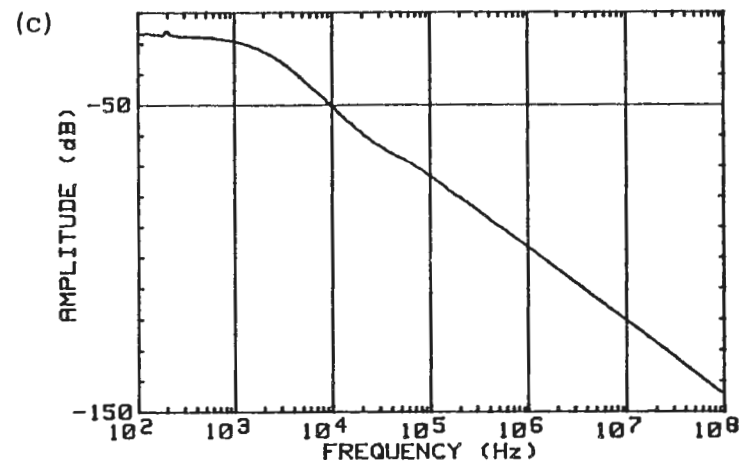
Nonlinearities, such as arcing and sparking at joints, are not represented with this test method because of the low currents injected. However, it has been demonstrated that extrapolated results from CW measurements are six to eight dB more conservative than higher level pulse measurements (Shortess, Cooley, and Melander, 1988 and Walen, 1988).

Swept CW measurement techniques for full vehicle lightning protection design demonstration are as follows:

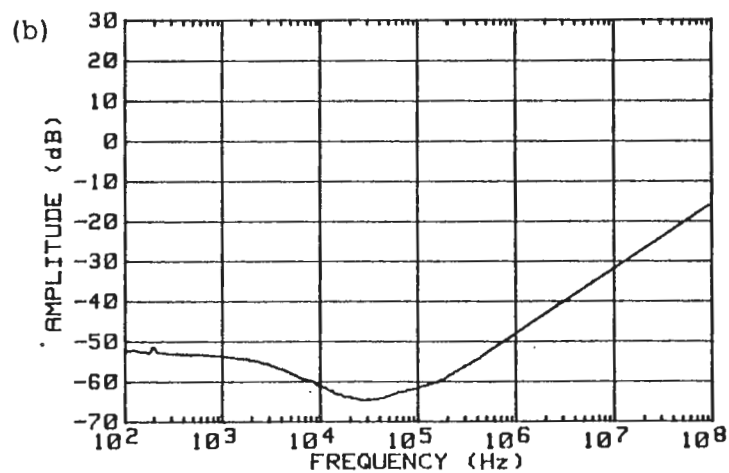
- Conduct time-delay-reflectometer (very low-level pulse) measurements to identify apertures and locations where EM coupling occurs.
- Conduct CW measurements to measure transfer functions from the external lightning current attachments to the EM coupling sources, and develop transfer functions from the sources to wiring and from the wiring to equipment.
- Combine the CW measured transfer functions with a model of the external lightning threat to obtain an overall computer model for EM response prediction.



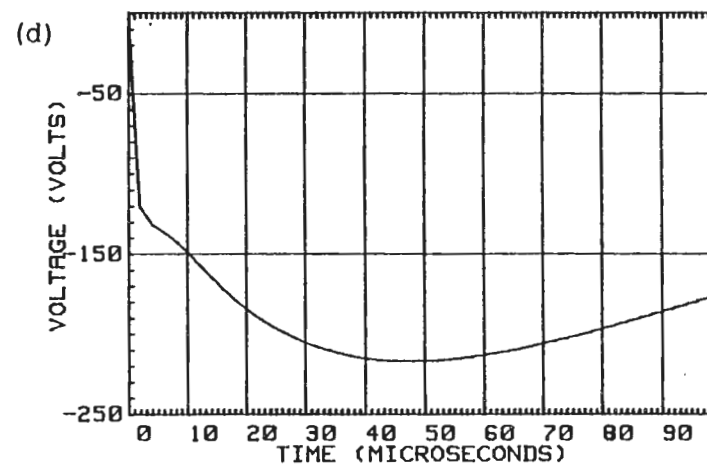
TEST POINT N3V1 - MEASURED DATA



N3V1 DATA CONVOLVED WITH THREAT WAVEFORM



TEST POINT N3V1 - SMOOTHED DATA



FOURIER TRANSFORMED PULSE RESPONSE

FIGURE 4.2-2. EXAMPLE TRANSFER FUNCTION ANALYSIS SEQUENCE

- Excite the computer models with the applicable severe lightning current spectrum and transform to time domain determining the full-threat level transients at critical circuits.
- Perform bench tests (induced cable current or pin voltage injection tests), using the predicted severe lightning pulses (possibly including timing from apertures and restrikes), at critical circuits on the aircraft.
- Determine protection design margins for the equipment. If the protection is inadequate, determine designs to reduce transients or reduce equipment susceptibility to the transients.

This method is also applicable to evaluating coupling coefficients for the high-voltage leader-attachment phase and nearby strike, as well as for the high-current phase by means of piece-wise linear simulation techniques. For these techniques, the generator, airplane, and return circuit are configured to represent the EM environment for each of the phases, one at a time. The coupling parameters included for the current and EM fields response of a given subsystem to the various aspects of the threat may be used to define the appropriate threat waveforms for a bench test. These time-ordered threats may then be imposed upon the component or subsystem in the proper sequence.

4.2.2. Low-Level Fast-Rise Pulse

Under this method, current pulses are generated by means of a pulse generator that is tightly coupled to the test aircraft (see figure 4.2-3). For this application, the test aircraft return circuit inductance and pulse generator capacitance combine to provide a critically damped circuit. This results in a current pulse having a high rate of rise with a relatively short decay time. The parameters are chosen so that the peak rate-of-rise matches the rate-of-rise for severe lightning currents while the peak amplitude matches the peak amplitude for low levels of lightning current. The generator decay time is much faster than natural lightning.

This simulation provides stress levels that match the severe rate-of-rise of natural lightning. This is appropriate when the principal EM coupling mechanism is mutual inductance or mutual capacitance, as is the case for balanced circuits isolated from structure. For composite structures where the coupling mechanism may be due to resistive drops, the simulated currents are not high enough to match the expected levels of moderate lightning strikes.

Since such a generator produces less than a moderate lightning stress on much of the equipment in an aircraft, the survival of equipment under the test condition only provides confidence in the capability to survive minor lightning strikes. The low-level of current injected will not be sufficient to cause non-linear responses, functional upset, or damage responses.

Careful analysis and interpretation of the test data is necessary to provide confidence in the capability to survive severe lightning strikes. This analysis must include interpretations of the coupling mechanism (I-dot or I) for each cable run and equipment area on the aircraft. The coupling mechanisms are necessary to extrapolate measured voltages and currents to higher severe

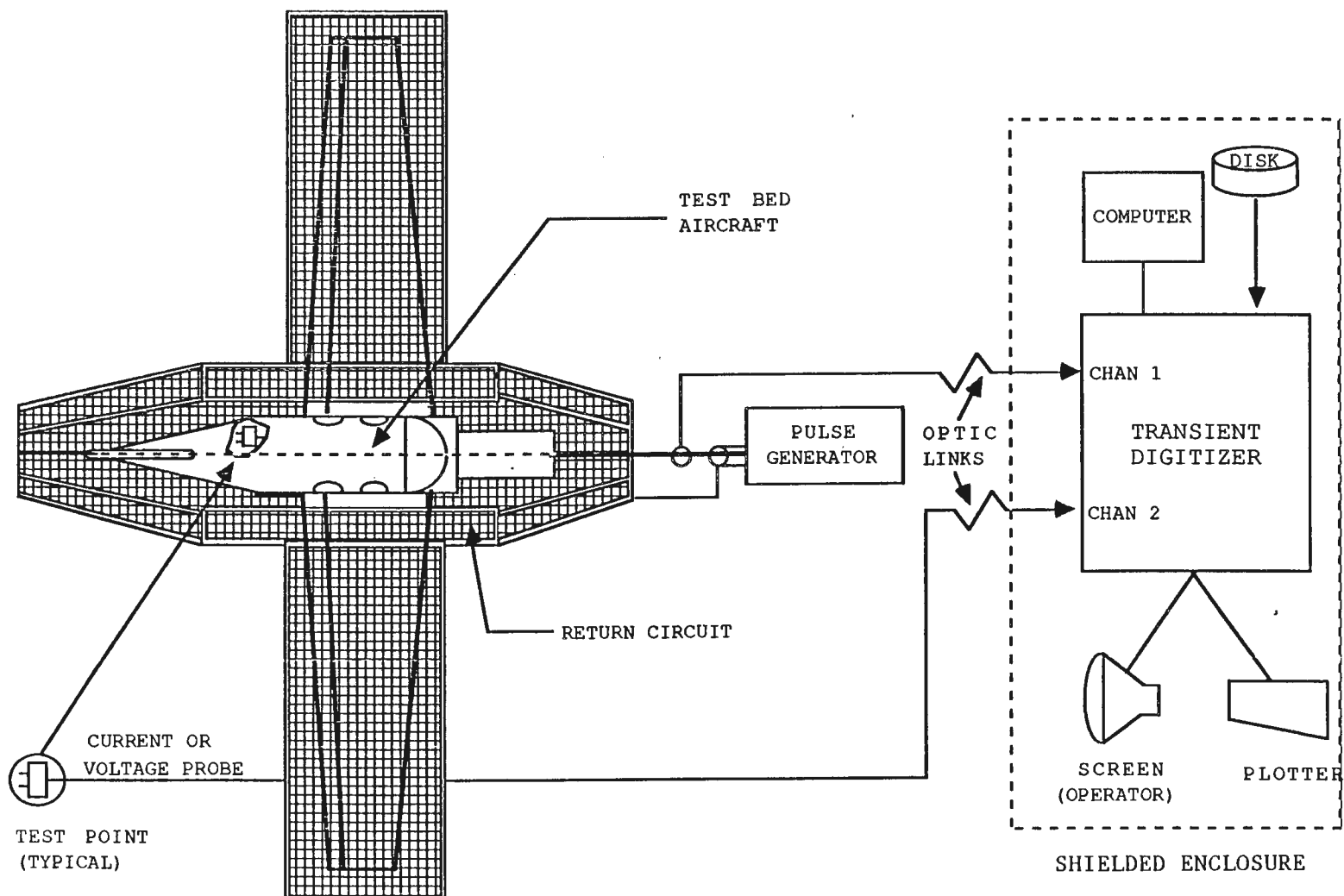


FIGURE 4.2-3. TYPICAL PULSE SIMULATION TEST SETUP

lightning responses. The I-dot driven responses scale according to the rate-of-rise parameters while the resistive driven responses scale according to I parameters. For many responses it will be found that they are driven by both I and I-dot coupling mechanisms. Since the generator circuit is tightly coupled to the aircraft under test, it may not be possible to separate the I and I-dot coupling from the data. This is because the internal coupling in the aircraft provides a different scale factor for I and I-dot at different test points, due to internal RLCM parameters, as opposed to the fixed I/I-dot parameters built into the generator. Therefore, the separation of the test results into purely I or I-dot will not be possible without considerable test efforts using different generator parameters.

If the generator waveform parameters match lightning parameters except for peak amplitude, then the response can be scaled by a multiplicative constant. For any other generator or choice of waveform parameters this will not be the case.

Based upon linearity arguments, Walko (1974) developed such a test based upon the use of a very low-level double exponential current pulse with a peak magnitude as low as 200 A. If rise and fall time constants match the values for lightning, then this current pulse theoretically contains all of the frequency components of a 200,000 A waveform in proper proportion, thereby allowing transients to be scaled linearly. The pulse is passed through the aircraft structure (from nose to tail, for example). The resulting transients produced on critical wire runs inside the aircraft are monitored by connecting voltage/current measurement instrumentation across the circuit to be monitored. The peak values of the measured transients are then extrapolated upward to the severe lightning current level. In the case of a 200 A, 6 x 70 microsecond test pulse, the peak levels are multiplied by 1000 to determine the anticipated transients for a 200,000 A strike with the same waveform. Since the test wave shape is the same as the threat waveform, the I-dot and the peak current, I, have the same scaling factor. This is illustrated for 200 kA and 200 A double exponential current pulses in figure 4.2-4.

Unless the generator rise and fall time constants match lightning parameters, it is probably better to use the basic measured responses as representative of a low to moderate lightning stress and not try to extrapolate the measured responses to higher levels.

The most accurate approach to extrapolation of the voltage and current measurements to higher levels of stress is through an analytical model based upon electrical equivalent circuits. This approach to modeling is similar to the analysis required for the CW method. The major difference is that the measured data for comparison with the models is derived from the pulse measurements.

The digital recording equipment used to acquire data from pulse testing typically has a dynamic range of only 40 dB. This is several orders of magnitude less than the dynamic range obtainable when measuring in the frequency domain.

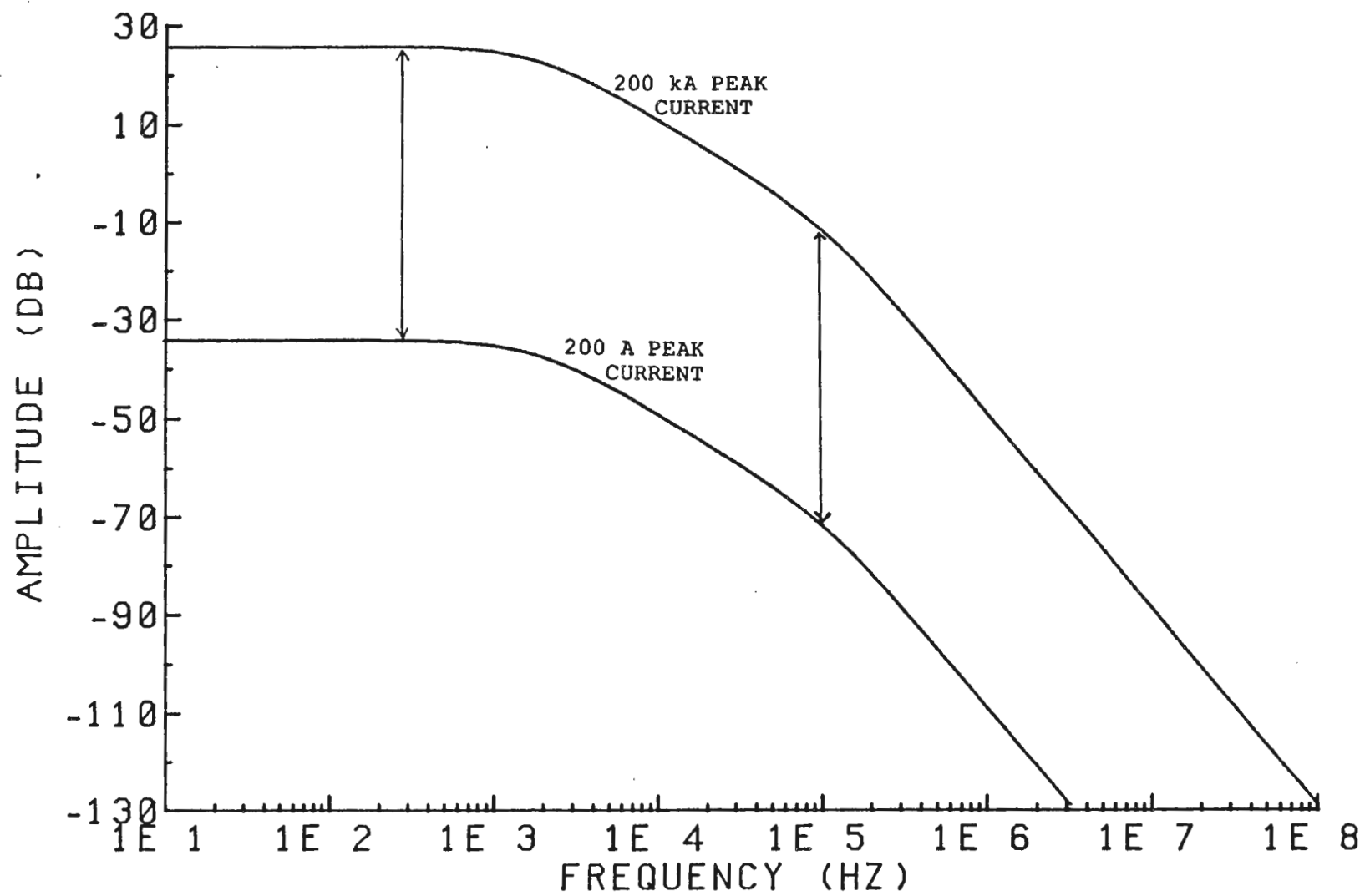


FIGURE 4.2-4. COMPARISON OF FREQUENCY SPECTRA

Low-level pulse measurement techniques for full vehicle lightning protection design demonstration are as follows:

- Conduct time-delay-reflectometer (very low-level pulse) measurements to identify apertures and locations where EM coupling occurs.
- Conduct CW measurements to measure transfer functions from the external lightning current attachments to the EM coupling sources. Develop transfer functions from the sources to wiring and from the wiring to equipment.
- Combine the CW measured transfer functions with a model of the low-level pulse generator to obtain an overall computer model for EM response prediction of the test responses.
- Compare the measured pulses with the model predictions to provide confidence in the computer models. Determine the error bars in the model responses from the measurement comparisons.
- Excite the computer models with the applicable severe lightning current spectrum and transform to time domain determining the full-threat level transients at critical circuits.
- Perform bench tests (induced cable current or pin voltage injection tests), using the predicted severe lightning pulses (possibly including timing from apertures and restrikes), at critical circuits on the aircraft.
- Determine protection design margins for the equipment. If the protection is inadequate, determine designs to reduce transients or reduce equipment susceptibility to the transients.

It should be noted that second and third bullet items utilize results from CW testing. Bullet items utilizing the CW method provides a higher degree of confidence in the modeling of the various coupling mechanisms. The pulse data could also be used to refine the models but there would be lower confidence in the extrapolation using a model that had been refined with the pulse test alone. This is due to the difficulty of separating the various RLCM parameters from the time domain test responses.

4.2.3. High-Level Fast-Rise Pulse

Under this method, current pulses are generated by means of a large pulse generator that is tightly coupled to the test aircraft and return circuit. There is considerably more energy required in the main pulse generator for this method than the low-level fast-rise test method. In the usual application, the test aircraft and return circuit resonate with the pulse generator capacitance to provide high rate-of-rise current pulses having a damped sinusoidal waveform. The generator and return circuit parameters are chosen so that the pulse rise time matches the rise time for severe lightning currents. The peak amplitude is near enough to the severe lightning current levels that the data extrapolations can be made with high confidence. The generator decay time can be adjusted to the same as natural lightning or much faster than natural lightning.

This simulation provides levels at and above the moderate lightning strike current amplitude. This test method is appropriate when the amount of analysis is to be minimized. The analysis provides high confidence in the survival of the aircraft and equipment under severe lightning strikes.

The high-level pulse simulation technique provides a level of stress that is somewhat greater than the expected value for lightning strikes. This test stress increases the confidence in the protection design over that of a low level test technique used alone. For example, portions of the protection system, such as cable shields and grounding schemes, could function at levels associated with low-level testing but be damaged at higher levels. The test conditions provide confidence in the survival of the aircraft against all but severe lightning strikes.

The principal advantage of this test method is that the amount of scaling or extrapolation is not very great. Careful interpretation of the test data is necessary in order to provide confidence in an aircraft survival of severe lightning strikes. However, there is greater confidence in scaling factors of five to ten (from 20 or 40 kA to 200 kA) than factors of 200 required for the low-level pulse method. The data analysis must of course include interpretations of the measurements to validate the measurement process. The data analysis effort is much simpler and more readily accepted than the analysis required for lower level test methods.

4.2.4. Severe Full-Threat Pulse

Unfortunately, it is difficult or impossible to generate the severe lightning currents in large aircraft as can be demonstrated by a relatively simple analysis. The primary reason for this limitation is due to the inductance and resistance of the return circuit and the airplane. For an ideal conductor and the return circuit in a coaxial configuration, the test circuit surge impedance is typically 50 ohms. For 200 kA, this requires 1 MV to drive the test aircraft and 1.3 MV for the charge voltage of the pulser. These values are achievable but require specialized high voltage construction practices. For a large aircraft, the circuit impedance is more inductive. The inductance is approximately given by:

$$L = 120 \ln(h/2d) \ h/c$$

where h is the aircraft length, d is the aircraft diameter, c is the speed of light, and the units of the constant are henrys per second. For a 50-meter aircraft 3 meters in diameter, this results in an inductance of approximately 40 μ henrys. It requires 8 MV to drive 200 kA/ μ s into this inductance. Such a generator is not presently achievable. However, for a small general aviation size aircraft the inductance can be on the order of 3 to 4 μ henrys and only 600 to 800 kV is required to reach the 200 kA/ μ s rate-of-rise. While this is a large order for most general aviation manufacturers, such generators are well within the reach of several lightning test organizations.

4.2.5. Shock-Excited Pulse

The pulse techniques described above focus on directly attached current waveform simulation and associated EM fields. There is some concern that the high voltage and electric fields induced on an aircraft, just before a strike and during the aircraft charging, should also be considered in the simulation. The electric field mode can be simulated by using a spark gap between the aircraft and the pulse generator and between the generator and the ground. These spark gaps are to be placed at a structural extremity away from the pulser attachment. When the pulser is fired, the aircraft will be charged, through the pulser gap, toward the pulser voltage. The charging will be terminated when the aircraft-to-ground spark gap discharges to ground. This technique produces high voltage and electric field gradients on the aircraft to simulate the natural lightning effects.

This test method simulates the presence of corona, the leader attachment during the charging phase, and a return stroke as the aircraft discharges. The coupling mechanisms are more easily separated for analysis because the charging and discharging events are separate in time.

The major setup difference from prior techniques is that, in the shock-excitation test, the aircraft functions as the peaking capacitor. If the aircraft is isolated from ground, an additional peaking capacitor may be unnecessary. The discharge of the aircraft, when the output gap arcs over, should excite the internal circuits in much the same way as the discharge of an aircraft in a preionized lightning channel would when it is discharged by the return stroke.

Shock-excitation tests are unique because of the high-voltage aspects of the technique. The aircraft must be well isolated above ground. Specially designed high-voltage isolation pads placed under the wheels allow testing up to 400 kV on most aircraft.

A shielded high-voltage impulse generator is used as the excitation source. This shielding is required so that a clean output voltage waveform may be applied to the vehicle. To simulate the natural lightning charging and discharging of the aircraft, it is desirable to charge the aircraft to over one million volts with respect to the coaxially arranged return lines. At these voltages, realistic corona should be developed so that, when the aircraft discharges to ground, the transients produced will be representative of natural lightning. Essential elements of the test are fiber-optic data links which allow system transients to be monitored while the system is charged to very high voltages.

An important result of the shock excited test investigation (Clifford, Crouch, and Schulte, 1982) was that capacitive coupling ($C \, dV/dt$) was found to be the dominant coupling mechanism in some very important cases. In particular, the transients on high-impedance signal circuits used for single-wire computer logic circuit interconnections were found to be dominated by capacitive coupling. The situation is reversed for low-impedance circuits where inductive ($L \, dI/dt$) coupling dominates.

For these tests, the maximum values of the lightning parameters of interest (I , dI/dt , E , and dE/dt) are determined by the amount of charge stored on the aircraft before discharge occurs.

The major implication of these results for simulation testing is that sufficiently high voltages and rates of change of voltage must be present in the test to reproduce the natural lightning conditions. Since nonlinear corona and streamering effects are expected to play a role in the induced voltages and currents experienced by an aircraft struck by lightning, testing with low voltages (grounded vehicles) may not yield an accurate simulation.

The amplitude and duration of the oscillatory currents produced on the structure by the rapid discharge of the aircraft to ground (free response) are controlled by the dissipation factors in the test setup. These factors include arc impedance and corona streamering effects. When the test article is grounded, neither of these dissipation factors is present. It still remains to determine exactly what the correct values for the dissipation factors are, but additional laboratory studies, coupled with forthcoming flight data, should resolve those uncertainties.

4.2.6. Comparison of Test Methods

In a comparison of lightning simulation techniques on a composite test article (Cooley and Shortess, 1987), it was found that mean values of predicted currents due to severe lightning strikes for different techniques compared as follows:

- low-level swept CW +6 to +8 dB
- low-level pulse -1 to -5 dB
- moderate-level pulse baseline
- shock-excitation +2 to +6 dB

The moderate-level pulse results were used as a baseline. These data were considered more accurate than data measured using other techniques for the following reasons:

- The scaling factor from 7 kA to 200 kA had less uncertainty than for low levels of injected current.
- Non-linear effects were noted at several test points. Non-linear effects were not in evidence for the low-level tests.
- The spectral content was similar to the SAE-AE4L defined threat, Component A. This allowed a minimal amount of analysis because a determination of resistive or dI/dt response was not necessary.

The CW predictions were larger (by a factor of two) and therefore more conservative than the predictions resulting from the moderate-level pulse tests. The

low-level pulse results predicted responses about 1.5 times less than the moderate-level pulse. Shock-excitation was also a factor of two more conservative than the moderate-level pulse results.

4.2.7. Return Circuit Configuration

The return circuit for the lightning simulator and test article determines the field distribution around the object under test. More uniform field, voltage, and current distributions are produced by configuring the test article and return circuit as a coaxial transmission line, see figure 4.2-5. The electric and magnetic fields around the test object approximate field distributions for an aircraft in free space. An additional benefit is that this arrangement provides low circuit inductance and resistance which is required by some pulsers to produce a satisfactory waveform. This is particularly true for the RLC pulse generators.

Some special considerations must be given for the construction of a return circuit. For high-level testing, the return circuit must sustain high currents and withstand high voltages. The return circuit must be made of a low resistance material, such as wire mesh or solid sheet, and maintain good physical contact to be suitable for low-level testing. The attachment points must be low inductance, have a high current carrying capability, and shield the test object from the fields of the pulser switch. The configuration should be built to accommodate all test configurations, e.g., nose-to-tail and wing-to-tail current injection. The construction should be such that the spacing between the return circuit and the test object maintains a constant impedance transmission line and any changes in return circuit geometry should be gradual compared to a wavelength of the highest frequency of interest. This is essential to prevent reflections due to impedance mismatches.

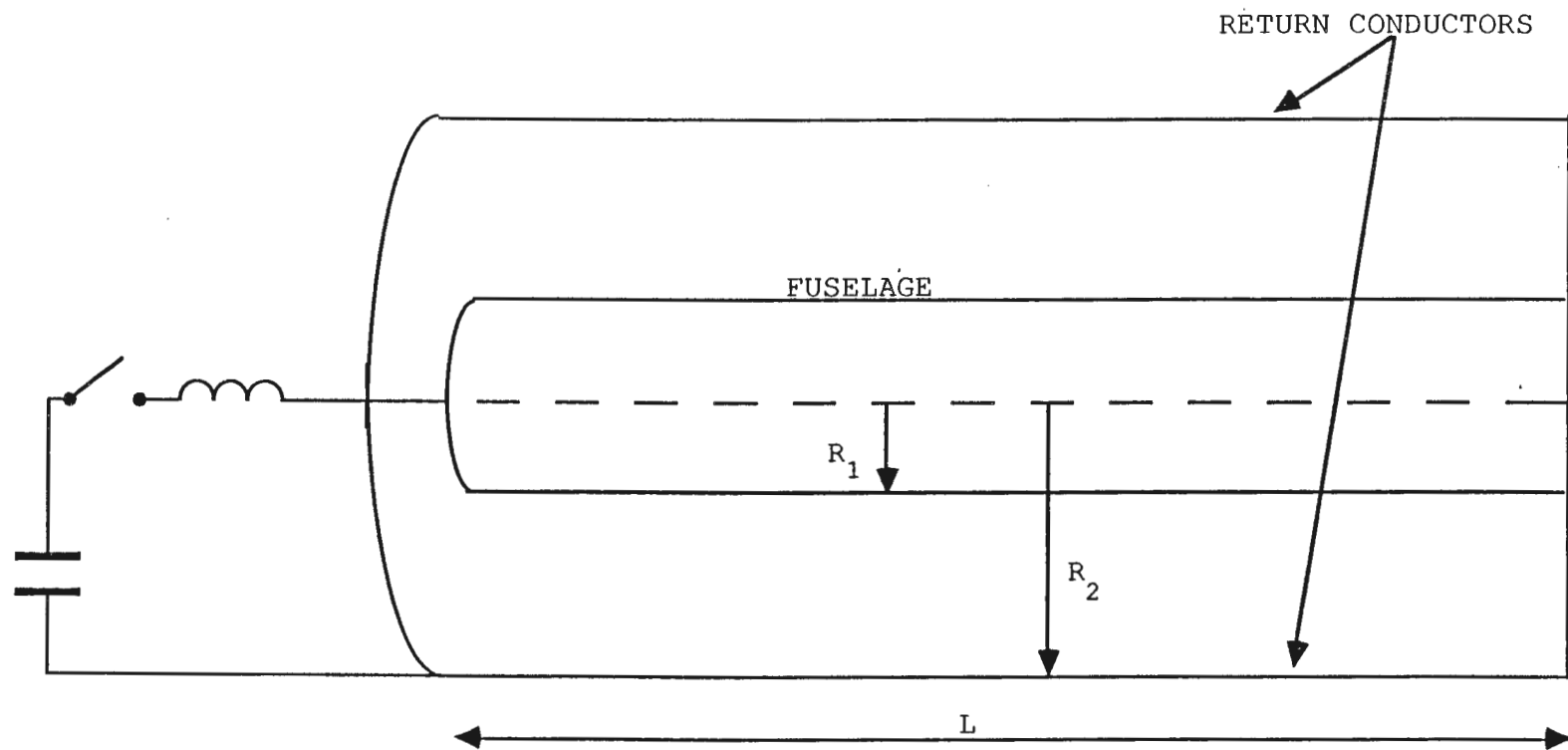
4.3. Data Quality Verification

Several recommendations are given in this section and can be used to assure that measured data is of high quality and is valid. These recommendations can be utilized for both CW and pulse tests.

Fiber optics should be used in both drive and signal transmission, if possible. The optics electrically isolate equipment to prevent ground loop currents. The data acquisition equipment should be placed in a screen room to provide shielding from pulser fields and to eliminate noise pickup. Pulser noise can be a major concern at high levels of current.

System calibrations must be performed daily to verify the correct operation of the CW measurement equipment and to obtain inherent system responses for proper data calibration. The measurements should be verified twice daily or at any time the system configuration changes. The data obtained from system calibrations are then subtracted from measurement data to remove inherent system responses. Calibration of some fiber optic units is necessary to define the amplitude of a known signal into a digitizer.

13-62



SIMPLIFIED AIRCRAFT GEOMETRY - THE AIRCRAFT FUSELAGE IS SHOWN AS A CYLINDER R_1 METERS IN RADIUS, L METERS LONG WITH RETURN CONDUCTORS SPACED AT AN EQUIVALENT RADIUS OF R_2 METERS.

FIGURE 4.2-5. SIMPLIFIED RETURN CIRCUIT

Noise measurements should be taken at each measurement location in each test configuration. The sensors are either a coaxial cable terminated in 50 ohms or a current probe with nothing through its center near test wiring as input to the fiber optics. These measurements can then be compared to the corresponding test point measurement to determine the signal-to-noise ratio. This is much easier to do in the frequency domain; however, a reasonable judgement can be made for a time domain measurement. A signal level 10 dB above the noise level is considered reliable data. Examples are shown in figures 4.3-1 and 4.3-2 of signal and noise measurements at the same location for CW and pulse tests.

End-to-end resistance measurements of the test setup should be made before and after each test configuration. This will be a measure of how well structural connectivity (joints, etc.) is maintained after the injection of current. An ohmmeter capable of accurately measuring a milliohm is necessary equipment for this check.

The input current should be monitored for every measurement. This current is used as a reference measurement during CW testing. During pulse testing, the input current must be checked to ensure proper pulser operation. The pulse amplitude will vary with each shot and should be recorded in the case of large variations. Data must be taken not only to characterize the simulator performance, but also to characterize the interaction and coupling of the simulated lightning environment with the test aircraft and the coupling effects on electronic/electrical systems.

Questionable current measurements can be checked in the following manner. With the current probe reversed, repeat the measurement and check the signal polarity. If the signal does not change polarity, then noise is being measured.

Validity checks should be performed by the experimenter during each test. These checks consist of comparing measured responses against theoretical models for their behavior to make sure the measured response curves are "reasonable." Any failure to produce "reasonable" data should be considered a failure in the instrument setup and/or test connections, and steps should be taken to correct the problem.

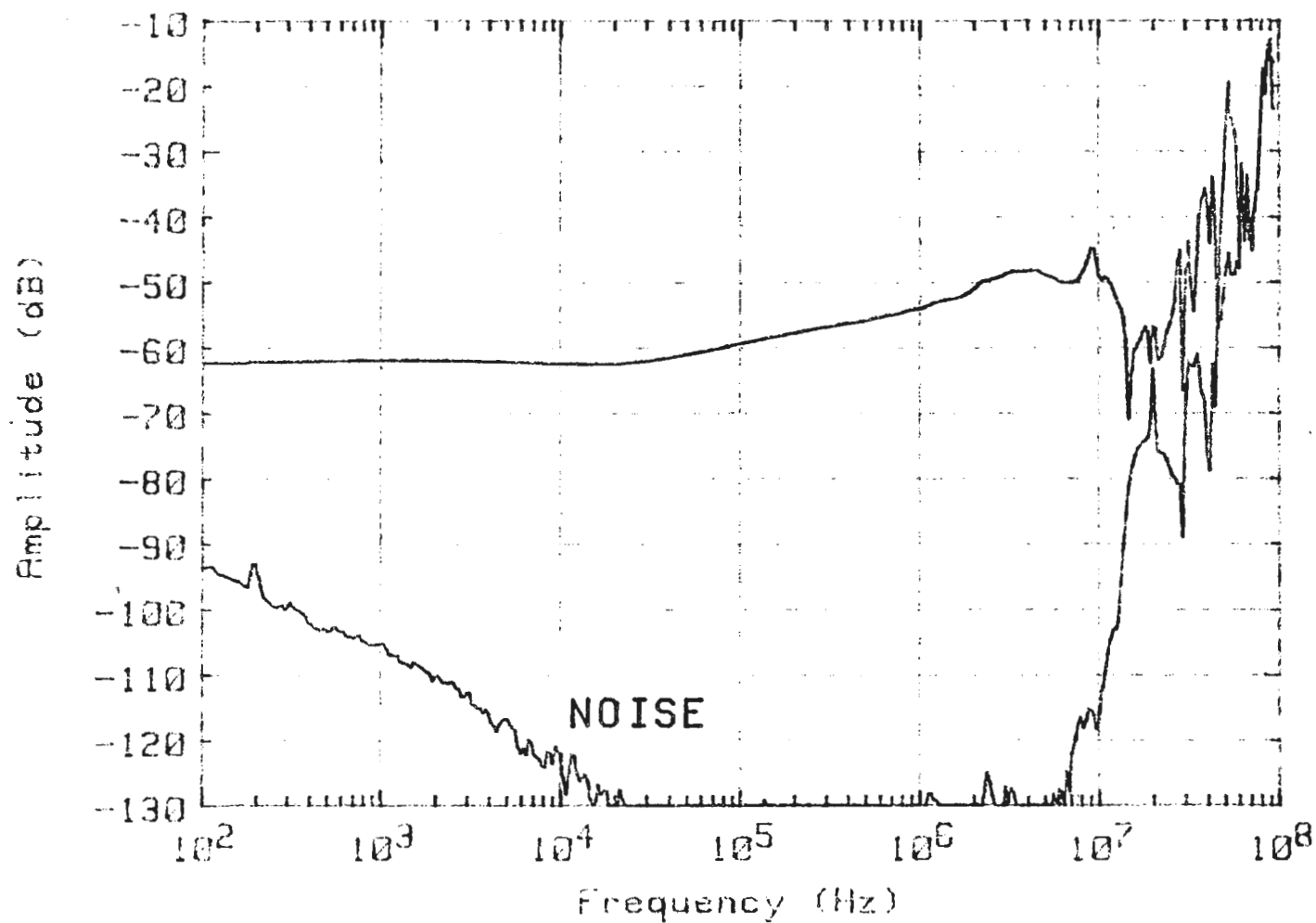
When using digital recorders, a vertical resolution should be set to obtain the measured response with the peak at least one-third of full scale to achieve adequate resolution of the signal response and to utilize the maximum bandwidth capabilities of the digitizer.

4.4. Bench Tests

Bench tests are used to test the susceptibility of individual components or subsystems to predicted levels. These predictions are based on measured data or analytic models for currents and fields at the location where the equipment would be installed.

Voltage and current transients induced in interconnecting wiring may be resistive, following the double exponential lightning source; a derivative, dI/dt , of the excitation source; or oscillatory, due to excitation of cable resonances.

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TEST ID. No.: 3V2W

(8-12-85): COMPARISON OF SIGNAL TO NOISE AT THE SAME LOCATION

FIGURE 4.3-1. COMPARISON OF SIGNAL AND NOISE IN THE FREQUENCY DOMAIN

13-65

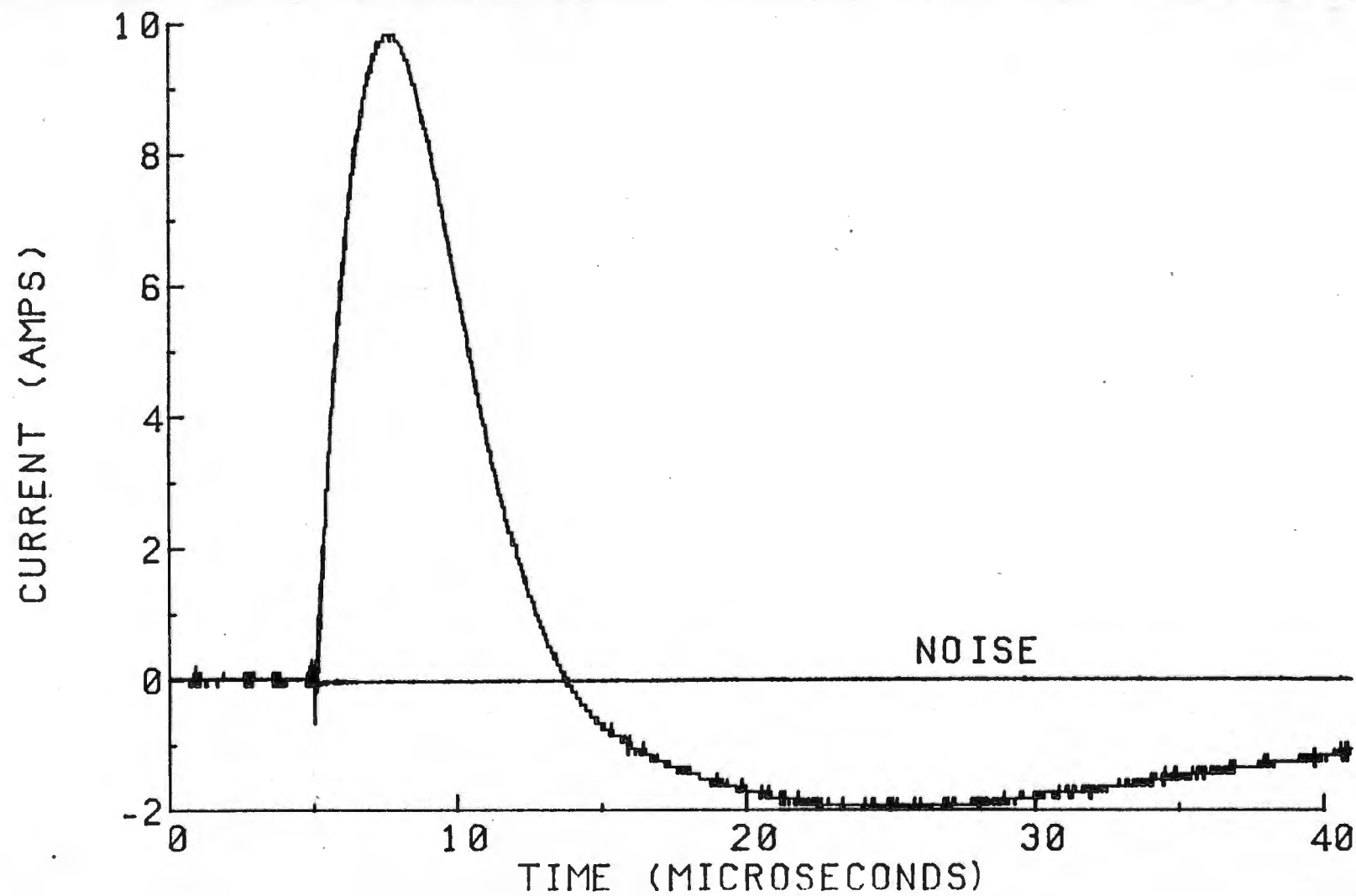


FIGURE 4.3-2. COMPARISON OF SIGNAL AND NOISE IN THE TIME DOMAIN

Waveforms, which define these responses, are shown in figure 4.4-1 (AC 20-53). These waveforms also represent the threat imposed on equipment at the cable interfaces.

Currents are injected into interconnecting cables by magnetic coupling using a clamp-on current transformer. Alternatively, electric and magnetic fields are injected into cables and enclosures by a parallel plate transmission line excited by a high current pulser. Power system transients are induced by direct series connection of a coupling capacitor and pulse generator across the power supply. A typical bench test setup for current injection is shown in figure 4.4-2.

Equipment is connected via appropriate cabling to simulate actual installed configurations. During the pulsing, the equipment is powered up and in an operational mode to simulate in-flight conditions. Equipment is monitored for damage or upset throughout the test sequence.

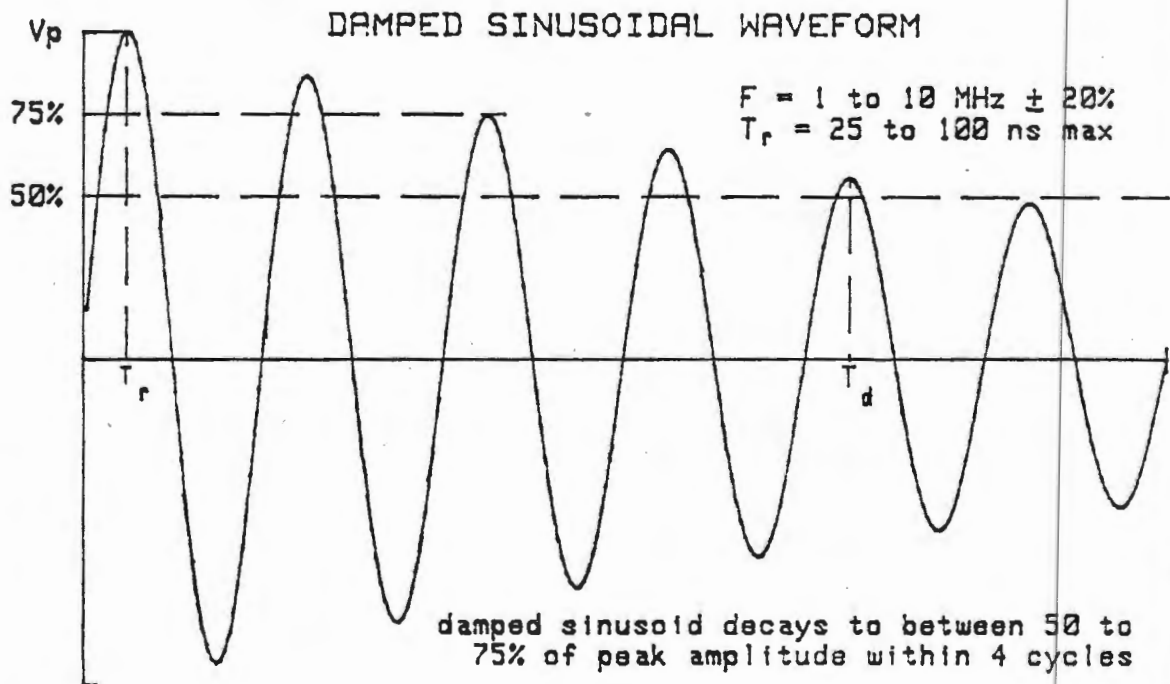
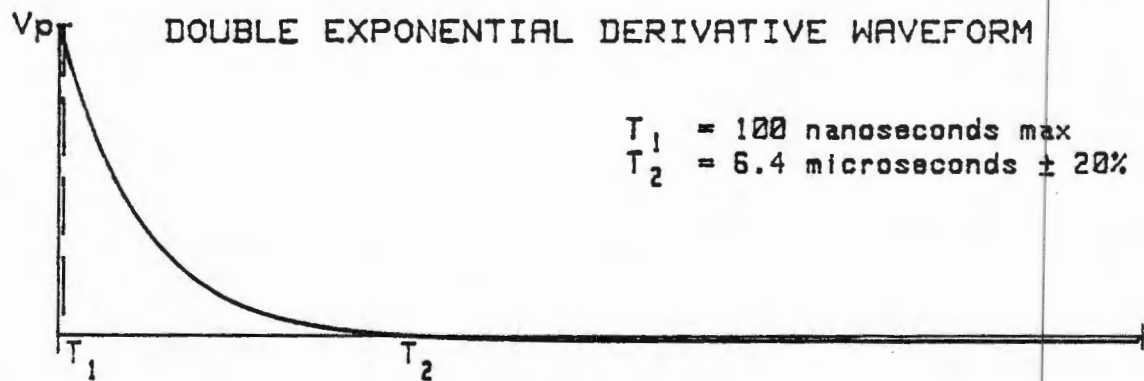
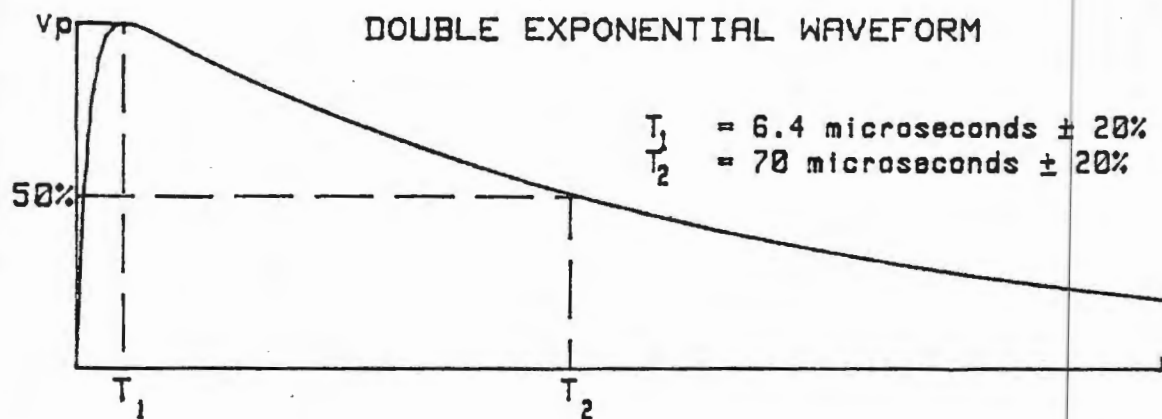


FIGURE 4.4-1. IDEAL BENCH TEST WAVEFORMS

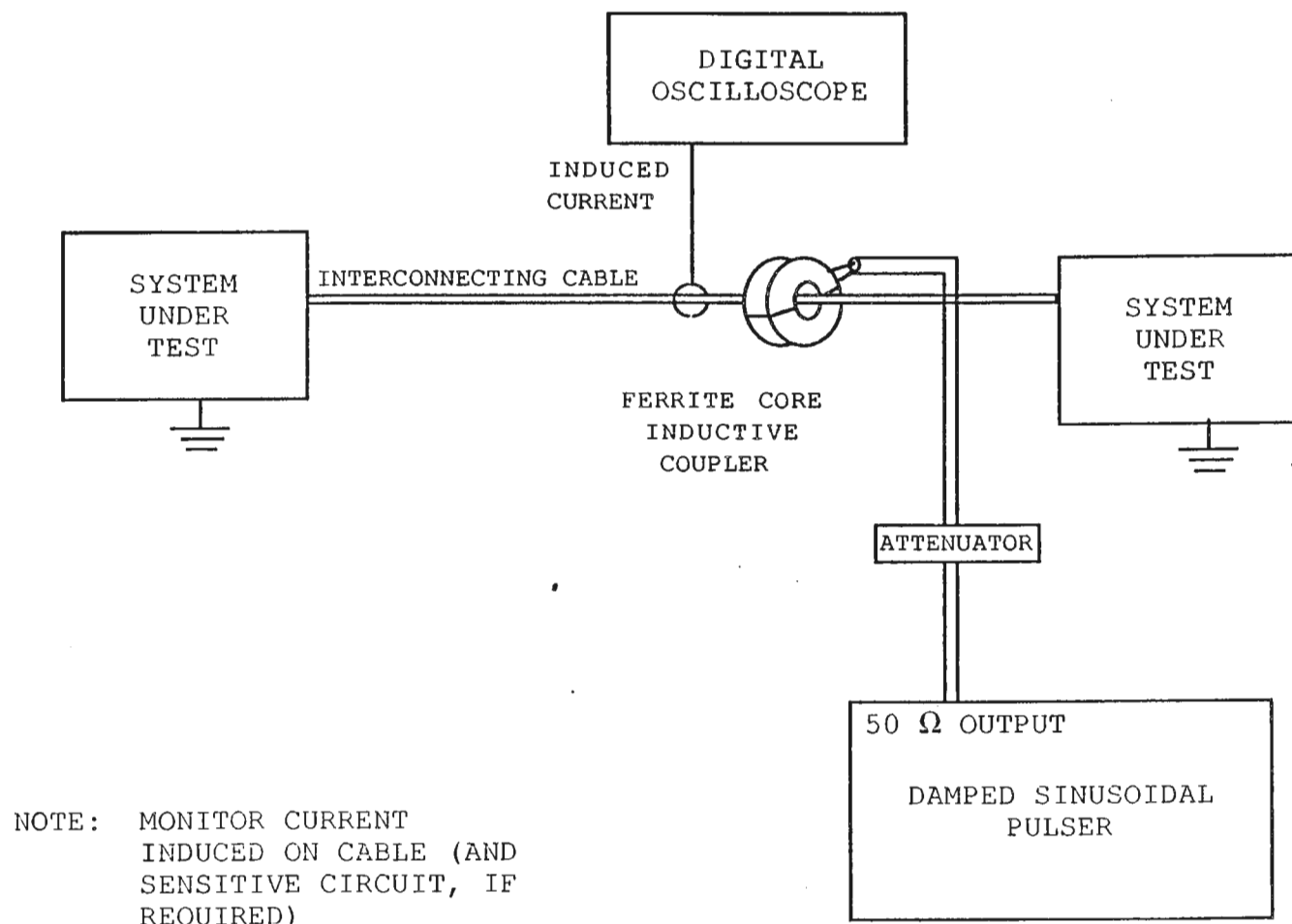


FIGURE 4.4-2. CABLE CURRENT INJECTION TEST CONFIGURATION

5. INDIRECT EFFECTS PROTECTION - ANALYSIS TECHNIQUES

Accurately predicting the coupling of lightning to equipment within a structure such as an aircraft is a complex process. Figure 5-1 illustrates the principal elements of EM coupling prediction for lightning indirect effects. These elements follow the flow of energy from external environments to internal equipment.

The external coupling region contains resistive and inductive terms. The early time, inductive term appears as magnetic fields around the aircraft. These fields will build up and change rapidly in accordance with the lightning current pulses. The late time, resistive term appears as a voltage along the structure. While both terms are always present, the relative size of each depends upon the lightning pulse shape.

The internal fields present depend on several factors: penetrations through apertures, antennas, external wiring, and joints; diffusion through the skin; shape factors; and intrinsic shielding provided by the structure materials. External EM fields will leak into the interior of the aircraft through openings in the structure such as windows, radomes, access panels, and doors. Electrical imperfections such as joints, gaps, and holes also allow the entry of some EM fields. In addition to the EM field coupling, there may be resistive voltage drops in the structure as lightning currents flow. Currents may also flow inside the structure as a result of structural interconnections which can affect the internal EM environment. Leakage through apertures contributes to fast rise voltages. Effects due to diffusion through the skin will have a slow rise time compared to lightning pulses. Diffusion effects become important at frequencies where the structure thickness is equivalent to or greater than an electrical skin depth of the material.

Sources of internal currents in the wiring include B-dot coupling, E-dot coupling, and resistive voltage drop (IR) coupling. The magnitude of these sources is dependent on local structure, currents, and fields coupled to the interior.

Voltages are coupled into aircraft wiring in several ways. If part of a circuit connecting electronic equipment connects to structure, there will be a voltage difference between the wires and the structure. Voltages are also coupled between wires by electric and magnetic induction, even if the wires are not connected to the structure. The effects of induction depend upon the time rates of change of the lightning currents and EM fields. Since the total structural voltage drop depends upon both the inductive and resistive terms, voltages in the wiring depend upon the lightning current time rate of change as well as the peak current values.

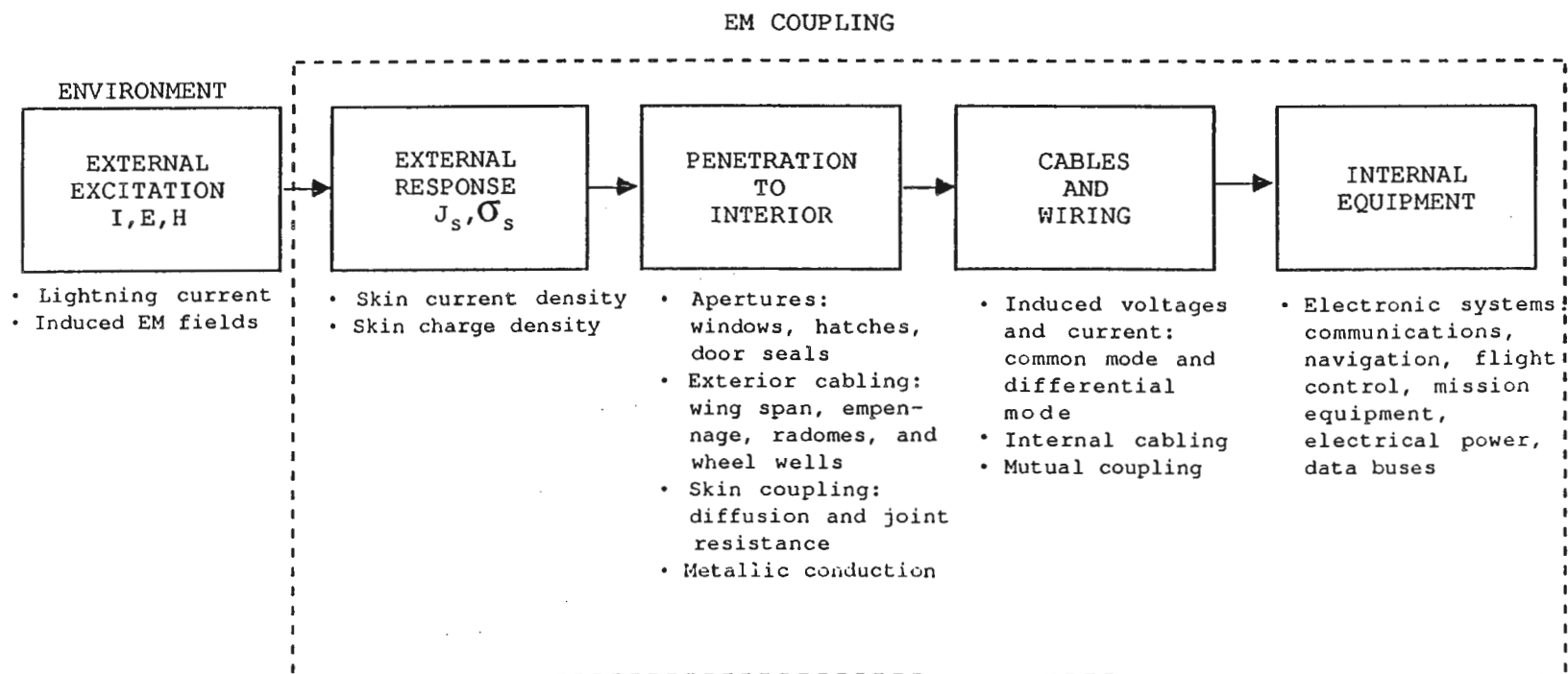


FIGURE 5-1. ELEMENTS OF EM MODELING AND ANALYSIS FOR LIGHTNING INDIRECT EFFECTS

For an aircraft made of aluminum, the coupled voltages are rarely important except when the lightning current flows through joints and hinges. However, the resistance of an advanced structural material such as G/E is many times that of aluminum. Voltages of a few tenths of a volt have no effect in an aluminum structure. Voltages that are larger by a factor of several hundred to a thousand times than those for aluminum, because of the lower conductivity of the composite materials can become very serious.

Coupling to electronic equipment is dependent on the termination and source impedances, the circuit configuration, and the wire bundles which connect to each box.

5.1. Modeling Methods for Composite Structures

In advanced technology aircraft, EM coupling models for lightning protection design must account for the composite structural materials and the fastener joining techniques. Many of the present models (such as the environment, external response, cable and wiring, and internal equipment) used for metal aircraft are useful for composites. The penetration to interior models require changes for particular aircraft and wiring configurations.

The modeling of energy penetration from the exterior to the interior has several common elements for any aircraft particularly in the apertures and exterior wiring and cables. The principal difference between metal and composite aircraft lies in the increased contribution of the IR in the fuselage and wing skins and in the increased current flow paths in the structure. The G/E structure has typically 1000-3000 times higher resistance than the similar aluminum metal structure. This higher resistance leads to strong interactions between the external and internal EM regions for composite structures.

For frequencies above a few megahertz, or for early time (i.e., on the order of a few microseconds), there is no difference in the lightning response between aluminum and graphite structures. This is because at high frequencies, the EM "skin effect" forces the currents to the outside of materials, to outside surfaces, and to sharp radius of curvature portions of the structure. Protection against the early pulse is essentially the same for both structural materials. Wiring must be kept near metallic structure to reduce the magnetic field B-dot coupling factors.

The main difference in energy penetration between graphite and aluminum structures is in the frequencies below the megahertz range, or for late time (i.e., on the order of a few tens of microseconds). For this range of frequencies, considerable current flows into the interior as a result of the range of thicknesses and materials used in aircraft structure. Since the current flow is resistive for G/E materials, the protection measure of keeping the wiring close to the structure will not have any protective effect. Furthermore, the IR voltage drop will be much larger for G/E than for aluminum.

5.2. Computer Modeling Codes

Analytic models are useful tools in determining the effectiveness of various protection designs. Analytic models can be used in parallel with testing to validate test results or to assess various protection methods for use on particular equipment.

There is no single analytic method which can provide accurate results at all levels of EM coupling, from lightning interaction with an aircraft to transients in the wiring. A multi-level analysis, as shown in figure 5.2-1, can be used to provide answers to the overall coupling problem. Using this approach, the exterior currents on the airframe due to a lightning strike are calculated first. Next, the interior fields and shield currents are calculated. These are determined by penetrations due to joints, apertures, and diffusion. The final level of analysis is to compute the transients at the component level. The input currents to each level are those calculated from the preceding level.

This type of analysis assumes that the coupling between levels is negligible. This implies that metal structure or wiring inside the aircraft does not significantly affect the exterior skin currents. This assumption is not valid for an aircraft consisting of mainly G/E skin. In this case, internal metal structure will carry the bulk of the current at low frequencies. Internal fields will be dominated by these currents rather than diffusion through the composite material. Models of mixed materials aircraft need to include internal structure in the external coupling model to accurately model current flow through the aircraft.

A description of some of the various computer codes available, their limitations, and the type of analysis to which they are suited is given below. Table 5.2-1 summarizes the capabilities of each program. The User's Manual for AC 20-53 gives additional details and references for some of these codes.

Surface currents have been calculated using finite difference time domain and method-of-moments techniques. Method-of-moment codes use a wire grid, surface patches, or strips to model the vehicle surface.

WIRANT, Numerical Electromagnetics Code (NEC), General Electromagnetic Model for the Analysis of Complex Systems (GEMACS), REDIST, THREDE, T3DFD, Thin Wire Time Domain (TWTD), Lumped Parameter Network (LPN), and transmission line models have all been used to determine surface currents. Care must be taken in the model definition to provide valid results.

WIRANT is a wire grid method-of-moments code which can be run on an IBM-XT or compatible personal computer, or a mainframe computer. The calculated currents and fields are output in the frequency domain. The code handles a frequency range which is suitable for lightning. It has been used to model several aircraft, e.g., Boeing 757, to determine EM coupling due to lightning. Results have compared favorably with experimental data. Plane wave excitation or current injection may be used to drive the models. Mixed materials may be incorporated into these models.

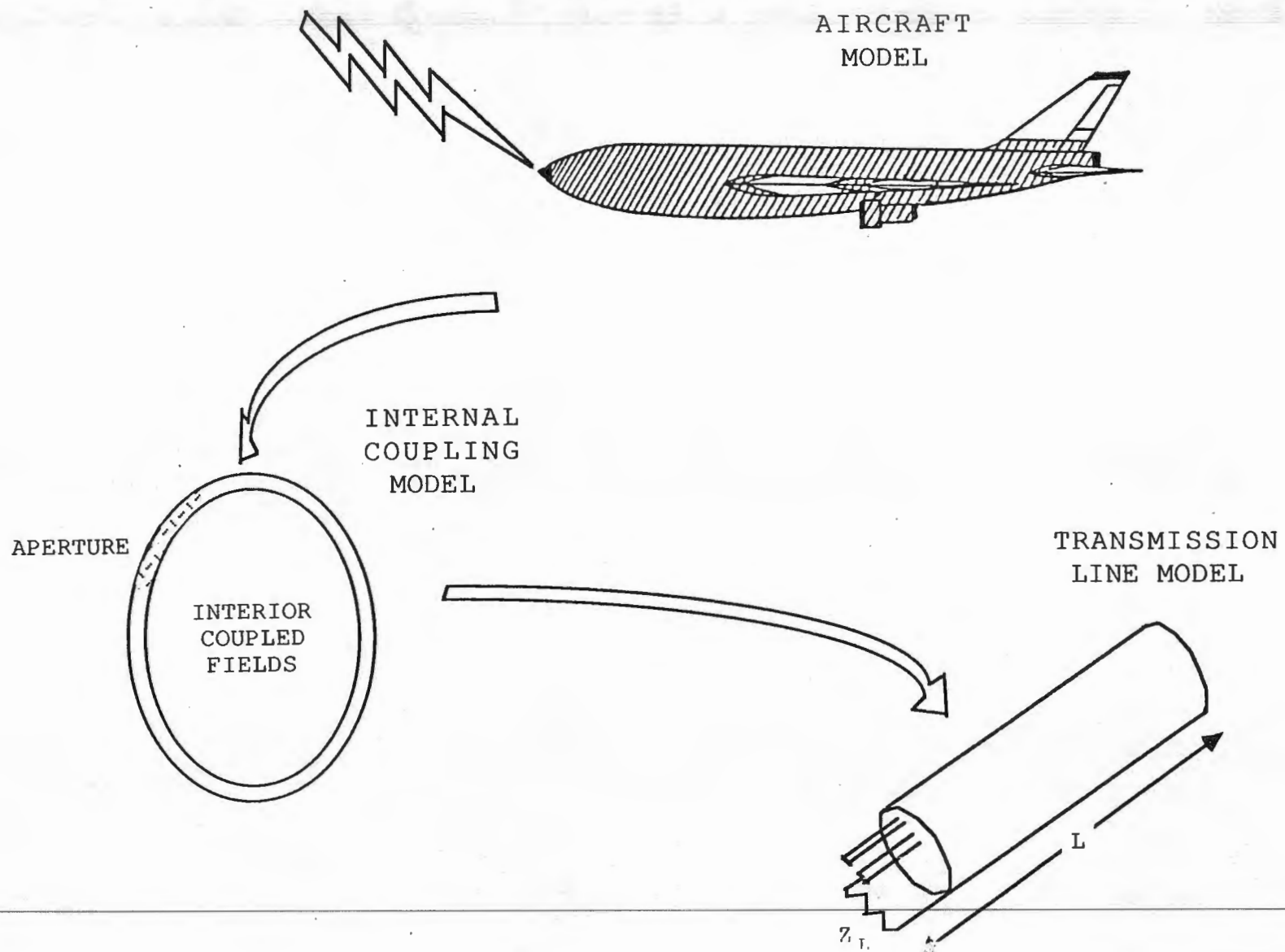


FIGURE 5.2-1. MULTILEVEL LIGHTNING COUPLING ANALYSIS APPROACH

TABLE 5.2-1. CAPABILITIES OF COMPUTER CODES FOR MODELING EM COUPLING TO AIRCRAFT

COMPUTER CODE	LIGHTNING THREAT	EXTERNAL COUPLING J_s, σ_s	INTERNAL COUPLING (APERTURES & DIFFUSION)	CABLE PENETRATION (SHIELDS & BUNDLES)	INTERNAL WIRING	PIN TRANSIENTS	EQUIPMENT UPSET
WIRANT	OK	OK	LIMITED SIZE	OK	N/A	N/A	N/A
THREDE	OK	SHORT TIME	OK	N/A	N/A	N/A	N/A
TRANSMISSION LINE	OK	OK	N/A	N/A	N/A	N/A	N/A
GEMACS	OK	OK	OK	OK	OK	N/A	N/A
T3DFD	OK	OK	OK	N/A	N/A	N/A	N/A
NEC2	OK	OK	N/A	N/A	N/A	N/A	N/A
LPN	OK	OK	ESTIMATE	N/A	N/A	N/A	N/A
TWTD	N/A	ESTIMATE RESONANCE	N/A	N/A	N/A	N/A	N/A
RESISTOR	OK	LATE TIME	N/A	LATE TIME	N/A	N/A	N/A
REDIST	OK	2-D LIMITS	2-D LIMITS	OK	OK	N/A	N/A
SCEPTRE	N/A	?	?	OK	OK	OK	N/A
SPICE	N/A	OK	N/A	OK	OK	OK	OK
TRANSSIM	N/A	?	?	OK	OK	?	N/A
SINLINE	N/A	N/A	N/A	CABLE/WIRE	OK	?	N/A
TRAFFIC	N/A	N/A	N/A	CABLE/WIRE	OK	OK	N/A
HANAP	N/A	N/A	N/A	N/A	N/A	OK	OK
TCAP	N/A	N/A	N/A	N/A	N/A	OK	OK
CIRCUS	N/A	N/A	N/A	N/A	N/A	OK	OK
SCORCH/ SUPERSAP	N/A	N/A	N/A	N/A	N/A	OK	N/A

EXTERNAL MODELS

INTERNAL MODELS

NEC is a method-of-moments code which uses segmentation by patches. The upper frequency limit is about 100 MHz. Both free space excitation and current injection can be modelled. This code requires a large amount of memory and can only be run on a mainframe computer.

GEMACS is a method-of-moments code which can also be run on an IBM-AT or mainframe computer. Predictions of external coupling to a model (segmented into a wire grid or patches), fields penetration due to aperture coupling, and the induced currents on a wire in a cavity behind the aperture can all be obtained. An example of the use of the code is given in Coffey and Hebert (1986). An entire session of the Applied Computational Electromagnetics Society (ACES) conference was devoted to the capabilities of this code (1988).

REDIST is a two-dimensional method-of-moments code which utilizes thin strips to form models. This code was developed on an IBM-XT compatible personal computer. It provides frequency domain currents and fields, both external and internal, and can be used for mixed material structures. Coupling to wiring can also be determined with this code. Limitations arise where the structural cross-section changes rapidly and when the structure thickness is greater than a skin depth of the material. Verification using scale models has provided good agreement (Geren, Melander, and Hall, 1987). Results have been compared to experimental data from a composite test bed aircraft (Cooley and Shortess, 1987).

THREDE is a three-dimensional finite difference computer code which runs on a mainframe computer. The primary limitation of this code for application to lightning is that most of the Low Frequency (LF) response is not included in the time domain response calculated by the code. Since a dominant coupling mechanism for composite structures is late time (after several microseconds) resistive drops in the structure, this code would not be suitable for modelling mixed material aircraft.

T3DFD is similar to THREDE, but allows mixed skin conductivities within the model.

TWTD utilizes a thin wire approximation and gives solutions in the time domain. It provides quick approximations of aircraft resonances. The code cannot treat current injection and so is not applicable to lightning.

LPN codes model physical structures with an equivalent circuit. This has the advantage of simple development and implementation. Much less memory is required for these simpler models. Currents can be determined by using a transmission line code. LPN theory may be applied to internal coupling provided calculations of the external coupling are available to be used as input to the internal model.

Transmission line models are useful to provide resonances and current densities. These models are usually simpler than models using a wire grid code. The SAE AE4L Orange Book and Simpson and Katzer (1988) give examples of the use of this type of code. Transmission line theory is applicable at all levels of coupling analysis.

Resistor models can be used to determine late time responses, due to structural IR drops, with fair accuracy. An example of this very simple model is given in section 6.2. Internal structure and cable bundles can be included. The resistance can be easily calculated by known physical parameters or measured values. Cooley and Shortess (1987) contains another example of this type of modelling.

TRAFFIC, SINLINE, SCEPTRE, TRANSSIM, and SPICE may be used in conjunction with results obtained from one of the codes described above which calculates external coupling to determine transients on internal wiring. TRAFFIC consists of a library of codes and is a part of PRESTO (see PRESTO User's Guide). It provides frequency domain output for wiring transients. SPICE and SCEPTRE are time domain codes and can thus include non-linear elements.

HANAP calculates pin transients in the frequency domain and design margins using a mainframe computer. Threshold Circuit Analysis Program (TCAP) can perform similar calculations for simple circuits on a hand calculator (Pryzby and Plumer, 1984). For non-linear analysis, CIRCUS, a code within the TRAFFIC library, will calculate design margins. SCORCH and its predecessor, SUPERSAP, may be available as a commercial product to calculate pin transients.

6. SUPPLEMENTAL EXAMPLES

6.1. Coupling to Wiring

An example of a simple two-dimensional model is used to illustrate coupling to wiring within a mixed materials structure. The model, shown in figure 6.1-1, is a box which has three metal sides and one G/E side. The conductivity and thickness of each side is indicated on the figure. The thicknesses used are for typical aircraft structure. The code RESIST was used to evaluate this model.

Numbers 1 through 6 inside the box indicate the shield locations evaluated. Each model run contained a single wire at a different location. The output was given as a transfer function of the shield current relative to the input current for the entire structure. Figure 6.1-2 contains the transfer functions for each shield location.

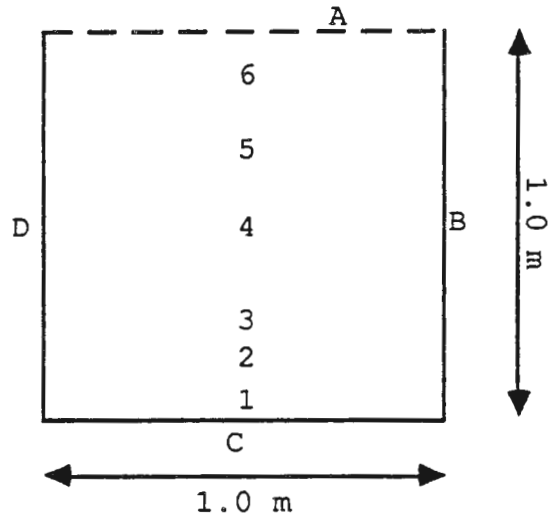
A worst case estimate of shield currents can be obtained by multiplying the calculated shield transfer function by the threat, in this case a double exponential lightning spectrum, and Fourier transforming the resultant spectrum to the time domain. Current pulse responses for each shield location are shown in figure 6.1-3. The peak currents vary by an order of magnitude. Induced shield currents are small for locations near the metal bottom of the box and increase steadily as the shield is moved toward the graphite lid.

These estimates of current represent worst case (upper bound) strikes, because the entire 200,000 A is assumed to flow through the box and shield. This would not be the case if the box were part of an aircraft. In order to obtain a more realistic estimate of internal shield currents, a vehicle external model must be analyzed to obtain the division of current between the modeled box and the remainder of the vehicle. This transfer function is then multiplied by the shield transfer function (calculated previously) and the lightning threat to obtain the shield current relative to the aircraft current.

Leakage through the graphite can be illustrated by looking at the magnetic fields calculated with this model. Figures 6.1-4 and 6.1-5 show magnetic field contour lines for the box at 1 kHz and 1 MHz. At 1 kHz, the G/E is transparent to the fields. The fields are excluded from the box at higher frequencies due to the skin effect. This indicates why early time responses for graphite or metal structures are similar.

After determining the shield currents, the current and voltage transients at the component level can be calculated using an appropriate method. A transmission line formulation is used here.

REDIST MODEL
METAL BOX WITH GRAPHITE LID



Shield locations denoted by numbers 1-6.

WALL	STRUCTURAL PARAMETERS	
	THICKNESS (m)	CONDUCTIVITY (mhos/m)
A	2.3E-3	1.5E4
B	1.3E-3	3.5E7
C	1.3E-3	3.5E7
D	1.3E-3	3.5E7
SHIELD	2.54E-4	3.5E7

FIGURE 6.1-1. REDIST EXAMPLE MODEL GEOMETRY

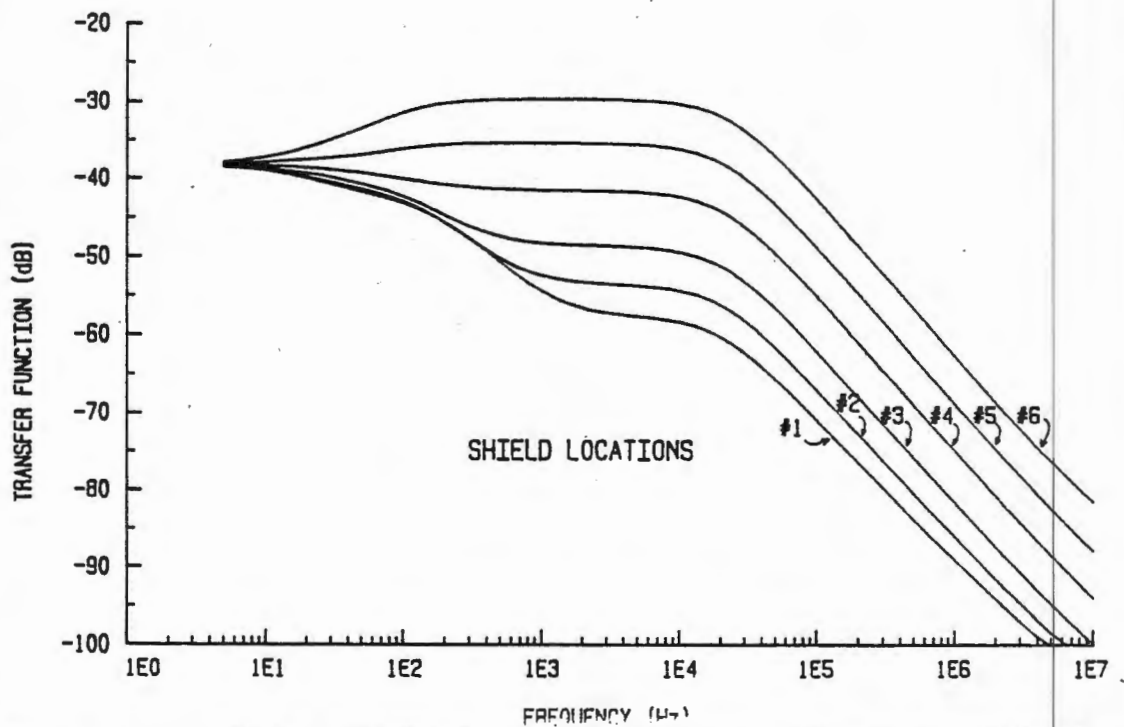


FIGURE 6.1-2. SHIELD TRANSFER FUNCTION FOR METAL/GE BOX FOR SEVERAL SHIELD LOCATIONS

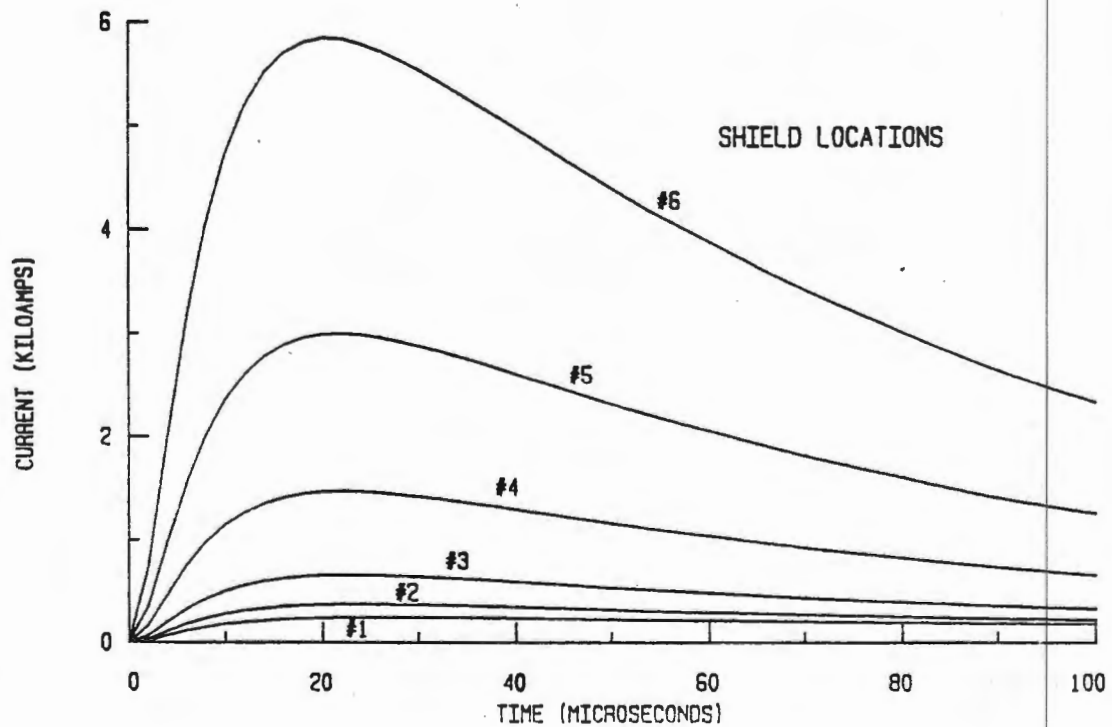


FIGURE 6.1-3. SHIELD CURRENT FOR METAL/GE BOX FOR SEVERAL SHIELD LOCATIONS FOR 200 kA ATTACHED LIGHTNING

CONTOURS OF CONSTANT VECTOR POTENTIAL

$F = 1E3 \text{ HZ}$

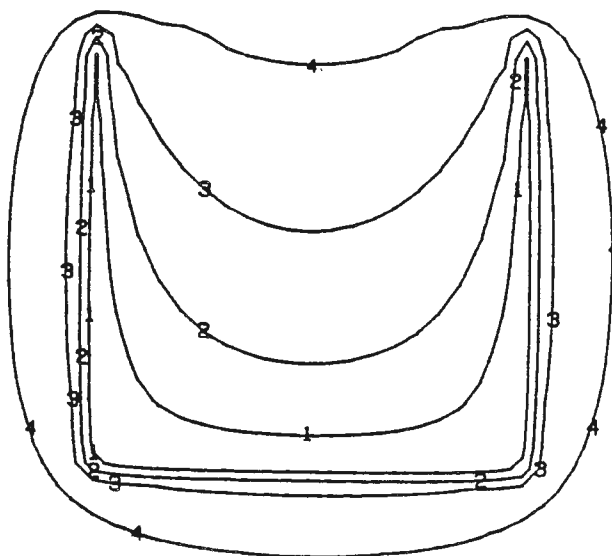


FIGURE 6.1-4. MAGNETIC FIELD CONTOURS FOR METAL/GE BOX AT 1 kHz

CONTOURS OF CONSTANT VECTOR POTENTIAL

$F = 1E6 \text{ HZ}$

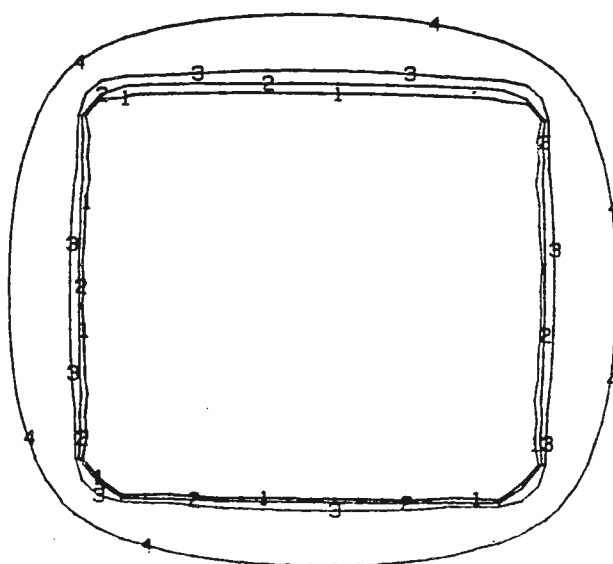


FIGURE 6.1-5. MAGNETIC FIELD CONTOURS FOR METAL/GE BOX AT 1 MHz

The coupling to the shields is a continuous magnetic or electric field interaction along the length of the shield. This method of calculation uses a continuous voltage source term running the length of the transmission line model to the equipment terminations at each end. Basic assumptions made for this model are:

- The shield transfer function (voltage relative to current) is given by $R_{\text{shield}} + j\omega \times L_{\text{shield}}$

where

$$R_{\text{shield}} = 1.78 \times 10^{-3} \text{ ohms per meter}$$

$$L_{\text{shield}} = 8.96 \times 10^{-10} \text{ henrys per meter}$$

These numbers were obtained from measurements made on braided aluminum shields.

- The termination impedance was assumed to be 100 ohms at each end.
- The voltage and currents were calculated at one end of a transmission line with a continuous voltage source assumed equal to the shield current times the shield transfer function.
- The line length, L , was assumed to be 10 meters.

The results can be seen in figure 6.1-6. The time domain voltage waveforms for each of the six shield locations are shown. The peak voltages can be estimated using the following equation.

$$\begin{aligned} V_{\text{peak}} &= V_{\text{oc}} / 2 \\ &= (I_{\text{peak}} \times R_{\text{shield}} \times L) / 2 \end{aligned}$$

where V_{oc} is the open circuit voltage and I_{peak} is the peak shield current.

For shield location #5 with a peak current of about 3 kA,

$$\begin{aligned} V_{\text{peak}} &= (3 \text{ kA} \times 1.78 \times 10^{-3} \text{ ohms/m} \times 10 \text{ m}) / 2 \\ &= 26.7 \text{ V} \end{aligned}$$

Larger voltages will be induced on longer cables and smaller voltages will be induced on shorter cables. This is due to resonance effects on the cables. Figure 6.1-7 illustrates the variation in induced voltages for several line lengths.

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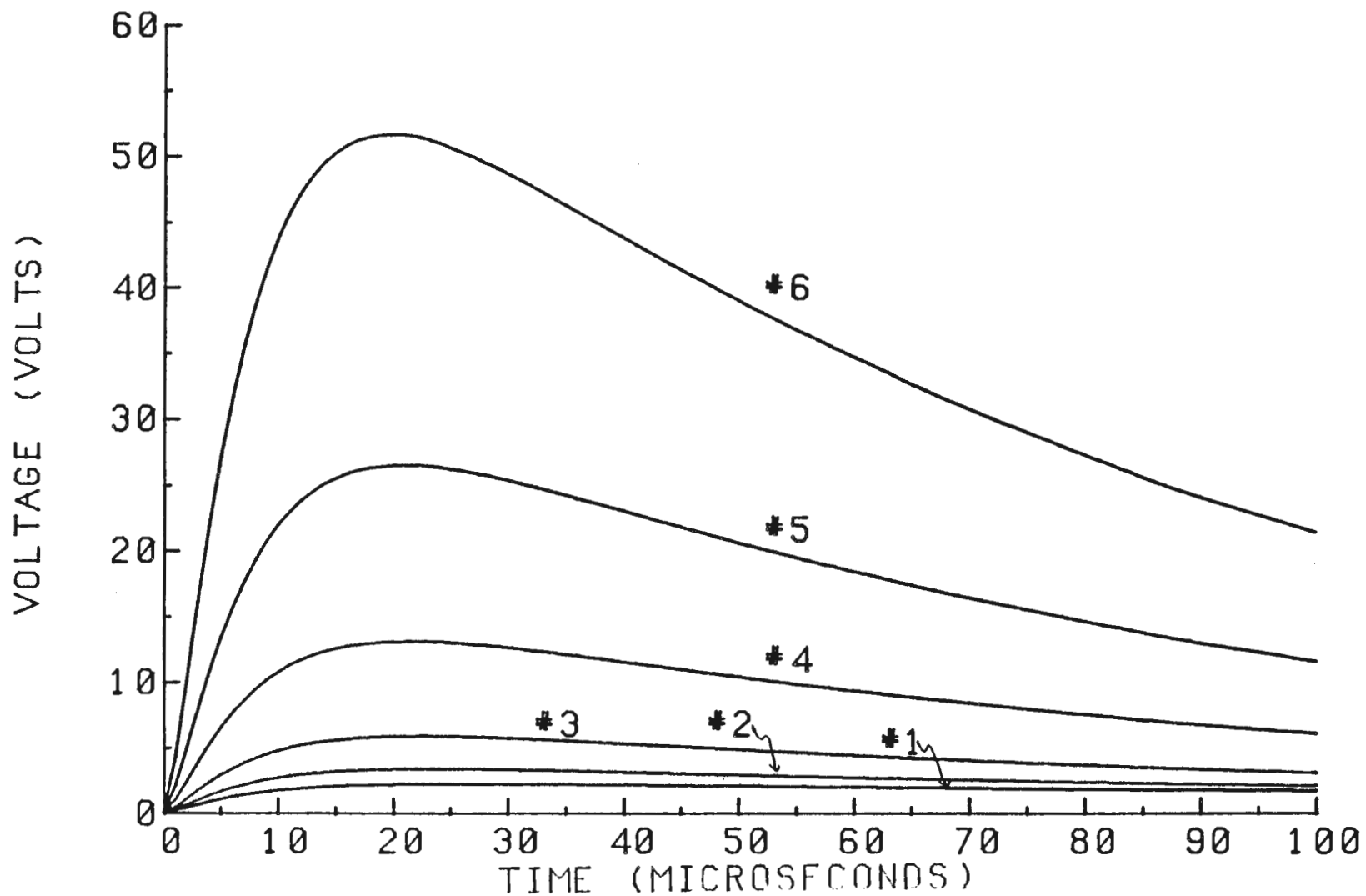


FIGURE 6.1-6. PIN VOLTAGE WAVEFORMS FOR SEVERAL SHIELD LOCATIONS

13-83

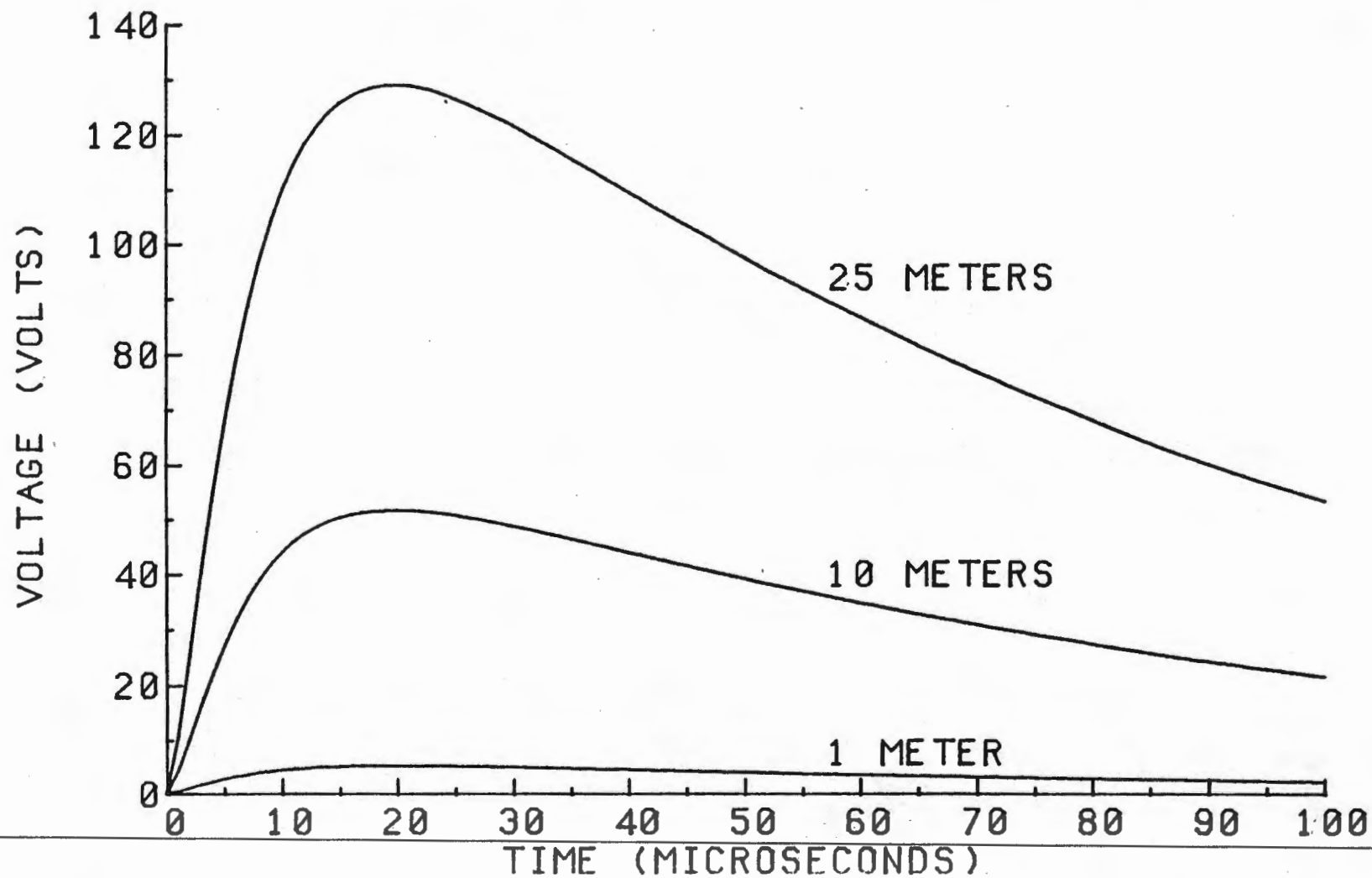


FIGURE 6.1-7. PIN VOLTAGE WAVEFORMS FOR SHIELD LOCATION #6 FOR VARIOUS LINE LENGTHS

6.2. Composite Wing Model

A dc resistance model of a composite wing used in NASA tests was constructed to determine the amount of current flowing through various sections of the wing. Results of this modeling study will predict energy flows into various structural members and fasteners. A dc model is appropriate for lightning studies since structural and fastener damage occurs because of the high energy levels. These high energy levels result primarily from large LF components of the lightning pulse.

An estimate of the dimensions and properties of the composite wing are listed in table 6.2-1. The spars were considered "I" beam sections 1/4 inch thick with the remaining values given in the table. The skin was taken as a product of average dimensions obtained from a NASA report, Pryzby and Plumer (1984). Both sides of the wing skin were considered in the total area. The braid running in the leading wing edge was assumed to be 1/2 inch braid which was taken to have 3 milliohm per meter resistance, as measured in previous studies. Fasteners were assumed to have a resistance of 10 milliohms each. These resistances are similar to predictions of various fastener resistances in Cooley (1985). The length of the wing is assumed to be 5.77 meters.

The structure resistance, R_s , is determined using the following equation.

$$R_s = L / (\sigma \times A)$$

where L is the length in meters, A is the area in square meters, and σ is the conductivity in mhos (or siemens) per meter.

This simple resistance model predicts current level as percentages of total current in table 6.2-1. The predictions are obtained simply by resistive division between each item and the entire structure. These are compared to measured values for late time (80 microseconds) into the pulse. The agreement is good considering that dimensions, conductivities, number of fasteners, and fastener resistances are not accurately known.

TABLE 6.2-1. COMPOSITE WING RESISTANCE MODEL RESULTS VS. MEASUREMENTS

ITEM	WIDTH (IN)	HEIGHT (IN)	AREA (IN ²)	CONDUCTIVITY (MHO/M)	STRUCTURE RESISTANCE (OHM)	TOTAL FASTENER RESISTANCE	STRUCTURE RESISTANCE W/FASTENERS	ITEM CURRENT (% OF TOTAL)	MEASURED CURRENT (3) (% OF TOTAL)
FRONT SPAR	1.00	2.00	0.88	28000	0.363	0.005 (1)	0.368	3.4	2% TO 5%
CENTER SPAR	2.50	5.00	2.38	28000	0.134	0.005 (1)	0.139	9.0	2% TO 5%
REAR SPAR	1.50	3.00	1.38	28000	0.231	0.005 (1)	0.236	5.3	2% TO 5%
SKIN	77.00	0.0875	6.74	28000	0.047	0.001 (2)	0.048	25.8	29%
BRAID					0.017	0.005 (1)	0.022	56.6	71%

TOTAL RESISTANCE
OF STRUCTURE = 0.012 OHM

FASTENER RESISTANCE
ASSUMED = 0.01 OHM

- (1) 2 FASTENERS ASSUMED
(2) 10 FASTENERS ASSUMED
(3) MEASURED DATA FROM PLUMER (1984)



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GLOSSARY

ACTION INTEGRAL. The action integral is a critical factor in the production of damage. It relates to the energy deposited or absorbed in a system. This energy cannot be defined without knowing the resistance of the system. The instantaneous power dissipated in a resistor is I^2R and is expressed in watts. For the total energy expended, the power must be integrated over time to get the total joules, watt-seconds. By specifying the integral of $i(t)^2$ over the time interval involved, a useful quantity is defined for application to any resistance value. In the case of lightning, this quantity is defined as the action integral and is specified as $i(t)^2 dt$ over the time the current flows.

ACTUAL TRANSIENT LEVEL. The actual transient level is the level of transients which actually appear at the system interfaces as a result of the external environment. This level may be less than or equal to the transient control level but should not be greater.

AIRCRAFT LIGHTNING INTERACTION. An encounter with lightning that produces sufficient current within or voltages along an aircraft skin or structure to pose a threat to the aircraft electrical/electronic systems, as a result of a direct lightning attachment.

ATTACHMENT POINT. A point of contact of the lightning flash with the aircraft.

CHARGE TRANSFER. The integral of the current over its entire duration, $i(t)dt$, in coulombs.

COMPONENT DAMAGE. Condition arising when the electrical characteristics of a circuit component are permanently altered beyond its specifications.

CORONA. A luminous discharge that occurs as a result of an electrical potential difference between the aircraft and the surrounding atmosphere.

CRITICAL. Functions whose failure would contribute to or cause a failure condition which would prevent the continued safe flight and landing of the aircraft.

DESIGN MARGIN. The difference between the equipment transient design levels and the transient control level.

DIRECT EFFECTS. Any physical damage to the aircraft or onboard systems due to the direct attachment of the lightning channel. This includes tearing, bending, burning, vaporization, or blasting of aircraft surfaces or structures, and damage to electrical/electronic systems.

EQUIPMENT TRANSIENT DESIGN LEVEL. The level of transients which the equipment is qualified to withstand.

EQUIPMENT TRANSIENT SUSCEPTIBILITY LEVEL. The transient level which will result in damage or upset to the system components. This level will be greater than the equipment transient design level.

EXTERNAL ENVIRONMENT. Characterization of the natural lightning environment with idealized waveforms for engineering purposes.

INDUCED VOLTAGES. A voltage produced around a closed path or circuit by changing magnetic or electric fields or structural IR voltages.

INDIRECT EFFECTS. Voltage and/or current transients induced by lightning in aircraft electrical wiring which can produce upset and/or damage to components within electrical/electronic systems.

INTERNAL ENVIRONMENT. The fields and structural IR potentials produced by the external environment, along with the voltages and currents induced by them.

LIGHTNING FLASH. The total lightning event in which charge is transferred from one charge center to another. It may occur within a cloud, between clouds, or between a cloud and the ground. It can consist of one or more strokes, plus intermediate or continuing currents.

LIGHTNING LEADER STROKE. The leader forms an ionized path for charge to be channeled towards the opposite charge center. The stepped leader travels in a series of short, luminous steps prior to the first return stroke. The dart leader reionizes the return stroke path in one luminous step prior to each subsequent return stroke in the lightning strike.

LIGHTNING STRIKE. Any attachment of the lightning flash to the aircraft.

LIGHTNING STRIKE ZONES. Locations on the aircraft where the lightning flash will attach or where substantial amounts of electrical current may be conducted between attachment points. The location of these zones on any aircraft is dependent on the aircraft's geometry and operational factors and often varies from one aircraft to another.

LIGHTNING RETURN STROKE. A lightning current surge that occurs when the lightning leader makes contact with the ground or an opposite charge center.

MULTIPLE BURST. A randomly spaced series of bursts of short duration, low amplitude current pulses, with each pulse characterized by rapidly changing currents. These bursts may result from lightning leader progression or branching and may be accompanied by or superimposed on stroke or continuing currents. The multiple bursts appear to be most intense at the time of initial leader attachment to the aircraft.

MULTIPLE STRIKE. Two or more lightning strikes during a single flight.

MULTIPLE STROKE. Two or more return strokes occurring during a single lightning flash.

PEAK RATE OF RISE. The maximum instantaneous slope of the waveform as it rises to its maximum value. Mathematically, the peak rate of rise of a function, $i(t)$, may be expressed as the maximum of $d[i(t)]/dt$.

RETURN STROKE. See lightning return stroke.

STRUCTURAL IR VOLTAGE. The portion of the induced voltage resulting from the product of the distributed lightning current, I , flowing through the resistance, R , of the aircraft skin or structure.

SWEPT STROKE. A series of successive attachments due to sweeping of the flash across the surface of the airplane by the motion of the airplane.

SYSTEM FUNCTIONAL UPSET. Impairment of system operation, whether permanent or momentary (e.g., a change of digital or analog state) which may or may not require manual reset.

TRANSIENT CONTROL LEVEL. The maximum allowable level of transients appearing at the systems interfaces as a result of the defined external environment.

TRIBOELECTRIC CHARGING. Static electricity produced on a structure from the effects of friction.

UPSET. See system functional upset.

LIST OF ACRONYMS

ACAP	Advanced Composite Airframe Program
ACES	Applied Computational Electromagnetics Society
AEHP	Atmospheric Electricity Hazards Protection
AE4L	SAE Subcommittee (Lightning)
AFFDL	Air Force Flight Dynamics Laboratory
AFWAL	Air Force Wright Aeronautical Laboratories
ALCM	Air Launched Cruise Missile
B-dot	Derivative of the magnetic field with respect to time
CW	Continuous Wave
DNA	Defense Nuclear Agency
DOE	Department of Energy
E-dot	Derivative of the electric field with respect to time
EM	Electromagnetic
EMC	Electromagnetic Compatibility
EMI	Electromagnetic Interference
EMP	Electromagnetic Pulse
ETDL	Equipment Transient Design Level
FAA	Federal Aviation Administration
G/E	Graphite Epoxy
GEMACS	General Electromagnetic Model for the Analysis of Complex Systems
HF	High Frequency
I	Current
I-dot	Derivative of the current with respect to time
kA	Kiloampere
LF	Low Frequency
LPN	Lumped Parameter Network
LRU	Line Replaceable Unit
LTRI	Lightning and Transients Research Institute
NASA	National Aeronautics and Space Administration
NASC	Naval Air Systems Command
NEC	Numerical Electromagnetics Code
NSWC	Naval Surface Weapons Center
R	Resistance
RAE	Royal Aircraft Establishment
RF	Radio Frequency
RL	Resistance/Inductance
RLC	Resistance/Inductance/Capacitance
RLCM	Resistance/Inductance/Capacitance/Mutual
SAE	Society of Automotive Engineers
TCAP	Threshold Circuit Analysis Program
TCL	Transient Control Level
TTL	Transistor-Transistor Logic
TWTD	Thin Wire Time Domain
UHF	Ultra-high Frequency
UK	United Kingdom
VHF	Very High Frequency





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HANDBOOK-VOLUME II DIGITAL SYSTEMS VALIDATION

CHAPTER 14 HIGH ENERGY RADIO FREQUENCY FIELDS (IMPACT ON DIGITAL SYSTEMS)



PREPARED BY:

COMPUTER RESOURCE MANAGEMENT, INC.
950 HERNDON PARKWAY, SUITE 360
HERNDON, VIRGINIA 22070

PREPARED FOR:

FEDERAL AVIATION ADMINISTRATION
TECHNICAL CENTER
ATLANTIC CITY INTERNATIONAL AIRPORT, NEW JERSEY 08405

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1. INTRODUCTION

1.1. Background

The introduction of a new generation of aircraft (fixed wing, rotary wing, and powered lift) advanced technologies, composite structures, and sophisticated high-integrity, integrated, software-based, digital flight control and avionic systems has confronted the aviation community with numerous flight safety issues and concerns. These new aircraft are making use of fly-by-wire flight control systems in which direct mechanical or hydraulic linkages are being replaced by solid-state digital electronic systems controlling engines, fuel systems, and electrohydraulic/electromechanical actuators. Previously, flight control and avionic systems utilized analog technology which was not responsive to transient disturbances unless they exceeded the analog device damage level. Unlike their analog predecessors, digital electronics are very susceptible to transient effects, as well as to discrete-frequency radiation. Digital device performance can be adversely affected (upset) before the device damage level is reached.

A number of European military aircraft fatal accidents have been attributed to High Energy Radio Frequency (HERF). A digital fly-by-wire military Toronado aircraft and crew were lost during a tactical training strafing attack in Germany. The loss was attributed to HERF when the aircraft flew through a high intensity Radio Frequency (RF) field. The civil/military aviation industry has very limited experience or data directed to accidents caused by electromagnetic transients and/or radiation. The present criteria, specifications, and procedures are being reevaluated. The HERF fields apparently upset the digital flight control system of the Toronado which was qualified to a very low Electromagnetic Environment (EME) standard.

While composite materials may offer significant advantages in strength, weight, and cost, they provide less electromagnetic shielding than aluminum. The use of solid-state digital technology in flight critical systems creates major challenges to prevent transient susceptibility and upset in both civil and military aircraft. Therefore, the Civil Aviation Authority (CAA), United Kingdom (U.K.) and the Federal Aviation Administration (FAA), United States (U.S.) voiced concern relative to emerging technology aircraft and systems.

The present criteria, specifications, and procedures appear to be inadequate for industry and the certification authorities for three reasons: (1) the changes in aircraft construction from aluminum to composites, (2) the increased criticality of software-based digital flight control and avionic systems, and (3) the increased levels of RF energy over a wide range of frequencies. The CAA and FAA therefore decided that the impact of each of these areas on the total aircraft RF susceptibility hardening and protection should be assessed.

1.2. Objectives and Scope

Currently, the U.S. civil aircraft manufacturers use the electromagnetic susceptibility guidelines as referenced in the Radio Technical Commission for Aeronautics (RTCA) document RTCA/DO-160B, section 20. Category Z equipment qualified to this standard is demonstrated to perform satisfactorily when subjected to a 1 Volt per meter (V/m) electric field intensity for frequencies between 35 MHz and 1215 MHz (a 2 V/m standard is imposed for the Very High Frequency (VHF) communication band from 118 to 136 MHz). A similar standard is used for frequencies between 15 kHz and 35 MHz but is specified in different terms, with a slightly higher standard for the High Frequency (HF) band between 2 and 30 MHz. The European Toronado aircraft was designed and protected according to a similar specification.

The U.S. military qualification standards for RF susceptibility are contained in MIL-STD-461B, section 19. This standard calls for an electric field strength 5 to 20 times greater than that called for in the RTCA document. It also extends the frequency range to at least 10,000 MHz and possibly to 40,000 MHz. This document implies that an electric field of at least 200 V/m may be encountered at specific frequencies which are specified by the procuring command. Testing methods are described in MIL-STD-462. The U.S. Navy, in qualifying aircraft for a carrier flight deck, has defined high electric field intensities greatly exceeding the values of MIL-STD-461B. A number of current military aircraft have encountered Electromagnetic Compatibility (EMC) problems, including the UH-60 helicopters. EME considerations for equipment design and procurement are described in MIL-STD-235-1A.

There have been no recognizable problems reported on civil transport airplanes to date. However, fly-by-wire technology is in use today on some Boeing Model 757-200 and 747-400 airplanes, on the Airbus Industrie A-320, Pratt and Whitney 4000° engines, and other U.S. civil aircraft (fixed and rotary wing). In addition, civil and military transmitter systems are continually being upgraded to operate at higher frequencies and to produce higher effective electric field intensities. The Spectrum Engineering Division of the FAA Technical Center and the Aircraft Engineering Division in the Office of Airworthiness initiated a research effort with the Department of Defense (DOD), Electromagnetic Compatibility Analysis Center (ECAC), Annapolis, Maryland, to survey the Contiguous United States (CONUS) and its possessions for civil and military ground, airborne, and shipborne HERF sources. Simultaneously, a solicitation for the CAA and its European counterparts to aggressively conduct a similar survey and analysis was requested. Following the surveys by ECAC, a validation phase (airborne measurements) of selected transmitters and one airport was conducted by Ohio University, Athens, Ohio.

United States research initiatives include:

- Conducting research into worldwide civil and military ground, airborne, and shipborne emitter radiation which may be a threat to emerging aircraft and systems.
- Sharing knowledge of each country's (i.e., United States, United Kingdom, and France) civil and military high powered transmitters.

- Developing the best approach to transfer this information to the airframe and systems manufacturers, RTCA, Society of Automotive Engineers (SAE), European Organization for Civil Aviation Electronics (EUROCAE), International Civil Aviation Organization (ICAO), etc.
- Developing an international susceptibility standard.
- Investigating possible regulatory actions (i.e., Notice of Proposed Rulemaking (NPRM), Advisory Circular (AC), User's Manual, etc.).
- Considering the airworthiness, certification, and operational issues and implications.

In addition to these initiatives, the Flight Safety Research Branch at the FAA Technical Center is participating in an Electromagnetic Radiation (EMR) measurement test being conducted by the U.S. Air Force Special Missions Operational Test and Evaluation Center (SMOTEC), Hurlburt Field, Florida. The primary purpose of this test is to collect EMR data at selected locations to enable safe air transport of munitions. Experience has shown that when an electromagnetic field impinges on a munition, it can fire the electroexplosive devices, destroy transistors, or cause electronic circuits to malfunction. Examples of the multitude of threats considered in this chapter are illustrated in figure 1.2-1.

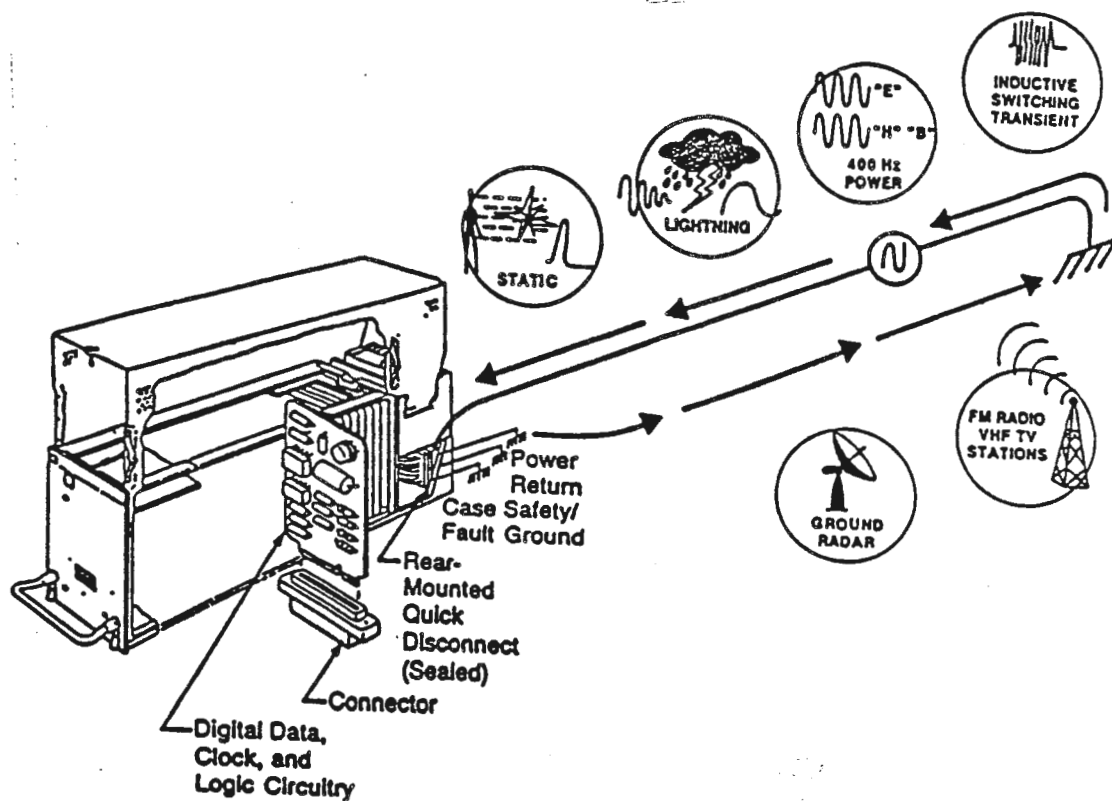


FIGURE 1.2-1. INDUCED RADIO FREQUENCY THREAT

2. ELECTROMAGNETIC ENVIRONMENT SURVEY AND ANALYSIS - UNITED STATES

2.1. Guidelines

In an effort to standardize prediction techniques, the FAA and ECAC documented the techniques used in the form of guidelines (DOD/ECAC, 1987) for predicting maximum EMR levels at aircraft exposed to high-powered emitters. Those guidelines were used in the calculation of maximum peak and average electric field strengths at the aircraft as a result of ground, airborne, and shipborne RF emitters. The guidelines for calculating peak and average field strengths are presented as step-by-step instructions starting with the data on the emitters in the EME and ending with an envelope graph representing maximum field strengths. The methods for calculating field strengths in the worst-case scenario and in the takeoff and landing scenario are defined and discussed. Simplifying assumptions are made and, where appropriate, justifications are shown. The minimum distances used in the U.S. environmental analysis are presented. The results obtained by following these guidelines will be peak and average field strengths, at worst case distances between emitters in the EME and aircraft. The total survey and environment analysis is described as the airport, air-to-air, ground, and shipborne environment.

Scenarios for the survey and analysis guidelines are as follows:

- A worst-case civil or military emitter, and for comparison, a takeoff or landing at specific airports.
- A worst-case considering the maximum EMR levels at an aircraft exposed to any ground, airborne, and shipborne emitter in the United States.
- The EME that would be experienced by an aircraft on takeoff and landing at six major U.S. airports.
- Integration and comparison of the worst-case emitters from the United Kingdom, France, and other countries as their information becomes available.
- Integration and comparison of the American, British, and French airport data.

2.2. Results

The results are presented in peak and average field strengths for the aircraft emitter sources surrounding airport, air-to-air, ground, and shipboard areas. The survey would review the emitters from 10 kHz to 40 GHz, identify worst-case conditions, and plot survey and study results in RF envelope histograms. To make the data manageable and unclassified, the granularity of the curve has been smoothed.

The following assumptions were made to simplify the task and the histograms:

- Specification of minimum distances.
- Continuous illumination.
- Mainbeam gain.
- Modulation independence.
- Constructive ground reflections at HF.
- Noncumulative field strengths.
- Near-field correction.

A typical analysis of a specific emitter considered the specifications of each, as indicated in table 2.2-1.

TABLE 2.2-1. FIXED AIRPORT EMITTER

HEIGHT - FINDING RADAR	
Operating Frequency	2700 MHz
Peak/Average Output Power	3.5 MW/2.9 kW
Antenna Size	8 ft x 30 ft
Antenna Gain	39 dBi
Distance	500 ft
ELECTRIC - FIELD STRENGTH	
Uncorrected Peak	6050 V/m
Uncorrected Average	175 V/m
Near Field	700 ft
Near Field Correction Factor	3.5 dB
Corrected Peak	4100 V/m
Corrected Average	120 V/m

2.3. Airport Environment

The airport environment consists of fixed and mobile emitters located on the airport. The airport is defined by a five nautical mile (nmi) radius from the geographic center of the airport. Minimum distance calculations are for airport fixed emitters and mobile emitters. By regulations, the minimum distance between a fixed emitter and the taxiway or runway is 250 feet (ft). Table 2.3-1 identifies some fixed emitters at the airport.

TABLE 2.3-1. FIXED AIRPORT ENVIRONMENT

Marker Beacon
Localizer
VOR
Glide Slope
Ground Controlled Approach Radar
Distance Measuring Equipment
TACAN
Microwave Landing System
Airport Surface Detection System
Non-Directional Beacon
Airport Surveillance Radar
Air Route Surveillance Radar
Weather Radar
ATCRBS Interrogator
VHF and UHF Communications/Telemetry

For fixed airport emitters above, predicted maximum peak and average field strengths were calculated and plotted as in figures 2.3-1 and 2.3-2.

The mobile emitter environment consists of all emitters that are not in a fixed location within a minimum distance of 50 ft. These include those emitters on ground support vehicles as well as those in aircraft. Aircraft emitters may operate while on the ground and taxiway, and prior to takeoff. Table 2.3-2 outlines those mobile emitters.

For mobile airport emitters above, predicted maximum peak and average field strengths were calculated and plotted as in figures 2.3-3 and 2.3-4.

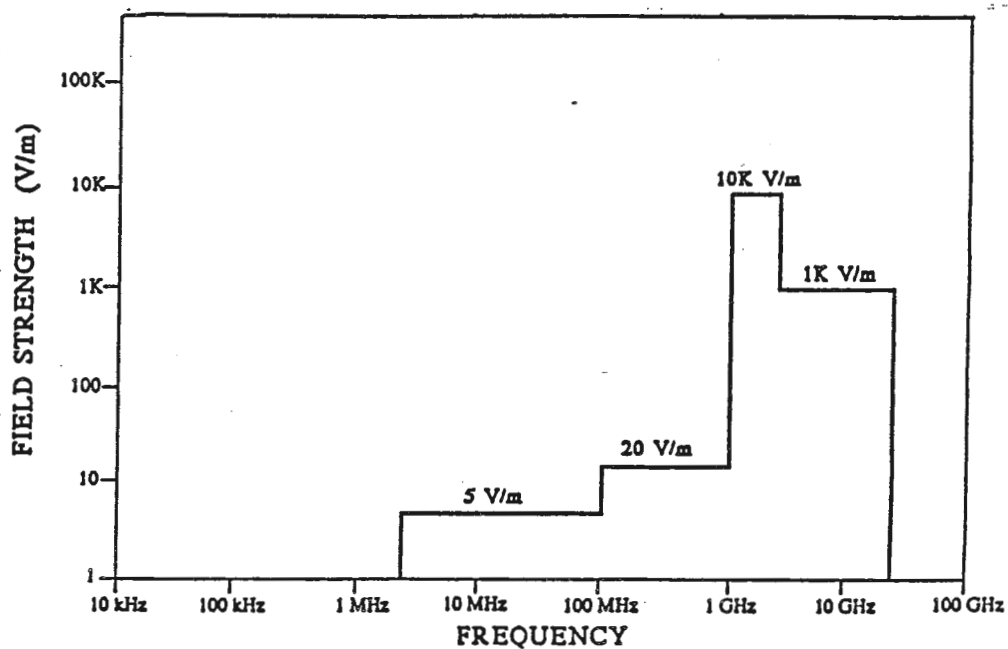


FIGURE 2.3-1. PREDICTED MAXIMUM PEAK FIELD STRENGTH LEVELS FROM CIVILIAN AIRPORT FIXED EMITTERS

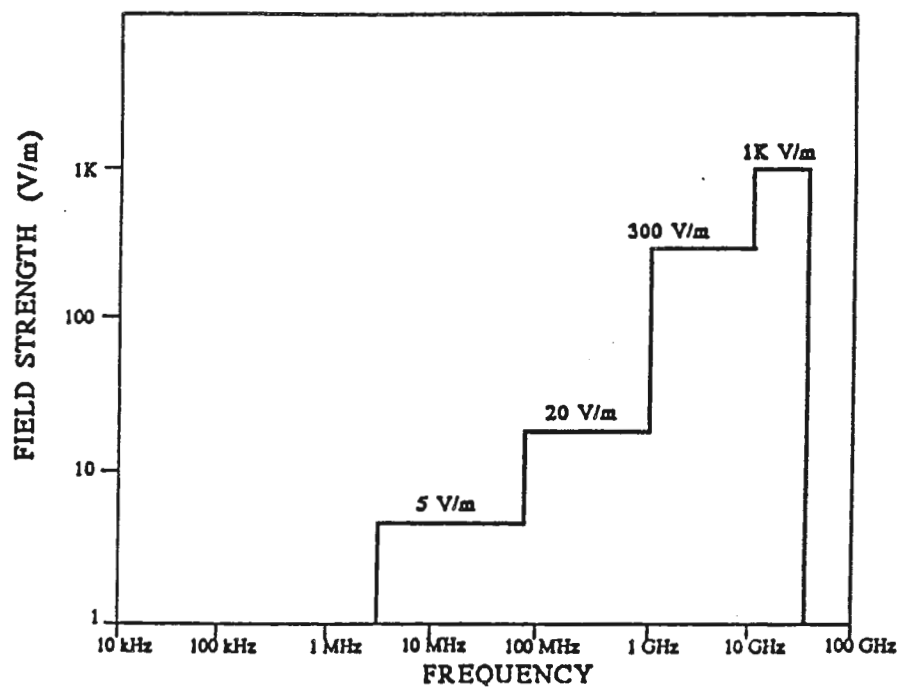


FIGURE 2.3-2. PREDICTED MAXIMUM AVERAGE FIELD STRENGTH LEVELS FROM CIVILIAN AIRPORT FIXED EMITTERS

TABLE 2.3-2. MOBILE AIRPORT ENVIRONMENT

HF/VHF/UHF Communications
 Tacan
 Doppler Navigational Radar
 Radio Altimeter
 Weather Radar
 ATCRBS Beacon

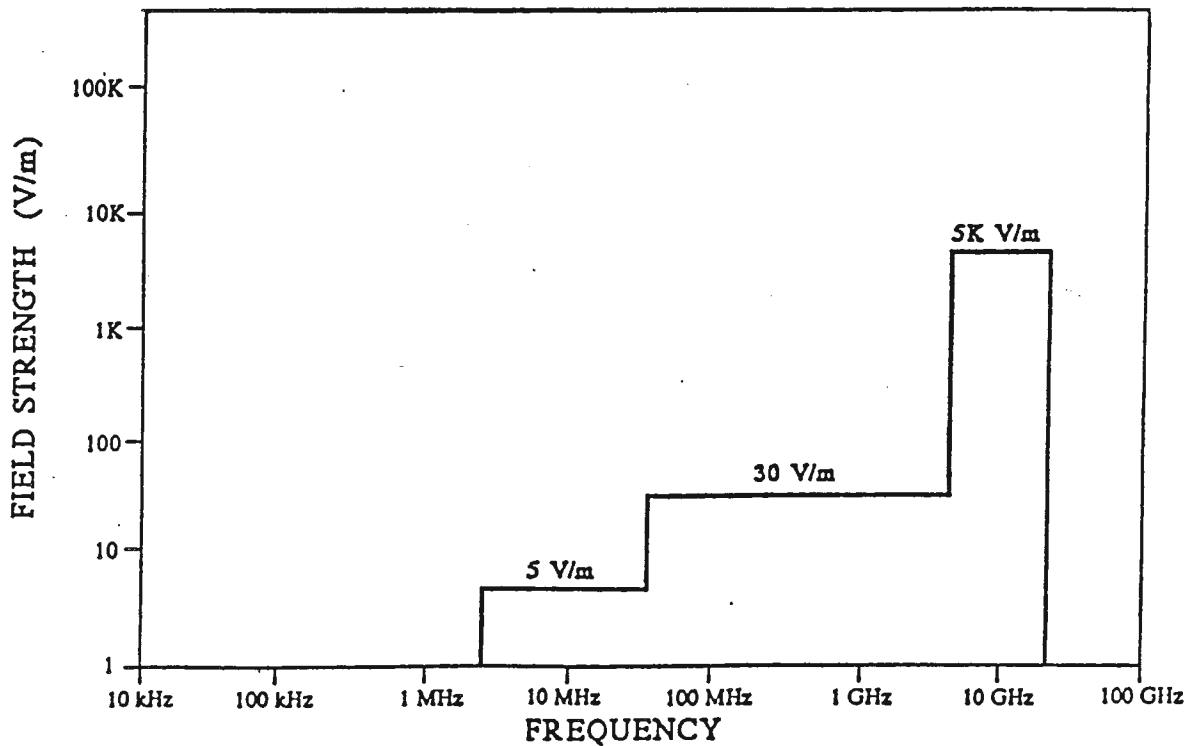


FIGURE 2.3-3. PREDICTED MAXIMUM PEAK FIELD STRENGTH LEVELS FROM CIVILIAN AIRPORT MOBILE EMITTERS

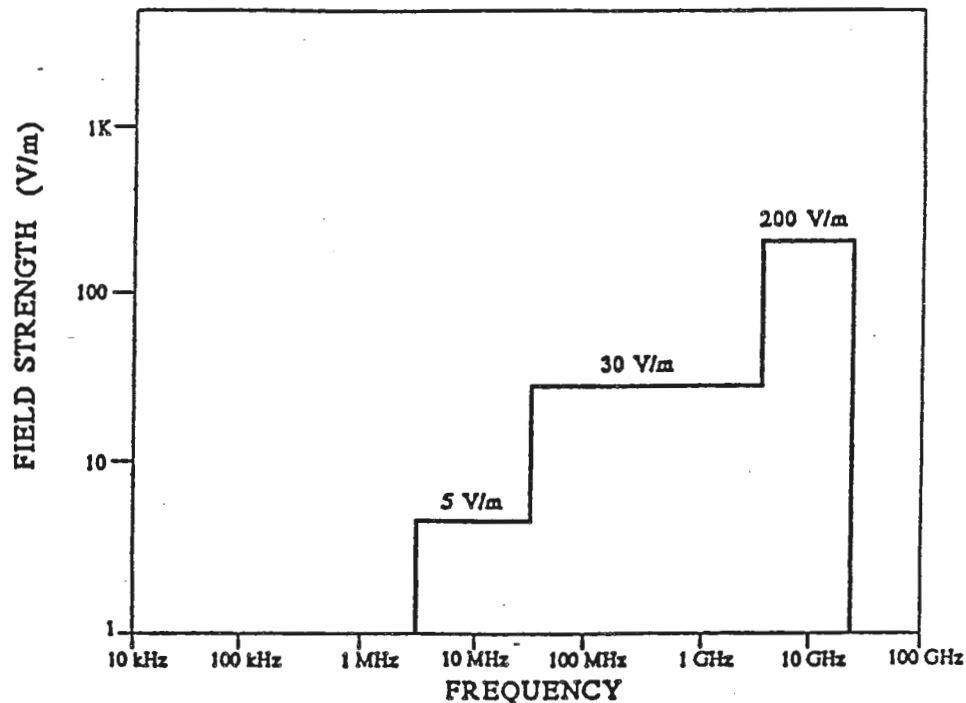


FIGURE 2.3-4. PREDICTED MAXIMUM AVERAGE FIELD STRENGTH LEVELS FROM CIVILIAN AIRPORT MOBILE EMITTERS

2.4. Air-to-Air Environment

In the air-to-air environment, normally, only military aircraft are considered. An interceptor aircraft may approach as close as 50 ft to an unidentified commercial aircraft while trying to establish its identity. Very high powered military Electronic Counter Measures (ECM) systems will be assumed as not operating at this close range. Non-interceptor aircraft, Airborne Warning and Control System (AWACS) or ECM aircraft would not intentionally approach closer than 500 ft but could be operating high-powered emitters. Predicted peak and average field strengths are calculated and plotted for the interceptor and noninterceptor aircraft in figures 2.4-1 through 2.4-4.

2.5. Ground Environment

The ground environment is the largest of the four environments since it includes all fixed and ground mobile emitters not located on an airport. These emitters are located at a distance greater than 5 miles from the center of the airport. The ICAO aircraft minimum obstruction clearance is 500 ft above ground level. The 500-ft minimum distance was used for field strength calculations; therefore, this places the aircraft within the near field of the emitting antenna. The ground equipment includes those types in table 2.5-1.

Predicted maximum peak and average field strengths are calculated and plotted for the ground environment equipment in figures 2.5-1 and 2.5-2.

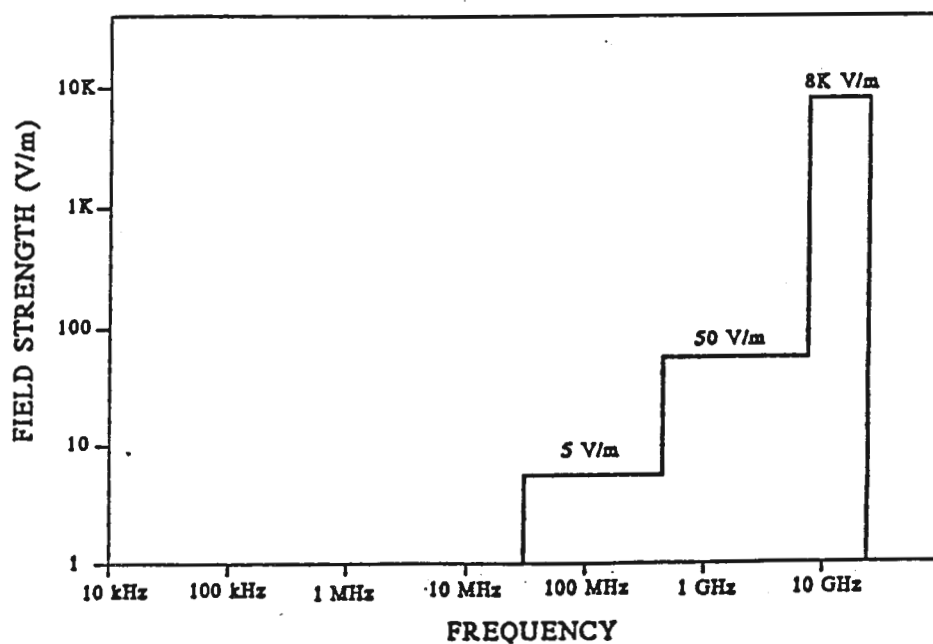


FIGURE 2.4-1. PREDICTED MAXIMUM PEAK FIELD STRENGTH LEVELS FROM INTERCEPTOR AIRCRAFT EMITTERS

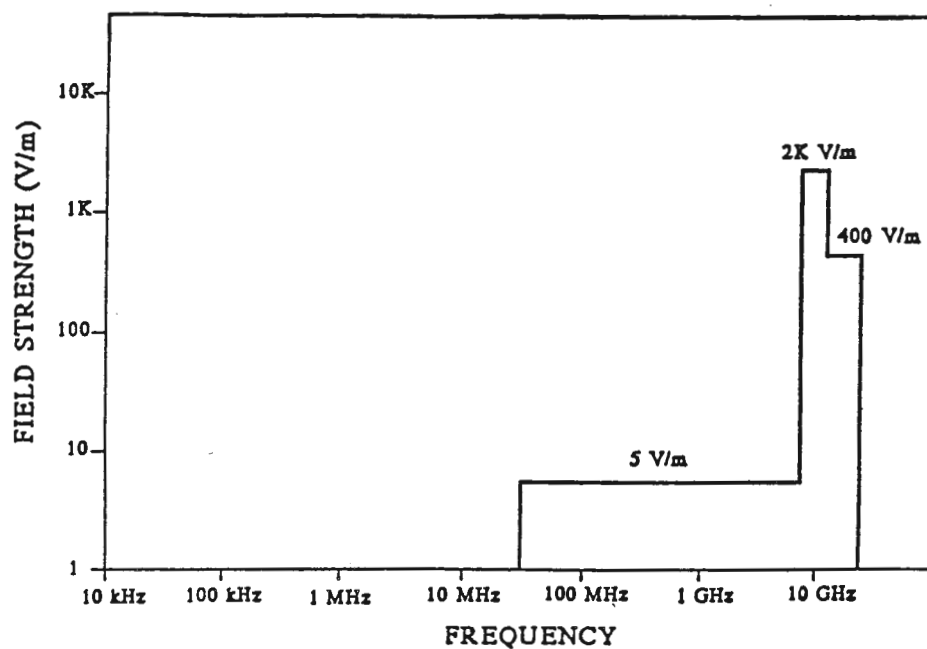


FIGURE 2.4-2. PREDICTED MAXIMUM AVERAGE FIELD STRENGTH LEVELS FROM INTERCEPTOR AIRCRAFT EMITTERS

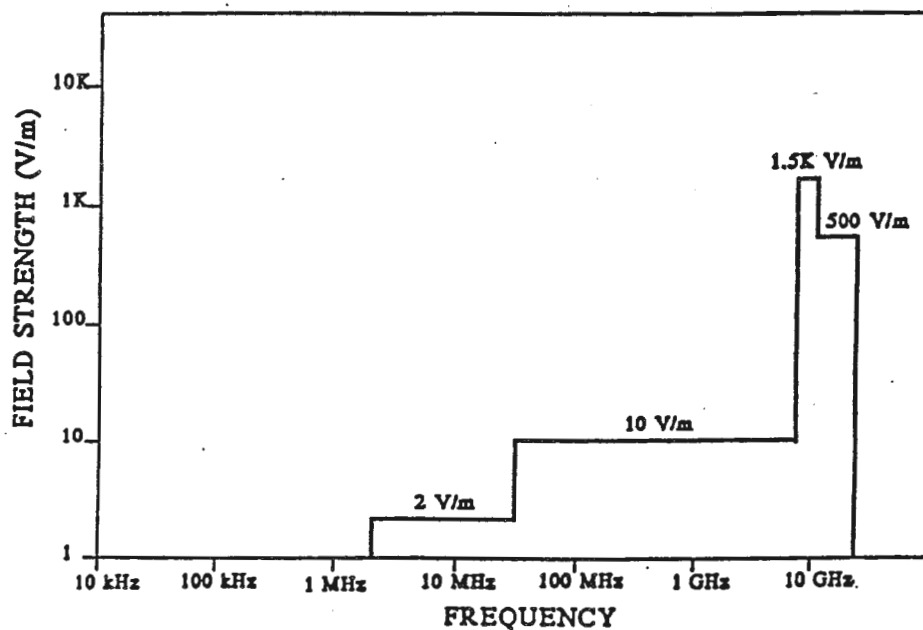


FIGURE 2.4-3. PREDICTED MAXIMUM PEAK FIELD STRENGTH LEVELS FROM MILITARY NON-INTERCEPTOR AIRCRAFT

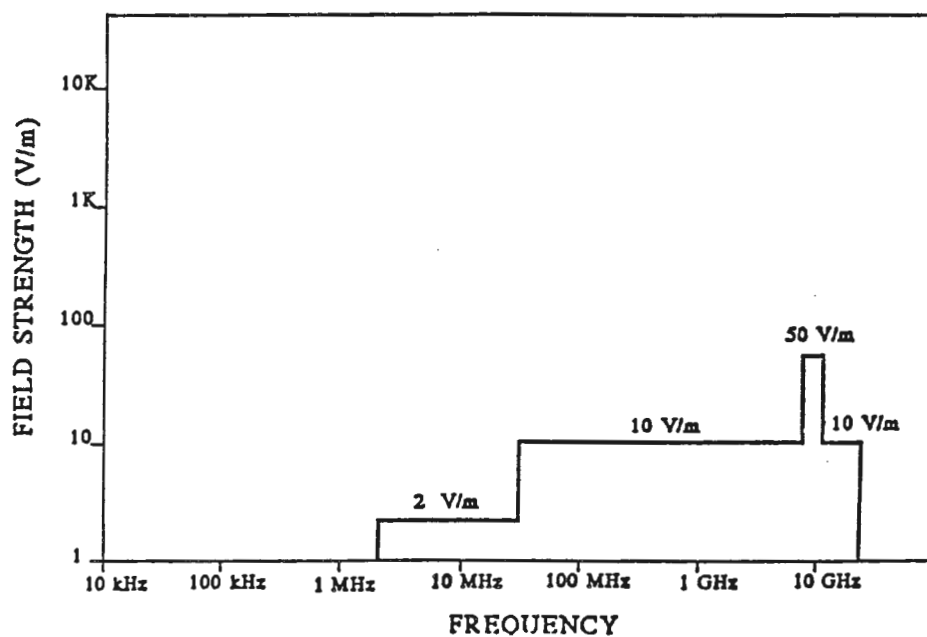


FIGURE 2.4-4. PREDICTED MAXIMUM AVERAGE FIELD STRENGTH LEVELS FROM MILITARY NON-INTERCEPTOR AIRCRAFT

TABLE 2.5-1. GROUND EQUIPMENT

Radars
Earth Terminals
Sounders
Troposcatter Communications
Commercial AM, FM, and TV Broadcast Transmitter
Test and Training Equipment

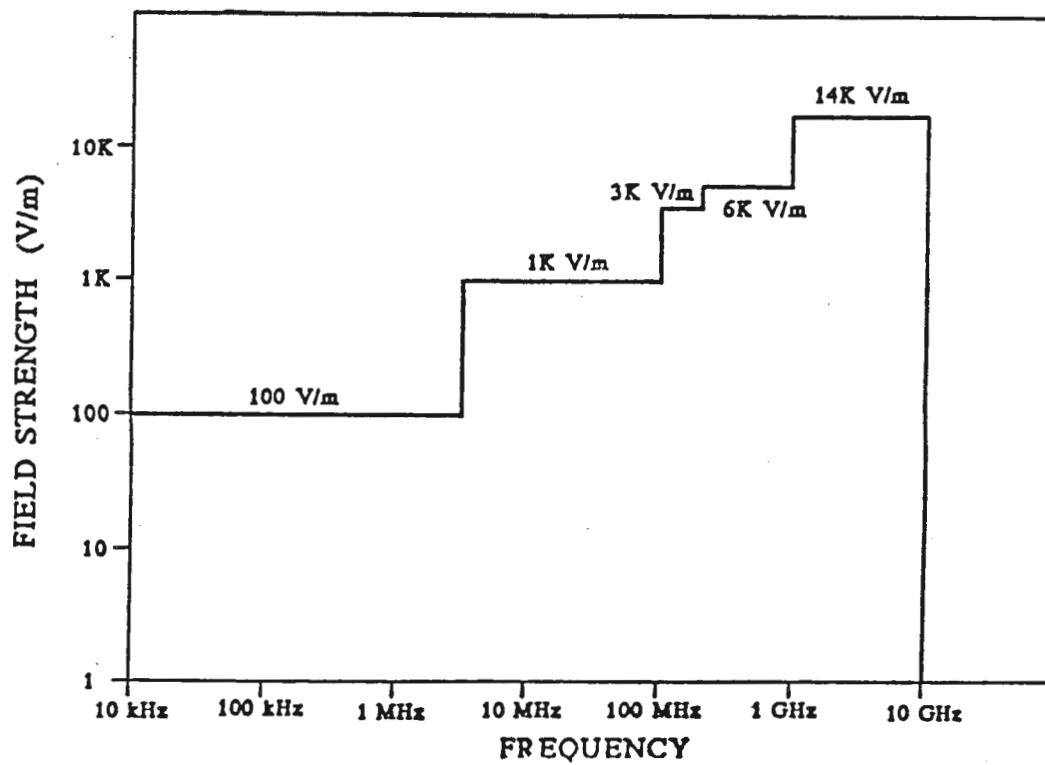


FIGURE 2.5-1. PREDICTED MAXIMUM PEAK FIELD STRENGTH LEVELS FROM GROUND EMITTERS.

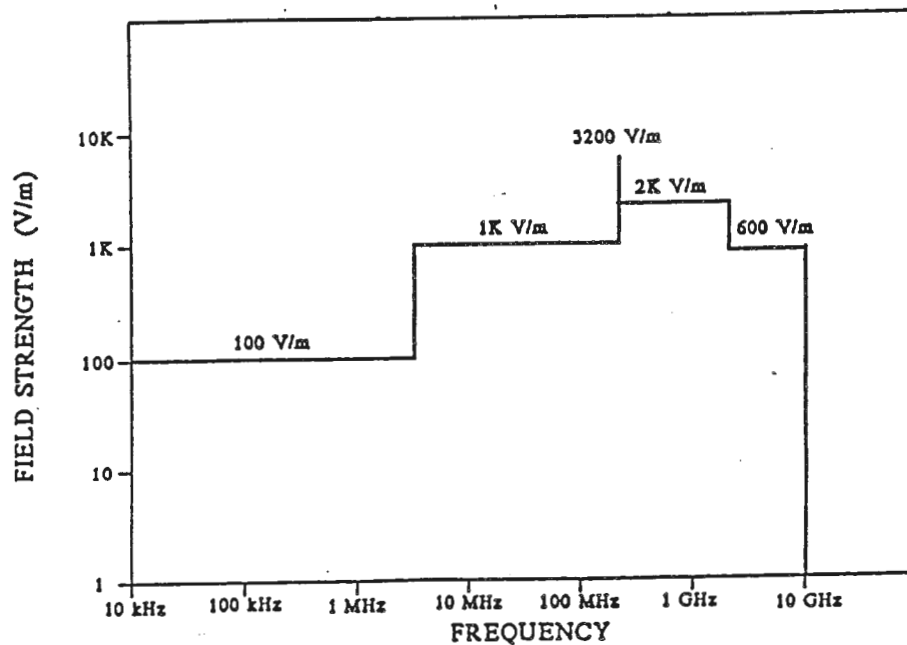


FIGURE 2.5-2. PREDICTED MAXIMUM AVERAGE FIELD STRENGTH LEVELS FROM GROUND EMITTERS

2.6. Shipboard Environment

A number of U.S. and international airports are located near harbors, shipping lanes, etc.; therefore, approach to landing and takeoff paths may cross over commercial and/or military ships, i.e., oil transports, cruise liners, aircraft carriers, and support vessels. Some of these ships, i.e., aircraft carriers, contain very high-powered emitters at high frequencies. A minimum distance of 1000 ft was considered for the field strength calculations. The shipboard equipment includes those types listed in table 2.6-1.

Predicted maximum peak and average field strengths are calculated and plotted for shipboard equipment in figures 2.6-1 and 2.6-2.

TABLE 2.6-1. SHIPBOARD EQUIPMENT

Search Radars Track Radars IFF/SIF HF and UHF Communications

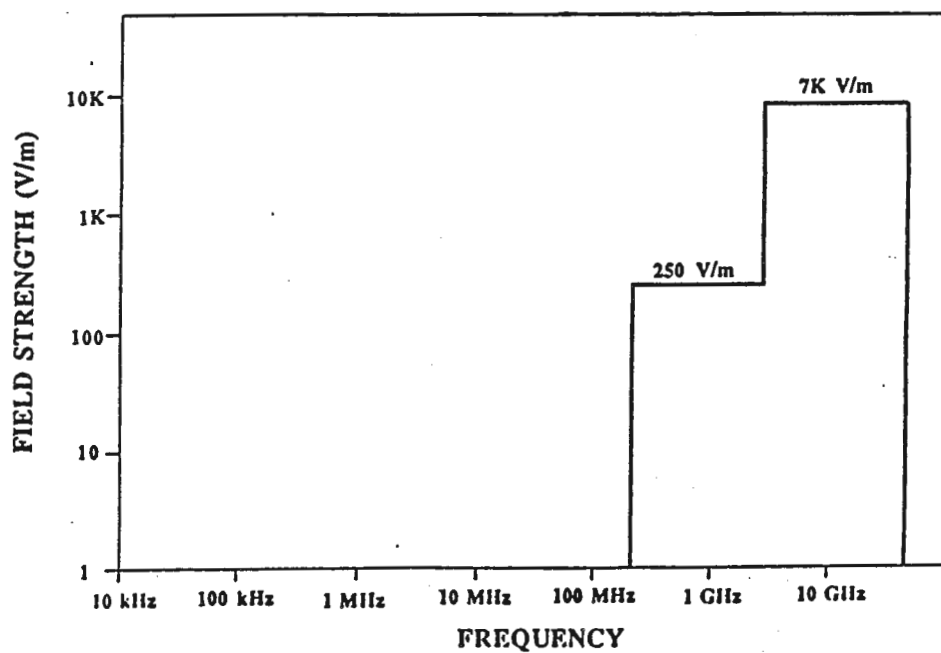


FIGURE 2.6-1. PREDICTED MAXIMUM PEAK FIELD STRENGTH LEVELS FROM SHIPBOARD EMITTERS

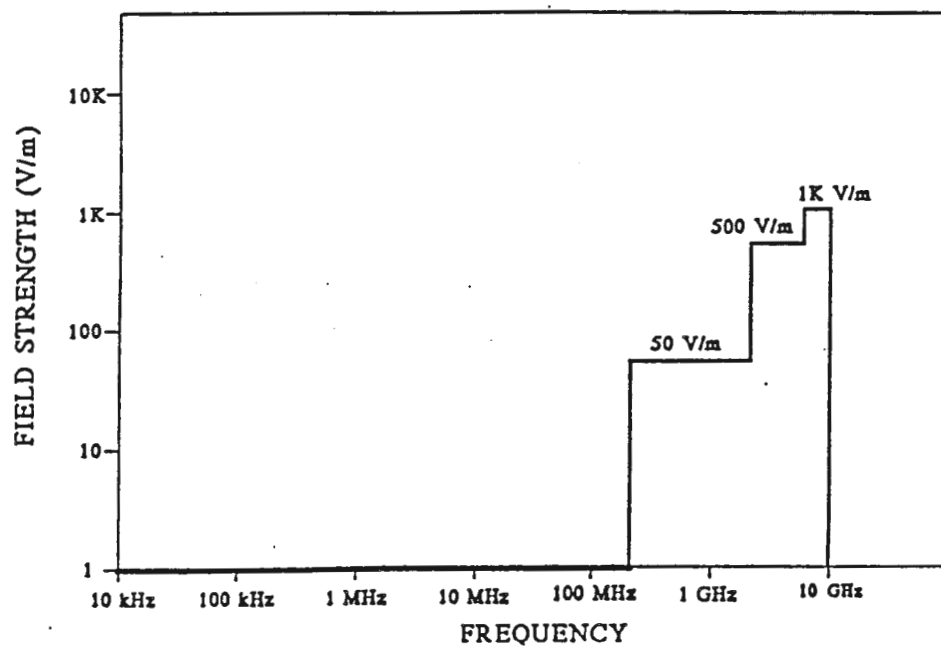


FIGURE 2.6-2. PREDICTED MAXIMUM AVERAGE FIELD STRENGTH LEVELS FROM SHIPBOARD EMITTERS

2.7. Composite of United States Emitters

The airport, air-to-air, ground, and shipboard peak and average field strengths were integrated into electric field strengths for the total U.S. environment. The U.S. environment includes both civil and military emitters. Predicted maximum peak and average field strengths are calculated and plotted for composite U.S. emitters in figures 2.7-1 and 2.7-2.

2.8. Composite of United States Compared to NATO STANAG 3614AE

Tracking of the total U.S. environment and the North Atlantic Treaty Organization (NATO) STANAG is remarkably close. The U.S. environment is not as granular as the NATO environments. In the early FAA and ECAC analysis, the charts were more granular, and the decision was made to present the data in a fashion as not to give enough information that would compromise the classified military emitters. Figures 2.7-1 and 2.7-2 are more granular than figure 2.8-1.

2.9. Takeoff and Landing Scenario

For minimum distance determination on and around airports, the worst-case condition is the engine out situation for a two-engine aircraft. The aircraft performance and FAA criteria dictates that the aircraft climb to and maintain a minimum altitude prior to initiating a go-around. Figure 2.9-1 shows the FAA criteria and the minimum altitudes that the aircraft must maintain.

The minimum distance between an aircraft and an emitter is dependent on the location of the emitter and the altitude of the aircraft. Where appropriate, the height of the emitting antenna above ground should be deducted from the aircraft minimum altitude to obtain the minimum distance from that specific emitter.

Based on the above criteria, minimum distances for field strength calculations can be estimated as follows:

- As expressed in the airport analysis above, a minimum distance of 250 ft is described near airport runways or taxiways.
- In an engine-out situation, the aircraft must be at an altitude of 35 ft at threshold (end of runway), climbing at a minimum gradient of 2.4 percent.

This places the aircraft at an altitude of 300 ft at a distance of 1.84 nmi from the end of the runway.

- The aircraft will be at an altitude of 300 ft to a distance of 4.5 nmi from the end of the runway.

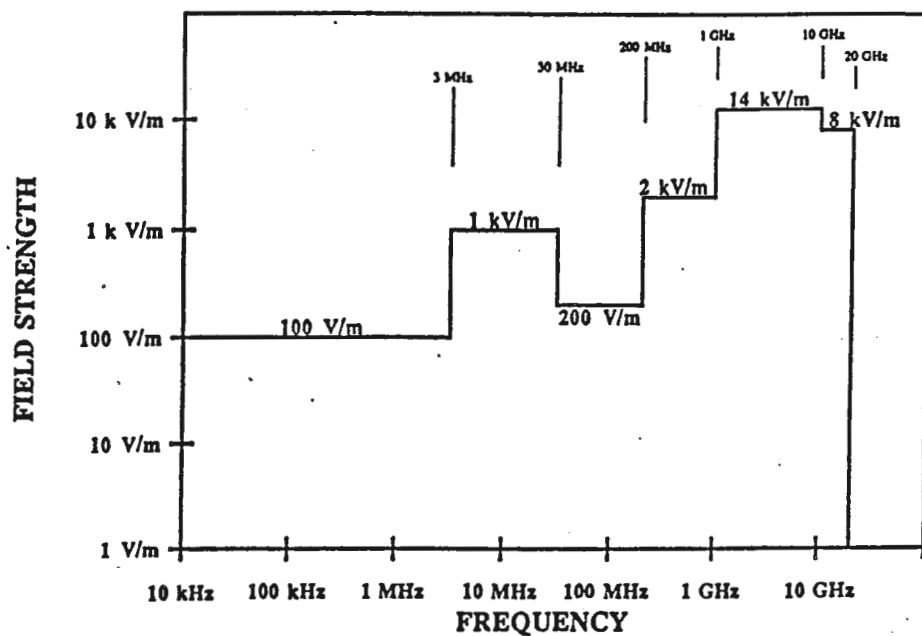


FIGURE 2.7-1. PREDICTED MAXIMUM PEAK FIELD STRENGTH LEVELS COMPOSITE OF UNITED STATES EMITTERS

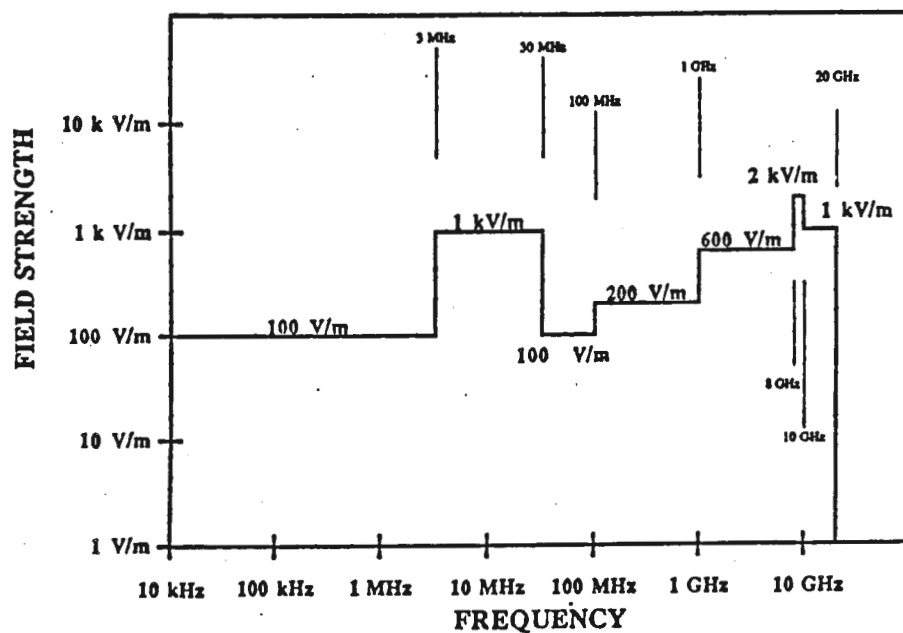


FIGURE 2.7-2. PREDICTED MAXIMUM AVERAGE FIELD STRENGTH LEVELS COMPOSITE OF UNITED STATES EMITTERS

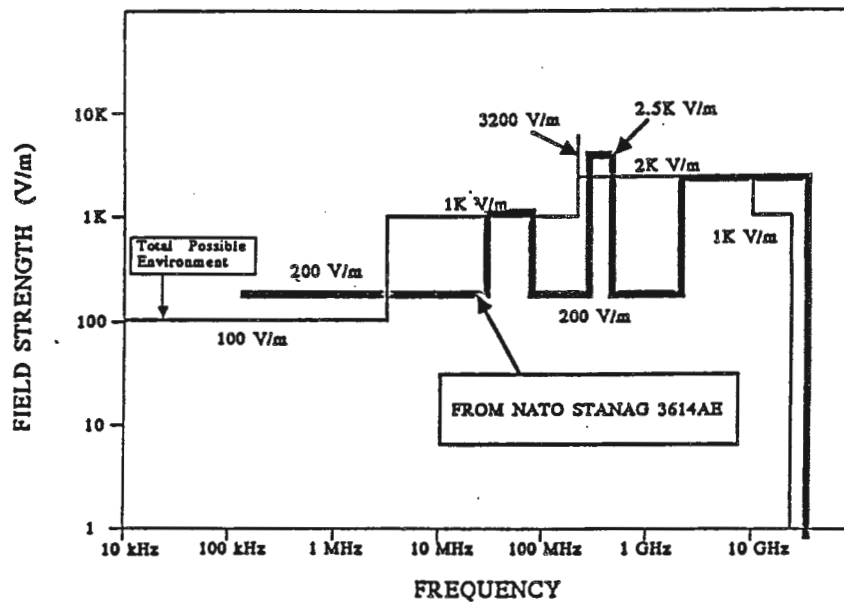


FIGURE 2.8-1. PREDICTED MAXIMUM AVERAGE FIELD STRENGTH LEVELS UNITED STATES COMPOSITE VS NATO STANAG 3614AE

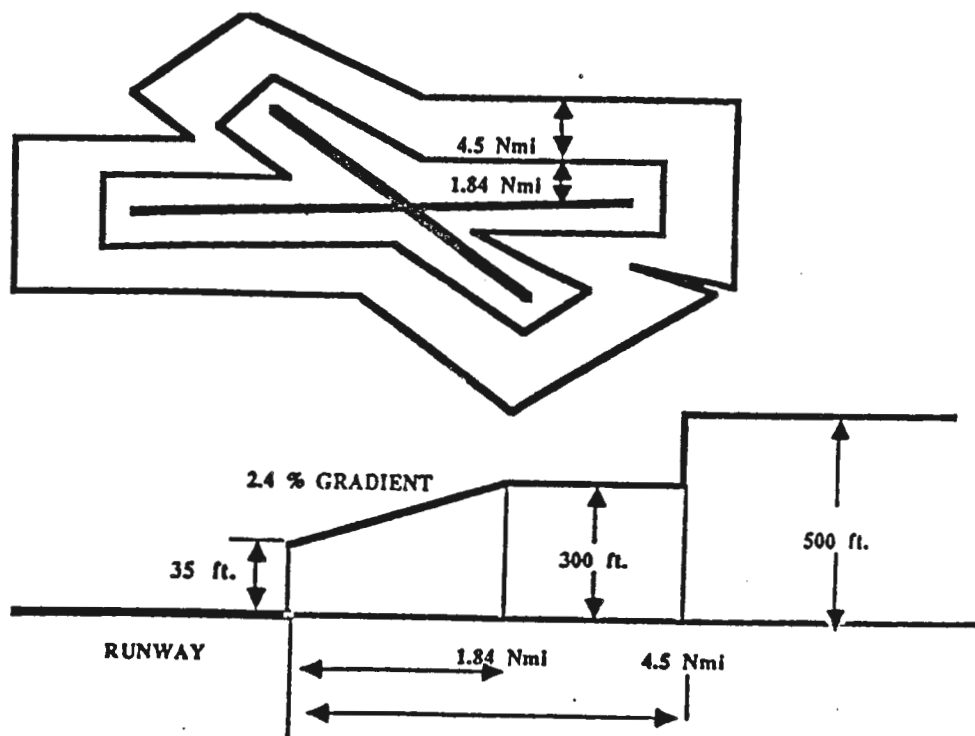


FIGURE 2.9-1. CRITERIA FOR SUSCEPTIBILITY THRESHOLD

- At 4.5 nmi, the minimum altitude is specified as 500 ft.

Note: In the shipboard environment, the minimum distance was considered to be 1000 ft, but the airport takeoff and landing scenario should be considered for those airports near the shipping lanes, which would reduce the minimum distance from 1000 ft to 300 ft. Recalculation of the peak and average field strengths in the shipboard environment is in progress.

The six airports that were examined in the United States are Baltimore, Honolulu, New York (Kennedy), Los Angeles, Seattle, and Denver (Stapleton). They were selected as being representative and typical of U.S. airports with certain unique local environments, industry, seaports, etc.

It is possible that the field strength levels may be higher at other airports than for the six airports selected, but there is a confidence level for those data being representative.

Peak and average field strengths were calculated for each airport, resulting in 12 graphs. Two composite graphs were developed from the 12 individual graphs. They are compared and overlaid in figures 2.9-2 and 2.9-3.

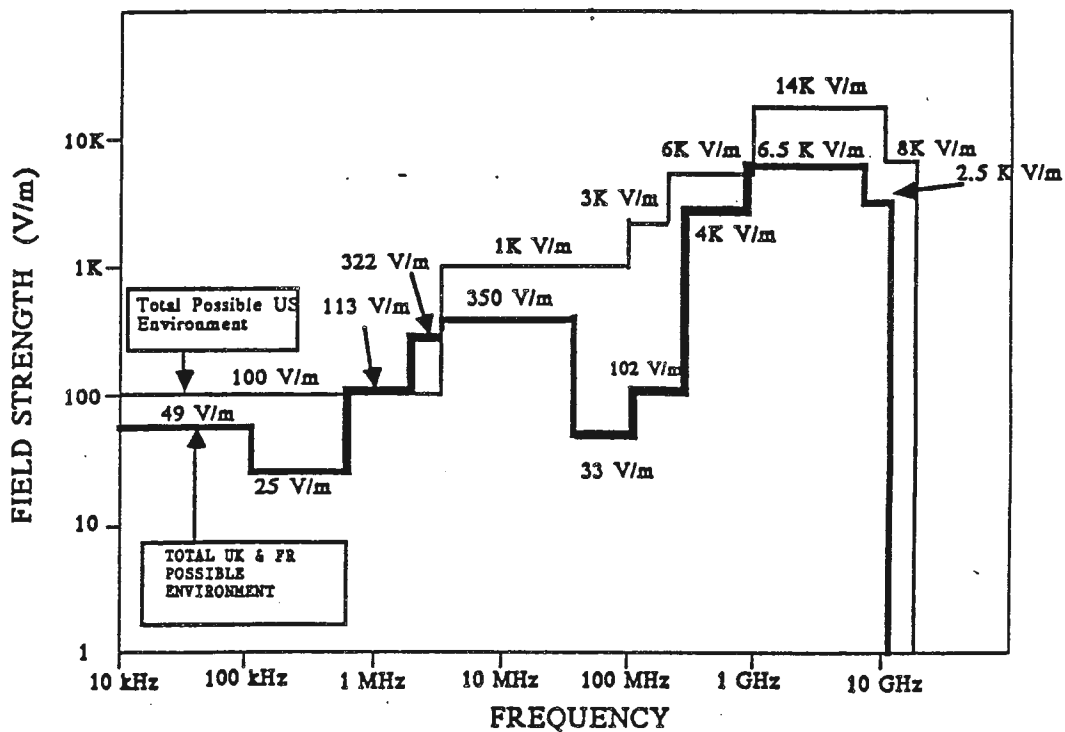


FIGURE 2.9-2. PREDICTED MAXIMUM PEAK FIELD STRENGTH LEVELS COMPARISON UNITED STATES COMPOSITE WITH SIX AIRPORT COMPOSITE

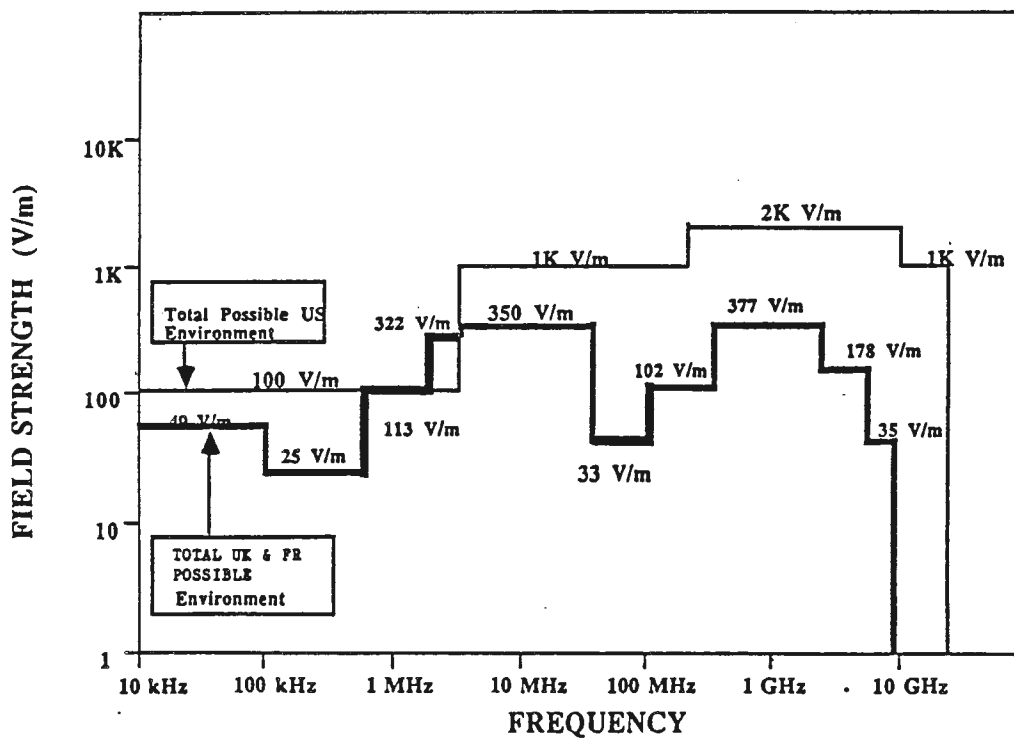


FIGURE 2.9-3. PREDICTED MAXIMUM AVERAGE FIELD STRENGTH LEVELS COMPARISON UNITED STATES COMPOSITE WITH SIX AIRPORT COMPOSITE

3. EME SURVEY AND ANALYSIS - WORLDWIDE

3.1. Guidelines

The CAA, U.K., initiated requests for other European countries to participate in this HERF survey and to study each country's own emitter environments. Initially, only the United Kingdom and France responded and aggressively supported this total effort. Their results are presented herein. In September 1987, 27 countries were invited to support and participate in an International Conference in Washington, D.C. Ten countries attended including the United States, United Kingdom, and France. Canada and Sweden were the only other countries in which preliminary surveys were conducted. Sweden's early results were presented at this conference by the Swedish representative.

In April of 1988, a similar international conference was conducted in Brighton, England. The same 27 countries were invited to participate and present their survey data. Australia and West Germany provided an overview of their preliminary data.

British, French, and Swedish technical teams used the FAA and ECAC format as guidelines for their survey and analysis. France, associated with its survey and analysis, conducted an extensive airborne (rotorcraft) measurement program for selected emitters. The U.S. measurement and validation effort was conducted by the FAA, ECAC, and Ohio University. The summary of these efforts will be described later in the validation study currently in progress.

3.2. Results

Results are presented in a comparison of the American worst-case total environment to the composite of the British and French survey data. Figures 3.2-1 and 3.2-2 show the predicted maximum peak and average field strength levels for this comparison.

Figures 3.2-3 and 3.2-4 show composite worst-case environment for all three countries.

Figures 3.2-5 and 3.2-6 show U.S. airports (Baltimore, Honolulu, Kennedy, Los Angeles, Seattle, Denver (Stapleton)) integrated with the British airports (Heathrow, Manchester, Glasgow) and the French airports (Orly, Toulouse, Lyon, Nice).

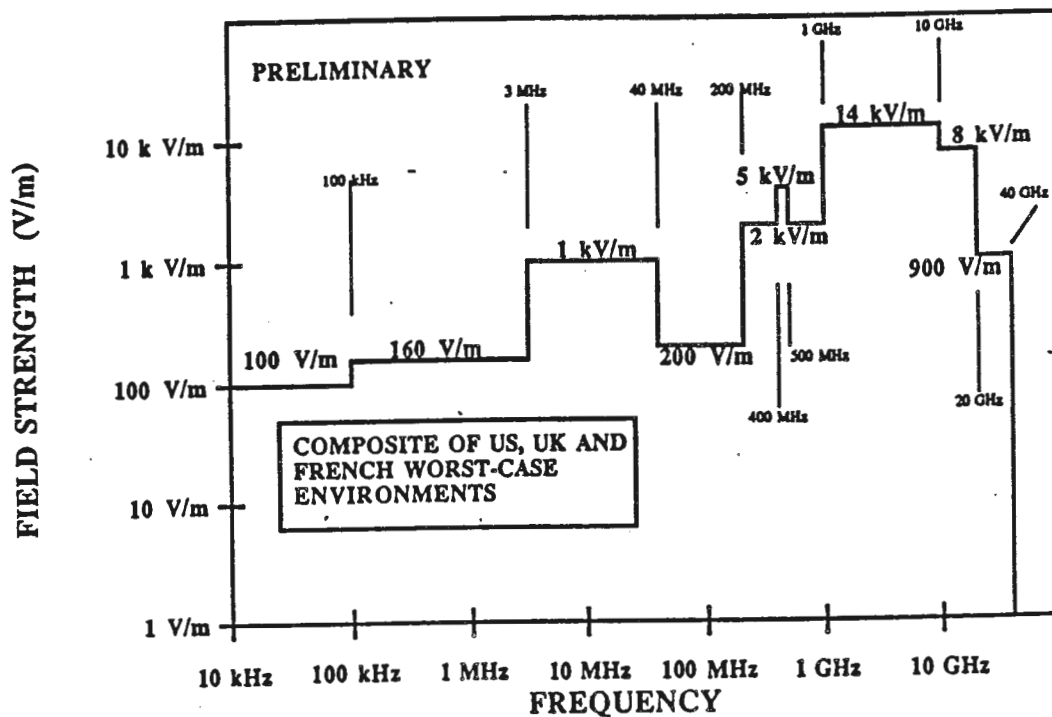


FIGURE 3.2-1. PREDICTED MAXIMUM PEAK FIELD STRENGTH LEVELS COMPARISON OF UNITED STATES COMPOSITE WITH UNITED KINGDOM AND FRANCE COMPOSITE

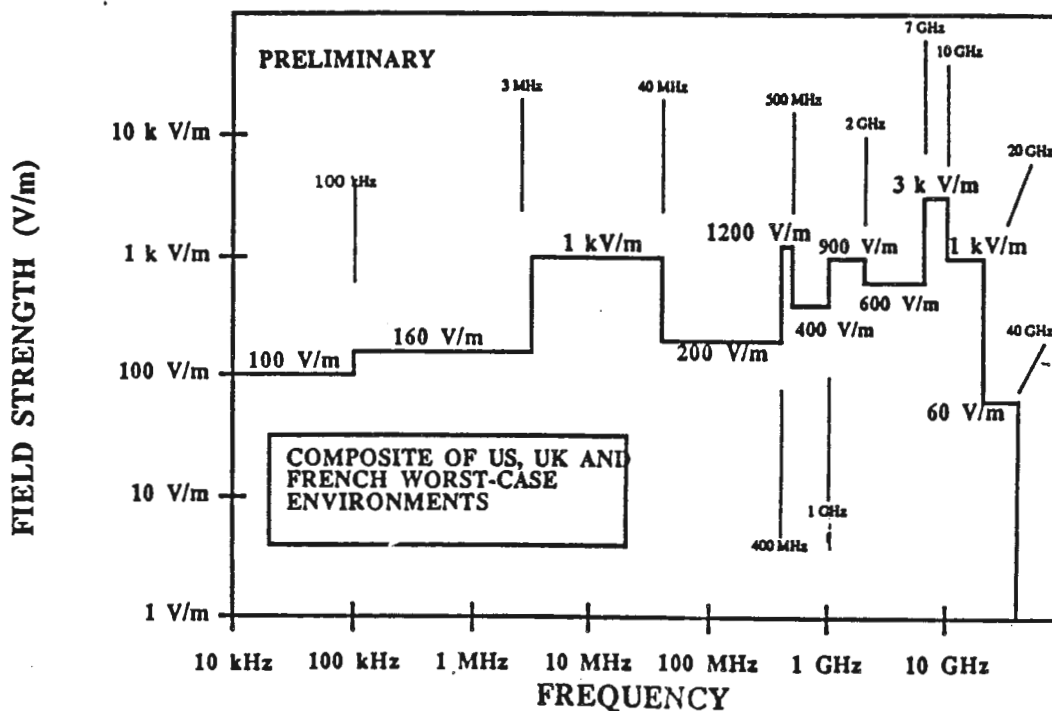


FIGURE 3.2-2. PREDICTED MAXIMUM AVERAGE FIELD STRENGTH LEVELS COMPARISON OF UNITED STATES COMPOSITE WITH UNITED KINGDOM AND FRANCE COMPOSITE

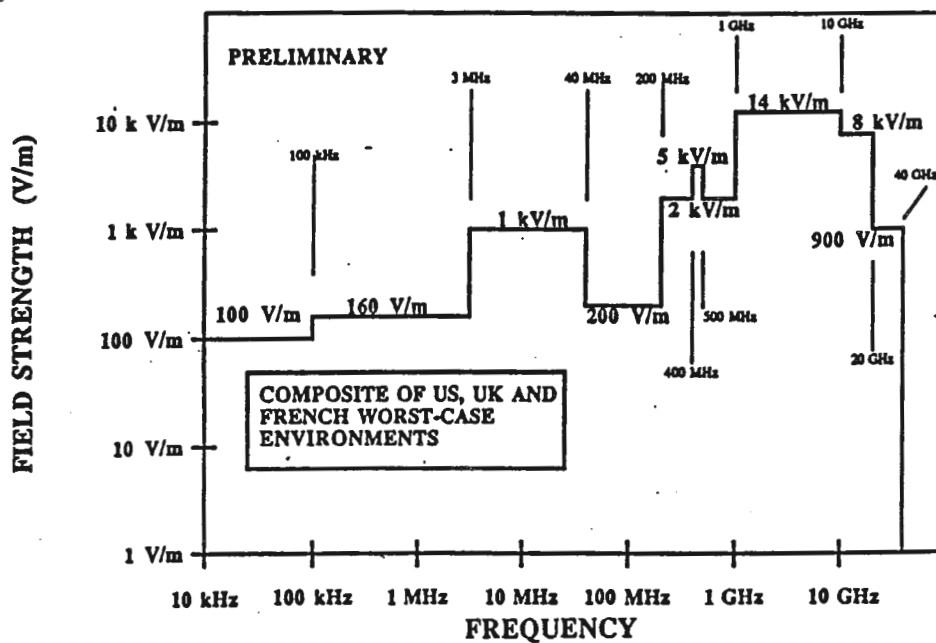


FIGURE 3.2-3. PREDICTED MAXIMUM PEAK FIELD STRENGTH LEVELS WORLDWIDE WORST-CASE ENVIRONMENT

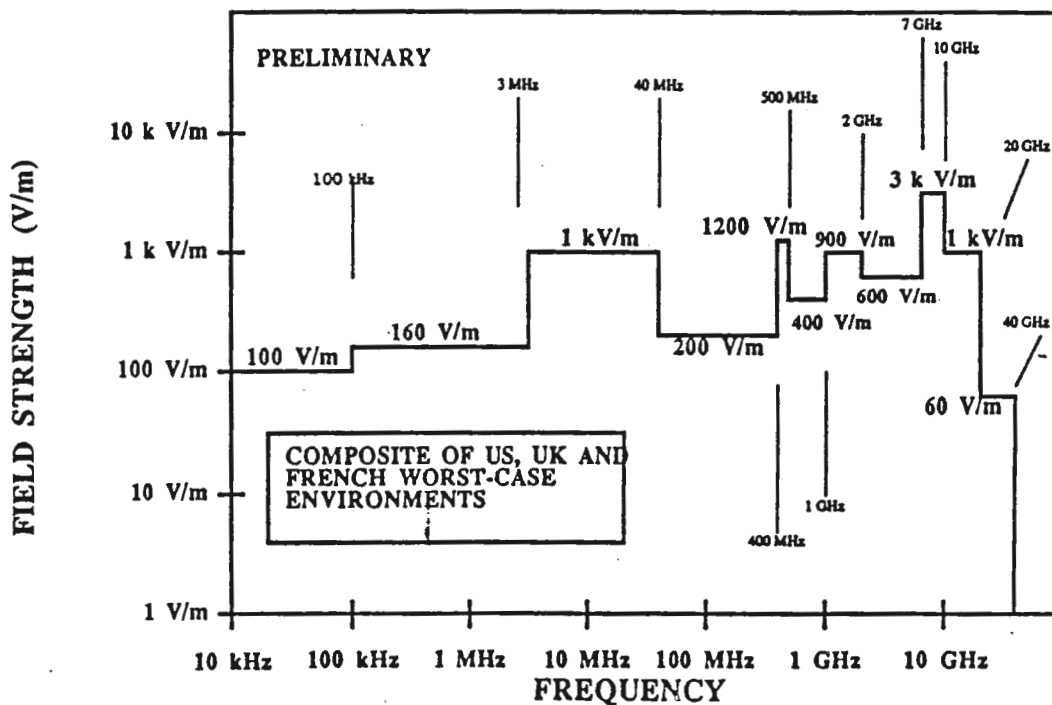


FIGURE 3.2-4. PREDICTED MAXIMUM AVERAGE FIELD STRENGTH LEVELS WORLDWIDE WORST-CASE ENVIRONMENT

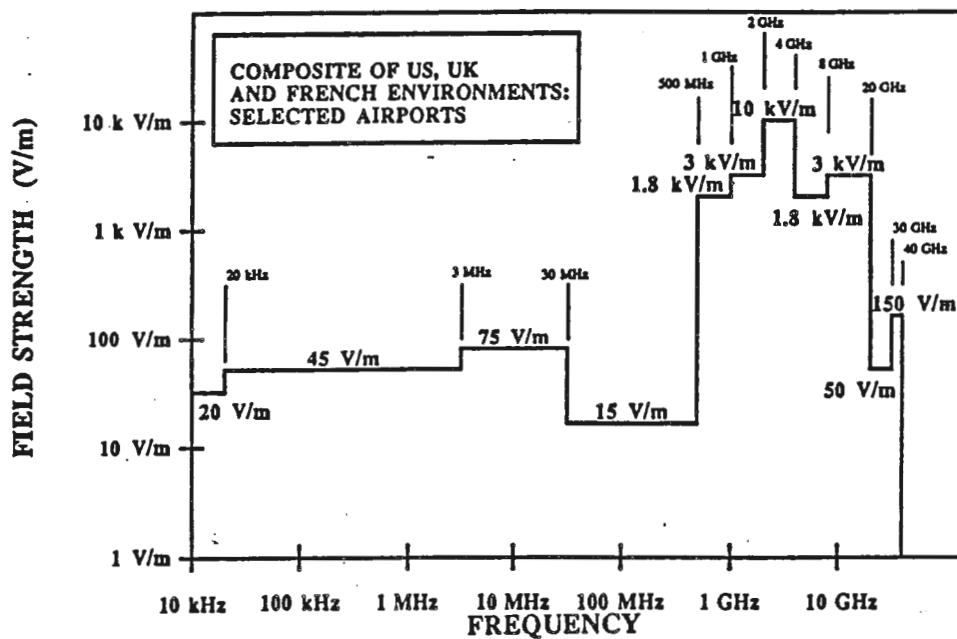


FIGURE 3.2-5. PREDICTED MAXIMUM PEAK FIELD STRENGTH LEVELS COMPOSITE OF INTERNATIONAL ENVIRONMENTS

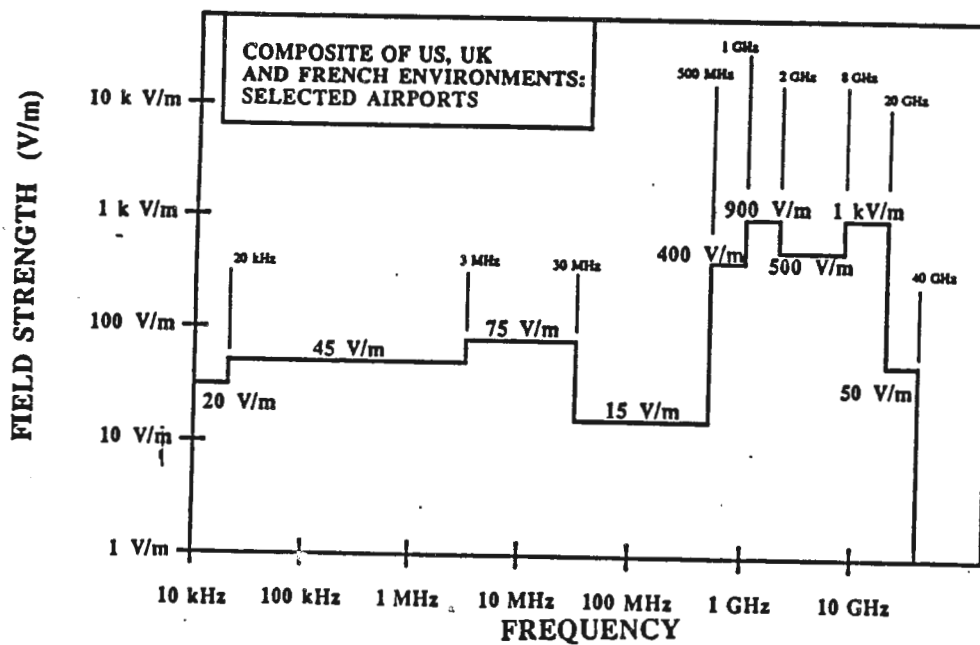


FIGURE 3.2-6. PREDICTED MAXIMUM AVERAGE FIELD STRENGTH LEVELS COMPOSITE OF INTERNATIONAL ENVIRONMENTS

4. COMMENTS

Based on the initial survey and analysis of our current environment, HERF fields are indeed a threat to state-of-the-art civil and military aircraft being placed into service throughout the world. Modern aircraft with composite construction, VLSI computers, and electronic devices for flight-critical functions have greatly magnified the threat of EMEs to safe flight operations.

The FAA, in concert with the DOD, U.K. CAA, French DGAC, and their European counterparts, are aggressively pursuing the research to meet this challenge. Following the survey and analysis, a validation phase of selected emitters will be conducted by Ohio University and ECAC. The guidelines established will promote uniformity of results from other members of the international aviation community so that all results can be combined and used to develop an international susceptibility standard.

RTCA was solicited by the FAA to establish a HERF subcommittee to update RTCA DO-160B, Section 20 to address the FAA EME as developed. A HERF Working Group was established under the RTCA Special Committee (SC-135) to review the equipment radiation susceptibility levels currently in DO-160B (which are inadequate for the FAA HERF environment) and then establish an industry-wide aircraft requirement for conducted and radiated susceptibility.

The FAA solicited the SAE-AE4 Committee for the development of procedures and guidance material, relative to HERF fields, which could be used during the aircraft certification process. Based on this request, SAE-AE4R, Aircraft Radiated Environment Subcommittee, was formed. The goal of the subcommittee was to draft an AC and attendant user's manual for guidance and information on HERF use in commercial aircraft.

SAE-AE4R subcommittee established three subcommittees: a Data Accuracy Subcommittee that will look at the exterior electromagnetic RF environment, a Design Approach Subcommittee that will develop design guidance for airframe manufacturers for the defined EME, and a Test and Analysis Methods subcommittee that will focus on validation techniques.

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APPENDIX A

GUIDELINES FOR DEVELOPING MAXIMUM PEAK AND AVERAGE FIELD
STRENGTH ENVELOPE GRAPHS FOR AIRCRAFT

BY ALEXANDER GROSS
OF THE
DOD ELECTROMAGNETIC COMPATIBILITY ANALYSIS CENTER
ON BEHALF OF
THE DEPARTMENT OF TRANSPORTATION
FEDERAL AVIATION ADMINISTRATION

SEPTEMBER 1987



APPENDIX A - GUIDELINES FOR DEVELOPING MAXIMUM PEAK AND AVERAGE FIELD STRENGTH ENVELOPE GRAPHS FOR AIRCRAFT

BACKGROUND

The U.S. Federal Aviation Administration (FAA) is concerned with the safety and integrity of new technology transport-category airplanes with fully flight-critical and flight-essential systems. One of their concerns is the protection of these aircraft from electromagnetic radiation (EMR) produced by external high-energy ground and airborne radio-frequency (RF) emitters. The FAA is in the process of developing proposed requirements and standards to address this concern.

As part of this effort, the Electromagnetic Compatibility Analysis Center (ECAC) was requested to predict the maximum EMR levels at aircraft in the United States for two specified scenarios: a worst-case total U.S. scenario and, for comparison, a take-off and landing scenario at specific airports. The worst-case scenario considered the maximum EMR levels at an aircraft exposed to any emitter in the U.S. The second scenario considered the EMR that would be experienced by an aircraft on take-off and landing at six major U.S. airports.

Those results of this study will be used by the FAA in the development of U.S. requirements and standards relating to the effects of transmitted RF energy on flight-critical control and avionics systems aboard modern aircraft. Since aircraft are required to fly throughout the world, an international RF susceptibility standard is also desired. It is suggested that maximum EMR level predictions should be made by other members of the international aviation community to support an international standard. In an effort to standardize the prediction techniques, ECAC has been requested by the FAA to document the techniques that were used by ECAC in the form of guidelines for predicting maximum EMR levels at aircraft exposed to high-powered emitters.

OBJECTIVE

The objective of this paper is to present those guidelines that were used by ECAC in the calculation of maximum peak and average electric field strengths at transport-category aircraft with fully electronic flight-critical and flight-essential systems as a result of ground and airborne RF emitters. These guidelines are presented to promote uniformity of results from other countries so that all inputs can be combined.

APPROACH

The guidelines for calculating peak and average field strengths are presented as step-by-step instructions starting with the data on the emitters in the electromagnetic environment (EME) and ending with an envelope graph representing maximum field strengths. The methods for calculating field strengths in the worst-case in the worst-case scenario and in the take-off and landing scenario are defined and discussed. Equations used to estimate missing technical parameters, to calculate the near field of an antenna, and to calculate the field strengths are provided. Simplifying assumptions are made and, where

appropriate, justifications are shown. The minimum distances used in the U.S. environmental analysis are presented. The results obtained by following these guidelines will be peak and average field strengths, at worst-case distances between emitters in the EME and aircraft.

PROCEDURES

GENERAL ASSUMPTIONS

The calculation of maximum EMR is based on several assumptions. These are:

1. The minimum allowable separation distance between civil aircraft and an RF emitter is used to calculate the maximum electric field strength at the aircraft. The minimum distance is a worst-case distance that is determined by the scenario being considered and by the environment within that scenario.
2. The civil aircraft is being continuously illuminated by the mainbeam of the emitting antenna.
3. All calculations consider only free-space loss between the emitter and the aircraft. Above 30 MHz, no consideration is given to reflecting surfaces or obstacles that may exist in the vicinity of either the emitter or the aircraft. Below 30 MHz, only the ground reflected wave is considered and assumed to combine constructively with the direct wave, effectively doubling the field strength.
4. Field strength calculations for aircraft locations within the near field of the emitting antenna are made using the near-field correction factors discussed later in this paper.
5. The resulting field strength is independent of modulation or any other equipment-related factor.
6. The cumulative effect of two or more emitters illuminating the aircraft is not considered.

DETERMINATION OF THE MINIMUM DISTANCES FOR THE WORST-CASE TOTAL ENVIRONMENT SCENARIO

The total environment can be separated into four parts: the airport environment, the aircraft-to-aircraft environment, the ground environment, and the shipboard environment. While on the ground, the aircraft will be exposed to emitters in the airport environment. While in flight, the aircraft will encounter emitters in the aircraft-to-aircraft, ground, and shipboard environments. The minimum distance for an aircraft in one of these environments is defined as the closest the aircraft can approach an emitter in that environment. Each environment and its minimum separation distance will be discussed in turn. The minimum distances are summarized in table A-1.

TABLE A-1. SUMMARY OF MINIMUM DISTANCES FOR THE WORST-CASE SCENARIO

ENVIRONMENT	MINIMUM DISTANCE
FIXED EMITTERS ON AIRPORT	250 FT
MOBILE EMITTERS ON AIRPORT	50 FT
INTERCEPTOR AIRCRAFT EMITTERS	50 FT
NON-INTERCEPTOR AIRCRAFT EMITTERS	500 FT
GENERAL GROUND EMITTERS	500 FT
SHIPBOARD EMITTERS	300 FT

The Airport Environment

The airport environment consists of fixed and mobile emitters located on an airport. For this discussion, the airport includes an area defined by a five-nautical-mile radius from the geographic center of the airport. The minimum distance at which field strengths are to be calculated differs for fixed and mobile emitters.

Fixed Emitters. The fixed airport environment consists of all fixed emitters located on an airport. In general, the minimum distance between a fixed emitter and the taxiway or runway is defined by regulations. This minimum distance is used as the minimum separation distance between the aircraft and the fixed airport environment.

Table A-2 lists the typical fixed emitters in the U.S. airport environment. Not all of these emitters are located on every airport, but they are located on some airport in the United States. Based on FAA regulations, fixed emitters are located no closer to an airport runway or taxiway than 250 ft. A minimum distance of 250 ft, therefore, was used in the calculation of the field strengths from the fixed emitters on a U.S. airport.

Mobile Emitters. The mobile airport environment consists of all emitters that are not in a fixed location. Emitters, such as commercial VHF radios on ground support vehicles, are part of the mobile environment. Emitters, such as HF and UHF communications, TACAN, Doppler Navigation Radars, radio altimeters, aircraft weather radars, and ATCRBS beacons, aboard other aircraft are also included in the mobile airport environment. These aircraft emitters may be operating while the aircraft is on the ground and waiting to take off.

A minimum distance of 50 ft was used in calculating field strengths from mobile emitters on the airports in the U.S. analysis.

TABLE A-2. TYPICAL FIXED AIRPORT ENVIRONMENTAL EMITTERS

MARKER BEACON
LOCALIZER
VOR
GLIDE SLOPE
GROUND CONTROLLED APPROACH RADAR
DISTANCE MEASURING EQUIPMENT
TACAN
MICROWAVE LANDING SYSTEM
AIRPORT SURVEILLANCE RADAR
AIR ROUTE SURVEILLANCE RADAR
WEATHER RADAR
ATCRBS INTERROGATOR
VHF AND UHF COMMUNICATION AND TELEMETRY

Aircraft-to-Aircraft Environment

This environment consists of emitters onboard other aircraft at distances much closer than normal in-flight separation. Note that the emitters on other civil aircraft were included in the mobile airport environment. Therefore, only military aircraft need to be considered in this environment.

The military aircraft fall into two categories: interceptor and noninterceptor aircraft. An interceptor aircraft may approach as close as 50 ft to an unidentified commercial aircraft while attempting to establish the identity of, and provide instructions to, the aircraft. It can be assumed that very high powered military emitters such as electronic countermeasure (ECM) systems will not be operating at this close range. Noninterceptor aircraft would not intentionally approach closer than 500 ft from another aircraft but may be operating high powered emitters such as early warning radars and ECM systems. In the U.S. study, ECAC used these distances when calculating electric field strengths.

Ground Environment

Of the four environments, the ground environment is the largest since it includes all fixed and ground mobile emitters that are not located on airports. Emitters such as sounders, troposcatter communications, commercial broadcast, radar astronomy, land mobile, test and training equipment, radars and transmitting earth terminals are part of the ground environment.

The International Civil Aviation Organization (ICAO) aircraft minimum obstruction clearance is 500 ft above ground level. Therefore, the minimum distance

at which field strengths are calculated is 500 ft. Note that for most high power emitters, this places the aircraft within the near field of the emitting antenna.

Shipboard Environment

This environment includes emitters on commercial and military ships that may be located in harbors near airports. Some of the high-powered emitters that are included are: search and track radars, fire control radars, satellite communications, HF communications, and electronic countermeasure systems. A minimum distance of 1000 ft was used in the U.S. analysis based on the locations of U.S. harbors.

DETERMINATION OF THE MINIMUM DISTANCES FOR THE TAKEOFF AND LANDING SCENARIO

The takeoff and landing scenario is concerned with the environment on and around airports, where the aircraft is expected to be near the ground. The worst-case condition is the one-engine out situation for a two engine aircraft. In this situation, the aircraft must climb to a minimum altitude and must maintain that altitude. Figure A-1 shows the FAA criteria and the minimum altitudes that the aircraft must maintain. Based on this criteria, the minimum distances for field-strength calculations are as follows:

1. While on the airport runways or taxiways, the minimum distance is 250 ft, as discussed under the airport environment section, above.
2. The aircraft must be at an altitude of 35 ft at threshold (end of runway), climbing at a minimum gradient of 2.4%. This places the aircraft at an altitude of 300 ft at a distance of 1.84 nautical miles from the end of the runway.
3. The aircraft will be at an altitude of 300 ft to a distance of 4.5 nmi from the end of the runway.
4. At 4.5 nmi, the minimum altitude is specified as 500 ft.
5. The minimum distance between an aircraft and an emitter is dependent on the location of the emitter and the altitude at that location as given in 1 through 4 above. Where appropriate, the height of the emitting antenna above ground should be deducted from the minimum aircraft altitude to obtain the minimum distance from that specific emitter.

CALCULATION OF FIELD STRENGTHS

The conventional measure of the intensity of a radiated electromagnetic wave is a measure of the intensity of the electric field in volts per meter. (Note that the distances in the above discussion were in English units since this is common for the aviation industry. The discussion of electric field strengths will be in metric units with volts per meter being the unit used in international regulations.) The field intensity of a radiated wave in free space falls off in direct proportion to the distance from the transmitting antenna. For practical purposes free space propagation is realized if the following conditions are

fulfilled: No large obstacles intervene between the antenna and the aircraft along an optical line of sight; no alternate transmission path can be followed by a substantial fraction of the radiated energy (this is the same as Assumption 3 in the Introduction section); the intervening atmosphere has a constant index of refraction, so that no bending of the wave occurs; and the intervening atmosphere does not absorb energy from the wave at the frequency used. For this analysis, it is assumed that these conditions are fulfilled between an emitter in the environment and an aircraft.

The emitter antenna will be either an aperture, phased array or linear antenna. Phased array antennas are assumed to have similar radiating properties to an aperture antenna of the same physical dimensions. Elliptical antennas are approximated by rectangular antennas for these calculations.

Calculation of Field Strengths in the Far Field

Aperture Antennas. The power density obeys the inverse square law starting at a certain distance from the antenna where the antenna appears to be a point source. This is where the "Fraunhofer Region," or far field, begins. The distance to the far-field boundary for aperture antennas can be designated as:

$$D_f = 2 L^2 / \lambda \quad (1)$$

where:

D_f - distance to the far-field boundary, meters

L - Maximum dimension of antenna, meters

λ - wavelength, meters

In the Fraunhofer Region, the radiation is in a diverging beam shape, where the intensity is maximum at the beam center, and decreases away from the beam center as the angle of divergence increases. If the antenna is made larger or the antenna illumination is made more uniform, the radiated beam is made narrower and the beam center power density is made higher. The power density at the beam center in the far field is given by the equation:

$$P_D = \frac{PG}{4\pi d^2} \quad (2)$$

where:

P_D - power density, watts/meter²

P - transmitter power, watts

G - transmitter antenna gain, unitless

d - distance, meters

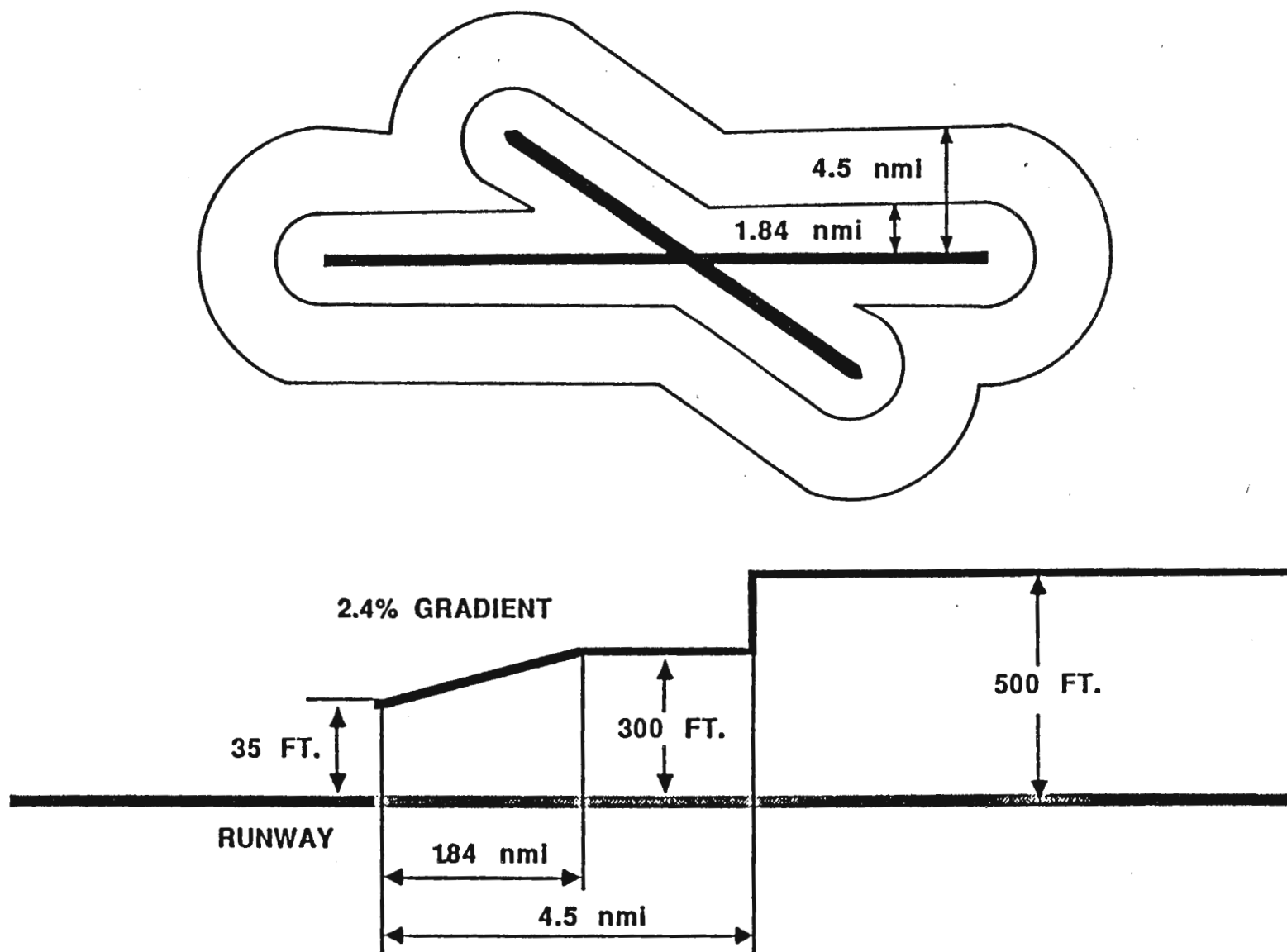


FIGURE A-1. FAA CRITERIA FOR A ONE-ENGINE OUT SITUATION FOR A TWO-ENGINE AIRPLANE

Power density is converted to field strength (FS) using the impedance of free space in Equation 3:

$$FS = \sqrt{P_d (377)} \quad (3)$$

Linear Antennas. Equations 2 and 3 can be used to calculate the electric field strength at given distances from a linear emitting antenna. The far-field boundary is designated as 4λ .

Calculation of Field Strengths in the Near Field

Power densities in the near field are calculated using the far-field equation (Equation 2) and a near-field gain reduction factor (χ), or:

$$P_d = \frac{PG\chi}{4\pi d^2} \quad (4)$$

where all values are previously defined. Power density is converted to field strength using Equation 3.

The near field is the region within the far-field boundary, as defined by Equation 1 for aperture antennas and by 4λ for linear antennas. It should be noted that for the aircraft environment scenarios described above the aircraft is within the near field of most emitters using aperture antennas and in the far field for all of the emitters using linear antennas.

Near field gain reduction factors were calculated for the ECAC study using US Department of Defense approved techniques. A summary of these techniques for aperture antennas is presented below. No near field gain reduction techniques are required for linear antennas.

Near-field Gain Reduction for Rectangular Aperture Antennas. For rectangular aperture antennas the vertical and horizontal axis may not have the same illumination taper. Therefore, the gain reduction for each axis must be determined, then summed (in dB) to determine the total gain reduction.

Using the antenna axial illuminations, the main beam axis near-field gain reduction for a selected distance, d , from the antenna may be determined in the following manner:

1. Normalize the selected distance:

$$\Delta_h = d / (L_h^2 / \lambda)$$

and

$$\Delta_v = d / (L_v^2 / \lambda)$$

(5)

where:

Δ_v, Δ_h - normalized distance associated with vertical and horizontal antenna dimensions, respectively.

- From table A-3, find the near-field gain reduction factors (χ_h and χ_v) using the axis illuminations and the Δ values. The total near-field gain reduction is the sum of the gain reductions, in dB, converted to absolute form:

$$\chi = 10^{-[(\chi_h + \chi_v)/10]} \quad (6)$$

TABLE A-3. NEAR-FIELD CORRECTION FACTOR FOR RECTANGULAR APERTURE ANTENNAS

FOR UNIFORM ILLUMINATION		χ , dB FOR ALL OTHER ILLUMINATIONS				
Δ	χ , dB	Δ	cos	cos ²	cos ³	cos ⁴
0.016	17.50	0.010	----	14.0	12.6	11.5
0.020	17.25	0.015	14.0	12.4	11.0	9.8
0.021	15.95	0.020	13.2	11.2	9.7	8.6
0.023	16.75	0.025	12.2	10.2	8.8	7.8
0.026	14.75	0.030	11.4	9.5	8.0	7.0
0.032	15.65	0.035	10.8	8.8	7.4	6.4
0.037	13.50	0.040	10.2	8.2	6.8	5.8
0.042	15.00	0.050	9.2	7.2	5.8	4.8
0.056	11.95	0.060	8.4	6.6	5.0	4.0
0.065	13.10	0.070	7.6	5.8	4.4	3.4
0.093	9.0	0.080	7.0	5.3	3.8	2.8
0.14	11.0	0.090	6.6	4.7	3.4	2.4
0.4	2.2	0.10	6.1	4.2	3.0	2.1
1.0	0.3	0.15	4.3	2.6	1.6	1.2
		0.20	3.0	1.6	0.9	0.6
		0.25	2.0	1.1	0.6	0.4
		0.30	1.5	0.8	0.4	0.3
		0.35	1.1	0.6	0.2	0.2
		0.40	0.8	0.4	0.2	0.2
		0.50	0.5	0.2	0	0
		0.60	0.4	0.2	0	0
		0.70	0.2	0.1	0	0
		0.80	0.2	0	0	0
		0.9	0.1	0	0	0
		1.0	0	0	0	0

NOTE: If the value of Δ falls between two points on the chart, then the value of χ should be interpolated.

3. Multiply the power density at distance, d , obtained using Equation 2, by χ as shown in Equation 4. Use Equation 3 to obtain the field strength from P_D .

Near-field Gain Reduction for Circular Aperture Antennas. The method employed with the near-field gain reduction factor for circular antennas is different than that for rectangular antennas. The axial power density at a given distance, d , from a circular aperture antenna is determined as follows:

1. Calculate the reference power density by substituting $2L^2/\lambda$ for distance in Equation 2. Or:

$$P_{D0} = \frac{PG\lambda^2}{16\pi L^4} \quad (7)$$

where:

P_{D0} - reference power density, W/m^2

2. Convert distance, d , to normalized distance $\Delta = d\lambda/2L^2$.
3. Using the antenna illumination, obtain the value of χ , the near-field gain reduction value, from table A-4, and multiply it by the reference power density to obtain the corrected power density. That is:

$$P_D = \chi P_{D0} \quad (8)$$

4. Convert to field strength using Equation 3.

DEFINING AND DETERMINING THE VALUES OF NECESSARY PARAMETERS

The methodology for field strength calculations, discussed above, is dependent on the availability of technical information on the emitters. Specifically, the calculations require the following technical parameters: transmitter power, wavelength, antenna gain, antenna dimensions, and antenna beamwidths. These parameters are discussed below along with techniques for estimating parameter values if they are not available.

Peak Power

The peak power is used in calculating the peak field strength from the emitter. It is usually recorded as the maximum rated power at the output of the emitter before the antenna connections. It does not include system losses or losses due to equipment aging. If the peak power is not available, then no calculations can be made.

TABLE A-4. NEAR-FIELD CORRECTION FACTOR FOR CIRCULAR APERTURE ANTENNAS

Δ	χ FOR ALL ILLUMINATIONS ^{1,2}			
	UNIFORM	$(1-r^2)$	$(1-r^2)^2$	$(1-r^2)^3$
0.01	26.0	27.7	59.0	103.5
0.02	26.0	28.1	59.2	102.5
0.03	26.0	30.1	59.4	102.0
0.04	26.0	31.9	59.7	102.0
0.05	26.0	33.5	60.7	101.0
0.06	26.0	35.0	62.6	98.0
0.07	26.0	37.0	63.7	92.0
0.08	26.0	39.0	63.5	84.0
0.09	26.0	40.0	61.0	77.0
0.1	26.0	40.7	56.0	69.0
0.15	24.5	29.0	32.0	35.0
0.2	18.0	19.5	20.1	22.0
0.3	9.5	10.5	10.5	11.0
0.4	5.5	6.0	6.0	6.0
0.5	3.5	3.9	3.9	3.5

- NOTES:
1. If the value of Δ falls between two points on the chart, then the value of χ should be interpolated.
 2. For $\Delta > 0.5$, calculate field strengths using the far-field equation.

Average Power

The average field strength from the emitter is calculated from the average output power of the emitter. This is the maximum rated average power before the antenna connections. If this value is not available for a particular emitter, it can be calculated from the peak power and other parameters.

The average power is the product of the peak power and duty cycle. The duty cycle is a unitless number, less than or equal to 1.0, which is the product of the pulsewidth (PW) in seconds and the pulse repetition frequency (PRF) in Hertz. Emitter documentation often lists multiple values of PW and PRF or the PW and PRF as ranges. The duty cycle may be the product of some PW and some PRF in the range(s) but not necessarily the maximum value(s). If the precise duty cycle cannot be determined, then the maximum values may be used to obtain a conservative estimate of the duty cycle. For non-pulsed systems, the duty cycle is unity and the peak and average powers are the same.

Maximum Mainbeam Antenna Gain

As mentioned above, it is assumed that the aircraft is within the mainbeam of the emitting antenna for the purpose of calculating field strength. The mainbeam gain is the maximum possible gain of the antenna. If the gain is not available, it can be calculated from the area of the antenna aperture, the beamwidths, and the antenna illumination. The following procedures will show how to derive the antenna gain from one or more of these factors.

Calculating Gain of Aperture Antennas if Antenna Dimensions Are Known. If the dimensions of the antenna are known, then the gain can be calculated by application of some simplifying assumptions. Equation 9 is the basic formula for calculating gain:

$$G = \frac{4\pi^2 A \rho}{\lambda^2} \quad (9)$$

where:

G - gain, unitless

A - area of antenna aperture, square meters

ρ - antenna efficiency, unitless

F - Antenna gain factor, unitless

λ - wavelength, meters

The antenna efficiency is a rating of how well the antenna emits the input power. Antenna efficiency is usually between 0.5 and 0.9, with a typical value of 0.55. If no other information is available, the typical value of 0.55 will be used for ρ . The product of antenna aperture and efficiency is known as the effective aperture, A_e , of the antenna. The antenna factor, F, is dependent on the type of aperture illumination used. If the aperture illumination is not known, uniform aperture illumination will be used because it will produce a conservative value of gain., For uniform illumination, $F = 1$.

Calculating Gain if Antenna Beamwidths Are Known. An approximation of gain from the known values of the horizontal and vertical beamwidths is:

$$G \approx \frac{20,000}{\theta_h \theta_v}$$

where:

θ_h, θ_v - horizontal and vertical beamwidths, degrees

This assumes that the illumination is uniform and the antenna efficiency is 0.55.

Calculating Gain if Antenna Beamwidths and Dimensions Are Known. If the antenna dimensions and beamwidths are known, a more precise value for gain can be calculated. Assume an antenna efficiency of 0.55. Calculate the antenna aperture area. Then determine the gain factor from table A-5 for rectangular apertures and table A-6 for circular apertures. Substitute these values into Equation 9.

To determine the gain factor for rectangular apertures:

1. Use table A-5. Note that the horizontal illumination may be different than the vertical illumination, with a different gain factor for each dimension.
2. Observe between what values θ_h falls after substituting the appropriate values of L_h and wavelength.
3. Find the gain factor, F_h , from the table.
4. Repeat steps 2 and 3 for θ_v and L_v to obtain F_v .
5. Obtain the gain factor for the antenna from the product of the horizontal and vertical gain factors. That is, $F = F_h F_v$.

For circular apertures:

1. Use table A-6.
2. Observe the range of values of θ after substituting the appropriate values for diameter and wavelength.
3. Find the gain factor, F , from the table.

Calculating Gain if Antenna Illumination is Known. By itself, antenna illumination can only determine the gain factor (F). Either beamwidth or antenna dimension (for each axis, if the aperture is rectangular) is also required to calculate gain using the above procedure. Only one of the parameters is required because the other parameter can be derived using tables A-5 and A-6.

If all three parameters (i.e., illumination, beamwidths and dimensions) are known, then:

1. The Gain Factor(s) can be obtained using table A-5 or A-6.
2. Calculate θ using the formulas in the last column of the selected table and compare with the actual beamwidth(s). If the corresponding information does not agree, further checking on the available technical parameters may be required.

TABLE A-5. BEAMWIDTHS AND GAIN FACTORS OF RECTANGULAR ANTENNAS WITH VARIOUS ILLUMINATIONS

ILLUMINATIONS	GAIN FACTOR, F	θ , DEGREES
Uniform	1.00	$50.4\lambda/L$ to $68.7\lambda/L$
Cosine	0.81	$68.7\lambda/L$ to $83.0\lambda/L$
Cosine Squared	0.667	$83.0\lambda/L$ to $95.0\lambda/L$
Cosine Cubed	0.575	$95.0\lambda/L$ to $110\lambda/L$
Cosine Fourth	0.515	$110\lambda/L$ to $116\lambda/L$

TABLE A-6. BEAMWIDTHS AND GAIN FACTORS OF CIRCULAR ANTENNAS WITH VARIOUS ILLUMINATIONS

ILLUMINATIONS	GAIN FACTOR, F	θ , DEGREES
Uniform	1.00	$58.5\lambda/L$ to $72.8\lambda/L$
$(1-r^2)$	0.75	$72.8\lambda/L$ to $84.2\lambda/L$
$(1-r^2)^2$	0.56	$84.2\lambda/L$ to $94.5\lambda/L$
$(1-r^2)^3$	0.4	$94.5\lambda/L$ to $103.5\lambda/L$
Cosine Fourth	0.515	$110\lambda/L$ to $116\lambda/L$

Median Gain. The median gain is sometimes used instead of mainbeam gain when calculating field strengths if antenna rotation is a factor. The median gain is the gain averaged over the entire rotation. Usually, median gain is used if an object is in the mainbeam of the antenna for too short a period of time. Yet, an aircraft may be in the mainbeam of the antenna for sufficient time to degrade or damage sensitive flight-critical systems aboard the aircraft. For example, an aircraft approaching for a landing at a typical velocity of 170 nautical miles per hour in the same direction that an antenna for an air search radar is rotating at a typical rate of 10 rotations per minute will be in the mainbeam of the antenna for one second. Damage to electronic equipment can occur during exposures of a few milliseconds. It is, therefore, recommended that mainbeam gain be used in field strength calculations instead of median gain.

Beamwidths and Antenna Dimensions

Beamwidths. In and of itself, beamwidths are not directly necessary for the calculation of field strengths, yet beamwidths serve to check or derive a value for gain, and to derive a value for antenna dimension.

Antenna Dimensions. The antenna length and width for rectangular apertures (L_h and L_v), or diameter (L) for circular apertures is required in the calculation of field strength in the near field. Given that gain is known, the antenna dimension can be calculated based on specific assumptions. Fewer assumptions are required if more information about the antenna is known.

If only the gain is known: assume $F = 1$ and $\rho = 0.55$. Solve Equation 9 for A , the antenna aperture area. Now assume a circular aperture and derive the diameter. The diameter is the dimension used in field strength calculations.

If the horizontal and vertical beamwidths are also known: assume uniform illumination for each axis. For rectangular aperture antennas, derive the value of L_h and L_v using table A-5. For circular aperture antennas, derive a value of L from table A-6. Substitute the appropriate values of F and A into Equation 9 and derive the efficiency, ρ . If the efficiency is not between 0.5 and 0.9, repeat these steps with a different assumption for illumination (one or both axis for rectangular apertures) until the efficiency condition is met.

If beamwidths and illuminations are known: Derive the value(s) of L from table A-5 or A-6. Substitute the appropriate values of F and A into Equation 9 and derive the efficiency. If the efficiency is not between 0.5 and 0.9, check the values of the parameters for correctness.

Peak and Average Field Strength Predictions

Using the methods described above, the electric field strengths can be calculated from the data collected on the emitters in a particular environment. The distance at which the field strengths are to be calculated is dependent on the environment in question. A separate set of results for each of the environments discussed above is recommended. Using peak power, the maximum peak field strengths can be calculated for the emitters. Average transmitted power

is used to calculate the maximum average electric field strength. The resultant electric field strength predictions can be tabulated in preparation for producing envelope graphs.

PRESENTATION OF RESULTS

The most convenient method of presenting the maximum electric field strength predictions is in the form of envelope graphs. First, the results of the analysis are plotted on log-log paper. The abscissa is frequency from 10 kHz to 100 GHz. The ordinate is field strength in volts per meter (V/m) with a range from 1 V/m to some power of 10 that is greater than the greatest value to be plotted on the graph. The resultant graph is a histogram. An example of a histogram is given in figure A-2. (Note that the value plotted in figures A-2 through A-6 and table A-7 are random and are merely used for illustration purposes.)

The "envelope" of the graph is a discontinuous line that identifies the level at which all of the values on the graph are beneath. The data in figure A-2 with an envelope over it is shown in figure A-3. Figure A-4 is just the envelope of the values. As can be seen, the values that produced the electric field strength envelope cannot be determined from figure A-4. The values of the levels in the envelope graphs are shown in table A-7. Note that such tables are required for the establishment of an international standard.

The results of both the total environment analysis and the takeoff and landing analysis is a composite of several environments. The composite graph merely follows the maximum value of the component envelope graphs. For example, the envelope graph of figure A-4 is combined with the envelope graph of figure A-5 to produce the composite graph is figure A-6. For the total environment, as many as 12 envelope graphs are produced and combined into two composite graphs, one for peak and one for average field strength. For the takeoff and landing environment, two graphs for the environments around each of the airports are produced. In the U.S. case, six airports were examined resulting in 12 graphs. The two composite graphs that are produced for the takeoff and landing scenario are comprised of these 12 individual graphs.

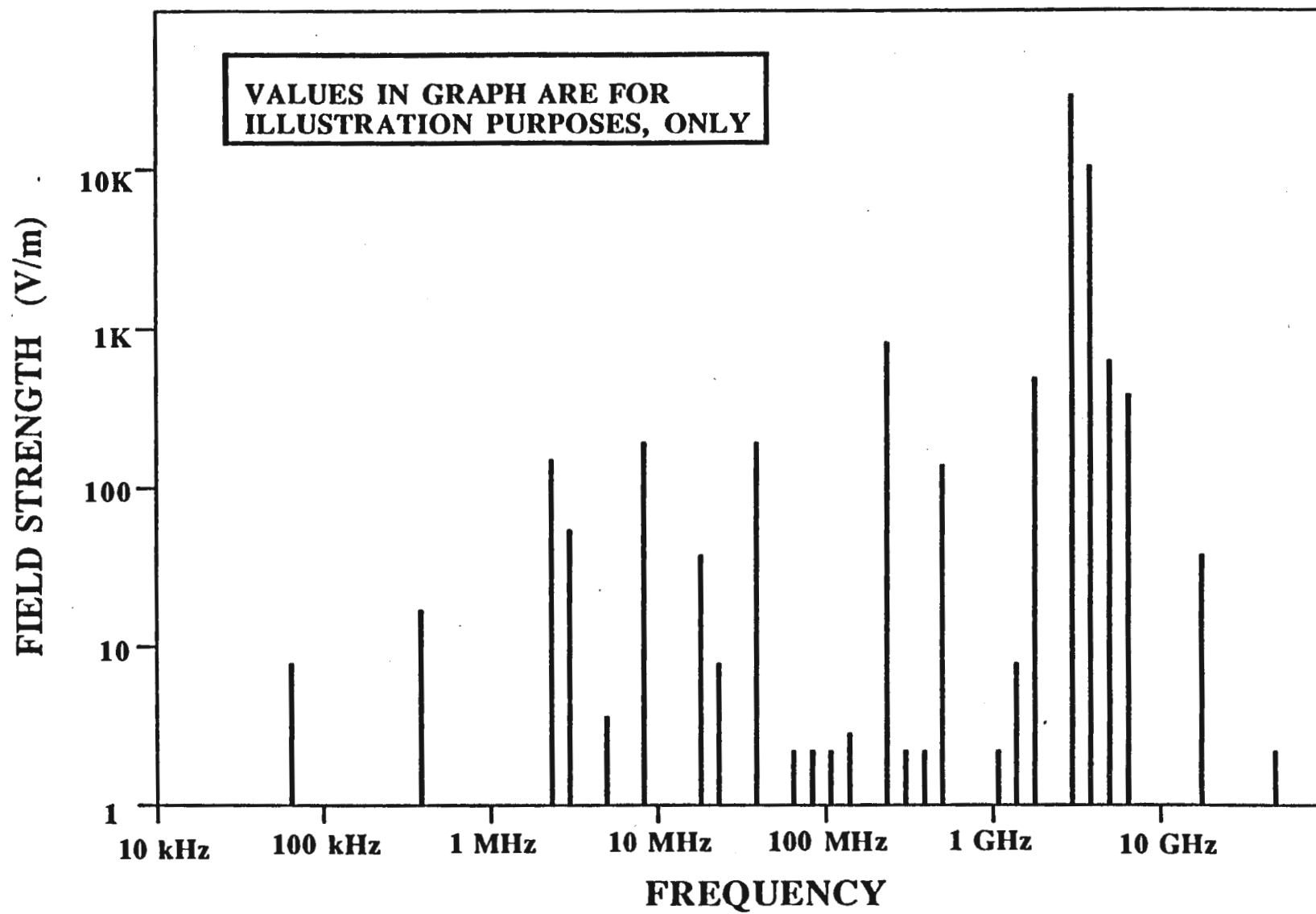


FIGURE A-2. EXAMPLE OF ELECTRIC FIELD STRENGTH VERSUS FREQUENCY GRAPH ON A LOG-LOG SCALE

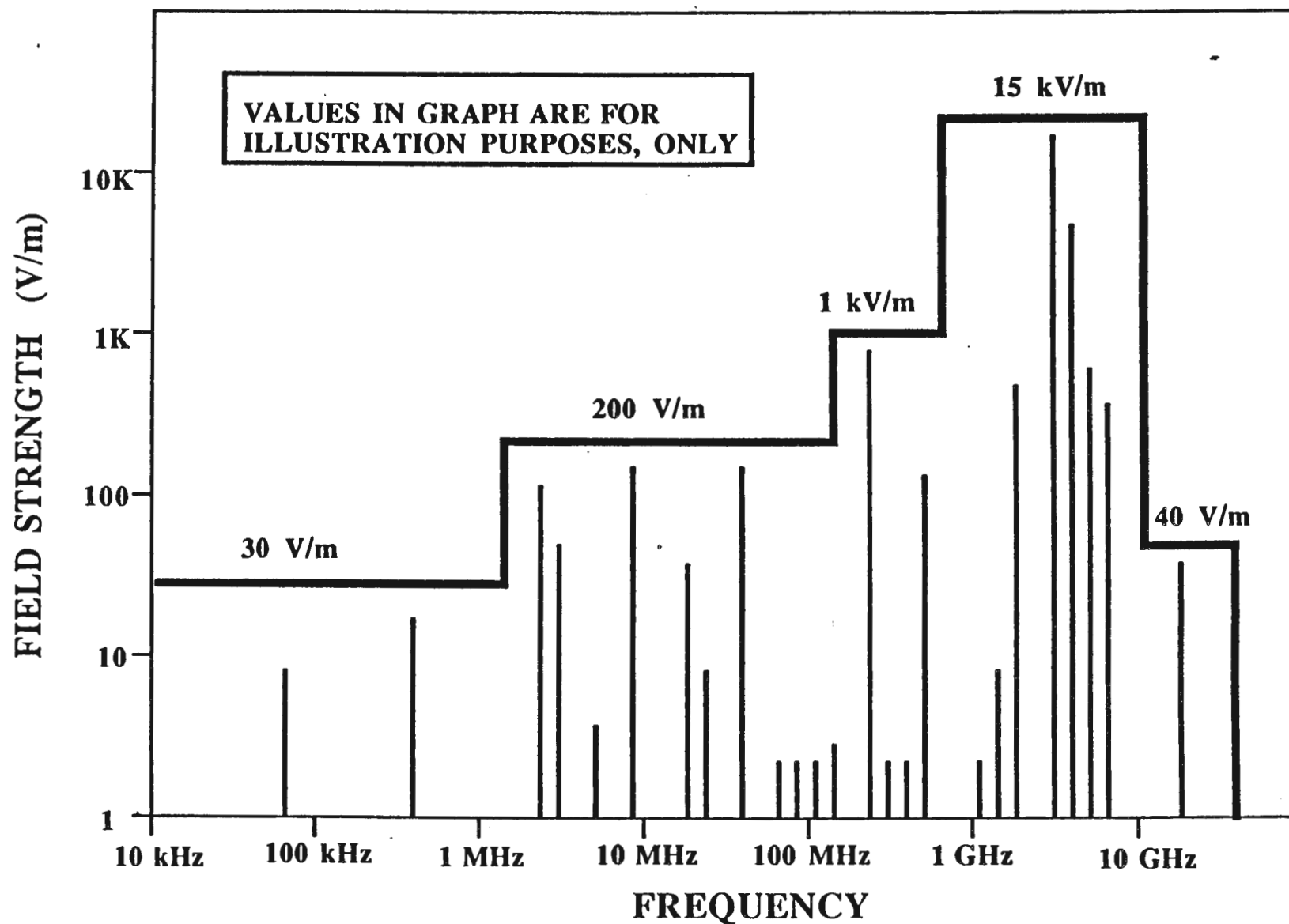


FIGURE A-3. EXAMPLE OF ENVELOPE OF FIELD STRENGTH VERSUS FREQUENCY GRAPH

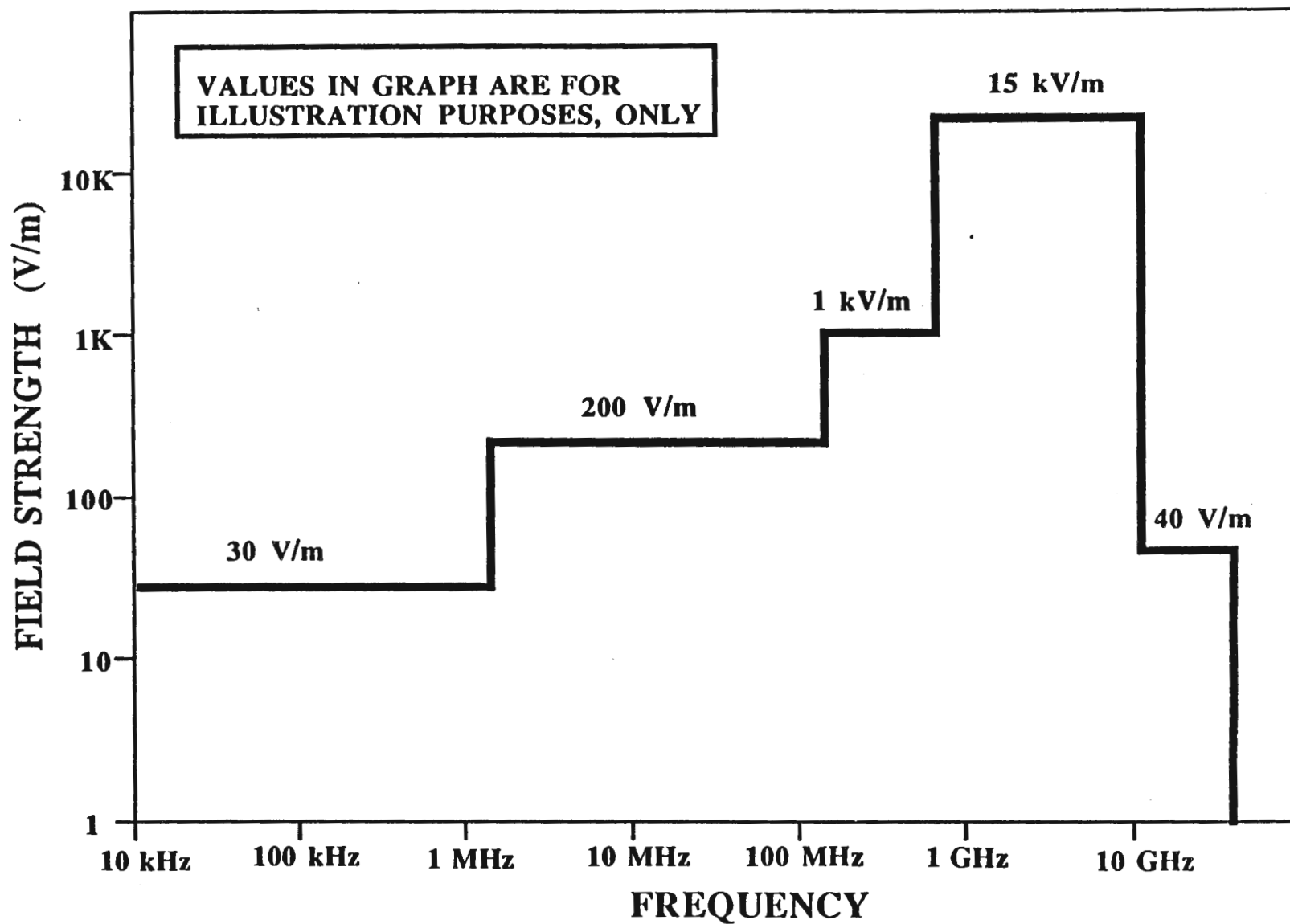


FIGURE A-4. EXAMPLE OF FINAL OUTPUT

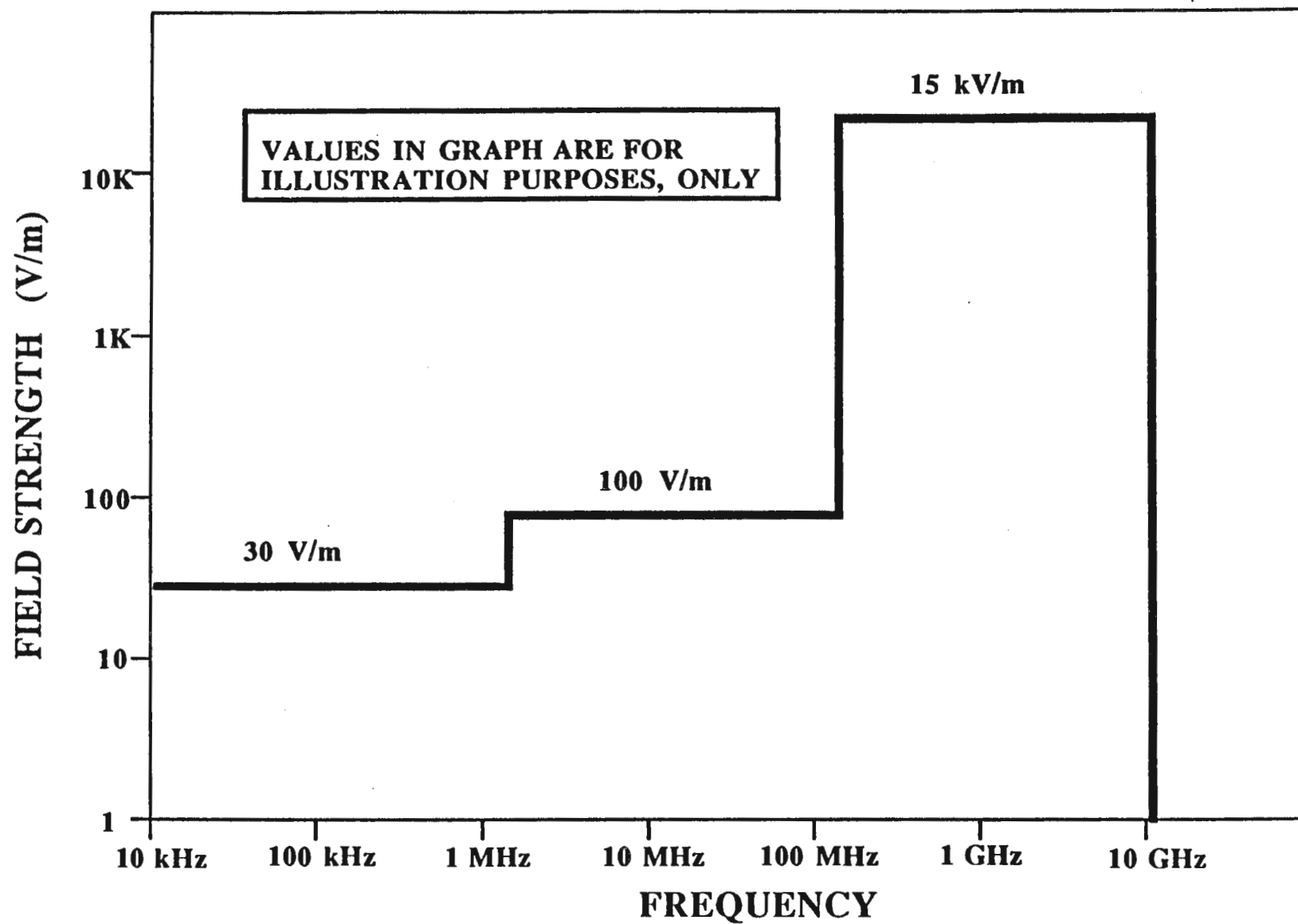


FIGURE A-5. EXAMPLE OF ENVELOPE GRAPH OF A SECOND ENVIRONMENT, TO BE COMBINED WITH FIGURE A-4 TO FORM A COMPOSITE GRAPH

14-47

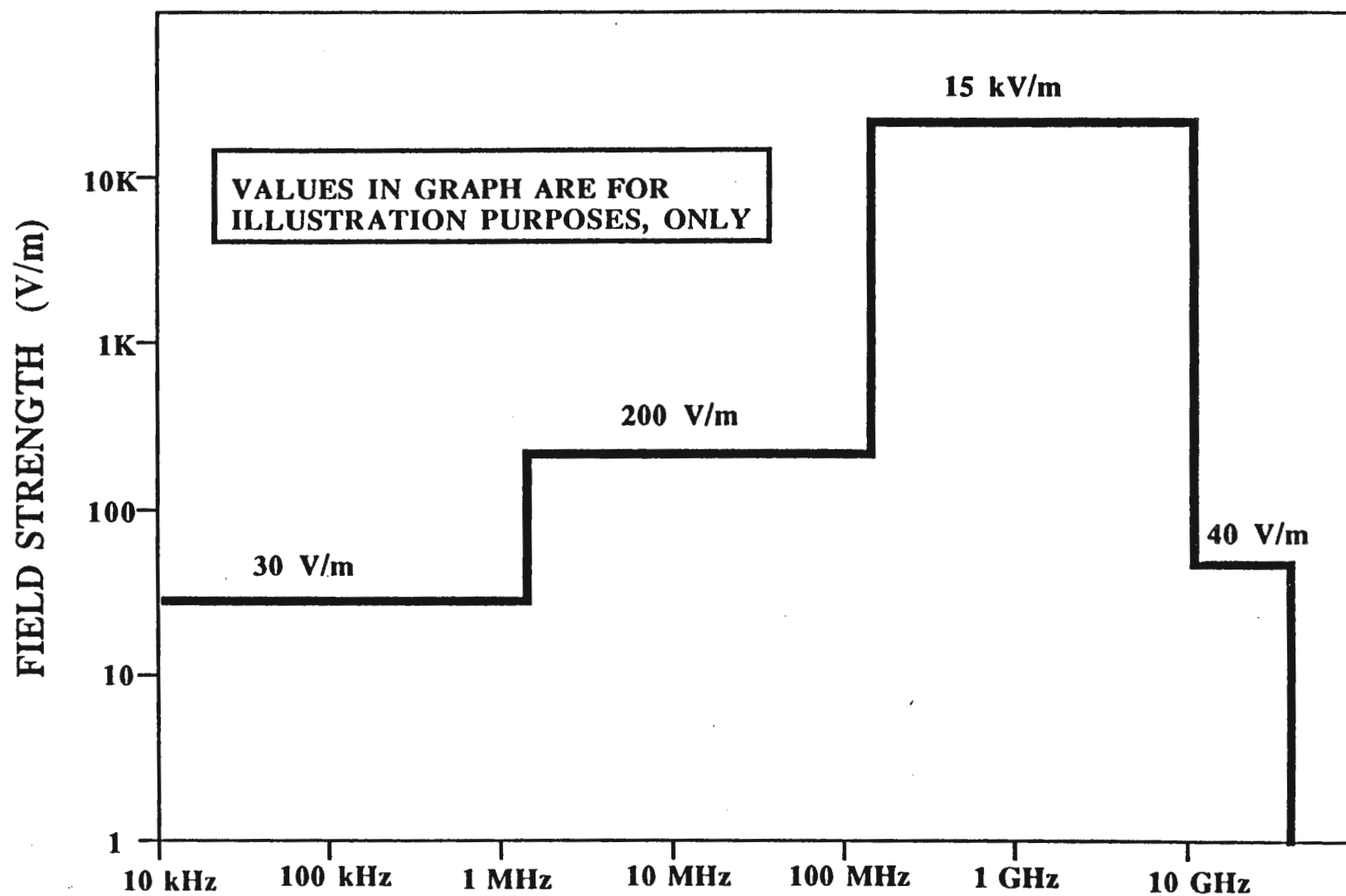


FIGURE A-6. COMPOSITE OF TWO ENVIRONMENTS SHOWN IN FIGURES A-4 AND A-5

TABLE A-7. SAMPLE ELECTRIC FIELD STRENGTH LEVELS USING VALUES IN FIGURE A-4

FREQUENCY	FIELD STRENGTH (V/m)
10 kHz - 2 MHz	30
2 MHz - 100 MHz	200
100 MHz - 800 MHz	1000
800 MHz - 10 GHz	15,000
10 GHz - 40 GHz	40

NOTE: Values in table are for illustrative purposes, only.

SUMMARY

The intention of this chapter was to provide the guidelines that were used to calculate the maximum electromagnetic radiation level that would be experienced by an aircraft flying in the United States. These guidelines are presented in an effort to promote uniformity of results from other members of the international aviation community in order that all results can be combined and used to develop an international RF susceptibility standard. Two aircraft scenarios were discussed: a worst-case scenario and a takeoff and landing scenario. The logic used by the United States to develop minimum distance criteria for each scenario was presented. The techniques used to calculate the far field and near field electric field strengths were described. Emitter parameters and methods that can be used to estimate values for parameters that are not available were discussed. Finally, how to present the results in the form of envelope graphs (useful because the results can be displayed in a simple chart without identifying the emitters involved) and field strength tables was shown.

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MIL-STD-462, Measurement of Electromagnetic Interference Characteristics.

RTCA DO-160B, Environmental Conditions and Test Procedures for Airborne Equipment.

LIST OF ACRONYMS

AC	Advisory Circular
AM	Amplitude Modulation
ATCRBS	Air Traffic Control Radar Beacon System
AWACS	Airborne Warning and Control System
CAA	Civil Aviation Authority
CONUS	Contiguous United States
dB _i	Decibels with respect to one milliamper
DGAC	Directorate Generale Aviation Civile
DOD	Department of Defense
ECAC	Electromagnetic Compatibility Analysis Center
ECM	Electronic Counter Measures
EMC	Electromagnetic Compatibility
EME	Electromagnetic Environment
EMR	Electromagnetic Radiation
EUROCAE	European Organization for Civil Aviation Electronics
FAA	Federal Aviation Administration
FM	Frequency Modulation
ft	feet
HF	High Frequency
HERF	High Energy Radio Frequency
ICAO	International Civil Aviation Organization
IFF	Identify Friend or Foe
kHz	kilohertz
MHz	megahertz
NATO	North Atlantic Treaty Organization
nmi	nautical mile
NPRM	Notice of Proposed Rulemaking
RF	Radio Frequency
RTCA	Radio Technical Commission for Aeronautics
SAE	Society of Automotive Engineers
SIF	Selective Identification Facility
SMOTEC	Special Missions Operation Test and Evaluation Center
STANAG	Standardization Agreement (NATO)
TACAN	Tactical Air Navigation
TV	Television
UHF	Ultra High Frequency
U.K.	United Kingdom
U.S.	United States
VHF	Very High Frequency
VLSI	Very Large Scale Integration
V/m	Volt/meter
VOR	VHF Omnidirectional Range



Department of Transportation
Federal Aviation Administration

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HANDBOOK-VOLUME II DIGITAL SYSTEMS VALIDATION

CHAPTER 15 ELECTROMECHANICAL ACTUATOR SYSTEMS (ELECTRICAL SYSTEMS CERTIFICATION ISSUES)



PREPARED BY:

COMPUTER RESOURCE MANAGEMENT, INC.
950 HERNDON PARKWAY, SUITE 360
HERNDON, VIRGINIA 22070

PREPARED FOR:

FEDERAL AVIATION ADMINISTRATION
TECHNICAL CENTER
ATLANTIC CITY INTERNATIONAL AIRPORT, NEW JERSEY 08405

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1.0 Electromechanical Actuation Systems Technology

1.1 Introduction and Scope

This chapter covers the technological description of electromechanical actuation systems (EMAS) including technology trends in these devices and certification issues involving actuators, electric power, and digital engine controls. The scope of this chapter, illustrated in figure 1.1-1, covers actuators, engine controls, and electrical power. The present electromechanical actuation technology is reviewed in section 2; particularly actuator devices for use on advanced technology aircraft. Technology and issues related to digital engine controls are included in section 3. Electric power systems technology is discussed in section 4 with emphasis on the technology trends of future generation aircraft.

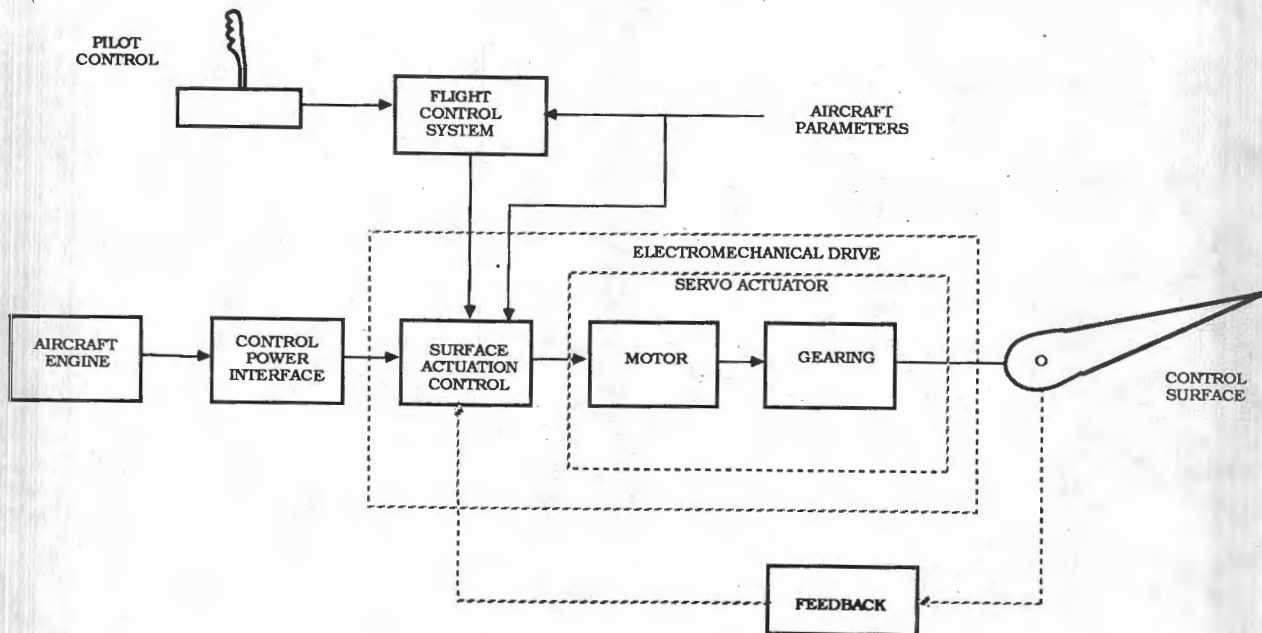


FIGURE 1.1-1. ELECTROMECHANICAL ACTUATION ELEMENTS

1.2 Emerging Technology

Areas of emerging advanced technology aircraft include the V-22 tilt rotor, all weather aircraft. This aircraft was originally intended for use by the military, but could find application in commercial operations, perhaps as a local interurban feeder to large airport hubs like the Los Angeles, Dallas-Ft.

Worth, New York, Chicago, and Washington, D.C. areas. Many other potential application areas could be found within the U.S. and internationally.

Another important area of future application includes faster commercial transports, made either by the U.S. or international consortiums. There is renewed interest by Aerospatiale in developing the British/French Concorde into a larger faster model. The National Aeronautics and Space Administration (NASA) has initiated research on an aerospace plane that has captured the interest of U.S. industry and has achieved limited congressional funding.

Any of these applications will require systems having higher reliability than today's EMAS while operating in far more severe thermal, shock, and vibration environments.

1.2.1 Electromechanical Actuation Systems for V-22

One view is that the V-22 aircraft is a flying actuator (refer to figure 1.2-1 from McManus 1985). Actuation systems are required for thrust control, flight control and stability augmentation, and configuration change-over from vertical to horizontal flight.

This aircraft may tolerate failures for a very short time during configuration changes or even during normal flight for some systems. Requirements for reliability and redundancy for commercial service should be more severe than for more conventional configurations of transport aircraft, due to flights over populated areas and the unique variable configuration.

1.2.2 All-Electric Airplane

The use of electric power rather than hydraulic power for actuator systems has several advantages (Leonard 1983, AiResearch 1976). An industry trend toward high voltage (270 volt dc) distribution, proposed and investigated by the Naval Air Development Center and several major airframe companies, may make it possible to operate actuators and control systems without hydraulics (Perkins and Marek 1977). This trend is based on the advent of permanent magnetic motors providing high power in a small space. The maintenance and repair of such all-electric systems would be simplified. For example, no hydraulic system bleeds or pressurization would be needed when maintaining or repairing an actuator.

Since electric power is more easily distributed, redundancy could be implemented by having several actuators rather than a single one driving a torque tube connected to several surfaces.

Three types of all-electric actuators that have been developed and evaluated include linear, rotary, and rotary-hingeline. Although electromagnetic actuation systems could eventually surpass electrohydraulic reliability, the results to date indicate that electrical actuator reliability has not been much improved over hydraulic actuators.

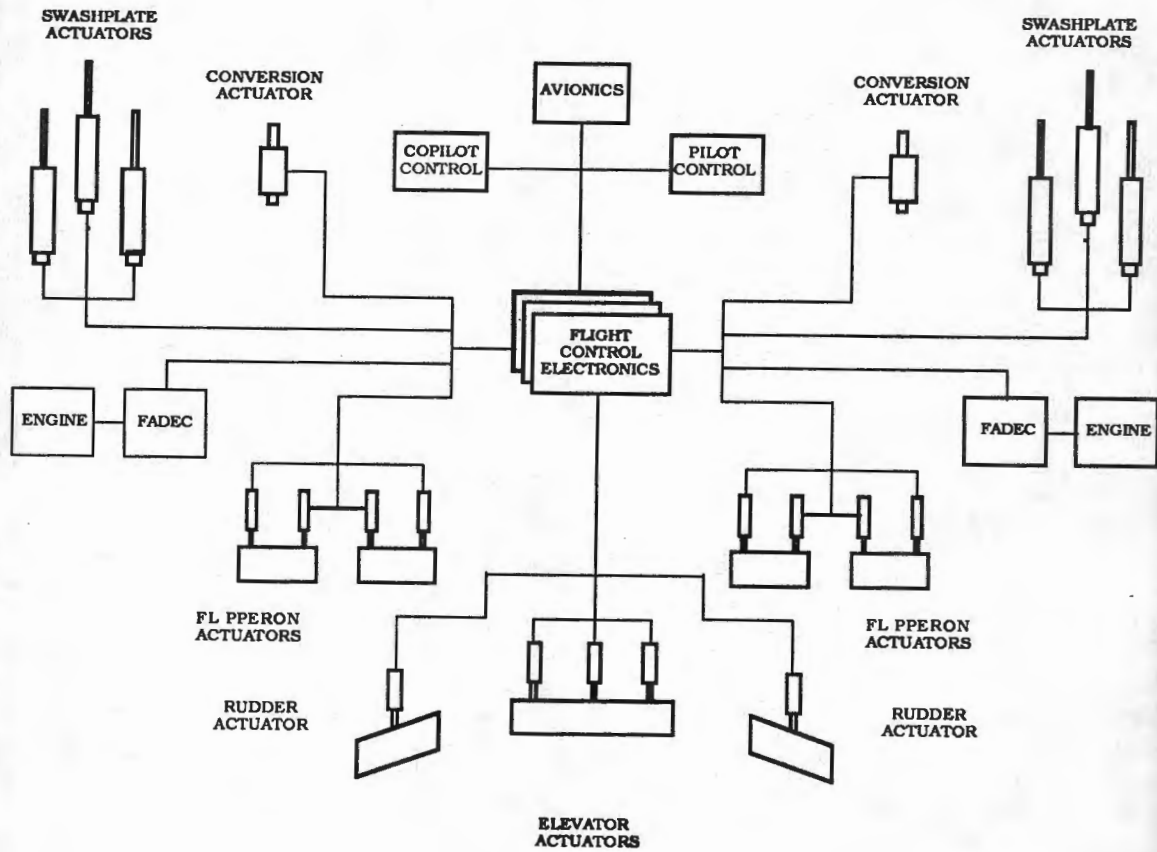


FIGURE 1.2-1. V-22 ACTUATION SYSTEMS

1.3 Technology Needs for an Aerospace Plane

An aerospace plane, described in Jorgensen 1983, is capable of flying intercontinental distances in commuter aircraft times. This imposes the most severe requirements for reliability and redundancy. Safe flight and immediate landing following a critical system failure are not possible from low earth orbit altitudes. In addition, a principal economic advantage of this aircraft lies in having flight operations with a round trip in a single crew day. This requires a low factor of unplanned maintenance. A two-hour turn-around is anticipated for the aircraft.

Flight operation at low earth orbit or suborbital altitudes requires flight critical systems having two orders of magnitude greater reliability than for present aircraft. Achieving these levels of reliability makes fault detection, isolation, and reconfiguration a necessity.

1.4 Electromechanical Actuation Systems Trends

In support of aircraft technology trends, such as those highlighted in this section, actuation system technology needs more reliable devices. One way of

accomplishing increased reliability is by using simpler devices. Current research and development is attempting to increase reliability as well as find ways for effectively applying computer technology. This is particularly true for the direct drive hydraulic actuators and electric actuators described in section 2. Military system applications have higher operating pressures for hydraulic systems and thus smaller and lighter actuator systems. This should also affect commercial actuation systems. The trend toward high voltage electric distribution is aided by a shift toward 270 volt dc power as an answer for providing an uninterruptable source of dc power for onboard computers. Newer engine control applications require additional computers on the engines. These additional computers provide automatic engine optimization (fuel versus power) during flight.

In addition to the development of the electromechanical actuation devices, research efforts by NASA Langley are also underway. NASA research will develop and demonstrate the means for controlling sensitive computer devices that must operate in severe aircraft electrical environments (i.e., electrical noise from onboard systems and from external sources of noise such as Electromagnetic Interference (EMI), lightning, and High-Intensity Radiated Fields (HIRF) environments). A multiyear advanced technology demonstration program is planned by NASA Langley to develop and demonstrate the technology for application of fiber optic devices as needed for the control of onboard computer devices. The program emphasizes Fly-By-Light and Power-By-Wire (FBL/PBW).

Fiber optic technology holds the promise for assuring that sensitive digital data is immune to electrical interference from external sources. The NASA program will include development and flight test of fiber optic devices to provide interconnects for digital control signals. This will be in the form of a fiber optic network data bus. Present computer devices may be interfaced to the network by bus interface devices that convert the electrical signals to optical signals. The electrical power to move surfaces and actuators will likely be provided by electrical means as for present systems.

2.0 Smart Servos and Actuators

A wide range of devices are currently available to operate the control surfaces on advanced technology aircraft. Improvements in hydraulic control devices and control configurations for fault detection and isolation greatly improve the reliability and performance of these devices. A realm long held by hydraulic actuators is being invaded by all-electric devices for control and actuation. This is due to the discovery of rare earth permanent magnetic materials. These materials allow operation of high power electric motors with small size and weight, and low standby power requirements.

2.1 Electrohydraulic Actuators

Hydraulic actuators operate via pressure applied to a power ram piston that converts the pressure to a linear or rotary motion. The positioning of a control surface is accomplished by directing the force into a mechanical linkage to the surface. The force is altered by directing pressure onto either side of a piston.

A simplex actuator, shown in figure 2.1-1, relies on only one hydraulic fluid and a single power ram piston. This configuration is widely used for ground based equipment and for some non-critical aircraft applications. It is not reliable enough for aircraft control in flight critical systems. While other parts of the actuator may be fault tolerant, any failure of the power ram or hydraulic system will disable the actuator. Therefore, for flight critical applications, power rams have two pistons in tandem, as shown in figure 2.1-2. Each piston is supplied from a separate hydraulic system. These actuators are designated as dual tandem actuators.

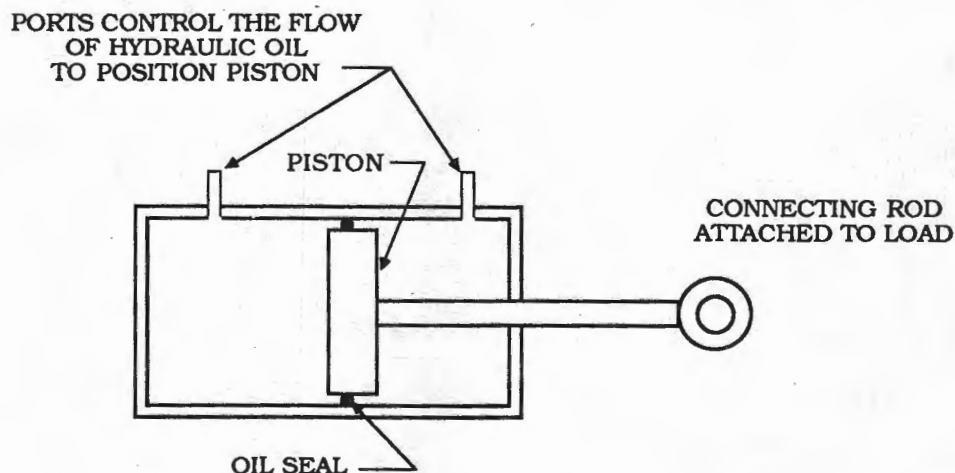


FIGURE 2.1-1. SINGLE POWER RAM HYDRAULIC ACTUATOR

SERVOVALVES CONTROL THE FLOW
OF HYDRAULIC OIL TO POSITION
BOTH PISTONS

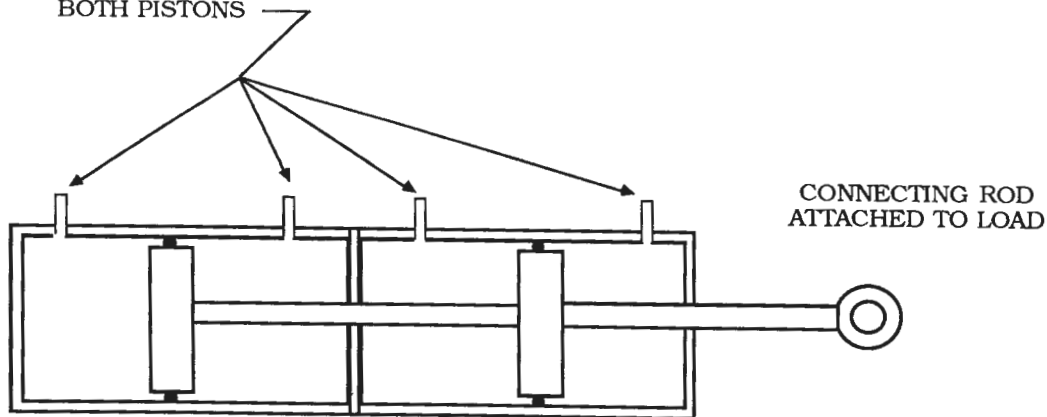


FIGURE 2.1-2. DUAL TANDEM POWER RAM ACTUATOR

A typical operational configuration for controlling a dual tandem actuator is illustrated in figure 2.1-3. These actuators use several stages to convert and amplify an electrical signal or mechanical input into useful motion. While current dual tandem actuators may differ from this diagram in some manner, the diagram conveys an understanding of the operation.

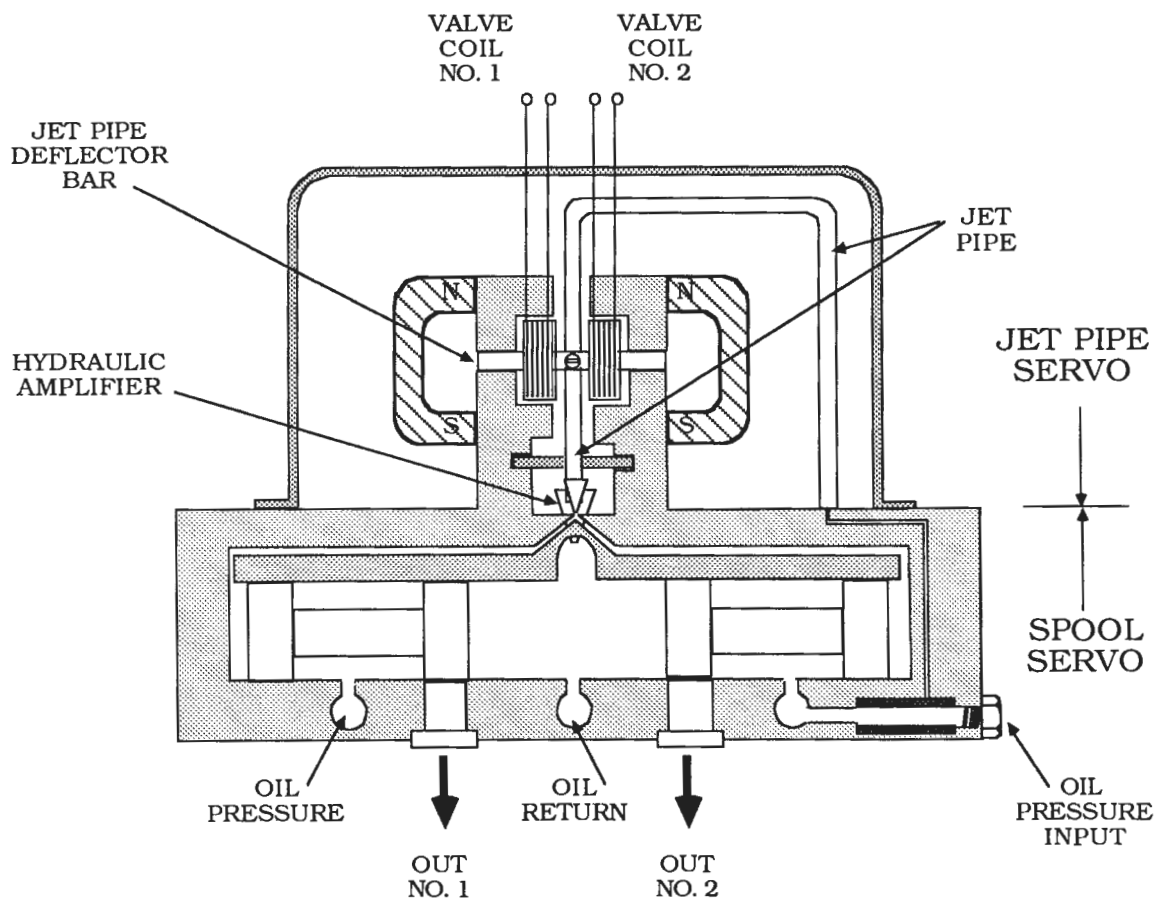
The first stage normally consists of three or four jet pipe or flapper valves that convert the input into a differential pressure for driving the second stage servovalve spool. The first two stages are often combined into a single unit called a two-stage electrohydraulic servovalve. With these devices, the spool position is controlled by feedback (normally mechanical) of the spool position to the first stage. The schematic and operation of a two-stage electrohydraulic control valve is shown in figure 2.1-4.

The second stage supplies hydraulic fluid to modulator pistons or servo ram pistons which in turn position the dual main control valve. The second stage may alternatively position the main control valve mechanically. Considerable force can be developed by these actuators (tens of thousands of pounds for large actuators). Because of limited power gain, the force to move the main control valve is also large. This two-stage control of the hydraulic flow to the main ram piston is the normal mode of operation. Closed loop control of the power ram position is often used. This control may be implemented by mechanical, analog, or digital processes. For mechanical feedback, the linear variable differential transformers (LVDTs) are replaced by mechanical linkages.

One experimental class of actuators consists of systems for eliminating the main control valve. These systems use more powerful 2-4 stage electrohydraulic control valves, or even electric motors, to operate the main control valve directly, as shown in figure 2.1-5.



Fault detection and isolation systems vary widely among actuator systems. These variations are the result of design decisions and design philosophy. Each example described in detail in the following sections represents a system at a different stage of technology. Current technology includes the in-place operational systems; research includes systems under development and laboratory prototypes. In these examples, the designers have the goal of increasing reliability of the actuation systems. Several fighter aircraft actuation systems are discussed. These systems are for application where extreme reliability is required because these systems must operate without failures in single thread fly-by-wire flight control systems.



The principal of operation is as follows:

Hydraulic fluid flows out of the jet pipe to the first stage of the hydraulic amplifier.

The first stage consists of the transmitter orifice on the end of the jet pipe and the two receiver orifices below the transmitter orifices.

When the jet pipe is in position shown, the two receiver orifices have equal pressure and the piston will remain balanced between the two output ports.

By deflecting the jet to the right or left, the pressure will increase on the between the right and left orifices. This results in motion of the second stage spool and therefore a hydraulic flow to the actuator.

The jet piston is controlled by the force of generated by the electric currents flowing in the coils #1 and #2 and the magnets on the left and right and the mechanical feedback spring

FIGURE 2.1-4. OPERATION OF ELECTROHYDRAULIC SERVOVALVE

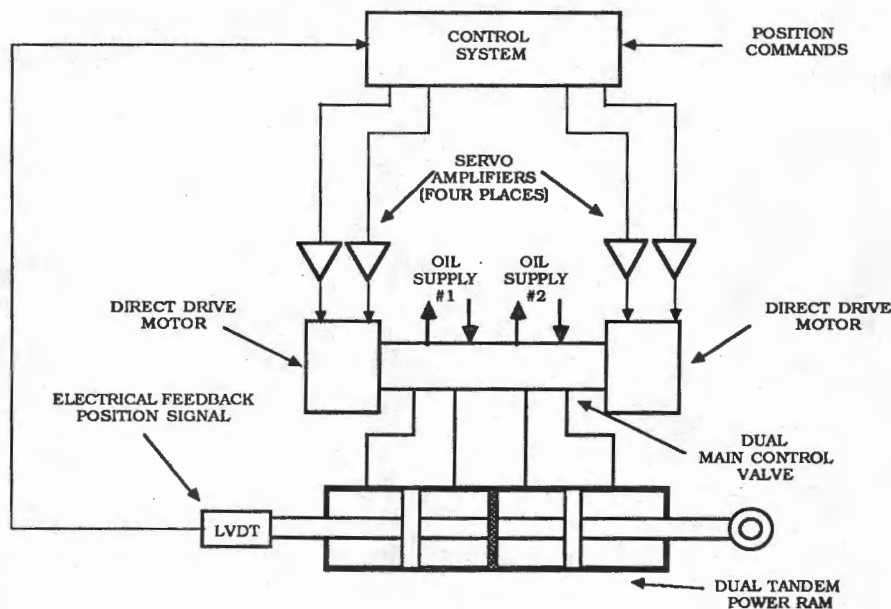


FIGURE 2.1-5. DIRECT DRIVE DUAL TANDEM ACTUATOR

The detection of component failure (and reconfiguration) must be considered in advanced fly-by-wire systems. One consequence of component failure may be the loss of the aircraft.

The most reliable hydroelectric actuators are built with the dual-tandem concept described in the previous section. This class of actuators uses more stages for amplification. Consequently, smaller first stage servovalves can be used, reducing the hydraulic power loss of the actuator. A major benefit of the main control valve is that it isolates the effect of load from the servovalves and produces a dynamic response having considerable stiffness. The loss of an electrohydraulic servovalve results in degraded performance (reduced response bandwidth) rather than total failure.

Current research is examining the removal of the main control valve in the dual tandem design. It is hoped that this modification will lead to a more reliable actuator having fewer parts. Although removal of the main control valve simplifies the design, more powerful two-stage servovalves are required. This design results in higher power loss and higher failure transients. In addition, less power is available for shearing metal chips that may jam the main ram piston.

A breakdown of failures for actuation systems was given in Kenmir 1972. This paper is relatively old, but indicates why the research trends are toward direct drive actuators. The following data indicates the percentage of operational failures due to actuator component parts:

▪ Tandem ram, control valves, bypass valves, and stabilization system	32%
▪ Servovalves and solenoid valves	45%
▪ Secondary Actuators	3%
▪ Transducers and connectors	20%

Increasing the system reliability by means of a dual channel will double the number of parts and thus double the total number of failures in a given time interval. The bulk of this increase is in servovalves and solenoid valves. Eliminating these parts would allow a significant increase in system reliability while having about the same number of failures requiring maintenance as for a single channel.

One approach to simplifying the design uses electric motors to control the main servovalve. Powerful dc motors, such as those presently used on direct drive experimental actuators, are needed to operate control valves with normal hydraulic pressures. The power loss of a standard two-stage design would be too large for a conventional servovalve to move the main control valve. Electric motors using rare-earth materials have the drive capability needed in a small size, weight, and power. At higher pressures, i.e., much greater than the normal 3000 pounds per square inch (psi), leakage around the piston becomes a significant problem. The dynamic performance is slower for this class of actuator compared to the direct drive or the conventional two-stage actuators.

The material in sections 2.1.2 and 2.1.3 is largely drawn from Baker and Bonnice 1988, and Jenney et al. 1989. These sections describe details of several selected actuators that include the following:

- F-16 Integrated Actuator
- F-18 Stabilator Actuator
- V-22 Swashplate Actuator
- Dynamic Controls F-16 Direct Drive Actuator
- Bell-4Valve actuator
- Boeing/Moog Digital Integrated Servo Actuator Controller (DISAC)
- Dynamic Controls All Digital Actuator
- Dynamic Controls F-15 Flutter Suppression Actuator

The first three, F-16, F-18, and V-22, are in use on recently developed military aircraft. They represent the best actuators currently in use. The rest of the listed actuators are experimental designs and indicate a variety of approaches for improving the performance or reliability of current designs. The Boeing Moog DISAC, Bell-4Valve, and F-16 direct drive actuators have eliminated the

main control valve. These designs are intended to improve reliability and performance. The Boeing Moog actuator is an experimental prototype developed to test microprocessor-based control, fault diagnosis, and failure management. The Bell-4Valve was developed for a flight test program. The F-16 direct drive actuator is a proof-of-concept prototype. The last two are unique conceptual prototypes built to demonstrate technology and evaluate new approaches for extended range of performance; as prototypes they have little built-in fault detection and isolation.

Table 2.1-1 shows a summary of the Fault Detection and Failure Management (FDFM) systems for the first six of these actuators (Baker and Bonnice 1988). For these systems the FDFM system is able to detect and reconfigure the actuators for servovalve, electrical, and sensor failures. No reconfiguration capability exists to address failures of the main control valve and power ram. The most likely common mode failure for the main control valve and power ram is jamming. The upper stages that drive these components are designed with sufficient shearing force to shear any metal chips that may jam the operation. An automatic reconfiguration capability exists for direct drive electrical motors. Since the most common failure modes are open or short circuits in the motor coils, the actuators are designed with several coils on a single solenoid and will continue operation with several coil failures. The shearing force and the dynamic response may be degraded due to a failure of one of the two motor coils.

Also not addressed in the systems examined are failures of the FDFM system components; including the fault detection and reconfiguration logic. This is especially true for logic circuits implemented hydraulically, or in analog circuitry where there is no redundancy. If the hydraulic or analog circuitry fails, the result could indicate a failure in error, or detect no failure at all. The latter case allows the actuator to continue operation but creates the situation where a subsequent failure may not be acted upon and a flight critical situation may arise without warning. In the case of false detection (isolation of a good component), the operation of the actuator may be shut off or degraded because of the failure in detection logic. Unless unnecessary loss of the actuator is critical to safe flight operations, false alarms are preferable upon failure detection. Digital implementations of the logic are more tolerant of faults since the logic is distributed to more than one processor.

In many cases, failures of fault reconfiguration components, such as bypass and solenoid valves and switches, are impossible to detect. This is because they are on-off devices and are not completely testable in flight. For example, if a valve or switch that is normally off fails in that position, the failure will not be detectable until the device is turned on. Failures that cannot be detected because they do not affect the present system operation are termed latent failures. Only when a component is commanded to turn on or off, and fails to act, will the failure be detectable. Since only one of the actuators investigated provided backup or redundant components for these devices (a backup coil for the DISAC actuator bypass valves), they are apparently reliable enough that fault detection is not required. Preflight testing of these devices would reduce the likelihood of flying with a failed reconfiguration device of this type.

TABLE 2.1-1. FAILURE DETECTION AND MANAGEMENT ON SIX SELECTED ACTUATORS

Component Failures	Actuators					
	F-16	F-18 stabilator	V-22 swashplate	Bell-4 Valve	DISAC	Dynamic Controls direct drive
Servo amplifier and servovalve or direct drive motor coil	Direct comparison of servo amplifier current implemented in analog logic. Switches to standby amplifier and coil.	Servo amplifier current is compared to the output of a digital model. Eliminates failed electrical channel.	Servo amplifier current is compared to the output of a digital model. Hydraulic flow to the corresponding servovalve is shutoff.	Servo amplifier current compared to the output of a simple analog model. The corresponding EHSV is disengaged.	Not described in reference.	Cross-strapping design eliminates the need for explicit FDFM.
Servovalve or direct drive motor	Hydraulic comparison of output pressures (i.e., different pressure) from the first-stage servovalves. Switches to active standby servovalve if one of the primary servovalves has failed.	Hydraulic pressure output of paired first stage servovalves is compared. Hydraulic supply to the pair of servovalves with a discrepancy is shut-down.	Measured position of the second stage spool is compared to a digital model. Hydraulic pressure to the failed servovalve is shutoff.	Direct comparison of measured spool position implemented in analog logic. Control system accommodates small failures; hydraulic flow shutoff when failure detected.	Measured position of the second stage spool is compared to slow and fast models of the EHSV. Bypass valve actuated.	None
LVDT	None?	Position of the servo ram is compared to a model. Electric channel disabled.	Self-test. Hydraulic flow to the corresponding servovalve is shutoff.	Detected by servovalve detection test.	Self-test. Use redundant LVDT by switching micro-processor control?	Comparison of the two error signals to each motor. Command to the corresponding motor removed.
Other	Other failures detected by comparing the actual system output to the measured output. Solenoids are activated to mechanically center the power ram.	Loss of hydraulic power activates a bypass/damping valve.			Position sensor on the bypass valves allows preflight testing to detected bypass valve failure.	

Some failures are handled using the capability within some actuators and do not require an explicit FDFM system. For example, the Bell-4Valve and the F-18 actuators can accommodate hardover servovalve failures with closed loop control. For these actuators, the control system will cause the other three servovalves to move in opposition to the failed servovalve. The result is a force fight while maintaining an acceptable performance. Similarly, the cross strapped amplifier design on the Dynamic Controls direct drive actuator for the F-16 simply offsets the effect of one amplifier failure with an opposite current in the other amplifier. Force fighting is characteristic of many such passive failure reconfiguration schemes. The advantage of this approach is that no logic or reconfiguration devices are required. The disadvantage is that excess capacity, greater than for normal operation, is required demanding more power than necessary. In addition, force fighting will degrade performance, while increasing mechanical wear and fatigue, thus increasing the likelihood of subsequent failures. Unless the fault detection thresholds account for the effects of force fighting, false alarms will occur.

Fault detection and isolation logic is implemented in three basic ways: hydromechanically (F-16); in analog circuitry (F-16, Bell-4Valve, and Dynamic Controls direct drive); and in digital software (F-18, V-2, and DISAC). For digital implementations, the logic can be in the central Flight Control Computer (FCC) or a local microprocessor. The trend in technology is from hydromechanical to digital logic. The disadvantages of hydromechanical logic are additional cost, power, size, weight, and hydraulic complexity. More complex single thread systems are generally less reliable or at least require increased maintenance. In addition, hydromechanical logic can only be used for direct redundancy failure detection and must be simple in order to be more reliable than the devices monitored. Furthermore, this logic is inflexible because any modifications require a major effort.

Analog logic overcomes most of these objections. However, analog logic is primarily limited to direct redundancy comparisons and self-test; only a limited analytical redundancy capability is possible. Digital implementation of FDFM offers the best potential. Digital systems can use all of the detection approaches including self-test, direct redundancy, and analytical redundancy. In addition, digital implementations can fine tune the logic during the operations and maintenance phase, as may be needed. Benefits of a local microprocessor, rather than placing all FDFM in a central FCC, include reducing the overhead of the FCC, less cabling to the FCC, distributed processing, and better digital control response time. However, the local environment on the microprocessor may be a serious limitation for achieving high reliability.

While digital implementations offer the best performance, the F-18 and V-22 actuator systems had constraints imposed on them by the design of the fault tolerant computer system. Interfacing with the quadruplex FCC on the F-18 required separate electrical sensors and actuators for each of the four channels. Each of the four redundant components or sensors interfaced with one FCC. No crosstalking between systems was allowed. This approach eliminates using a direct comparison of sensor outputs for actuator fault detection. For this system, the response to channel faults is to eliminate the entire electrical channel if a single sensor or component fault is detected. The V-22 system is similar to the F-18 except a triplex FCC is used. The DISAC actuator also has

limited the communication between channels. These architectures with a "brick wall" between channels benefit from a simpler FDFM system at the expense of additional components and increased failure rate. Reliable communication between channels may be possible with increased complexity and increased risk in achieving reliability goals.

The performance of FDFM systems is determined by the performance of each individual detection test. Self-test, used to detect failures on the DISAC actuator and LVDT failures, is only able to detect certain specific failures. Direct and analytical redundancy approaches should result in better detection performance due to the ease and accuracy with which component responses can be modeled. In practice (Jenney 1977), the thresholds set for fly-by-wire systems are often 30 to 50 percent of the maximum level for hardover failure. As a result only failures near hardover conditions are detected. The remaining failures are compensated for by the control system. Setting lower levels results in an unacceptably high false alarm rate. False alarms are apparently more important to the designer than concern over missed failures (Biafore and Grieszmer 1983, and Weinstein 1987).

Following are three areas of potential improvements pointed out in Baker and Bonnice 1988:

- Reduced false alarm rate.

The detection of failures in flight often cannot be duplicated on the ground. The two most common results during maintenance are "error condition could not be duplicated under test" and "component tests ok".

- More efficient FDFM design.

Several practices tend to increase the need for maintenance and decrease reliability. The first practice is simply adding a single sensor to a component for fault diagnosis. Then a faulty component is indistinguishable from a sensor failure. The result is that good actuator components may be removed as failed. However, if analytical redundancy or some other method was used to detect sensor failure, the reliability could be increased.

A second practice is including excess capacity to overcome failures by force fighting. While this approach may be the only one possible for a class of failures, several actuators use this approach simply to reduce the complexity of the FDFM system. In these cases, the result is increased size, weight, and power requirements. In addition, the increased wear from force fighting will ultimately increase the rate of failures.

Actuation designs could also be more efficient if fewer sensors were used. In existing systems, the loss of a component results in the shut down of an entire channel. Reducing the number of sensors through analytical redundancy, for example, could reduce the number of channels from four to three. Of course the additional burden in computer resources and communications must be weighed against the overall benefit.

- Control system reconfiguration.

None of the actuators reviewed alter the control system in any way following a failure. In addition to altering the control system to regain performance, reconfiguration could offer additional fault recovery options. Inner loop sensor failures might be compensated for by altering the feedback loop. Inner loop feedback improves the dynamic response of an actuator, but is often not absolutely necessary for continued emergency flight and landing. Effectiveness of control reconfiguration for failure management needs to be evaluated for each actuation system.

2.1.2 Detailed Examples of Current Technology

2.1.2.1 Standard F-16 Integrated Servo Actuator

Figure 2.1-6 is a schematic diagram for a standard F-16 actuator (Baker 1988). This actuator is used for controlling the horizontal tail, flaperons, and rudder of the F-16 aircraft. Because of the critical function of this actuator, it has considerable redundancy and fault detection and reconfiguration provisions. It is designed to accommodate dual electrical and hydromechanical failures. The actuator operates with a 3,000 psi supply pressure. The actuator has a force output of 36,000 pounds, a maximum slew rate of 5.5 inches per second, and a nominal frequency response having a 3 Hertz (Hz) bandwidth. This actuator is designed to accept two failures in the electrical system and one hydromechanical failure and continue to operate. Upon detection of a third failure the actuator will move to a centered position.

In the standard configuration (figure 2.1-6), the secondary actuator controls the position of the main control valve. The secondary actuator is controlled by three jet pipe servovalves. Failure detection and reconfiguration is accomplished hydromechanically using comparator spools to monitor the servovalve driving pressures and to drive current commands. A first failure of SV1 or SV2 will cause transfer of control to the standby servovalve SV3. A first failure of SV3 will lock the valve onto SV1 and SV2 control. Mechanical feedback loops are used for control of the servovalves, main control valve, and actuator ram. Solenoid valves are used to remove or apply pressure to the servovalves and the fail-center mode upon request from the actuator control electronics.

Note that the torque motors are not included inside the mechanical feedback loops. This results in the actuator position being commanded by current supplied to the servovalve coils.

The dual fail operation for electronic control is provided by each servovalve having two electrically separate windings. Either winding is capable of full servovalve control. These windings are driven by the FCC in active standby mode so that a first failure of a servo amplifier will be corrected within the flight control computer without activating any voting logic. The actuator control electronics incorporates analytical redundancy. This redundancy uses a mathematical model for the actuator position, including the dynamic rate plus position feedback signals. Comparison of the mathematically calculated position to the position measured using LVDT that is incorporated into the actuator

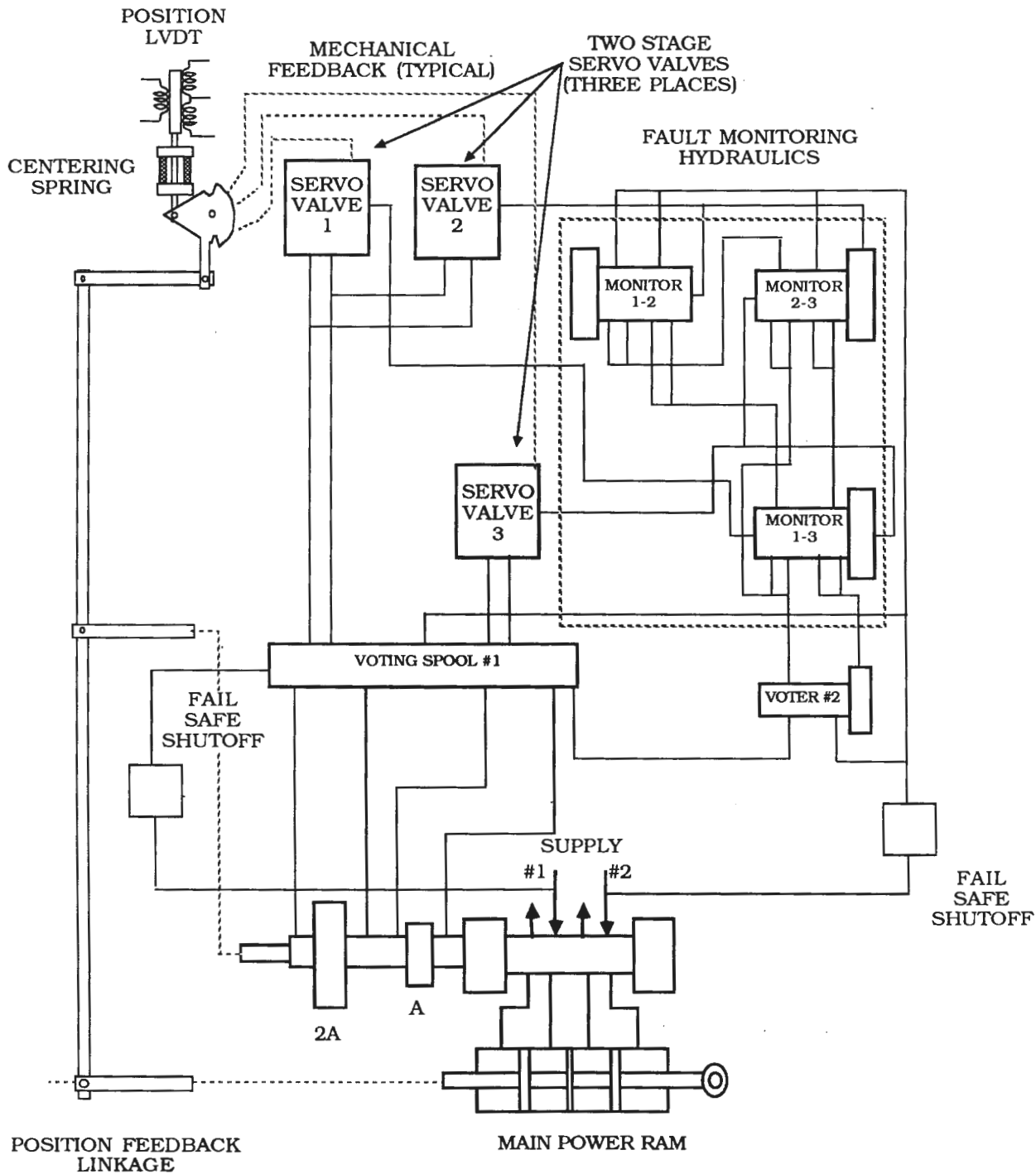


FIGURE 2.1-6. F-16 INTEGRATED SERVO ACTUATOR

determines an error signal. This error signal is used to determine if the actuator has failed. If the actuator has failed, move the piston to the centering mode of operation. In the centering mode, actuator position is determined by a centering spring attached to the position feedback linkage from the main actuator and the main control valve.

2.1.2.2 F-18 Actuator

The F-18 stabilator actuator (Harschburger 1983), illustrated in figure 2.1-7, has considerable redundancy and failure management. There are four single stage electrohydraulic servovalves which are paired to allow failure detection by comparison of their output pressures. This direct detection design uses four differential pressure transducers on each pair of servovalves. Failures of the servovalves, detected by these sensors, are contained by means of a solenoid valve which shuts off the hydraulic supply to that pair of servovalves.

Each of the servovalves is commanded electrically by four coils with the coil on one servovalve connected in series with a coil on each of the other servovalves. Each coil string is driven by a separate amplifier. Outputs from the four digital flight control computers are interfaced to a single amplifier supplying a string of coils. With this flux summing arrangement, the currents from the four separate amplifiers are effectively added.

Amplifier and coil circuit failures are detected by comparing the current from each amplifier to that calculated by a digital circuit model in the FCC. When a failure is detected, the amplifier for that coil channel will be disabled. This actuator design is able to meet performance specifications with two channel failures so that no failure management is needed. Detection of more than two channel failures results in the hydraulic flow to both pairs of servovalves being shut off, thus reverting to mechanical control.

Each pair of servovalves drives one piston of the dual tandem actuator. The two pistons are mechanically linked to a main control valve. This is motivated by the requirement for mechanical reversion in the event of a non-recoverable piston jam.

There are four LVDTs for sensing position for feedback on both the servo ram and the power ram. Each LVDT is connected to one of the four FCCs. Each FCC drives one of the servo amplifiers. Failure of an LVDT sensor or the command from the flight control computer is detected either by the servo amplifier current failure detection or by comparing the position of the servo ram with an analytical model. Sensor fault detection by direct comparison of sensor outputs is not possible because the digital quadruplex FCC architecture does not allow any cross-channel communication.

Common mode failures, resulting in loss of hydraulic power, are negated by a bypass valve that equalizes the pressure on either side of the ram piston, allowing the control surface to float.

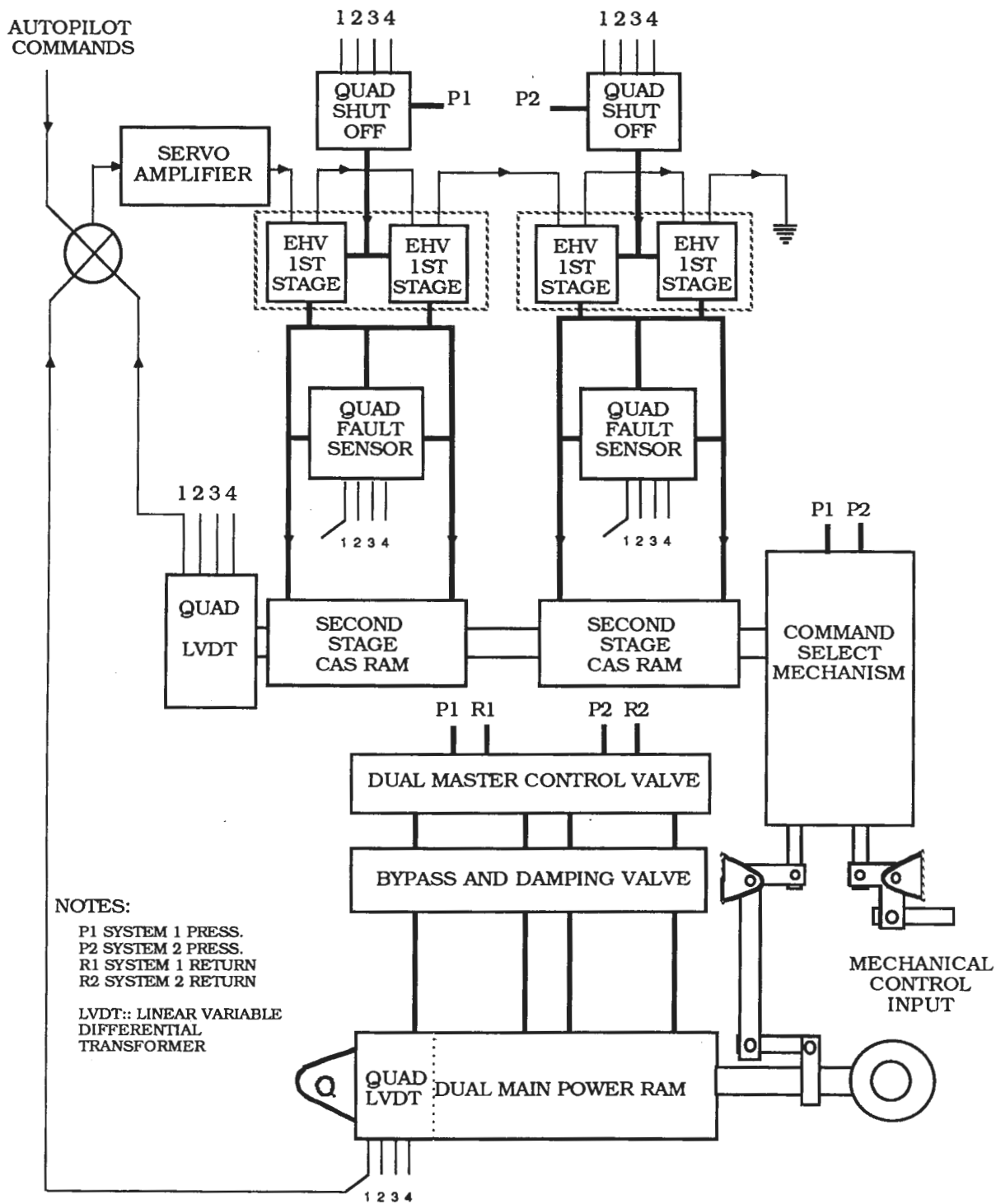


FIGURE 2.1-7. F-18 STABILATOR ACTUATOR

2.1.2.3 V-22 Actuator

The V-22 swashplate actuator (McManus 1985), illustrated in figure 2.1-8, is controlled by a triplex system. Three of these actuators are used on each main rotor. This system has considerable redundancy and failure management. The triplex uses self monitoring to detect failures within a local channel without depending on the other channels. However, the redundancy of the V-22 actuation system is not as great as for the F-16 quadruplex FCC. The triplex is configured as a duplex with failure reversion (a two-fail-operate system). There are two hydraulic supplies.

The actuator main control valve is driven with two unbalanced modulator pistons. Either of the two two-stage servovalves in system one are configured to drive one modulator piston with the third two-stage valve in system two driving the second smaller piston. Since the second piston has one-half the area of the first, this sets up a force fight. Since the drive currents are the same, the smaller piston loses. Each of the three servovalves are commanded separately by the three FCCs. In the normal no-failure condition, system one controls the main valve. Either FCC1 or FCC2 can perform the function. Any failure in FCC1 or FCC2 will be automatically detected by the comparisons between FCC1 and FCC2. Any failure in FCC3 will be unnoticed as long as either FCC1 or FCC2 is operating. Any two-failure situation will leave the system functioning with full performance.

Servovalve failures are detected using the position sensor on the second-stage spool of each servovalve. The current to each servovalve is measured and compared to the current calculated from a digital model in that FCC channel for detection of amplifier and servovalve failures. In the event of failure detection that channel will be shut down.

Each of the FCC channels contains a dual processor. The two FCC processors, A and B, in each channel monitor themselves. Either one can perform fault reversion. Some sensor inputs are compared across the channels to detect failures not covered by in-line monitoring. Single self-monitored sensors are used. These inputs are passed to the redundant processors (A and B) via a nonredundant sensor interface and single input/output (I/O) processor. Self checks by the two processors also check the I/O processor and sensors. Each processor calculates the output commands and expected response. Each processor can control engagement of the servo loops through discrete outputs. Commands from processor A are sent to the actuators by the I/O whose performance is checked by processor A and B using wraparounds. A watchdog timer is used to track each time frame; all actuators associated with an FCC channel will shut down upon failure detection.

The six LVDTs are used to monitor the position of the servovalves, the main control valve, and the power ram. Failures of LVDTs, including the triplex LVDTs on the main control valve and power ram, are detected using a self-test approach because no cross-channel communication is allowed for this FCC architecture. The response to failure of a servo amplifier, control coil, servovalve, or LVDT is the same: operate the shutoff valve or bypass valve to disable hydraulic flow to the servovalve in the failed channel.

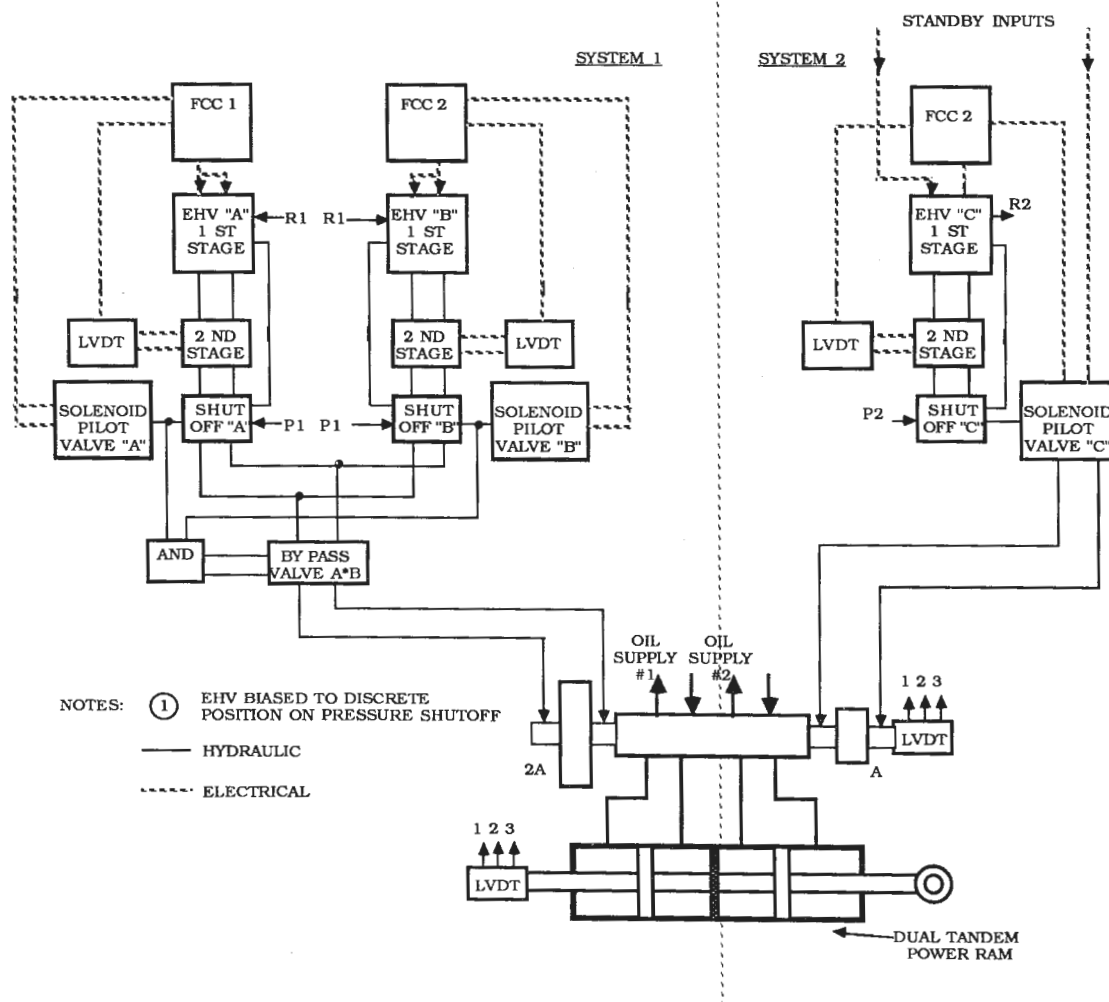


FIGURE 2.1-8. V-22 SWASHPLATE ACTUATOR

2.1.3 Detailed Examples of Advanced Technology and Research

2.1.3.1 Dynamic Controls' F-16 Direct Drive Actuator

The objective of this investigation, reported in Jenney 1989, was to implement and test a direct drive actuator for a current advanced technology fly-by-wire aircraft. The F-16 integrated actuator was chosen for the investigation because it was still in production and a sample flaperon/horizontal tail actuator was available for use.

A direct drive actuator directly modulates the flow of the main piston by hydraulic servovalves. This eliminates the requirement for a separate secondary actuator to position the main control valve.

Direct drive actuators operate the main servovalve using force motors having sufficient force. This technology was flight tested in 1981 on an F-4E at Edwards Air Force Base and has been subsequently refined in a laboratory investigation reported in Jenney 1989. The schematic diagram, shown in figure 2.1-9, illustrates a single-fail-operate direct drive actuator design. For the F-16 application a third channel of electronics was added to accommodate the dual-fail-operate requirement. Each channel was self-monitoring and independent of the others. Six feedback transducers were required to retain complete independence for redundancy purposes. The system designer placed the actuator inside a position feedback loop. This configuration is desirable because power is required only when the actuator is moving. Using the actuator in this manner also relaxes the requirement on linearity of the drive spool motion.

For this direct drive configuration, actuator performance degrades upon failure. Using electrical feedback for the direct drive actuator position also allowed inclusion of the force motors and control valves inside the position loop. This operation allowed simple reconfiguration following failure. Upon any single failure of the three self-monitoring control channels or both hydraulic supplies, transfer was made from the electrical-control mode to the self-centering mode. This transfer was accomplished with solenoid valves as for the normal F-16 actuator. With a multiple coil electrical control for each valve, the force applied is a direct function of the coil currents. When any channel fails, the position gain (and hence the flow gain) automatically changes. The actuator responds to position commands, with degraded performance, even if two of the three electrical channels fail.

Performance of this actuator could be maintained following failures by adding electronic elements. Using feedback of the spool position in an interloop would tend to keep the flow gain constant, with an increase of complexity. Gain changing after failure can be used to keep the valve performance constant following the loss of a channel. However, gain changing requires coupling between channels, which reduces control independence and could possibly introduce common mode failures. In the interest of high reliability, gain changing or interloop position feedback was not used for the prototype design.

The initial prototype design included analog failure monitoring. A second phase of the investigations (Jenney 1989) added microprocessor failure detection and correction. For either of these approaches, the failure monitoring would be checked pre-flight. These monitoring systems were designed so that failures in the monitoring system appear as single channel failures and could not cause hardover outputs.

The force motors were driven using pulse width modulation to reduce the heat sink requirements for the drive amplifiers. Because the coils are driven full-on or full-off, the power dissipated in the amplifiers is smaller than if operated in a linear mode. This minimizes the heat generation in the amplifiers and the size of the heat sinks. A potential disadvantage is the electrical noise generated by the current switching transients.

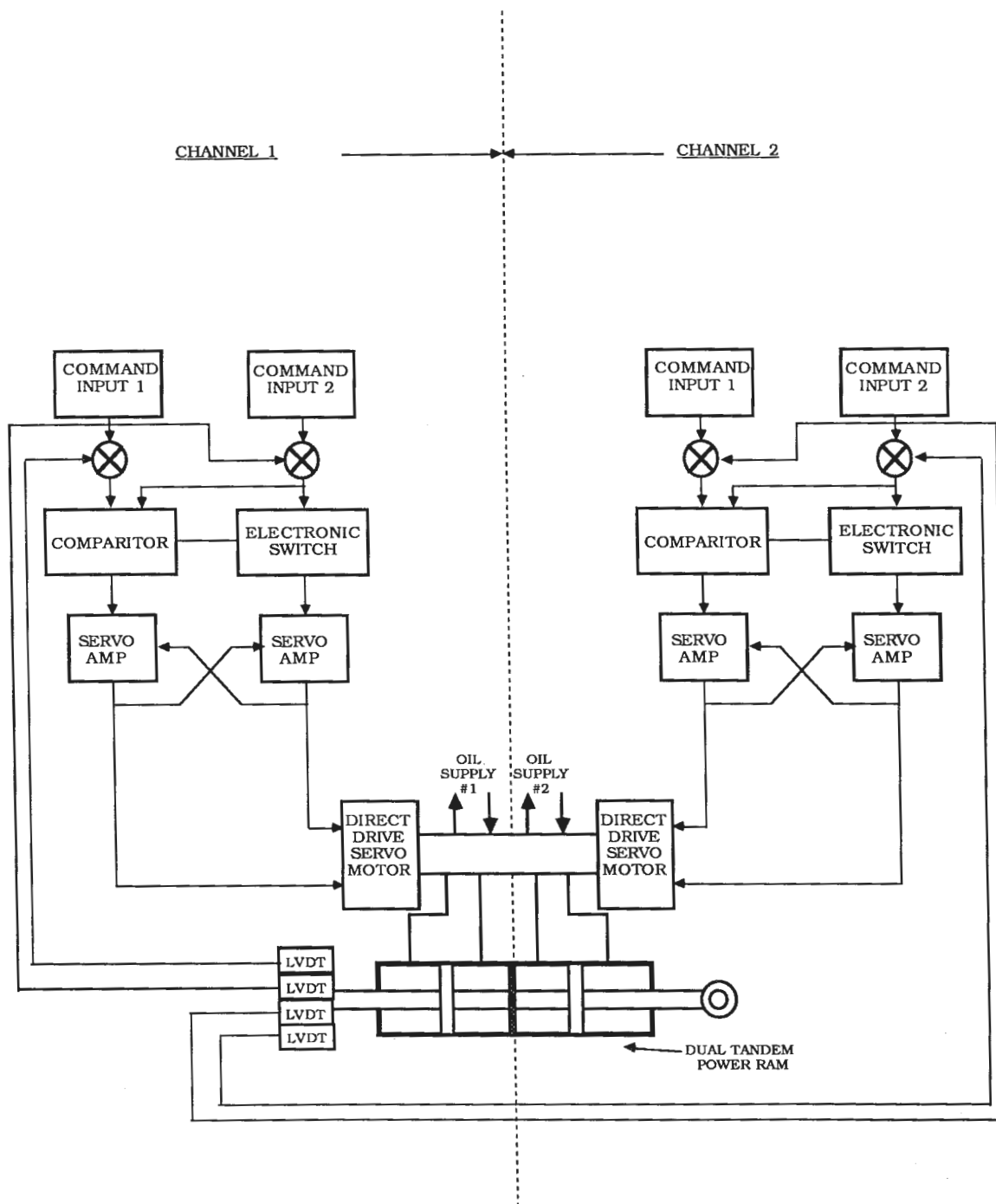


FIGURE 2.1-9. SINGLE FAIL OPERATE DIRECT DRIVE ACTUATOR

Figure 2.1-10 shows a block diagram of the two-fail-operate direct drive actuator designed for the F-16. All three channels operate normally, two for a single failure operation and one for the dual fail operate. Note that the three channels are self monitoring and no electrical mixing of the channels occurs. Since each channel uses two command inputs, the failure logic can detect input failures. These input signals are strapped together if failures are detected and corrected ahead of the control actuator system. Six feedback transducers allow each channel to be self monitoring. When a failure of a feedback transducer, summing amplifier, or command input is detected by the comparator, the inputs to the servo amplifier will be grounded. Two amplifiers are used for each channel. These two amplifiers are cross strapped together so when an amplifier fails, the other amplifier will be driven in the opposite direction, cancelling the output of the failed amplifier. If an amplifier fails resulting in a no-output condition, then the gain of the remaining amplifier will increase, tending to keep the channel force gain constant.

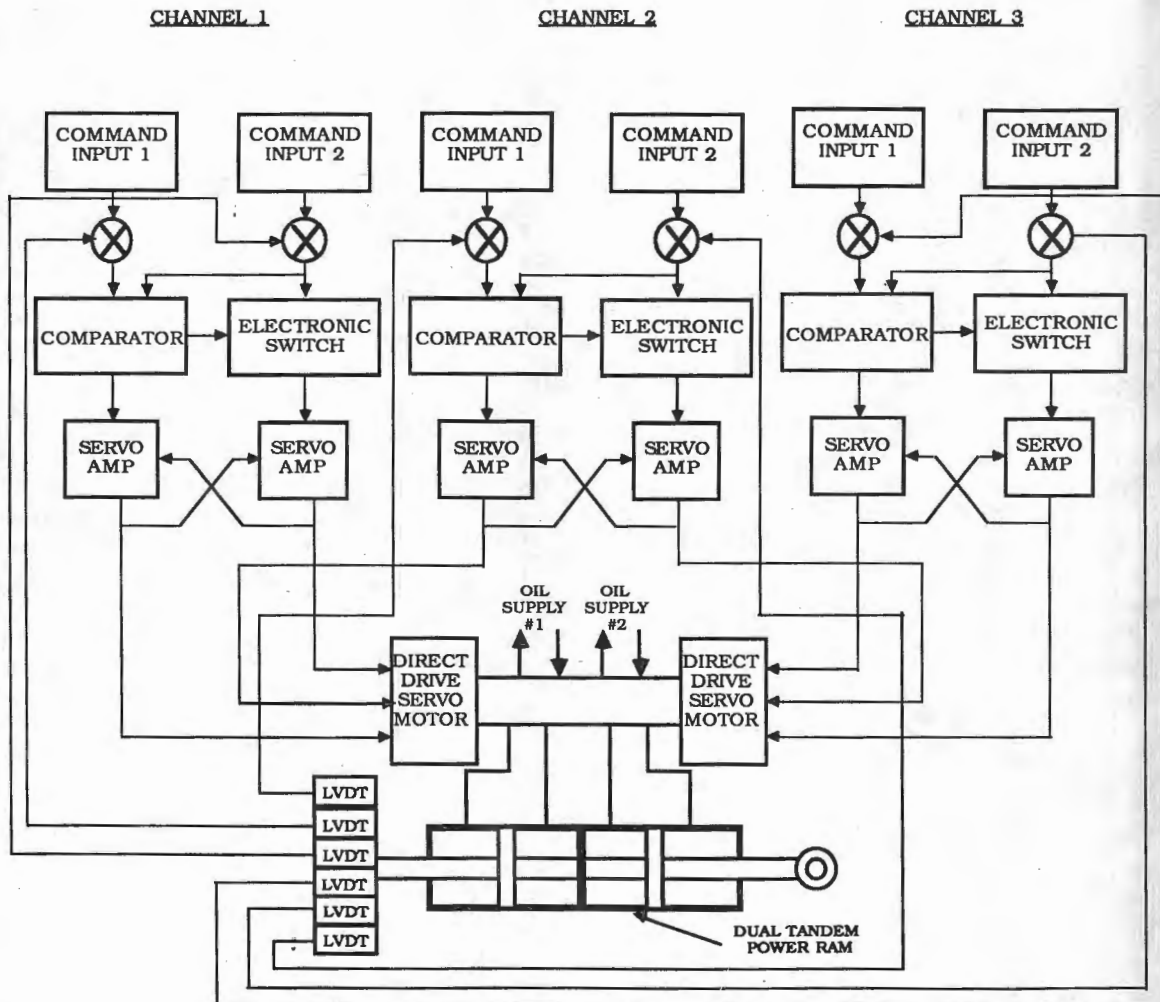


FIGURE 2.1-10. DYNAMIC CONTROLS' F-16 DIRECT DRIVE ACTUATOR

The direct drive control valve concept for laboratory investigations is based on a moving coil design. The coil assembly is suspended by a folded linkage incorporating both a centering spring and flexures for the pivot points. The coil assembly consists of three individual coils, electrically isolated from one another, with four layers of windings in each coil. The coils move within a magnetic circuit energized with a rare earth permanent magnetic material. Stainless steel 416 was used for the pole piece and outside cup. Tascore 21 was used for the permanent magnet material. The force output from the coil was 55 pounds with all coils energized at 1.2 amps. The resistance of each coil was nominally 4 ohms. Figure 2.1-11 shows the structure, coil, and housing. This is a wet coil design flooded with return oil. Use of a wet coil eliminates the need for a dynamic seal and provides cooling for the coil and damping for the coil motion. Since the oil is fed through a filter disk and only circulated locally, the problem of the magnetic structure acting like a magnetic oil filter does not occur.

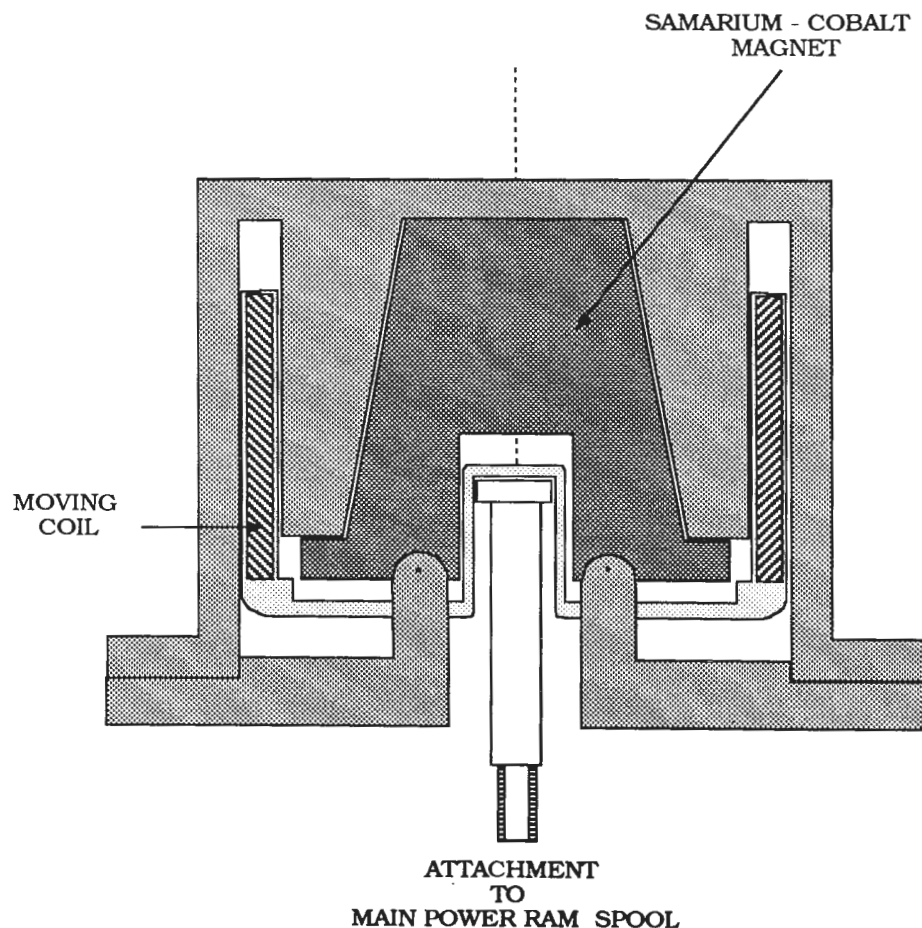


FIGURE 2.1-11. MOVING COIL TORQUE MOTOR

Two force motors were used for the F-16 actuator, one on each end of the main control valve. These motors were connected together in the valve package with a large flow passage so that the pressure could balance between the magnetic structures. The return pressure line of one hydraulic system supplied oil for both force motor coils.

The main control valve that both motors positioned was a three part design. The valve moved in a sleeve which was normally held stationary by a locking mechanism. Electrical feedback normally controlled the position. When mechanical feedback was used in the fail-operate mode, the sleeve was unlocked and driven by the mechanical feedback linkage. In this fail safe mode the driving electronics for the center spool are turned off and the control spool maintained in a center position by springs that center the force motor valves.

2.1.3.2 Bell-4Valve

The Bell-4Valve actuator (Abrams and Donley 1984, and Murphy and Haskins 1982 and 1984), illustrated in figure 2.1-12, uses four two-stage flapper valve servovalves to meter the hydraulic flow directly to the dual tandem power ram.

Any failures of the servovalve coil or drive wire are detected by comparing the drive current to that calculated from a simple analog model. If a failure is detected, then that particular valve is disengaged by operating a solenoid valve. Active servovalve failures are detected by directly comparing positions of the second stage spools of the four servovalves. For hardover failures, the control system causes a flow bypass around the piston of the failed servovalve.

Because the Bell-4Valve is able to accept a hardover servovalve failure, the detection thresholds are set so that only large failures are detected. The advantage of this approach is that false alarms are reduced. If both channels driving a piston fail, a bypass valve is activated for that piston. The FDFM system is implemented in analog logic.

2.1.3.3 Boeing/Moog DISAC

The DISAC prototype actuator (Waffner and Chenoweth 1984, and Chenoweth and Slaugh 1985), illustrated in figure 2.1-13, was developed to be controlled and managed by two microprocessors. It differs from the Bell-4Valve actuator in that only two servovalves are used.

In the primary mode, each microprocessor controls the servovalve and bypass valve for one channel. However, if one microprocessor fails a self-test, that processor drops control and the other processor takes control of both channels. Logic exists to keep both microprocessors from attempting to control the same component at the same time. Both processors access data from both channels by means of duplicate sensors. One sensor interfaces to each processor. In addition, both microprocessors are able to operate each servovalve and bypass valve using separate coils in each component.

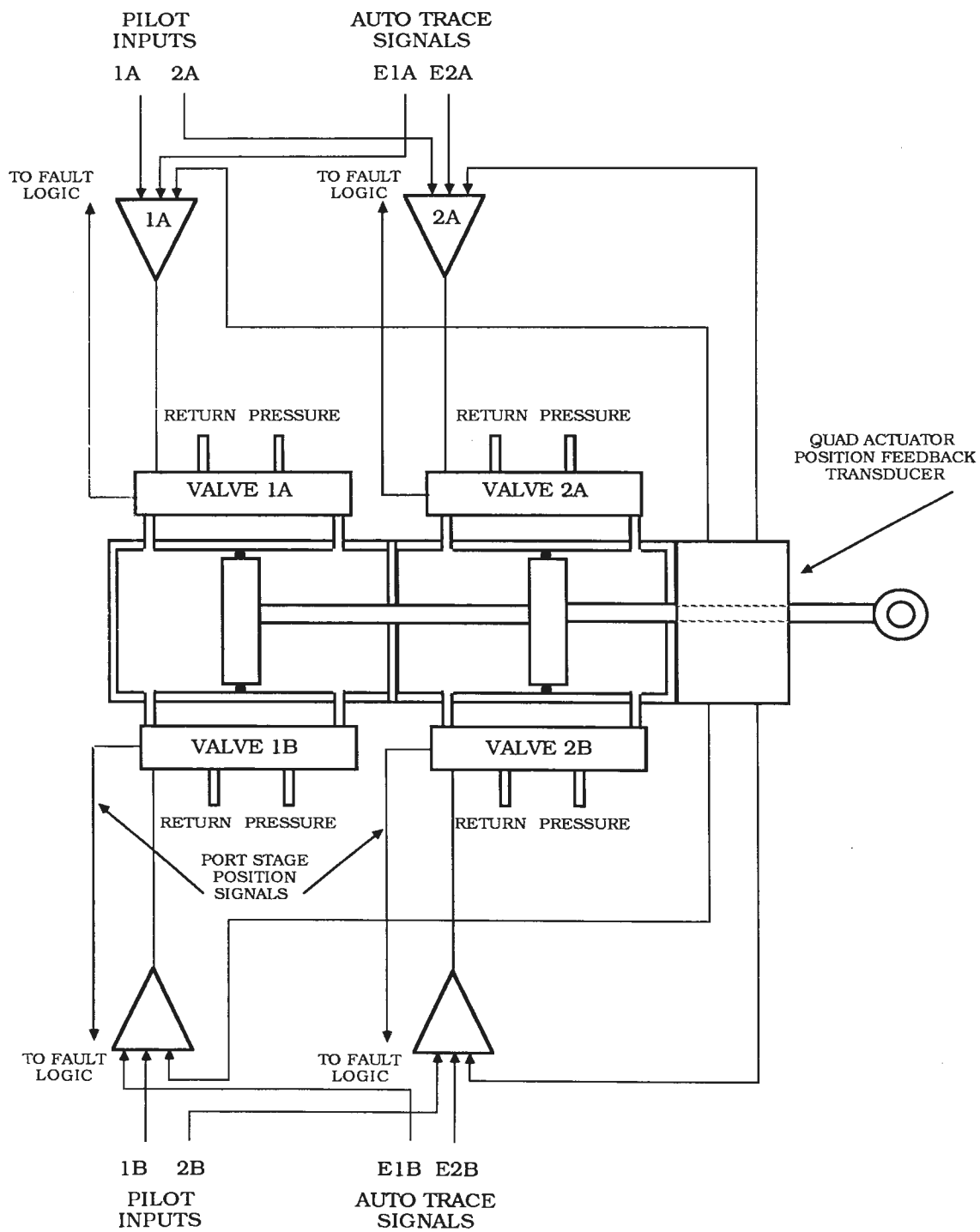


FIGURE 2.1-12. BELL-4VALVE ACTUATOR

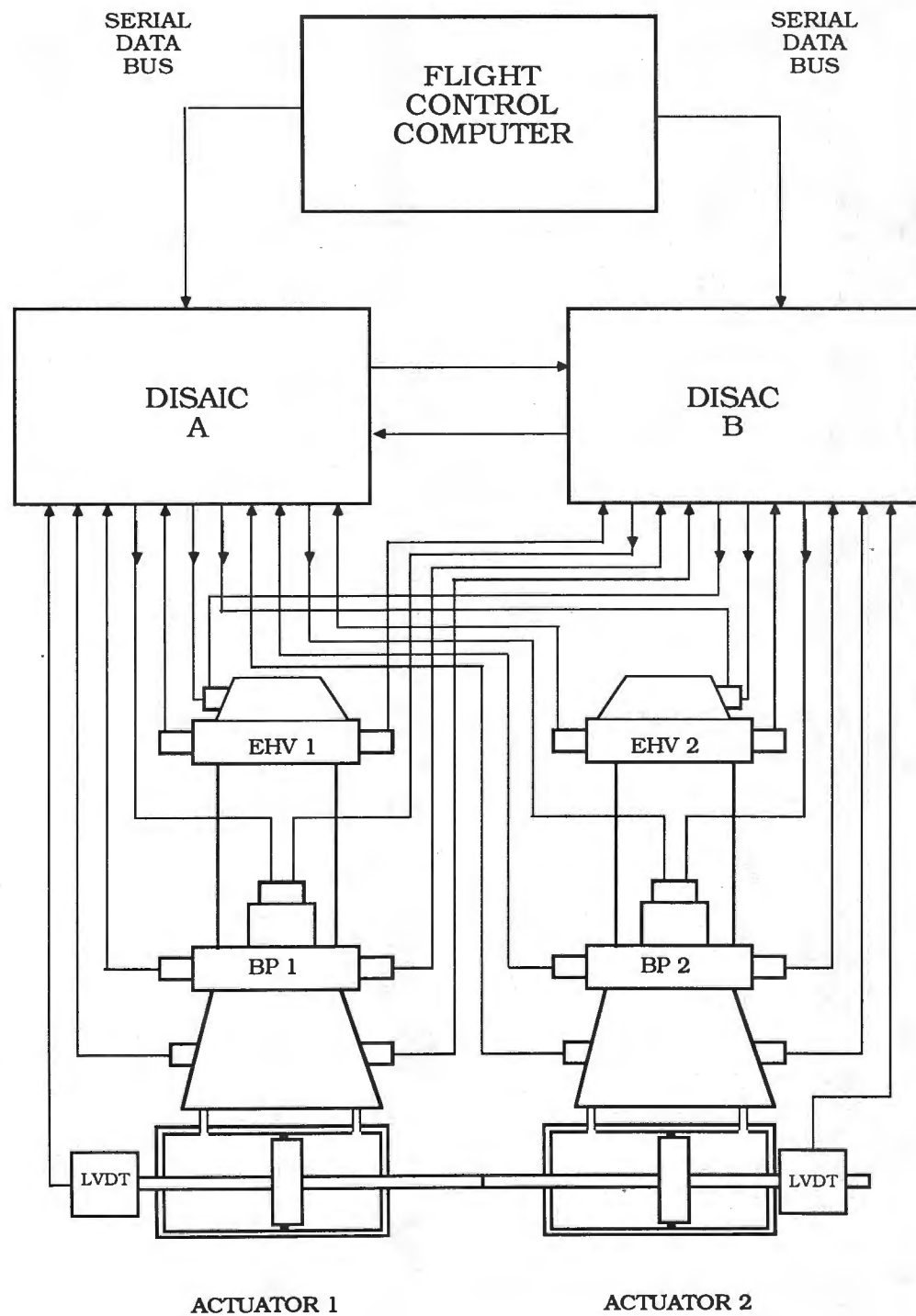


FIGURE 2.1-13. BOEING/MOOG DISAC ACTUATOR

Failure detection on this actuator consists of comparing the measured position of the second stage spools to fast and slow models of the servovalve to detect failures. LVDT failures are detected using self-test. The failure management response could be either to turn off the channel with the failed component or allow the other microprocessor to take over operation of the channel.

One unique feature of this actuator is the use of position switches on the bypass valve to verify its operation during preflight tests. Bypass valves failing open is a latent fault that cannot be observed during normal flight operations. Preflight testing at least verifies bypass valve operation prior to takeoff.

2.1.3.4 Dynamic Controls All Digital Actuator

Recent research and development, reported in Jenney 1989, illustrates innovation in implementation of a direct drive valve. This all-digital hydraulic servovalve has digital signals applied directly to poppet valves controlling hydraulic flow to the main piston. In this implementation the poppet valves control flow in binary increments. Each valve is driven on or off by the value of a single bit in a binary word. Thus, the overall flow is proportional to the binary value of a digital command. The flow is controlled by means of a piezoelectric (PZ) driven poppet valve.

The poppet valves used in this configuration had a step response 30 times faster than the electromechanical valves normally used for control of the main piston. The concept has potential for very fast response of the main actuator. The position control loop is closed with a microprocessor and position transducer. Results from a laboratory demonstration achieved 0.5 millisecond operating speed, although the PZ poppet valves failed to meet their speed objective by a factor of two too slow.

A schematic of the operating elements of the digital control valve and actuator are shown in figure 2.1-14. This implementation used four orifices. Four poppet valves are used for directional control. Valves V1 through V4 are connected from a cylinder poppet of the valve to hydraulic pressure return (operating these valves controls the direction of the piston). Opening valves V2 and V4 causes the valve to move to the right (as shown in figure 2.1-14).

A gallery is connected to the combined output flow from the flow modulation poppet valves and orifices. As shown in figure 2.1-14, the four orifices, labeled O1 to O4, are used for flow modulation. Orifice O1 is connected to the hydraulic supply while O2, O3, and O4 are connected to valves V5, V6, and V7. These three valves, in combination with the non-switched orifice O1, provide metered flow to the directional valves.

The rate that the actuator moves is determined by the size of the orifices and the number that are opened by the poppet valves. The sizes of the orifices are scaled so that the flow increases in powers of two (i.e., O3 causes twice the flow of O2, and O2 causes twice the flow of O1). Thus binary coding of the digital word connected to the three orifices allows eight steps from off to full

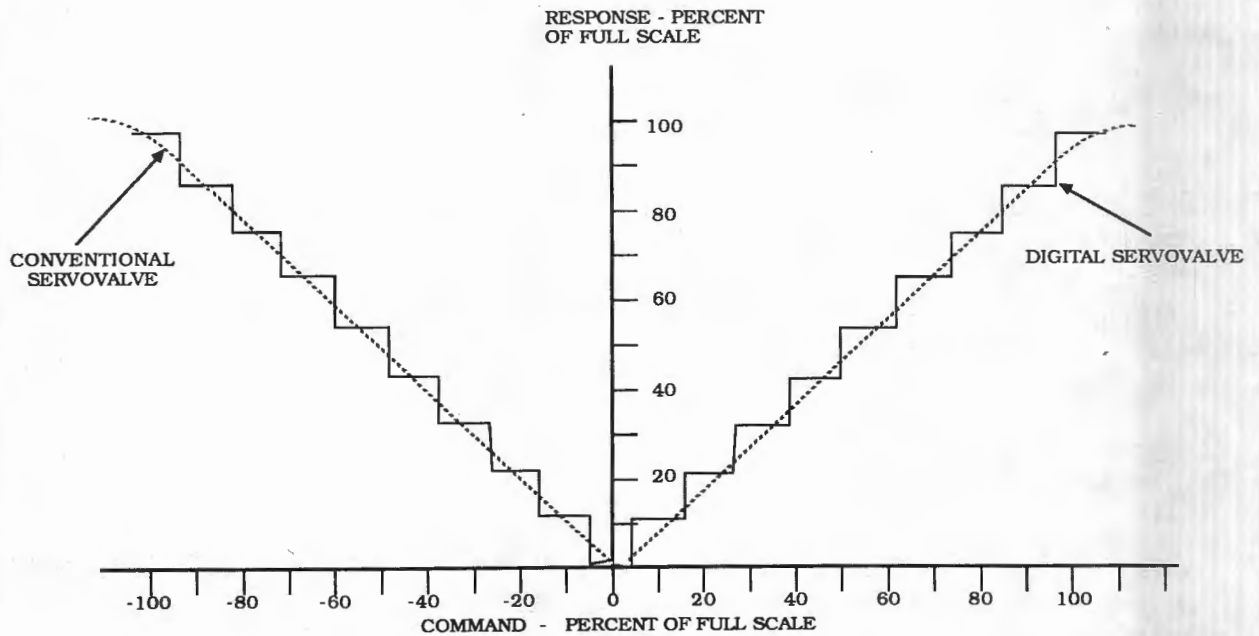
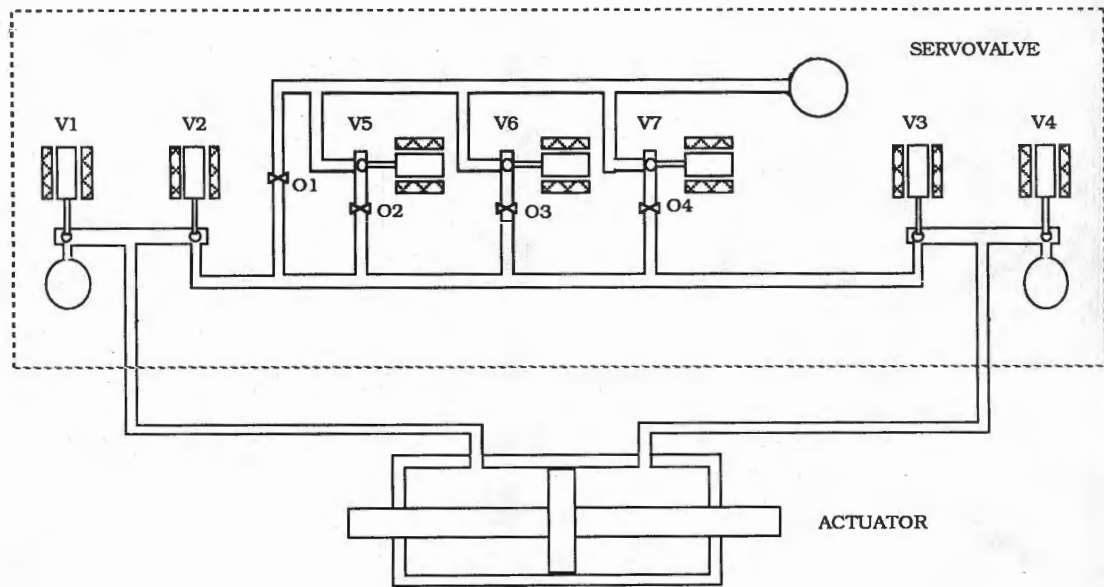


FIGURE 2.1-14. DYNAMIC CONTROLS' ALL DIGITAL SERVOVALVE ACTUATOR

on. With eight steps, flow linearity is maintained within ± 6.25 percent of a linear output. A production article could be built having additional steps and bits in the digital control word. Eight bits would provide 256 flow step combinations (better than 0.5 percent resolution); this is in excess of a more normal requirement of two to three percent linearity for actuator position control. Addition of one more poppet valve would provide a flow linearity of ± 3.1 percent, comparable to most analog servovalves.

From a reliability and redundancy viewpoint, this digital valve appears very promising. The valve motion is determined by fixed elements, the orifices. Control is afforded by digital devices that only operate in an off or on state. Further, if any one of the orifices or associated valves fails to operate, then the actuator functions at lower rates of motion. Degraded performance is usually more desirable than "hardovers" or "no response". The critical control element for establishing hydraulic flow is the poppet valve. Valve response times vary widely for electrical control of mechanical motion. This prototype demonstrates that great improvements in valve speed are possible.

A conventional magnetic solenoid valve is limited in response time by electrical and mechanical constraints. A survey of electromagnetic poppet valves, conducted as part of the research reported in Jenney 1989, provides some interesting results. The on-off response times for commercial poppet valves varied from 2.5 to 14 milliseconds and cost estimates were between \$1,500 and \$4,000 each. All of these valves failed to meet the required response of the control actuator. One reason for this can be found in the dynamics of the mechanical valve components. A typical solenoid valve consists of a spring, armature, and coil, as shown in figure 2.1-15. The motion of these elements requires the magnetic circuit to build up forces to accelerate the armature and open the valve in a

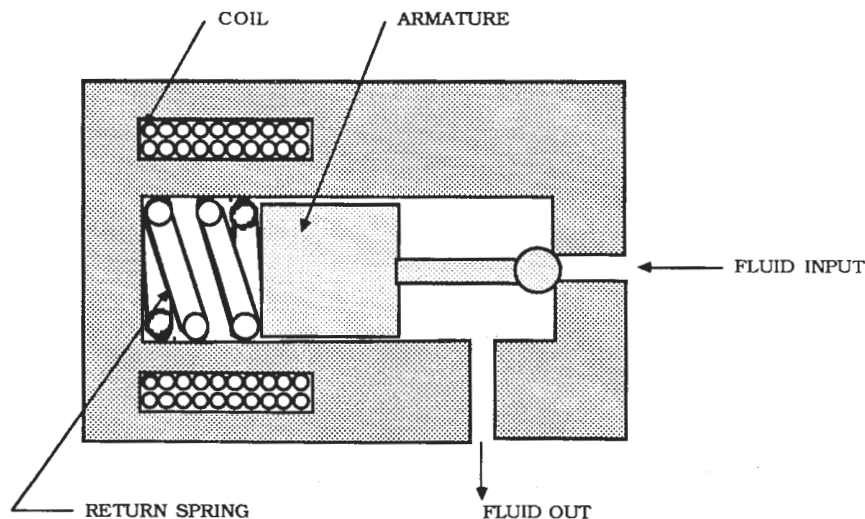


FIGURE 2.1-15. SOLENOID POPPET VALVE

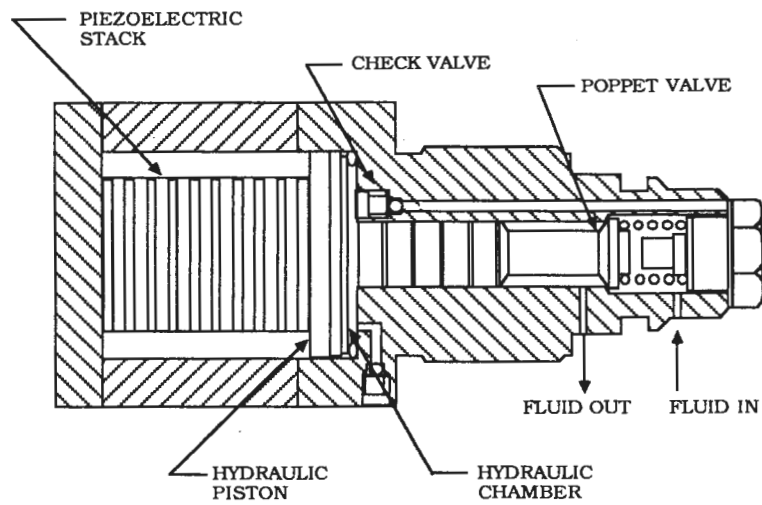
short time. The armature and coil must be sufficiently large to produce forces great enough to overcome the spring restraining force and inertia. For example, an armature weighing 0.022 pounds and travelling 0.02 inches in 0.001 seconds requires a driving force of 2.27 pounds. The spring preload to close a 0.30 inch diameter valve against a 3,000 psi pressure is 2.12 pounds. Thus the magnetic forces must exceed twice the spring force. An additional constraint is in the magnetic coil. A large number of turns on the coil will build up forces for the same current. However, the coil inductance increases with the square of the turns, requiring more time to build up the drive current. The RL time constant ultimately limits the current build up and speed of operation of the poppet valve.

A moving permanent magnet armature has also been tried for this application on other research projects and was found to have the same RL time constant limitation on rapid build up of control currents.

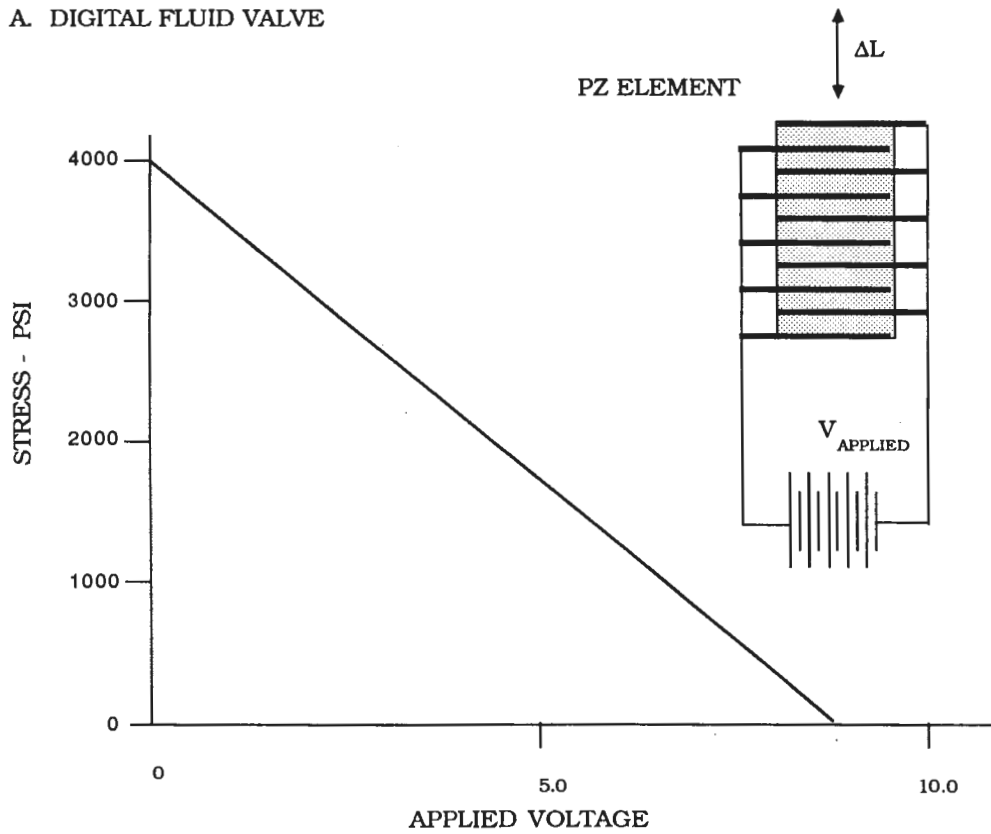
A review of PZ elements for application to poppet valves indicated feasibility of constructing a valve having a 1 millisecond operation time. A PZ material will distort when an electric field is applied to it. Conversely, an electric field will be generated when the material is distorted by a force. Figure 2.1-16 illustrates a poppet valve constructed from a multilayer PZ element.

The quantitative values for the relationship between the mechanical deformation and electric fields depends upon the material type, crystal axis orientation in the field, and physical configuration of the device. The output force capability of PZ materials is very high (on the order of 3,000 psi), although the strain is very small (on the order of 0.1 percent of the thickness). The electrical characteristic of PZ material is the same as a capacitor. Because of the high force to mass ratio for these materials, the strain can be achieved in a few microseconds. Typically, fields of 25,000 volts/inch are required to produce a strain of 0.001 inch. To reduce the driving voltage, PZ drivers are produced in thin layers, bonded together, and connected electrically in parallel. For a field of 25,000 volts/inch, a layer thickness of 0.020 inch only requires an applied voltage of 500 volts. When the material is preloaded, the strain will be determined by the modulus of elasticity and the stored energy. Since Young's modulus for a typical PZ material is about 25 percent that of steel, quite large preloads are no problem. The charge/discharge characteristic displays a large hysteresis: on the order of 20 to 30 percent for typical materials. This hysteresis limits the application of PZs to digital rather than analog control devices.

The laboratory investigation reported in Jenney 1989 achieved an operating digital controlled actuator having PZ poppet valves with 500 microsecond operation times in a hydraulic supply pressure of 3,000 psi. The microcomputer was an off-the-shelf Z80 having a real-time program processing time of 600 microseconds. This is very impressive compared to the fastest hydraulic poppet valve that has an operate time of greater than 5 milliseconds.



A. DIGITAL FLUID VALVE



B. PERFORMANCE OF PZ STACK

FIGURE 2.1-16. PIEZOELECTRIC SERVOVALVE

Since these devices operate very quickly and are easily driven from 270 volt dc with digital controllers, future aircraft should have numerous applications of PZ devices. The high speed and high force capability of the PZ actuators should be applicable to direct-drive force motors for controlling spool type servo-valves.

2.1.3.5 Dynamic Controls' F-15 Flutter Suppression Actuator

There is a trend toward increasing the operating pressures of military aircraft hydraulic systems. Higher system pressure allows the use of smaller, lighter weight actuators with smaller drive areas. This conflicts with the normal design technique of using a larger drive piston area to increase actuator stiffness to avoid flutter of the control surfaces. An increased stiffness is usually accomplished by increasing the ram piston area (i.e., by using an actuator having a higher force output and flow demand than required for maneuvers). An Air Force project (Jenney 1989) developed a prototype actuator having electronic control signals for changing the impedance at the flutter frequency and thus effectively damping control surface flutter.

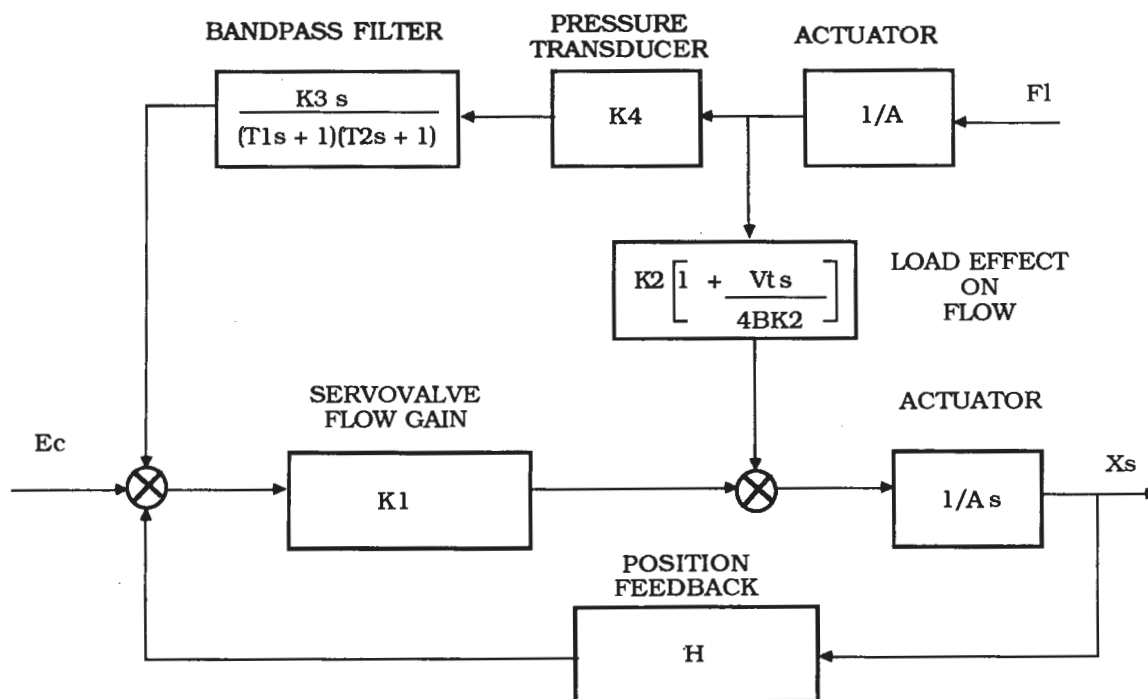
The control surface flutter phenomena are aeroelastic, self-excited vibrations in which airstream energy is absorbed by the control surface. The resulting surface motion consists of bending and rotation, each having a unique frequency and amplitude for a given airstream velocity. If either mode of oscillation is suppressed, classical flutter will not occur. The control surface rotational motion is primarily determined by the spring constant of the actuator and the rotational moment of inertia of the surface. Normally the rotational motion of a control surface is lightly damped. The normal technique for reducing flutter is to raise the resonant frequency by increasing the stiffness of the actuator.

An alternate technique for damping control surface flutter is to change the damping of the surface oscillation to be greater than critical damping, then the motion will be suppressed. This damping could be accomplished by load pressure feedback, thus changing the impedance of the actuator at the flutter resonant frequency. The technique of impedance modification has a good performance payoff by allowing actuators to be sized for maneuver requirements. As an example, the Tornado aircraft uses hydromechanical load-pressure feedback on the horizontal tail actuator for damping flutter.

Figure 2.1-17 shows the block diagram of the flutter suppression actuator. With no-load force, F_1 , the control loop is a conventional electrohydraulic position control system. The effect of the load force is to reduce the flow to the integrating actuator from the servovalve. This reduction is both a steady state and dynamic effect. The steady state effect is a reduction of the flow to move the actuator because of leakage. This load pressure sensitivity is represented by the term K_2 . A secondary effect of the load force feedback reduces the differential pressure across the servovalve (which causes a reduction of the valve output flow). This effect was ignored to linearize the model. This analysis assumption does not introduce large errors if the differential load pressures are much less than the differential supply pressure across the servovalve (within the constraint $P_1 < (2/3)P_s$). The dynamic effect of the load pressure is due to the compliance of the fluid in the actuator and depends on the rate of change of load pressure. The effect is to reduce the dynamic flow

gain of the servovalve (particularly when the flow is a function of actuator motion).

The load pressure feedback, shown in figure 2.1-17, results in a positive feedback loop. The bandpass filter in series with the differential pressure signal across the actuator drive area provides a lead over a double lag filter function. This filter effectively blocks the low frequency effect of feeding back load pressure. Thus, the static stiffness and steady state position accuracy of the main control loop are not affected.



Notes:

E_c = Command input (volts)

T_1, T_2 = Bandpass filter time constants

K_3 = Bandpass filter gain

K_4 = Differential pressure

K_1, K_2 = Servovalve flow gain

V_t = Actuator fluid volume

B = Bulk modulus of fluid

A = Actuator drive area

s = Laplace operator

FIGURE 2.1-17. DYNAMIC CONTROLS FLUTTER DAMPING ACTUATOR

2.2 All-Electric Actuators

Electromechanical actuator (EMA) designs have evolved into a proven technology with a proven record. The material discussed in this section draws upon the discussion of Hair 1985 (refer to AiResearch 1976 for early work). The operation of these actuators is based on the use of high density rare-earth magnetic material and brushless commutated motors, coupled to a mechanical transmission. The motor converts an electric current into torque, which is transferred through the gear box into mechanical motion. The output gearing is connected to the flight control surface with a proper torque speed characteristic for control surface motion.

There are three types of actuator configurations: linear, rotary, and rotary hingeline. For a dual-drive motor configuration, the motor output can be velocity summed or torque summed. The linear actuator uses a ball screw mechanism to transmit mechanical force. Rotary actuators use a planetary gear arrangement for surface motion. A rotary hingeline actuator integrates the actuator into the hingeline and thus into an aerodynamic surface envelope, eliminating the need for any fairing.

2.2.1 Actuation Concepts

An actuator must be designed for a specific level of performance. This performance obviously depends upon the control surface application for a particular aircraft. Actuators for the trailing edge will have different performance than for a rudder. Efficient structural load transfer is also a benefit of the hingeline configuration, as illustrated in figure 2.2-1. Each actuator is unique in terms of performance: stall load, no-load speed, operating bandwidth, stiffness, and rotational envelope.

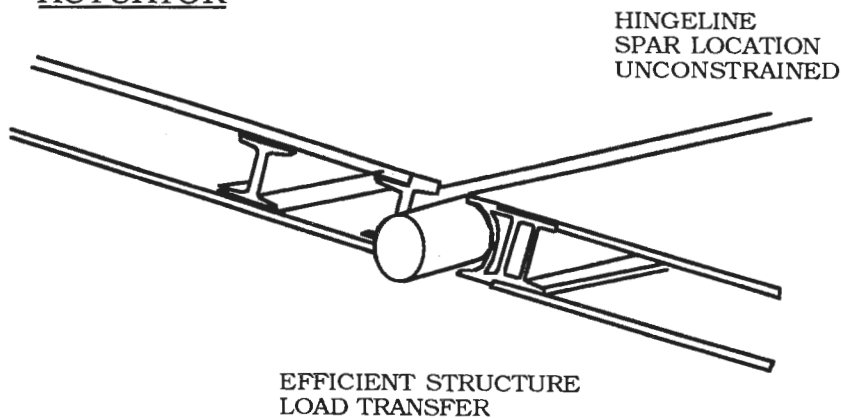
In terms of reliability, one of the first questions that needs answering is fault tolerance. Whether the actuator performance degrades gracefully with multiple faults (i.e., fail-op/fail-safe, fail-op/fail-op/fail-safe, etc.). These performance parameters depend upon the surface application and design limits. Once requirements are established, an electric actuator can be designed to a desired level of fault tolerance. Since faults in components occur more frequently than can be allowed for most applications, the level of fault tolerance determines the ultimate level of reliability for any given actuator. For a typical case, a triplex FCC system (with skewed sensors and reversion mode using analytical redundancy estimators for critical feedbacks) results in a two fault tolerant system. Obviously, redundancy must be applied to the actuation system.

Design to a level of performance is not enough. Performance, reliability, maintainability, cost, size, weight, and power parameters must be met for each application. All these requirements must be taken into the design process and evaluated for each change.

Electromechanical actuation affords considerable flexibility in redundancy designs. Multiple channels can be accommodated in an integrated package. The single motor design, shown in figure 2.2-2, provides actuation where redundant surfaces can be used for fault tolerance. The motor is mounted on one end and,

in the event of motor failure, surface motion is restrained by the brake mounted on the other end. Dual channel actuators, having a single output shaft driven by two motors, can tolerate single failures. A hingeline actuator tends to favor a dual motor for each channel, such as that shown in figure 2.2-3, due to the natural integration of the motors and gearbox. The motor/gearbox outputs may be summed as torques or speed. In the case of a torque summed actuator, both motors are connected along a common shaft. In the event of a failure, the failed motor is decoupled through a clutch and the remaining motor delivers one-half of the original torque. Similarly, a velocity summed dual-channel actuator allows operation at one-half speed, the failed motor being isolated with a brake.

ROTARY HINGELINE ACTUATOR



HYDRAULIC LINEAR ACTUATOR

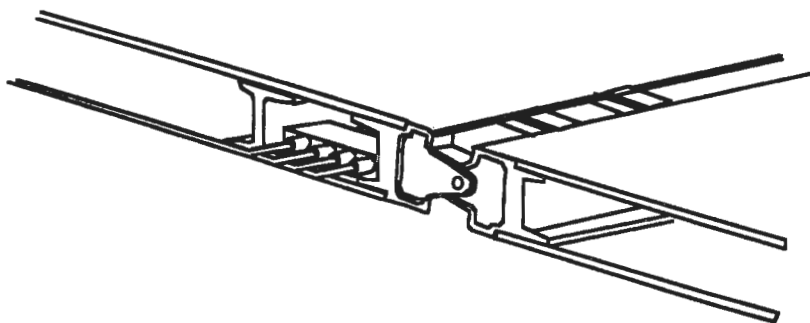


FIGURE 2.2-1. ELECTRIC ACTUATOR CONCEPTS

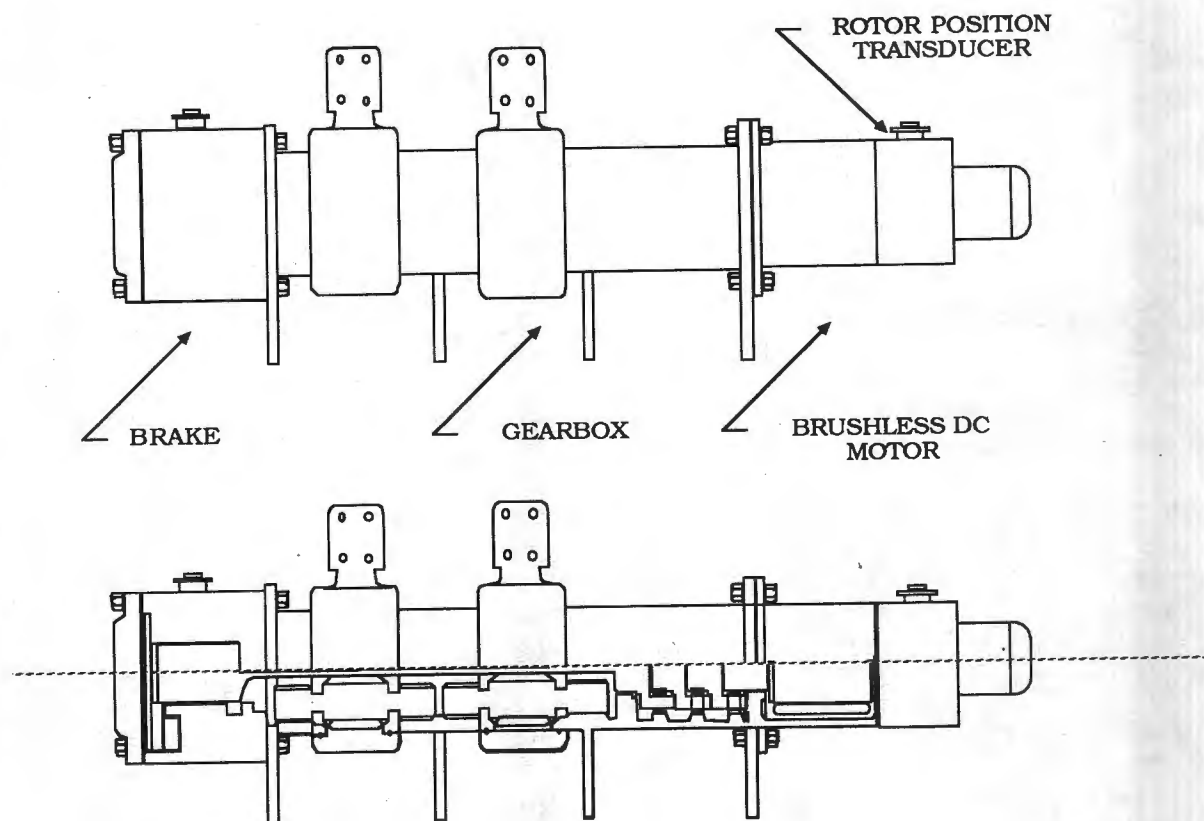


FIGURE 2.2-2. SINGLE MOTOR HINGELINE ACTUATOR

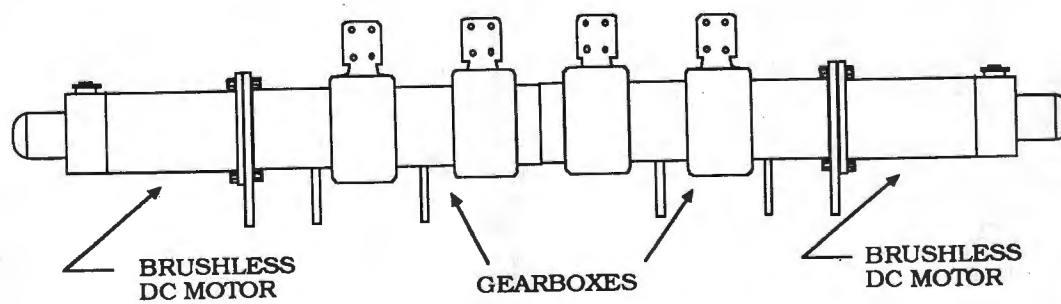


FIGURE 2.2-3. DUAL MOTOR HINGELINE ACTUATOR

Actuation performance under single faults is assured by sizing the actuator for 100 percent performance with loss of a single motor. This requires a 200 percent capability for a dual channel under no-fault conditions. Multiple channel configurations have lower size penalty. A four channel actuator, having four motors driving a common shaft, would be sized for three motors to provide the required torque. This would only require one-third increase in motor capability. If a single motor fails, then this quad configuration operates with full output.

2.2.2 Active Isolation

The ability to actively isolate motor and gearbox failures with brakes and clutches is a key advantage for EMAs over hydraulic systems. Although various computer logic and hydro-logic schemes can isolate failures in electrohydraulic actuators, jamming one piston of a dual actuator results in the complete failure of the actuator. The isolation, by brakes or clutches, of a failed motor or gearbox allows the EMA to continue with any single failure.

Thus active isolation results in tolerance for any single failure in a motor/channel. Typical of this case would be loss of power to one motor of a dual channel. A fault detection and isolation system would determine which motor was without power and, for a velocity summed actuator, engage the brake to isolate and contain the failure. If the brake were not engaged, then the motor would back drive the failed motor through the gearbox. A jammed motor could be isolated without the brake. A jammed motor is the critical failure for the torque summed actuator. When one motor jams, the clutch will be activated to release the jammed motor. This requires a signal from the fault detection system. However, the motor remains engaged in the event of a failure in the clutch electrical circuit. A subsequent second motor jam could not be acted upon with a failed clutch circuit.

2.2.3 Gearbox Technology

The gearbox is critical for operation of these EMAs. All the designs use some form of gears for mechanical transmission. Rotary actuator designs employ planetary gears in simple and compound arrangements to transmit the appropriate torque and speed. Linear actuators use ball and tube gears having similar criticality and constraints.

Planetary gearboxes have large gear reductions (on the order of 500 and 1,000 to one) and can be constructed in small volumes. The gearbox is the place where all loads and signals come together to form the final force output to the surface. This makes the gearbox the critical link. Without redundancy, an EMA relies on several gears to move the surface. This gear train should have redundancy; the cost of this redundancy is significant in terms of size, weight, and complexity.

2.2.4 Reliability

An Air Force sponsored study performed by Grumman (Hair 1985) developed some insight into the relative reliability of EMAs and electrohydraulic systems. For the study two actuators were compared for controlling an F-14's rudder surfaces. A prototype EMA was designed and tested as part of the effort so the results are

a valid point design. The selected electrohydraulic actuator was for the F-14 rudder actuation system. A failure modes and effects criticality analysis (FMECA) was conducted according to defined criticality classes I through IV (results are in table 2.2-1 [Hair 1985]). An initial observation is the increased number of failure modes for the EMA compared with the electrohydraulic unit. This increase is largely due to the increased number of parts, particularly in the gearbox. There is a need to reduce the number of parts in these actuators. Current research is on simpler actuators, development of direct drive valves, and reconfigurable flight control systems relying on simplex actuators.

TABLE 2.2-1. CANDIDATE ACTUATOR FAILURE MODES COMPARISON

ACTUATOR TYPE/ CRITICALITY CLASS	F-14 ELECTROHYDRAULIC	F-14 ELECTROMECHANICAL
CLASS I	3	11
CLASS II	10	12
CLASS III	3	0
CLASS IV	114	266
TOTAL MODES	130	289

The electrohydraulic actuator studied for the F-14 rudder system only exhibited three failure modes which resulted in Class I failures. These failures could result in the immediate forced landing of the aircraft and probable damage to the aircraft. These two modes are piston rod/rod end mounting structure failures and failures of the valve housing. Consequently, the probability of a failure is very low.

EMAs, on the other hand, exhibited 11 catastrophic failures. Two of these were structural mounting failures while the remaining ones were due to failures of nonredundant gears within the gearbox. Nonredundant gears are the major area for potential failures. The root of this problem is that the gears can fail in a number of ways, including wearing and shearing gear teeth, fracturing, and jamming. There are also multiple ways of failing the actuator controller. All of these failures combine to make the probability of failure larger than for electrohydraulic actuators. Since the structural failure modes have low probability of occurrence, the gearbox modes constitute the major contribution to increased failure probability of this EMA.

2.3 Actuation Reliability and Redundancy Technology

Fault diagnosis is the process of determining if a failure has occurred and, if so, what component or subsystem has failed. This information is sent to the

failure management system which determines an appropriate response to the failure. The tasks making up fault diagnosis and failure management are frequently referred to as failure detection, fault isolation, and system recovery and reconfiguration. The information flow for these tasks is illustrated in figure 2.3-1. In this section the contents of these tasks are discussed in general terms drawing from Baker 1988. This material provided a basis for further understanding of fault diagnosis and failure management.

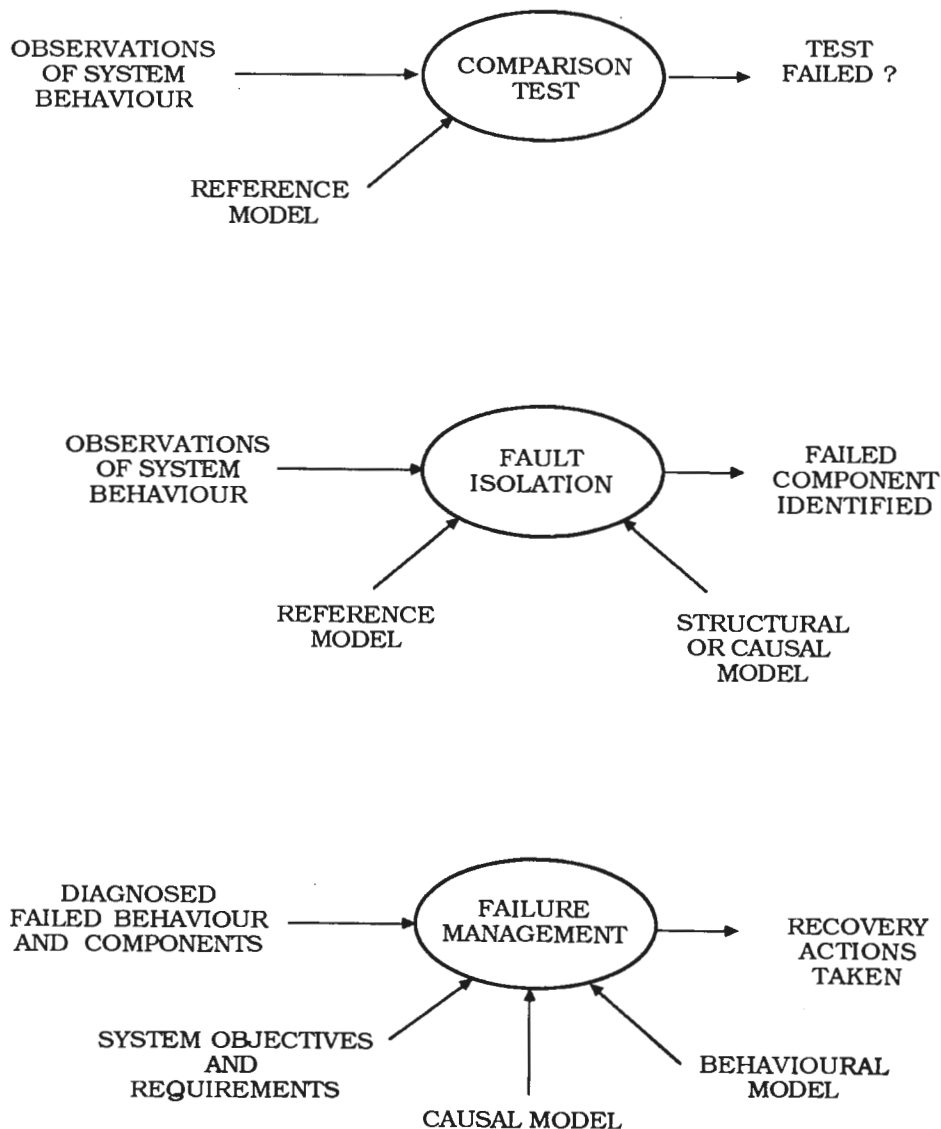


FIGURE 2.3-1. FAULT DIAGNOSIS AND FAILURE MANAGEMENT

2.3.1 Failure Detection

Failure detection is the operation of distinguishing between normal and abnormal behavior of a system. The detection process consists of a continuous cycle of monitoring, information processing, and comparison testing.

In general, a failure is detected by monitoring the behavior of a component, subsystem, or system of interest; converting the raw data into a useful form by filtering or numerical processing; and comparing the resulting measurement with the predicted normal values. A failure is detected when the comparison test inputs differ. The outcome of this comparison test is usually a binary value "ok" or "failed". This approach neglects simultaneous failures in the two channels and assumes the comparator did not fail. Common mode failures must be detected in other ways. For example, common mode failures can be detected by monitoring individual channel inputs and outputs and by monitoring the comparator in a calibration test.

Performance of the failure detection system depends upon the sensor measurements, reference models used to predict the measured values (or system state values derived from the sensors), and the comparison test. The sensors are an important part of the detection process because they are an additional source of failure that must also be managed.

Reference models can take on several forms. Use of hardware reference models is referred to as direct redundancy. Use of software models is referred to as analytical redundancy. In the case of software reference models, the most common models are analytical functional relationships. Using these models and information from the system other than direct measurements, a reference for the measured values is synthesized. Accuracy for these reference models can vary from approximate (to detect hardover failures) to highly accurate (to detect soft failures in performance optimization systems).

Either the normal behavior can be modeled or the failed behavior can be modeled. When modeling normal behavior, the reference model can be implemented in hardware or software. For hardware reference models, a duplicate set of hardware can be utilized to develop the normal behavior.

When modeling the failed behavior of a system, modeling all possible component failure modes is very challenging because the failures occur in many different ways and at different times. As a result the failure models are usually approximate.

One failure modeling method in use monitors each critical component or failure mode. The disadvantage of this scheme is that the failure modes must be defined in advance, before a system is fully developed. Nevertheless, there are acceptable systems that detect faults utilizing models of failed behavior.

Two failure models that are commonly used for fault detection are range checking and trend checking. Range checking declares a failure whenever an output exceeds a conservative estimate of the operating range for that variable. Trend checking declares failure when a variable has an abrupt change that is not physically possible.

Threshold detection is the most common fault detection method. For this method a threshold is set on the difference between the model and measured behavior. Errors occur from sensor noise so thresholds must be set high enough to account for these errors. Detection thresholds do not have to be constant. Some systems use scheduled thresholds.

Missed failures and false alarms plague all these fault detection systems. False alarms are of two kinds: removal of good equipment by faulty detection, and failure to remove faulty equipment.

System reliability depends upon the fault coverage of the detection scheme. Careful testing is necessary to verify that all possible faults can be detected by whatever detection method is used.

2.3.2 Fault Isolation

The process of isolating faults after a failure has occurred is called fault isolation. Faults may be isolated at a component, card, or channel. The results of this process are correct isolation, incorrect isolation, or no decision.

Approaches are local isolation, arbitration, and generate and test. Local isolation identifies failed components immediately upon test failure. Its implementation relies upon sensors monitoring specific components.

Arbitration uses comparisons of only two sources of information. Other information channels, such as from multiple sources, are tested two at a time and grouped by pairs. The failed components are then identified by using the test results from the pairs to isolate the failed component. The data for arbitration may be either direct data from redundant hardware or indirect data calculated from analytical models

The arbitration method is most efficient when used with sensors, as the outputs can be compared two at a time and isolated directly. When isolating other components or subsystems using arbitration, the sensed information used to isolate a failed component must be separately validated. For example, if individual sensors were used to isolate three redundant components, at least two sensors would be required on each component. With only one sensor per component, a sensor failure cannot be isolated from a component failure. A sensor failure on one of the three components can be isolated if the component being measured could be assumed not to fail at the same time, and if one of the sensors agrees with sensors on the other components.

Most other approaches to fault isolation include generation of a hypothesis of which component has failed and then testing the validity of the hypothesis. This requires identifying candidate failed components and testing to verify if the component failed. If the candidate has failed, isolate it; otherwise repeat the prior steps with another candidate.

Hypothesis testing requires a simulation of candidate component failures and generation of data for comparison with measurements on the system. This process is very challenging to implement for a large system.

There are several benefits and disadvantages of the generate and test method. This is a very powerful method, having fewer sensors and processors than other methods. Indirect methods have some ability to adapt to changes. However, this method has several weaknesses. There is uncertainty in the convergence of the process, the solution times are variable, and the candidate selection process is arbitrary. Computation requirements are greater for generate and test than for other methods of fault isolation.

2.3.3 Failure Management

Failure management logic evaluates failures and determines the proper response. Part of this logic must determine, in a very short time, where in the process to return control. Control should be returned at a point where a certain level of performance can be maintained.

The logic process and algorithms for fault management depend upon the process and the equipment still functioning. Few systems take fault isolation information and execute a predetermined sequence of actions.

Recovery usually means to shut down the processor and restart, bringing the system back to a known condition. Two processes are used: the first is reconfiguration to eliminate failed components, and the second is to recover control. Logic is also needed to restore control in a condition where the reconfigured system will operate properly.

The recovery process normally includes other actions taken to correct or minimize failure effects in addition to reconfiguration. These actions may include the following:

- Shutting down operation in a manner to provide a safe condition. Actuators may be locked holding control surfaces in a neutral position. Alternatively the actuators may allow control surfaces to float in the air stream.
- Opposing the effects of failure while the system is reconfiguring. For actuators, this may be a force fight with opposing valves or solenoid coils.
- Returning the system to a condition where control can be resumed. For actuators, this may happen automatically during reconfiguration.

Recovery is generally not needed for actuators as they reconfigure quickly; failures only cause small transients. The discussion in section 2.1 covers the reconfiguration methods for several actuators.

3.0 Digital Engine Control Reliability and Redundancy

3.1 Reliability Models for FADEC Systems

The utilization of full-authority digital engine controller (FADEC) systems for military and commercial aircraft is well under way (Newirth and Bosco 1980; and Barclay, Lenox, and Bosco 1978). The critical design requirement behind this development is achieving high reliability and availability. Current approaches use redundant sensors, processors, and actuators for meeting these reliability needs.

Design of redundant systems requires analytical tools for supporting the design process. The capability to quickly determine the effect of different design decisions on the reliability of the system is of great importance. Reliability models range from simplified parametric analysis of broad system level descriptions to detailed logic gate level models. There are numerous tools for building and analyzing reliability models including algebraic methods, computer simulation, hardware emulations, and logic analyzers. (Dotson 1988, January and August).

This section provides two examples of reliability models for electronic engine controllers, including a parametric analysis using system level models and a more detailed Markov model. An example is presented describing in detail the development of a Markov model for a dual redundant FADEC system. This model contains the detailed features of the design which impact the reliability. This detailed state model is simple to use and widely applicable. In the final subsection, additional examples are presented illustrating design tradeoffs that can be made using algebraic system level models. Although these models require parameters normally defined by more complex Markov models, they provide insight for system designers.

The operational state of a redundant system changes as the result of component failures, decisions made upon detection of faulty components, and reconfiguration resulting from control system actions to isolate and remove faulty components. Markov modeling techniques provide a natural method for determining the effects of random failures and assessing the effects on system performance. The classical fault tree (combinational) approach is less effective for dealing with changes to system configuration. Markov models easily accommodate the time varying probabilities and effects which occur in reconfigurable systems. Implementation of a Markov model is simplified because first order approximations can be used to define the state transition probabilities.

3.1.1 Detailed State Model For FADEC

The principal challenge in developing a Markov model is definition of the model states. As the number of components in a system increases linearly, the number of states in a model increases exponentially. The analyst's challenge is to construct a table of states that does not require too many entries. A complex

redundant system could be partitioned into less complex subsystems and modeled in segments. The results could then be combined logically to obtain results for the larger system (Smith et al. 1976). The approach demonstrated in this section is summarized from Gai, Harison, and Luppold 1983. This modeling approach uses prior knowledge about the system to construct a model having an acceptable minimum number of states in a single model. This is preferable to combining results from many models where any insight from the model on system behavior may be lost in the numerous subelements.

The organization of the components in a FADEC is shown in figure 3.1-1. Two processors form the central part of the system. Each processor, associated power supply, and set of dedicated inputs and outputs make up a single channel. Communication between channels is via a Universal Asynchronous Receiver Transmitter (UART) interface. This serial data channel operates at a low bandwidth compared to the processor internal operations. Outputs of both processors are symmetric, allowing either channel independently to control the fuel flow for the engine and control of stator vane angles. Inputs to the system are asymmetric and include the following:

- Power level actuator (PLA) sensors
- Engine pressure (P2, Pb, P5, P_{amb}) sensors
- Fan and turbine speed (N1, N2) sensors
- Temperature (T1, T2.5) sensors
- Temperature (TD) diodes
- Feedback sensors from fuel flow actuators (WFB)
- Feedback sensors from stator vane actuators (SVA_{fb})
- Interfaces from independent air data systems (AIR)

During normal operation, the primary channel retains control of the engine until the first failure. When a failure of a sensor or actuator associated with the primary channel is detected, the primary channel substitutes the secondary sensor if available. Failure of a sensor which is not dual redundant results in a transition to a control mode not requiring that sensor, and control continues to reside in the primary channel. Transfer of control to the secondary channel occurs only in the event of failure of the primary processor or its power supply.

The reliability requirement of this engine controller is three shutdowns in a million hours. This is a probability of failure of 0.333×10^{-6} .

Redundancy management for the system is based on self-tests for failure detection and identification. The self-tests for processors and memories include watch dog timers, parity checks, and memory check sums. Built in self-tests are used for all sensors and interfacing electronics, and wraparound tests are performed

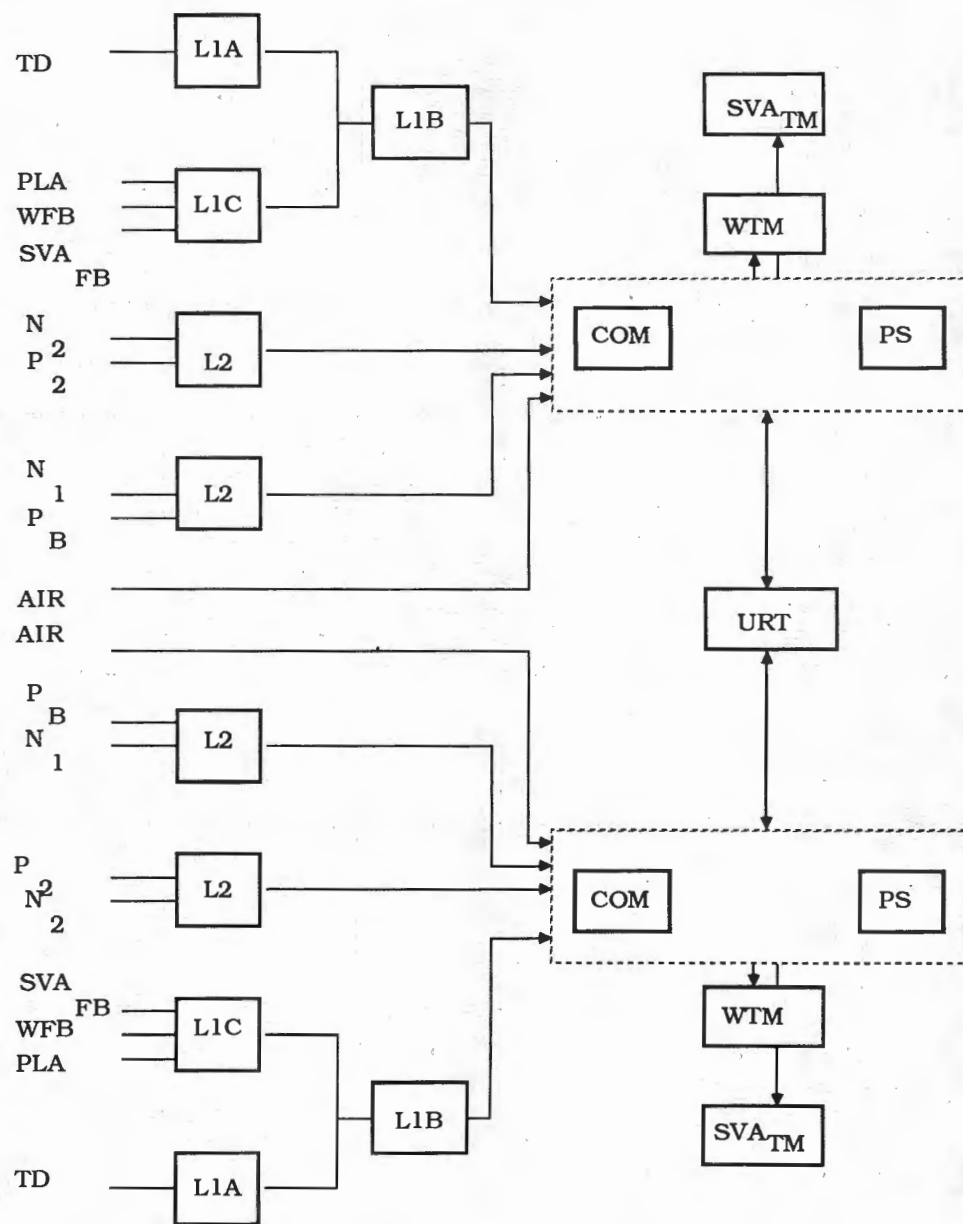


FIGURE 3.1-1. DUAL REDUNDANT ENGINE CONTROLLER

on output commands. In addition, each sensor input is tested for out of range conditions. Other redundancy management techniques include sensor comparisons and use of synthesized versions of sensor measurements as references. These analytical redundancy techniques are not considered in this discussion, but are normally implemented in operational controllers (refer to section 3.2).

The engine controller, shown in figure 3.1-1, operates in four distinct modes: Engine Pressure Ratio (EPR), Low Rotor Speed (N1), High Rotor Speed (N2), and Fuel Flow to Burner Pressure (W/P). The components that support these modes are listed in table 3.1-1. The preferred control method is EPR. For this mode the pressure sensor P5 is used for control. When P5 is not available, control is transferred to the N1 mode. When neither N1 or EPR are available, the N2 mode is used. Loss of N2 reverts to W/P mode.

TABLE 3.1-1. COMPONENTS REQUIRED FOR ENGINE CONTROL MODES

ENGINE CONTROL MODE	SENSORS REQUIRED												
	PLA	WTM	WFB	T2*	P2*	PB	P5	N2	N1	P* _{amb}	SVA _{TM}	SVA _{FB}	T2.5
ENGINE PRESSURE (EPR)	X	X	X	X	X		X	X		X	X	X	X
FAN SPEED (N1)	X	X	X	X	X			X	X	X	X	X	X
TURBINE SPEED (N2)	X	X	X	X				X			X	X	X
FUEL FLOW (W/P)	X	X	X			X		X			X	X	X

* This information is derived from available air data.

3.1.2 Markov Model

The choice of system states for modeling is made in a systematic manner. Group 1 consists of one state representing the normal mode of operation in which there have been no failures or false alarms. Group 2 consists of those states having one failure or false alarm. Group 3 consists of those states where there have been more than one failure or false alarm.

Group 1 states, being the normal situation, follow from the normal definition of the system diagram. Group 3 states can be aggregated into the inflight shutdown state for any component of a channel or for the sensors. Group 2 states provide the main modeling challenge.

For modeling group 2, first consider the states resulting from fault detection of a failure (or false alarm) of a primary actuator or sensor. The effect of

any detected failure is the same: a sensor or actuator is designated as failed and removed from the system. Thus, for each of the sensors or actuators, one state can be used to model detected failures. Failures or false alarms of dual redundant interface modules are treated in the same way. One state can model effects of covered failures or false alarms of nonredundant components.

If the effect of removing a component from the primary channel is different from the effect caused by removal of its counterpart from the secondary channel, then failures or false alarms of primary and secondary components must be modeled by separate states. The processors, power supplies, and interface modules are in this category since removing them from the primary channel causes the transition from EPR to N1 mode. Removing them from the secondary channel leaves engine control in the EPR mode. Finally, an uncovered failure of any component in the primary control channel is assumed to result in an in-flight shut-down. There are a total of 30 states in group 2. Since any of the states from group 2 can transition to the shutdown state, the 30 states, shown in table 3.1-2, are all that is needed for the model of the dual engine controller.

3.1.3 Transition Probabilities

The second step in model development is to determine the probabilities of transition out of the system states for groups 1 and 2.

The transition probabilities for changing states are required to quantify each of the steps shown by the model. They are calculated as being single independent failure events at each time step. This assumption is completely general because the time step interval can be arbitrarily small. The single step transitions for the FADEC under consideration are shown in figure 3.1-2.

Four of the transitions out of state 1 correspond to a change in control mode. These are transitions from the EPR mode to the N1 mode (states 6, 15, 17, and 19). These transitions are due to detected and corrected failures of the P5 pressure sensor, its interfacing electronics, or the computer or power supply which make it available to the system. The transition from state 1 to state 30 accounts for the uncovered failures of components in the primary engine control channel. Finally $p(1,1)$ represents the probability that no transition out of state 1 occurs in a single time step; this is computed as one minus the sum of the transition probabilities of all the other transitions.

As an example of the considerations used in calculating the probabilities, consider the transition out of state 2 (one PLA failure), as shown in table 3.1-3. This transition is due to failures in either of the PLA sensors or false alarms from the primary PLA. In order to calculate the probabilities, it is conservatively assumed that a covered failure of the primary channel caused the system to enter this state. As a result, the UART, processor, power supply, interfacing electronics, and PLA from the secondary channel must be included in the set of required components. Because of the detected failures and control reversion, the system is now vulnerable to failures of these additional components. This would not be the case if state 2 were the result of false alarms associated with a detected failure of the secondary PLA.

The transitions out of state 2 lead to state 26 (EPR mode with two failures), state 27 (N1 mode with two failures), or state 30 (inflight shutdown). The detailed investigation in Gai, Harison, and Luppold 1983 indicates 43 events are included in the transitions to state 26, five events lead to state 27, and 23 events lead to state 30. These computations are beyond the scope of this section and the reader is referred to Gai, Harison, and Luppold 1983.

TABLE 3.1-2. MARKOV MODEL STATE DEFINITIONS

STATE	DEFINITION
1	Normal mode, no failure or false alarms
2	One PLA failure or false alarm
3	One WFB failure or false alarm
4	One WTM failure or false alarm
5	One AIRINC failure or false alarm
6	P5 failure or false alarm
7	PB failure or false alarm
8	One N1 failure or false alarm
9	One N2 failure or false alarm
10	One TD failure or false alarm
11	One LIA failure or false alarm
12	One LIB failure or false alarm
13	One LIC failure or false alarm
14	One L2 failure or false alarm
15	L3 failure or false alarm on right side
16	L3 failure or false alarm on left side
17	COM failure or false alarm on right side
18	COM failure or false alarm on left side
19	PS failure or false alarm on right side
20	PS failure or false alarm on left side
21	UART covered failure or false alarm
22	UART uncovered failure
23	One SVA _{FB} failure or false alarm
24	One SVA _{TM} failure or false alarm
25	T2.5 or L4 failure or false alarm
26	EPR mode with two failures or false alarms
27	N1 mode with two failures or false alarms
28	N2 mode with two failures or false alarms
29	W/P mode with two failures or false alarms
30	Inflight shutdown condition

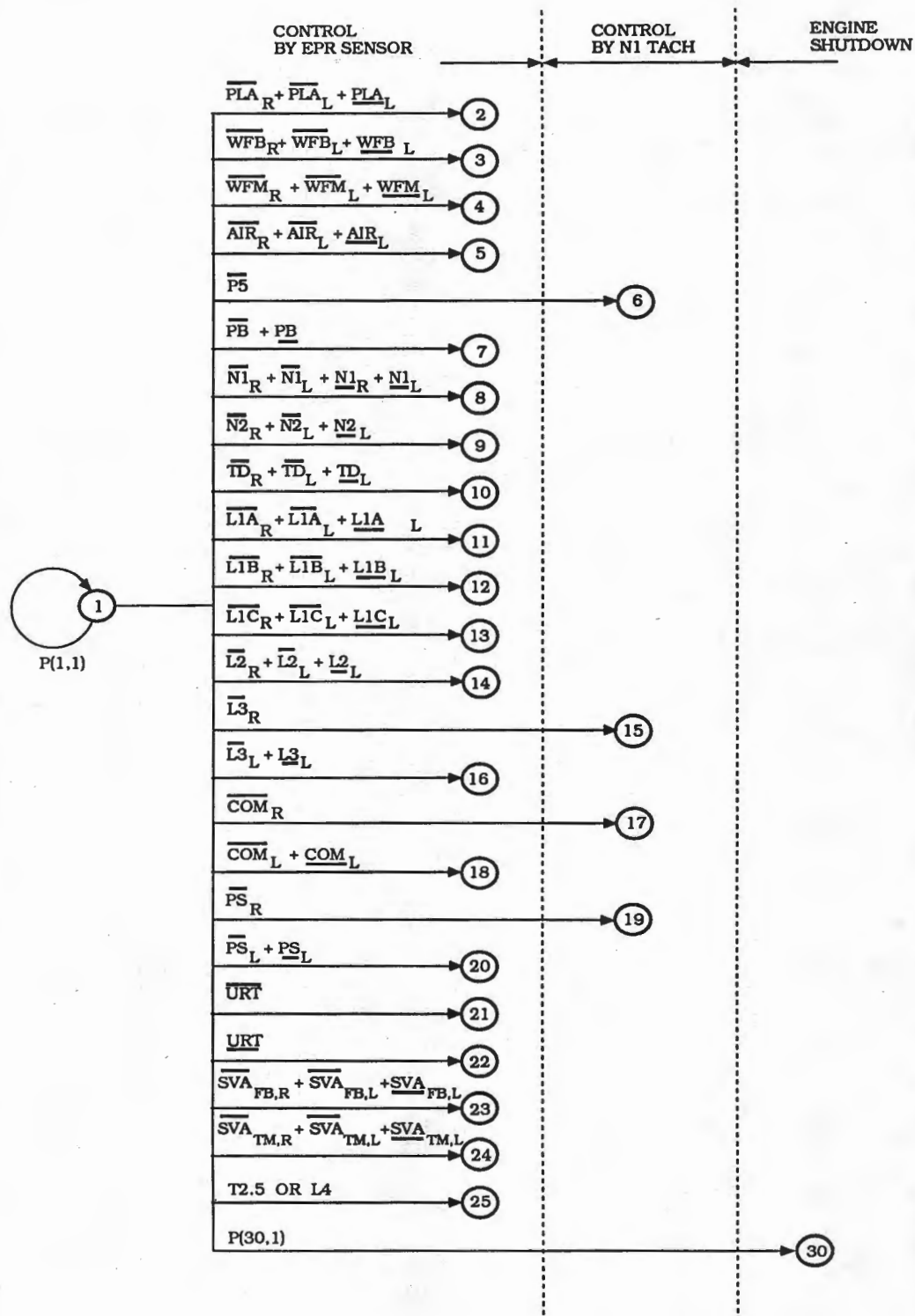


FIGURE 3.1-2. STATE TRANSITIONS FOR ENGINE CONTROLLER OUT OF STATE 1

TABLE 3.1-3. PARAMETERS FOR TRANSITIONS OUT OF STATE 2

Transitions to State 26	Transitions to State 27	Transitions to State 30
$\overline{WFB}_R + \overline{WFB}_L + \underline{WFB}_L$		$\underline{WFB}_R$
$\overline{WTM}_R + \overline{WTM}_L + \underline{WTM}_L$		$\underline{WTM}_R$
$\overline{AIR}_R + \overline{AIR}_L + \underline{AIR}_L$		$\underline{AIR}_R$
	$\overline{P5}$	$\underline{P5}$
$\overline{PB} + \underline{PB}$		
$\overline{N1}_R + \underline{N1}_R + \overline{N1}_L + \underline{N1}_L$		
$\overline{N2}_R + \overline{N2}_L + \underline{N2}_L$		$\underline{N2}_R$
$\overline{TD}_R + \overline{TD}_L + \underline{TD}_L$		$\underline{TD}_R$
$\overline{LIA}_R + \overline{LIA}_L + \underline{LIA}_L$		$\underline{LIA}_R$
$\overline{LIB}_R$		$\underline{LIB}_R + \overline{LIB}_L + \underline{LIB}_L$
$\overline{LIC}_R$		$\underline{LIC}_R + \overline{LIC}_L + \underline{LIC}_L$
$\overline{L2}_R + \overline{L2}_L + \underline{L2}_L$		$\underline{L2}_R$
$\overline{L3}_L + \underline{L3}_L$	$\overline{L3}_R$	$\underline{L3}_R$
	$\overline{COM}_R$	$\underline{COM}_R + \overline{COM}_L + \underline{COM}_L$
	$\overline{PS}_R$	$\underline{PS}_R + \overline{PS}_L + \underline{PS}_R$
		$\overline{PLA}_L + \underline{PLA}_L$
	$\overline{URT}$	$\underline{URT}$
$\overline{SVA}_{FB,R} + \overline{SVA}_{FB,L} + \underline{SVA}_{FB,L}$		$\underline{SVA}_{FB,R}$
$\overline{SVA}_{TM,R} + \overline{SVA}_{TM,L} + \underline{SVA}_{TM,L}$		$\underline{SVA}_{TM,R}$
(T2.5 or L4)		(T2.5 or L4)

3.1.4 Effect of Maintenance

Redundant fault tolerant systems can operate with one or more failed components and thus have higher reliability than simplex systems. This reliability enhancement only occurs if the redundant system is operated over short times relative to its mean time to failure (Clements 1980). This is because early in the operational life, not all the components are necessary for operation. As time passes and some components fail, more of the redundant components are added to the required set for system operation. As a result the failure probability of a redundant system is not constant, but increases with time. Therefore, the proper use of a redundant system is to operate it over a much shorter time than the mean time to failure. In addition, periodic maintenance is used to replace all the failed components, bringing the unit up to "like new" operational condition. Only in this way can a system be operated with low probability of system failure during the operational cycles.

The effect of the maintenance time interval on the system failure probability can be demonstrated with a simple model. Let $N(t)$ be the number of failures per hour at time t , where $t \ll T_s$, the mean time to failure for the system. If, during operation, maintenance is performed every T_m hours, where $T_m \ll T_s$, then the expected number of failures in t hours is

$$N(t) = (t/T_m) * P_{isd}(T_m) \quad [1]$$

The probability of failure of a dual redundant system with coverage C is given by

$$P_{isd} = (T_m/T_s) * [(1-C) + T_m/T_s] \quad [2]$$

Equations 1 and 2 imply that the number of failures in a million (10^6) hours with maintenance is

$$N_m = (10^6/T_m) * [(1-C) + T_m/T_s] \quad [3]$$

The parameter T_s , mean time for system failure, is determined from the Markov model solution. Since this is beyond the scope of this section, the reader may refer to Gai, Harison, and Luppold 1983 for details. The steps involve determining the model parameters for each of the state transitions in figure 3.1-2, and development of powers of a matrix, or alternatively, solutions of electrical circuit equations. The base line values for the component failure rates used for these calculations are shown in table 3.1-4.

The results of solving the Markov model for the probability of being in a given system state are as follows:

Operating in EPR Mode	0.995
Operating in N1 Mode	4.3×10^{-3}
Operating in N2 Mode	3.2×10^{-6}
Operating in W/P Mode	1.2×10^{-6}
Inflight Shutdown	4.0×10^{-4}

TABLE 3.1-4. BASE LINE COMPONENT FAILURE RATES

Component	Probability of Failure in 1 Hour
PLA	2.7×10^{-6}
WFB	2.7×10^{-6}
SVA _{FB}	1.0×10^{-6}
WTM	1.3×10^{-6}
SVA _{TM}	1.3×10^{-6}
AIR	2.5×10^{-6}
P5	4.6×10^{-6}
N2	1.0×10^{-7}
TD	1.6×10^{-8}
L1A	1.0×10^{-6}
L1B	2.1×10^{-6}
L1C	9.7×10^{-7}
L2	1.4×10^{-6}
L3	1.7×10^{-6}
COM	2.1×10^{-5}
PS	3.6×10^{-6}
URT	5.0×10^{-6}
N1	1.0×10^{-6}
PB	4.6×10^{-6}
T2.5 AND L4	2.7×10^{-6}

The probability of operating in the base line EPR Mode is very high (0.995); the probability of inflight shutdown is fairly low (4.0×10^{-4}). As a check, assume perfect coverage, $C=1$, and use equation 3 above to calculate a failure rate of 2.7 shutdowns in a million hours of operation.

Two other computational results are interesting. Figures 3.1-3 and 3.1-4 show the sensitivity of inflight shutdown to maintenance interval and fault coverage. These results indicate little sensitivity to maintenance interval for a 0.9 coverage. However, the maintenance interval is very important for lowering the probability of shutdown at higher levels of fault coverage. For a simplex system, inflight shutdowns are approximately eight in a million hours as shown by the curves in figure 3.1-3 for 0 coverage. Achievement of the 2.7 per million hours requires a fault detection and recovery system having fault coverage on the order of 0.95.

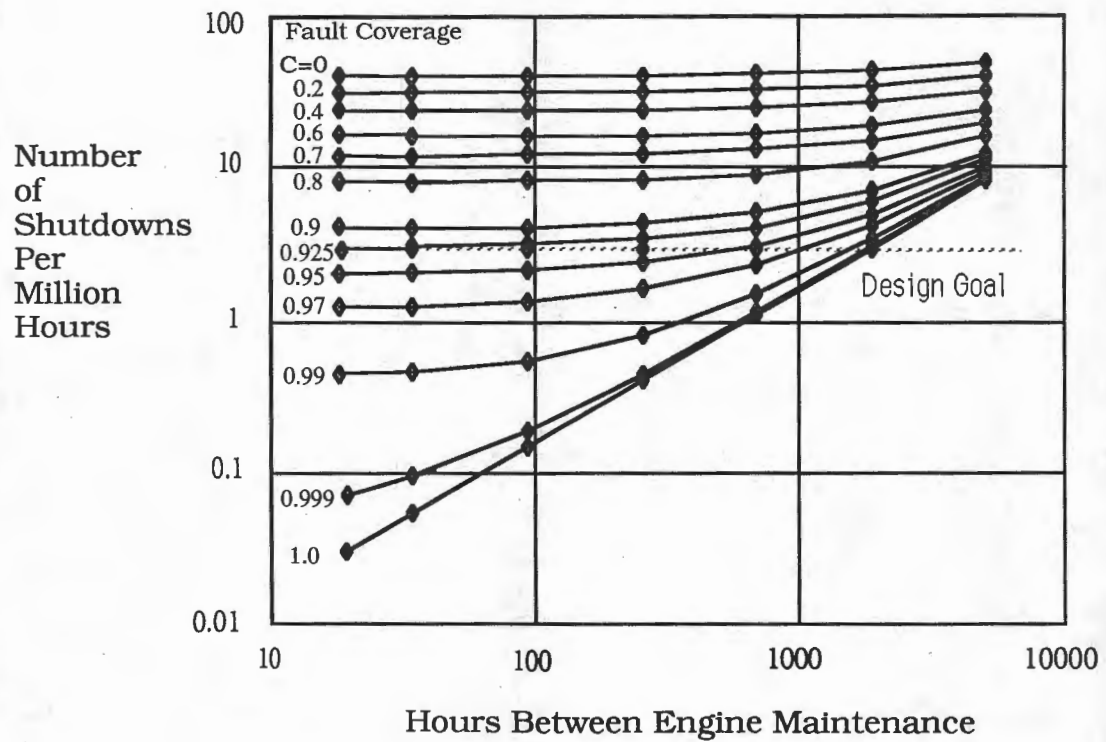


FIGURE 3.1-3. NUMBER OF SHUTDOWNS PER MILLION HOURS OPERATION - DUPLEX

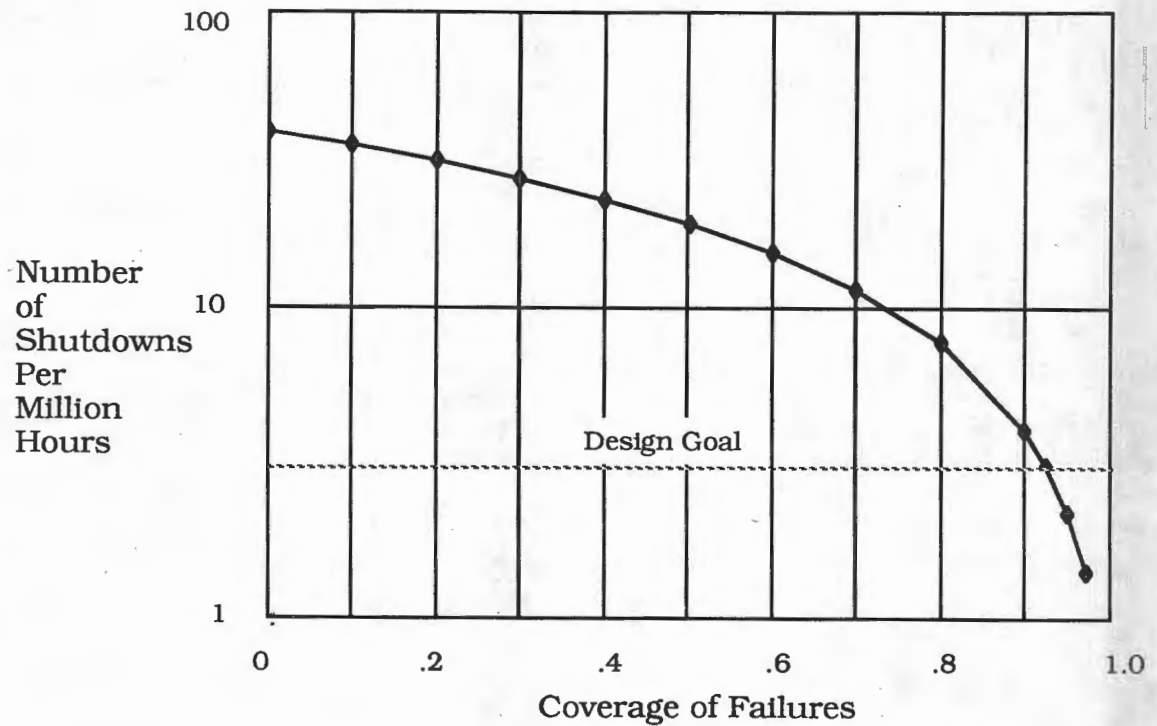


FIGURE 3.1.4. SENSITIVITY OF INFLIGHT SHUTDOWN TO COVERAGE

These results illustrate that attempting to improve system reliability by reducing the maintenance cycle limits the effectiveness of this approach unless very high levels of fault coverage can be achieved. For a controller having T_m of 150 hours and a mean failure time, T_s , of 15000 hours, coverage in excess of 0.99 is required to reduce the number of shutdowns by shortening the maintenance cycle.

3.1.5 Markov Models for System Safety and Availability

In the discussion of this subsection, the emphasis is on the two fundamental reliability objectives: system reliability and system availability. In general terms, flight safety is determined by having the minimum suite of equipment to insure no loss of life. Availability is the probability of being operationally ready at any point in time. In terms of safety, the probability of failure at a given time provides a measure of system safety. Reliability modeling is used to define and control the design for meeting the system safety and availability goals.

There is a fundamental conflict between high safety and high availability whenever redundant equipment is used for providing high reliability. For example, consider a system having a failure rate of 100 per 10^6 hours (MTBF is 10,000 hours) in operations having a two-hour flight duration. This system has a failure probability (P_f) of 2×10^{-4} and an availability (P_a) of 2×10^{-4} . The situation is quite different if a triplex (triple redundant) system is configured from these components. This new system has an improved failure probability of 8×10^{-12} (i.e., P_f^3), but a reduced availability of 6×10^{-4} (i.e., $3 \times P_a$) and MTBF=3333 hrs. Thus, while the triplex reliability is increased dramatically (nearly eight orders of magnitude) over the simplex system, the system availability has decreased by a factor of three.

This safety/availability conflict can be summarized simply. Addition of equipment in redundant or standby mode increases the likelihood that enough equipment will be working to provide safe flight. For adding redundant equipment, increasing the number of components increases the number of failures (not the system failure probability) in a given time. Since all the components are needed for dispatching the system, redundancy reduces system availability while increasing the useful operating time.

Since perfect reliability is not attainable, it is much more important to recover from failed components than to prevent them (Wulf 1975).

The probabilities can be calculated in broad terms using a system level Markov model having the following parameters:

L_t - Total Failure Rate	Rate of component failures
L_n - Nonfunctional failure rate	Failures that have no effect on the system operation
L_f - Functional failure rate	Failures that affect system performance

L_u - Uncovered failure rate

Functional failures that are not detectable or recoverable

L_c - Covered failure rate

Failures that may be detected, isolated, and recovered.

These terms are interrelated

$$L_t = L_u + L_f \quad [4]$$

$$L_f = L_u + L_c \quad [5]$$

For a single piece of equipment, coverage may be defined as

$$C = 1 - L_u/L_f \quad [6]$$

For a system or channel the coverage may be defined as

$$C = [\sum C_i * (L_c)_i] / [\sum (L_c)_i] \quad [7]$$

where C_i is the coverage for the i th component and $(L_c)_i$ is the corresponding failure rate.

Coverage is a term that requires careful attention to define properly. It is most accurately defined by testing the system by the fault injection techniques discussed in Chapter 5. In a multi-channel system, different coverage values are defined depending upon first or second failures. The first failure in a redundant system is more readily detected by the hardware and software implemented for this purpose (coverage often approaching one). Second failures in a duplex and third failures in a triplex being dependent upon self-test BIT and BITE mechanisms have a lower value of coverage (0.95 to 0.999).

The relationships defining system probability of failure in terms of component MTBF, component reliability (R), availability (Q), and coverage are as follows for duplex, triplex, and quadruplex systems (Hamilton Standard, August 1980):

▪ Duplex

$$P_{f1} = 2RQ(1-C_1) + Q^2C_1 \quad [8]$$

▪ Triplex

$$P_{f1} = 3R^2Q(1-C_1) + 3RQ^2C_1(1-C_2) + Q^3C_1C_2 \quad [9]$$

▪ Quadruplex

$$P_{f1} = 4R^3Q(1-C_1) + 6R^2Q^2C_1(1-C_2) + 4RQ^3C_1C_2(1-C_3) + Q^4C_1C_2C_3 \quad [10]$$

Where $R = 1 - \text{Exp}(-\text{MTBF} \times \text{hrs})$, $Q = 1 - R$, and C_1 , C_2 , and C_3 are the coverage of the first, second, and third failures respectively.

An important characteristic of coverage may now be illustrated. The likelihood of failure is extremely sensitive to the coverage of first and second failures. The first failure coverage in a triplex and the second failure coverage in a quadruplex is often considered to be unity. This is because detection techniques for first failures use cross-channel communications and direct comparisons. Changing this assumption on coverage slightly has a large effect. For example, changing the value of first failure coverage from 1.0 to 0.999 is summarized in the following figures for a system having a MTBF of 10,000 hours:

	P_f/hour ($C_2 = 1.0$)	P_f/hour ($C_2 = 0.999$)
Duplex	8×10^{-6}	8×10^{-6}
Triplex	1.2×10^{-9}	3×10^{-7}
Quadruplex	1.6×10^{-13}	4×10^{-7}

The duplex system is only shown for reference as it has no coverage for second failures. Such a change in coverage might result from ruling out cross-channel checking in a system design. The interesting point made by these figures is that the apparently small change in coverage for second failures has degraded the triplex and quadruplex to the extent that both these systems have approximately the same probability of failure.

Effects of periodic maintenance on the number of shutdowns can be developed parametrically using these algebraic models. The discussion in section 3.1.4 also used these system models for a dual system. The number of failures in a given number of hours is directly proportional to the time, provided that the probability of failure is small during the time interval. Following periodic maintenance, any system components that have failed are replaced and the system is in a "like new" state. As long as the maintenance time interval is small compared with the mean time between component failures, the number of failures can be determined from the probability of system failure.

For a triplex system an expression for the number of failures in a million hours, derived from equation 9 is

$$N_f = 10^6/T_m[3R^2Q(1-C_1) + 3Q^2C_1(1-C_2) + Q^3C_1C_2] \quad [11]$$

where T_m is the hours between periodic maintenance. The other terms are defined in equation 10. Figure 3.1-5 shows the result of these calculations for first failure coverage of 0.999 and second failure coverage ranging from 0 to 1. Comparison of the data with that in figure 3.1-3 illustrates the benefit for a triplex, even for small values of coverage for the second failure.

3.2 Analytical Redundancy Design for Improved Engine Reliability

Control system reliability requirements for the next generation of aircraft propulsion systems are leading engine control systems toward dual or even triple redundancy. Sensor redundancy can be minimized by use of analytical models that provide signals for a failed sensor. These signals are derived from aircraft parameters and measurements using other sensors. This minimizes the need for physically replicated sensor sets. Analytical redundancy used together with a

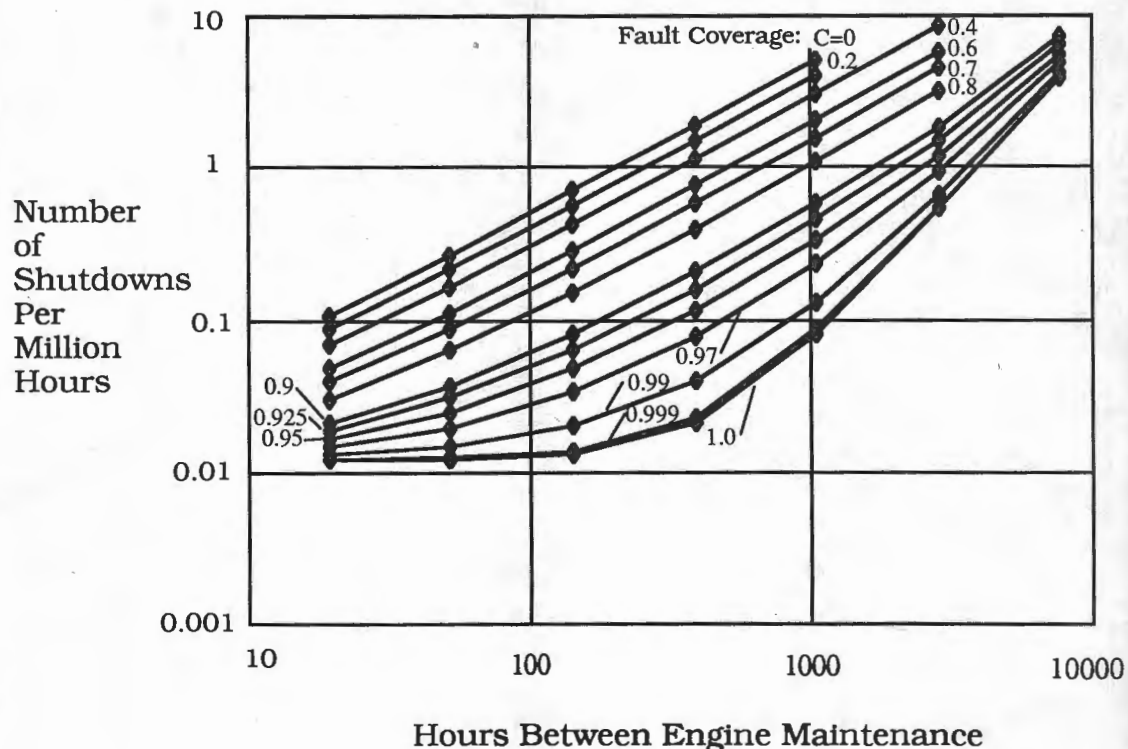


FIGURE 3.1-5. NUMBER OF SHUTDOWNS PER MILLION HOURS OPERATION - TRIPLEX

This section describes an analytical redundancy approach for increasing the reliability of aircraft engines reported in Swan and Vizzini 1988 and Brown and Vizzini 1986. (Also refer to Detroit Allison June, 1982 for a fighter aircraft study.) The original work was sponsored by the Naval Air Propulsion Center as the FADEC program (Vizzini and Toot 1980) and designated as a Failure Indication and Corrective Action (FICA) system. The initial work established the feasibility of this approach. The program represents an extension of the initial effort for a one-of-a-kind technology demonstrator to an operational product engine controller. Included in the new work is an analytical demonstration of operation over a full range of engine modes and parametric envelopes. Also included is a demonstration of fault isolation and recovery from failures in sensors and actuators. These investigations utilized a non-linear analytic model for a GE23A jet turbine engine to provide operational parameters and real-time modeling.

3.2.1 Analytical Redundancy Engine Control Concept

The control system layout, shown in figure 3.2-1, was used for the investigation reported in Brown and Vizzini 1986. It contains three main modules. The GE23A engine system predicts the outputs of engine actuators and sensors as a function of the sensed environmental conditions, actuator signals, and model updates. This engine model has been validated by prior work to represent the operation of a real engine, and is used to simplify the experiment and for providing known engine environments. The engine actuator inputs drive the non-linear engine model which provides sensor data for the control system. The engine module in the controller is an analytical model specifically designed to support the analytical redundancy system by providing model predictions for sensor outputs. The component tracking filter uses the sensor data and model values to compute differences between the sensed values and model values. This information is used to generate model updates and error information for failure detection and isolation logic. The failure-accommodation module logic computes the validated sensor information as a function of the sensor and actuator status, model values, and sensor data. In the event of actuator failure, this module provides the data for reconfiguring the engine control mode.

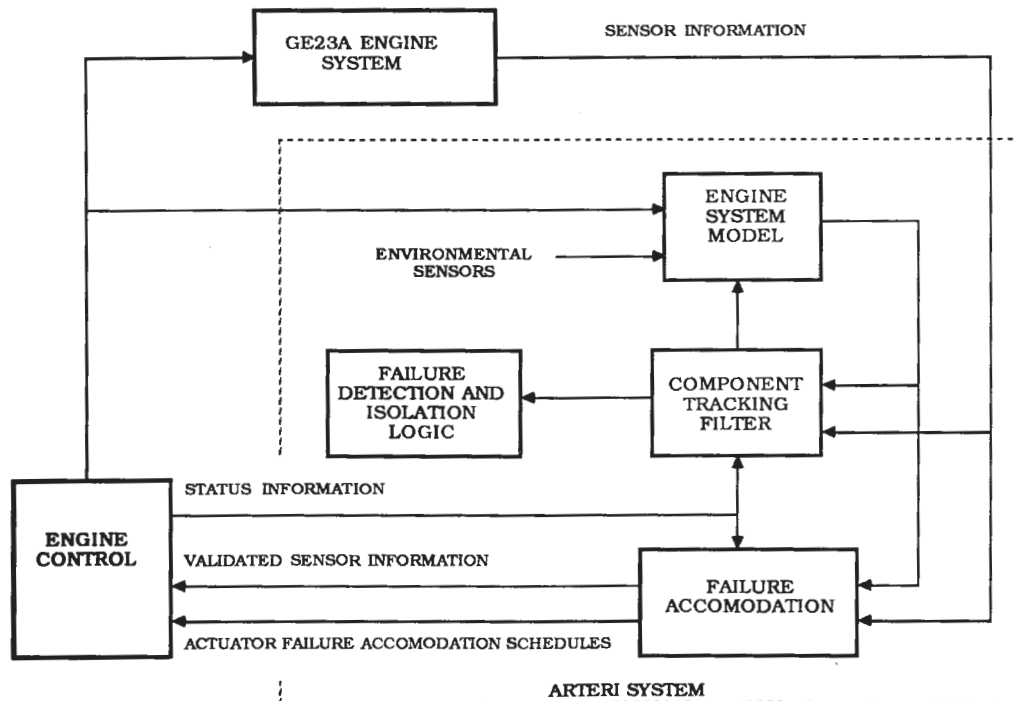


FIGURE 3.2-1. SCHEMATIC OF ANALYTICAL REDUNDANCY ENGINE CONTROLLER

3.2.2 Component Tracking Filter

The component tracking filter is the key element in this system. This adaptive filter updates engine module parameters as a function of differences between sensed and modeled values. This filter eliminates the need for manually tuning the engine model to an individual engine. Manual tuning was required for prior engine modules to accommodate differences between engines of the same type number.

There are two types of sensor failures: hard failures, where the sensor output violates physical limits on position or rate of change; and soft failures, where the sensor calibration gradually changes and slowly drifts, providing erroneous but believable values. Hard failures are the easiest to detect by parameter range checks or by range rate checks. Soft failures are much more difficult to detect.

The Analytical Redundancy Technology for Engine Reliability Improvement (ARTERI) system was designed to detect, isolate, and accommodate soft sensor failures while using the same sensor data to update the engine module parameters. The model accuracy needed for accommodating sensor failure is achieved by updating the engine module parameters to match the actual engine under control. This update process minimizes the errors between the sensed and measured values. Depending upon the rate of model update and the drift rate for failed sensors, the model may keep up with the failed sensor and corrupt the engine module. This makes fault detection and isolation difficult and may lead to high false alarms and removal of good equipment. All single-channel model-filter-based FDFM schemes have this same limitation. The ARTERI approach minimizes false alarms by integrating the functions of fault detection, isolation, and reconfiguration.

The approach to FDFM also uses the adaptive nature of the component tracking filter. Once the component tracking filter parameters were updated to match the actual engine parameters, the filter gain was reduced. Filter design requirements included the need to produce unique failure signatures, minimize corruption by soft failures, and provide a smooth transition to the model value. Smooth transitions were accomplished by decoupling the filter update process and by gain controls.

Sources of mismatch between the model and engine data are modeling inaccuracies and measurement uncertainties. Engine to engine differences occur from component quality and variation. Modeling approximations were necessary to meet the requirements of real-time operation. Measurement uncertainties are the random variation of a sensed parameter from its true value and the noise in electrical measurements. Compared with modeling inaccuracies, the measurement uncertainty is considered to be a small source of mismatch between modeled and measured data.

Two kinds of error result from model inaccuracy: steady state errors and dynamic errors. Steady state errors are the constant or near constant differences between measured and predicted values. These errors are due to parameter differences and biases in the measurements. Dynamic errors are the differences that occur during transient conditions such as during engine speed up and slow down. These errors are due to internal iteration loops, approximations for the component models, and high frequency sensor noise.

The component tracking filter, shown in figure 3.2-2, integrates four separate updates: state update, steady state error update, input update, and component update. Mismatch is minimized by a frequency decomposition of the sensed engine parameters and the model values. Actuator state updates are computed from position errors and a diagonal matrix of update parameters. This portion of the filter adjusts for steady state and dynamic errors. Engine model updates for steady state errors are computed as a function of time delayed differences between sensor measurements and predictions. Dynamic errors are treated as the difference between measured and predicted plus the sum of the steady state errors. Time constants for the filter lag are selected to be four to 10 times slower than the dominant engine response time. Engine state updates are computed as a function of the dynamic errors and an update matrix. The structure of this update is similar to a Kalman filter. However, only measured states are updated. The same structure is used for the component and updates loops. These updates drive the steady-state errors towards zero.

The structure of the component tracking filter allows the various filter updates to be computed at different sample rates. This reduces the throughput requirements of the filter. Since the state updates are related to the dynamics of the model, they are computed at the same 10 to 20 millisecond rate. The steady state error and input updates are low frequency loops computed at a rate four times slower. The lower rate is based on the assumption that the component performance levels change slowly over the life of an engine and component updates may be computed even more slowly.

For performance over the full engine operational envelope, the engine controller gains are scheduled as a function of power setting and flight condition. Initial gains are scheduled as a function of fan speed and sea-level operational conditions, thus accounting for various power settings. These first order curve fits are modified as a function of flight condition (altitude and mach number) based upon measured engine inlet environments. Inlet temperature and pressure measurements, divided by their sea level values, provide scale factors to correct the sea level model errors for speed, temperature, and pressure.

3.2.3 Failure Detection And Isolation

Errors computed from the component tracking filter form three separate error vectors. These vectors define an n-dimensional space, where n equals the number of vector components. When a failure occurs, the path that the vector follows is defined as that failure vector's signature.

The criteria established for computing the engine state update and input updates decouple the error vectors. When the error vectors are decoupled, the failure of one component will not affect the error signals for the others. The update criteria are designed as follows:

- Design the state update matrix so that the error produced by a failure in one sensor does not create errors for other sensors because of an update.

- Design the input update matrix so that the error produced by failure in one input does not create an error in other input channels because of the update.

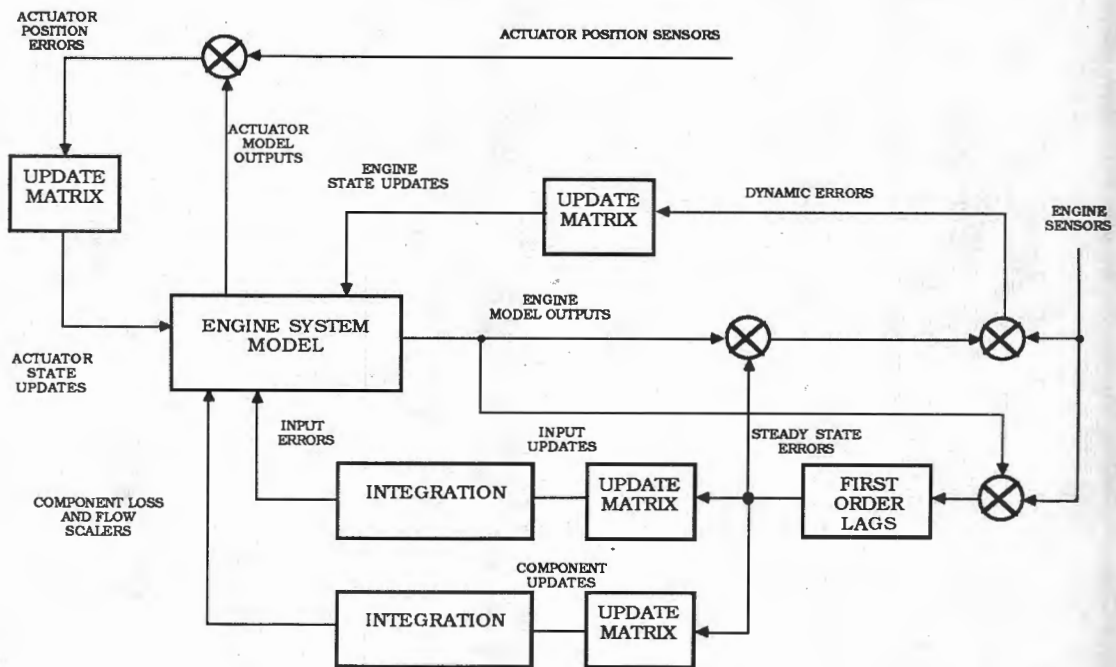


FIGURE 3.2-2. COMPONENT TRACKING FILTER

As a result of this decoupled design, engine state and input updates produce axial signatures. When failure occurs with this design, a change in error signal will only be observed in the failed channel. This design method is discussed at length in Brown and Swain 1987. The advantage of the axial error vector is that the error signatures only need to be tested; storage of the vector is not required. Out of bound indications will only appear in the error signal of the failed component. Bounds on the variation and rate of variation need to be established for each component. The individual error signals are compared with the established bounds to detect faults.

Examples of steady state and dynamic failure signatures for a soft failure in a fan speed sensor are shown in figures 3.2-3 and 3.2-4, respectively. Note that for these plots the threshold value for each channel is the full scale value (+FT and -FT) for each plotted parameter. Failure isolation can easily be accomplished with the data shown.

3.2.4 Failure Management

Failure management logic is based upon the following priorities: first, engine safety; and second, maintaining thrust and specific fuel consumption.

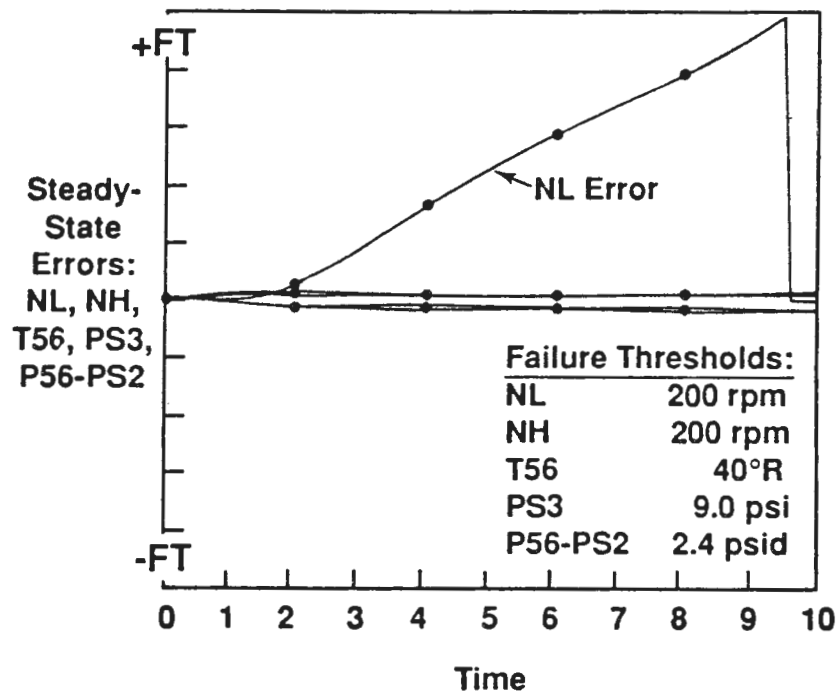


FIGURE 3.2-3. STEADY STATE ERROR DETECTION - FAN SPEED SENSOR

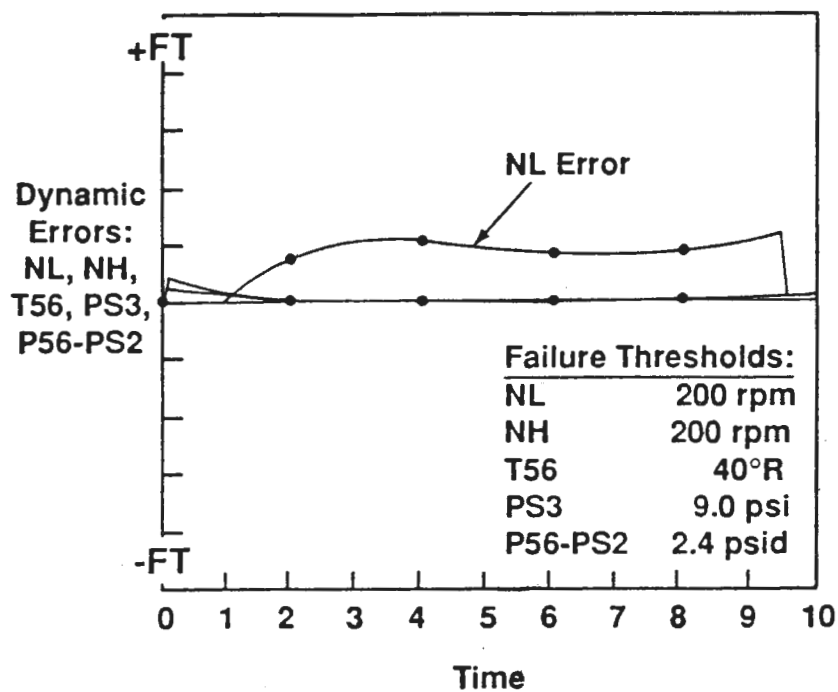


FIGURE 3.2-4. DYNAMIC ERROR DETECTION - FAN SPEED SENSOR

Failures of sensors and actuators must be detected and acted upon to isolate and correct engine control modes before any serious engine damage or loss in performance occurs. Corrective actions for sensor or actuator failures should not induce any large transients that could lead to loss of control. These corrective actions are accomplished by logic activated by error signals exceeding error threshold values.

Sensor failures are accommodated by analytical redundancy where the analytical sensor models replace the failed sensor. Smooth transitions are accomplished by computation of the valid sensor value from the sensor value, model value, and control gains. Actuator failures are accommodated by reconfiguring the control mode using fail-fix servovalves and failure accommodation schedules.

The primary failure usually observed for an actuator is a short or open circuit of a torque motor. When such a failure occurs, the fail-fix servovalve will cause the actuator to move to a preferred position. This position (open or closed) is selected to maintain safe engine operation and maintain performance as high as possible. The control schedules for the remaining actuators use precomputed schedules to minimize loss of engine performance. When safe operation cannot be maintained, the engine will be shut down. For other actuator failures, the corrective action is to shut off the torque motor allowing the fail-fix servo actuator to move to the preferred failure position.

Table 3.2-1 (Brown and Swain 1987) summarizes the results of the actuator failure accommodation studies. Note that for a compressor variable stator vane (VSV) failure, a fully open direction could result in blade flutter and possible engine damage, while a fully closed direction considerably reduces thrust.

3.2.5 Test Results

A number of tests were conducted to validate the design and implementation of the prototype engine controller. These tests included convergence tests to verify operation with different engines over the operational envelope; transient tests to verify the ability to track during large transient changes in engine operation including acceleration and deceleration; and fault detection, isolation, and accommodation tests to verify the controller ability to detect and correctly handle hard and soft failures.

Results for the soft failure tests, being the most difficult, are summarized in table 3.2-2. Figure 3.2-5 illustrates the resulting corrective action for a fan speed sensor failure. Note that the closed loop control for fan speed maintained constant speed prior to isolation of the failure. The controller tracks (maintains the demanded speed) from the failing sensor signals by increasing fuel flow. This is evident by noting the change in fan speed computed by the model. Also note that normal engine operation was resumed by the time the failure was detected. The results of accommodating a soft failure in the main fuel metering valve actuator sensor (WFM) is shown in figure 3.2-6.

The program (Brown and Swain 1987) summarized in this section developed an operational prototype engine controller having fault tolerance for both hard and soft failures. Operation of the prototype was validated by tests using a

non-linear simulation of a GE23A turbofan engine, although the design is adaptable to any engine.

TABLE 3.2-1. ACTUATOR FAILURE ACCOMMODATION TEST RESULTS

Actuator	Preferred Failure Direction	Effect on AB Operation	Effect on Dry Operation	Fault Accommodation Schedules
Afterburner Fuel Metering	Closed	Not Permitted	Normal Operation	N/A
AB. Exhaust Nozzle Area	Closed	Not Permitted	Maximum of 12% Thrust Reduction of 46% Reduction in Fan Stall Margin for Certain Altitude and Mach Number above 15,000 ft. and Mach 1.5 at IRP Increase in Flight Idle Thrust	Reduced Corrected Fan Speed Control Reference Schedules
VSV. Compressor Variable Stator Vane	Closed	Not Permitted	Shut Down Engine	N/A
WFM. Main Fuel Metering Valve	Closed	Not Permitted	Shut Down Engine	N/A

Tests on the prototype control of a simulated GE23A engine demonstrated the capability for achieving the following:

- Updating the control system engine model to match engine variations due to engine component quality and deterioration.
- Detecting and isolating hard and soft failures of environmental sensors, internal engine sensors, actuator position sensors, and engine actuation systems.
- Maintaining normal engine operation following single sensor failures.

- Maintaining safe engine operation following single failures of engine actuators.

Continued development and testing with a demonstrator engine is recommended.

Analytical redundancy appears to be viable for application to engine control and there should be commercial applications for this technology.

TABLE 3.2-2. SOFT FAILURE TEST SUMMARY

	Failures Simulated	Result
Engine Sensors	NL, Bias; Ramp $\pm$ 50 rpm/sec NH, Bias; Ramp $\pm$ 50 rpm/sec T56, Bias; Ramp $\pm$ 5° R/sec PS3, Bias; Ramp $\pm$ 3 psi/sec P56-PS2, Bias; Ramp $\pm$ 8 psid/sec	Detected, Isolated, and Accommodated
Actuated Sensors	VSV Bias; Ramp + 10%-st/sec WFM Bias; Ramp -5%-st/sec A8 Bias; Ramp +5%-st/sec	
Actuator	VSV Bias; Stuck Actuator, NL_{Rel} 105 - 95 WFM Bias; Stuck Actuator, NL_{Rel} 105 - 95 Uncommanded A8/Open, + 10 ma TMC	
Environmental Sensors	T2, Bias; Ramp +2.5° R/sec	
	PS2 Bias; Ramp -.25° R/sec	Detected, Turnoff Component Update

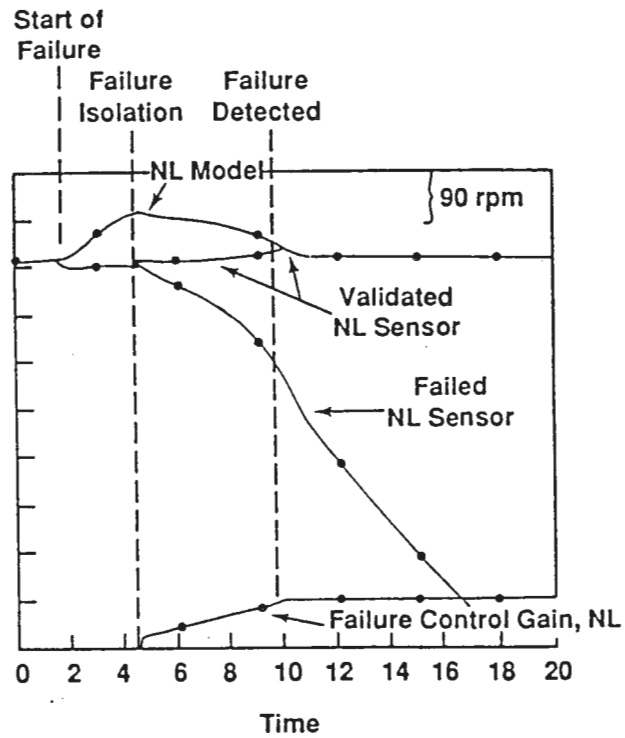


FIGURE 3.2-5. CORRECTIVE ACTION - FAN SPEED SENSOR

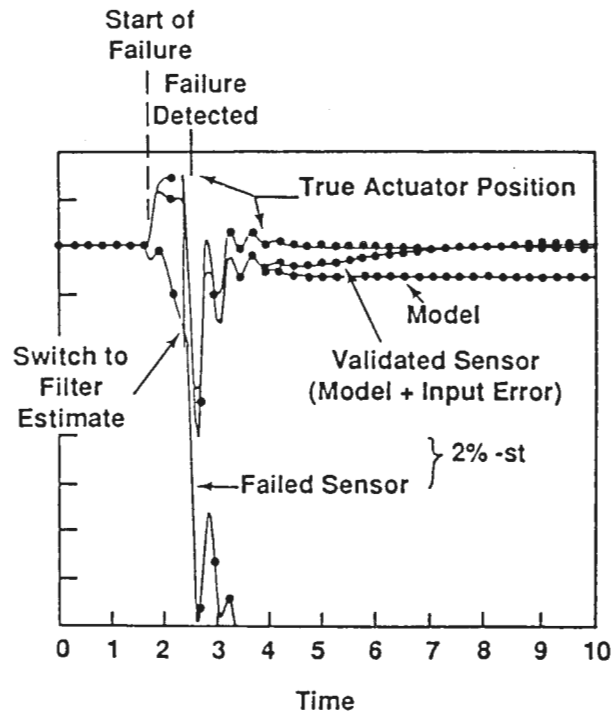


FIGURE 3.2-6. CORRECTIVE ACTION - FUEL FLOW ACTUATOR SENSOR

4.0 Primary and Secondary Electrical Power

4.1 Aircraft Power Systems

4.1.1 Power System Reliability Requirements

Electrical power system reliability directly affects the reliability of flight control and other flight critical avionic systems. The reliability of a single channel power system having a generator, generator controller, and load switches will not meet these reliability requirements. Hence, fault tolerance, including detection, isolation, and recovery, is an integral part of an aircraft power system. Material for this section is drawn from Mehdi and Leong 1985, Bret 1987, and Perkins and Marek 1977.

Requirements for reliability and redundancy, developed for the flight control and engine control systems, must be maintained for power generation and distribution. In a conventional power system, electrical generators driven by the aircraft engines provide the primary electrical power. A single generator on each engine can provide enough electrical power. A modern aircraft having computer equipment uses multiple generators and a backup auxiliary power unit (APU) to supply no-break power (at least no breaks longer than a few milliseconds). Flight critical systems operate from batteries charged from the generators.

Multiple generators may be needed on each engine to provide the reliability necessary for fly-by-wire flight controls, although a single generator could supply the electrical power. The F-16, for example, has a quadruplex redundant flight control system and a single jet engine. This aircraft also has multiple generators and batteries to provide an uninterruptible source of power.

Computer components are usually more susceptible to transients and noise on power lines than older technology. Added filtering is needed either in each component or in the power generation and distribution system.

Battery systems are needed on board the aircraft for operating critical systems in the event of an engine-out condition. Flight critical equipment must have a reliable source of electrical power to maintain safe flight and landing capability under extremely unlikely failure conditions.

Fast acting contactors and microprocessor generator controllers are needed for several reasons. Besides being quickly handled, faults in the electrical system must be handled so not to disturb power quality. Power transfer from one bus to another, such as from ground power to aircraft power, must not interrupt the supply. The normal contactors operate in a period of about 25 milliseconds. This delays the shut-down and turn-on time to about 50 milliseconds (DO-160 or MIL-STD-704). (Refer to Bret 1987 for a discussion of no-break power transfer.)

There is a trend in military power systems towards using 270 volt dc power for actuation systems. This places added requirements on power generation and distribution systems currently designed for 28 volt dc and 115 volt ac.

4.1.2 Electrical Power System Elements

Electrical power generation for a multiengine aircraft uses generators with constant speed drives mounted on the main engines. These generators can be isolated or paralleled for operation on the aircraft. Multiengine aircraft normally have parallel systems with three or more generators. Electrical systems for two-engine aircraft are usually operated in isolation. There usually are provisions, however, for tying the buses together if an engine or generator faults.

Failures can occur in the generator channel, distribution bus, avionic equipment, and wiring between these items. Protective devices and contactors segment the system for isolating faulted components. These contactors also supply power to the operating equipment.

Information measured from parts of the system provides fault identification and reconfiguration by control of these contactors. The electrical system is instrumented for measuring voltage and current at many locations. The instrument signals are fed to contactors in the power system control units including the generator control unit, bus control unit, and distribution center. Each group of contactors and instrumentation independently controls a portion of the system.

Because of independent control of the power system zones, faults in the system may allow an operating generator to reconnect or stay in parallel with a channel having an uncleared fault. This situation can be complicated if more than one failure happens or if a good generator is shutdown in error.

Fault tolerant power generation and distribution systems use fault detection and isolation at the local level. This tolerance may be accomplished by combining several techniques in the power system that include the following:

- Generator redundancy
- Power bus redundancy
- Functional load redundancy
- Power routing using bus interties
- Load shedding on prioritized basis

An aircraft power system may be configured in several ways to achieve high reliability. Power system redundancy may be accomplished by using several isolated channels or by reconfiguration using contactors.

An isolated channel approach, shown in figure 4.1-1, provides maximum independence between channels; it does not allow reconfiguration of power system elements for fault recovery. This approach provides isolation for power faults.

However, each single channel has only one power source, contactor, bus, and power control unit. The operation of the power controllers is discussed in section 4.2.1. Since the failure of any element leads to power failure, system reliability suffers. The reliability of a channel is the product of the element reliabilities. This isolated channel approach will meet flight control system reliability goals with the reliability of today's power systems.

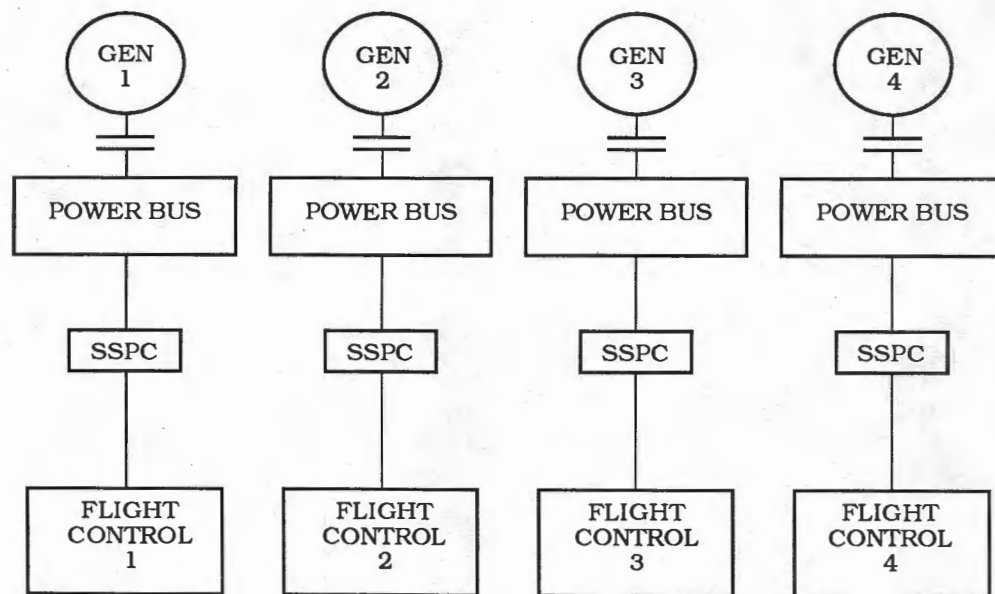


FIGURE 4.1-1. ISOLATED CHANNEL POWER SYSTEM

A reconfigurable approach, shown in figure 4.1-2, uses cross bus interties to provide power source redundancy. This approach does not have a "brick wall" isolation for flight control channels. This redundancy approach needs careful design and testing of the fault detection and fault isolation contactors. This example shows a three-generator power system supplying a four-channel flight control system. The use of two main power buses and contactors allows any generator to power any flight control system.

The reliability of the reconfigurable system is higher than the isolated channel system. Although in this system, correcting the effects of power system faults needs reconfiguration of working elements for maintaining power. The reliability of these contactors and their controls must be higher than those used for the system in figure 4.1-1. If the contactor controls fail, then the system will not operate properly. Present aircraft use manual contactors as a backup, if not the primary, for system configuration. Some systems use automatic load shedding (disconnecting non-critical loads) to keep critical equipment working.

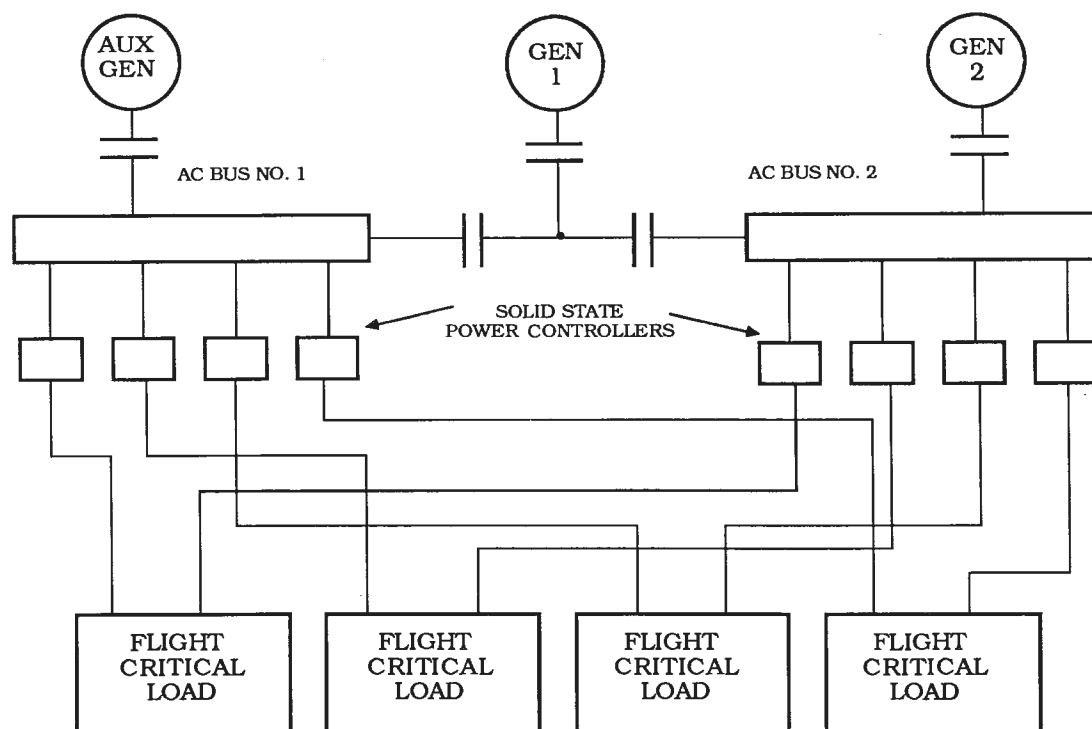


FIGURE 4.1-2. RECONFIGURABLE POWER SYSTEM

The most serious faults in any system are those affecting all of the channels at the same time. These common mode failures are often the hardest to protect against. The two-bus system, shown in figure 4.1-2, has two contactors between the main buses that give some common mode protection against a simultaneous bus fault and single contactor failure. Such a failure would prevent the isolation of the main generators. Careful system analysis and design can assure that such unlikely events have a lower probability of happening than any other event causing loss of flight control.

Power system design documents should provide fault trees and FMECA that include common mode effects and the normal component failure rate analysis.

4.1.3 Technology Trends

Advanced technology avionics have increased electric power demand and need power sources having improved reliability and power quality. The increased capability provided by application of electronic equipment to aircraft results in more equipment onboard, although the power demands of single systems have not increased. Large scale integrated semiconductor circuits are more susceptible to electrical transients and measures must be taken to protect them. A properly designed avionic system can operate in a very noisy environment with present aircraft power quality. The trade-off between protecting each unit of equipment

versus improving the power system source often comes out in favor of improving the source. This is particularly true of large commercial aircraft.

Uninterruptible power supplies (UPS) are needed for operating electronic equipment in the event of an interruption, or total loss, of electrical power generation. Flight critical systems need a back-up electrical power supply if all the engines or generators fail on a two-engine aircraft. Uninterruptible power can be supplied from batteries that fill in the gaps made by switching generators and by loss of generation.

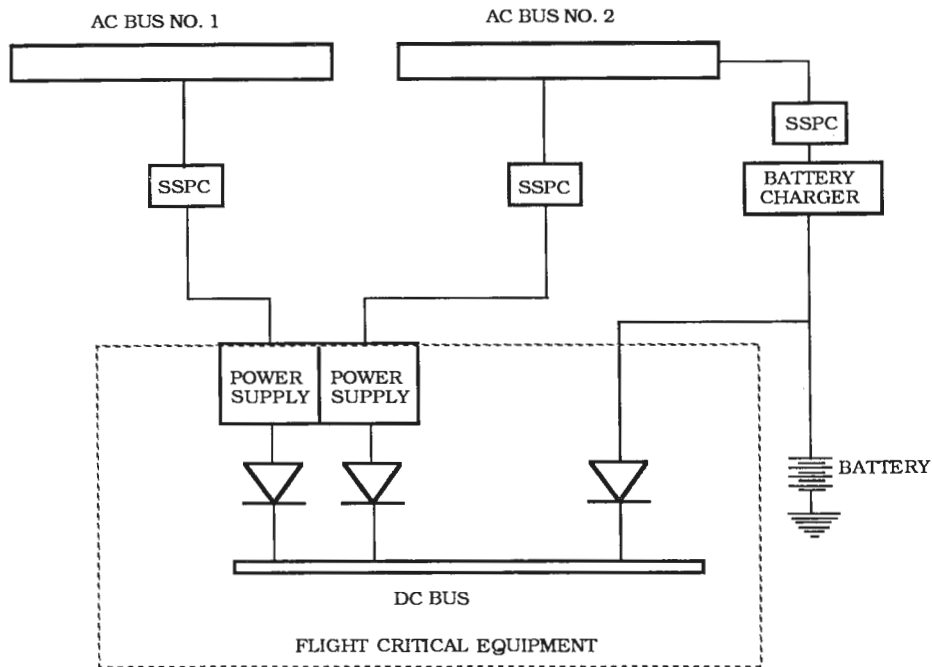
Two basic types of electrical power are available onboard an aircraft: ac and dc. Ac power is used for high energy electrical devices such as motors for landing gear and hydraulic pumps. Dc power could be used by electronic equipment, although present equipment is often configured for ac power. Some equipment operates from 28 volt dc and generates other dc levels internally. There is a trade-off between supplying power that the equipment most easily uses or putting power conversion equipment inside each component. For large aircraft the trade-off is on the side of supplying appropriate power to the equipment and minimizing power conversion in the equipment.

A current military aircraft system design trend to 270 volt dc power generation and distribution has several advantages over ac power. The major one being the elimination of the complex control and synchronization needed to parallel ac generators and to maintain frequency control with engine driven generators. An important consideration is supplying a UPS for computer equipment.

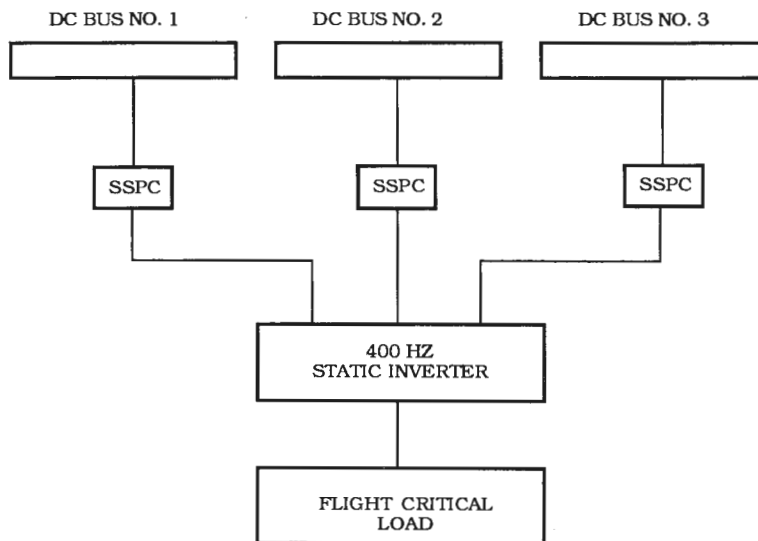
One method of providing uninterruptible power is by use of batteries supplied by rectifiers from ac generators. When a battery is used for primary power, other sources of power may easily be paralleled with diodes, as shown in figure 4.1-3. The battery is fully charged under normal conditions by a charger on the main power supply. Upon failure of the transformer rectifier unit (TRU), the battery will handle the load. This is called a battery float UPS. Batteries can handle the complete flight critical part of avionics and flight control for a period of thirty minutes to an hour.

Lead acid batteries are preferable to nickel cadmium because of the long time on float. Lead acid batteries may be fully charged with a 28-volt supply. But, nickel-cadmium batteries designed for 28 volts need a charger operating at about 36 volts. When discharging, the nickel-cadmium battery voltage drops quickly to 28 volts and stays there over the entire discharge time. Under normal float operation, the nickel-cadmium battery powered equipment must handle 36 volts. TRUs are typically unregulated and the higher voltage will pass to ac equipment as well.

Another UPS method is supplying dc directly from dc generators and using inverters to provide ac power where needed. Multiple sources of power can supply a flight critical bus. Figure 4.1-3 illustrates such a configuration for a TRU supplying a flight critical load.



A. BATTERY FLOAT UNINTERRUPTIBLE AC POWER



B. STATIC INVERTER VERSION OF UNINTERRUPTIBLE AC POWER

FIGURE 4.1-3. INTERRUPTIBLE POWER SUPPLY METHODS

4.2 Power Systems Detailed Examples

4.2.1 Fault Protection

The complexity of power generation and distribution systems for attaining high reliability needs automated fault protection. The following concepts provide isolation and protection against power system faults:

- Differential current protection
- Solid-state power controllers (SSPCs)
- Electromechanical power controllers
- Thermal circuit breakers

Generator feeders and main buses use differential current protection. This type of protection uses balanced transformers and voltage detection. The system provides fast and accurate protection levels. Operation is by a difference in current between two ends of a power feed, one being the load end and the other the bus end. Thus, any fault in the feed will be detected as a difference in the measured currents. Protection zones can be positioned to detect and isolate faults.

SSPCs replace the older thermal circuit breakers. SSPCs provide the same over current protection of the older units and provide a fast acting on-off switching control of electrical loads. Figure 4.2-1 shows an application where an SSPC replaces two circuit breakers and a relay for load switching. Fast fault clearing times can be achieved by using field effect transistors as the power switch. Test results from Mehdi and Leong 1985 indicate the fault isolation performance for five-amp 115 volt ac controllers with field effect transistors. When switched onto a 2,400 amp fault, the maximum current let through by these contactors was only 82 amps; the devices switched in about 10 microseconds.

Electromechanical power controllers or remote controlled circuit breakers perform the same function as SSPCs. Because they are mechanical devices, the switching speeds are slower and the devices are larger. These devices are suitable for high currents, as with larger motors.

4.2.2 High Voltage dc Generation and Distribution

This section provides an overview of the technology for high voltage dc generation and distribution. This technology was developed under a Navy program (Perkins and Marek 1977) and defined an Advanced Aircraft Electrical System (AAES).

Key features of this technology are high voltage dc generation, microcomputer generator controls, and SSPCs. These components were designed and built to provide electrical power for an A-7E fighter aircraft. A prototype system, built and tested for a single channel, supplied the system parameters. Trade studies were conducted on the benefits of applying this technology to an entire aircraft.

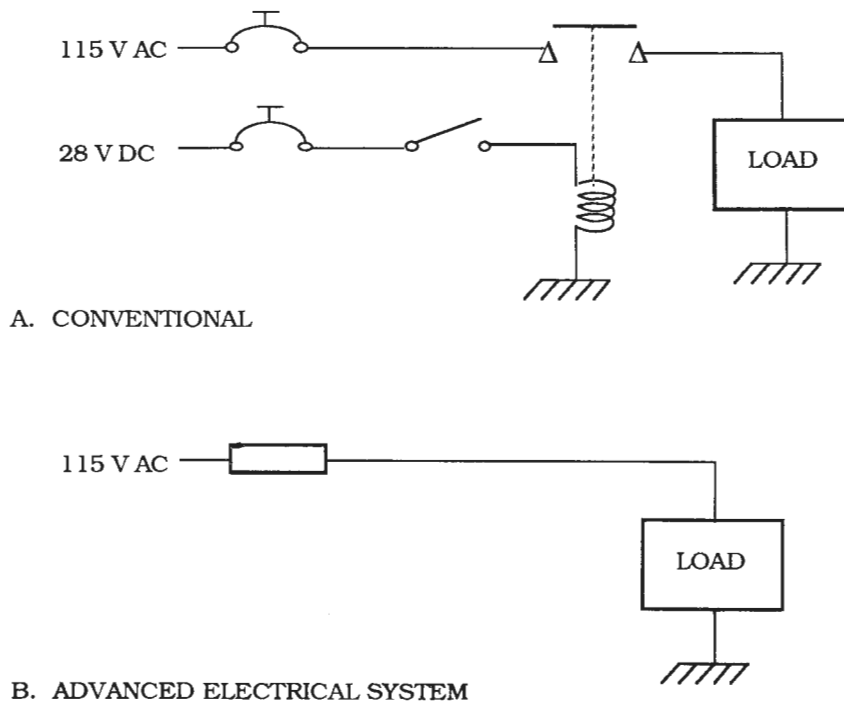


FIGURE 4.2-1. SOLID STATE POWER CONTROLLER SWITCHING

The AAES performed better than the conventional system in several areas. The areas of improvement included the following:

- Full system built-in test (BIT) showed failures down to the avionics unit and detected faults related to wire, connector pin, and terminations.
- The prototype included load management and power conditioning in the AAES although these functions are not feasible in the conventional system.
- The AAES design resulted in improved reliability by use of redundancy and distributed load management; neither is possible in the conventional system.
- Size, weight, and maintainability were all improved over the conventional system.

A trade study conducted to compare the High Voltage dc (HVDC) system with the conventional electrical system further illustrates the advantages of this system. An HVDC power system was designed for an A7-E aircraft using the prototype HVDC equipment parameters. The prototype AAES needed power conversion from 270 volt dc to standard 28 volt dc and 115 volt ac power. Since the penalty of distributing the three power levels biased the prototype parameters, the trade study included a "True HVDC" and "Projected HVDC" design. The AAES power system block diagram is shown in figure 4.2-2. Also shown are the military designations

for the hardware in the design from Perkins and Marek 1977. This prototype design uses samarium cobalt permanent magnet type generators for the 270 volt dc main and emergency generators.

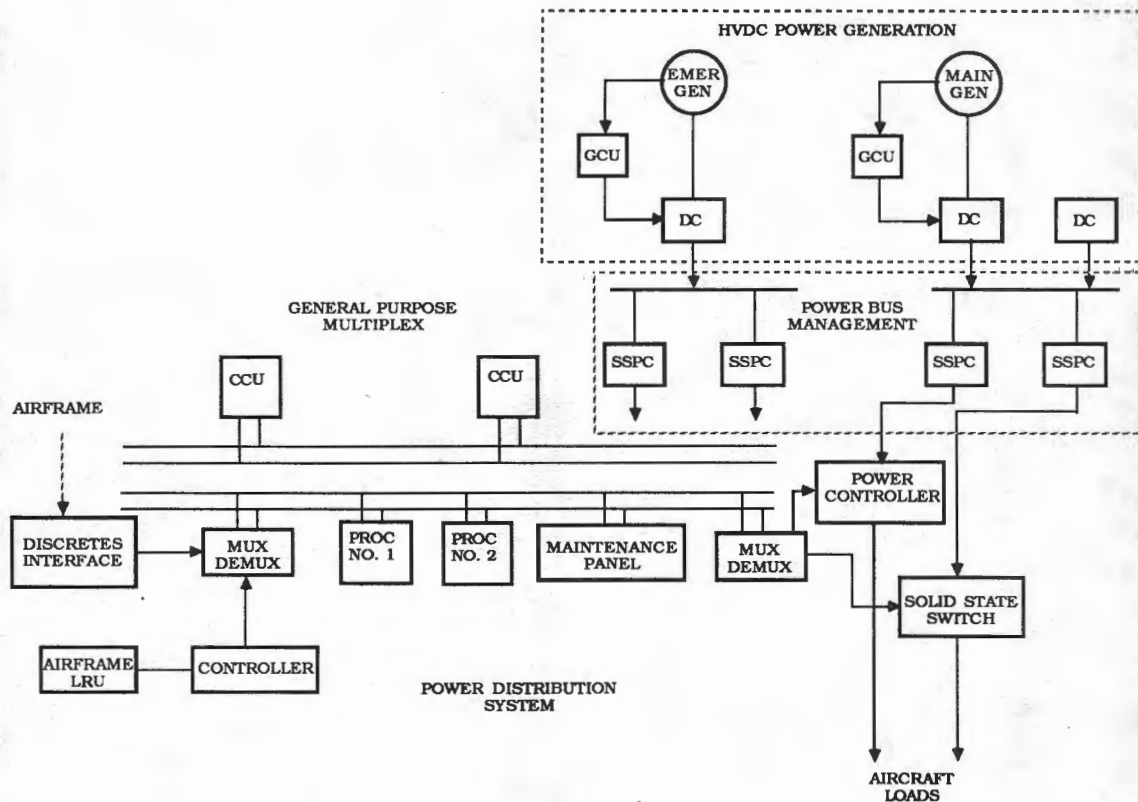


FIGURE 4.2-2. ADVANCED AIRCRAFT ELECTRICAL SYSTEM DIAGRAM

Comparative data from the trade study are presented in table 4.2-1; the "Projected" column shows the projection of the trade study to a new avionics suite that does not need 28 volt dc and 115 volt ac. If the avionics equipment were powered directly from 270 volt dc, then the 28 volt dc and 115 volt ac power conditioning and distribution could be minimized. Thus the "Projected" data is important for new aircraft.

Several benefits for an HVDC power system are given in Perkins and Marek 1977.

To the Equipment Manufacturer:

- Reduced voltage transients
- Uninterruptible power
- Smaller voltage variations
- Improved protection
- Smaller size and weight

TABLE 4.2-1. HVDC VS CONVENTIONAL POWER

PARAMETER	CONV	PROTOTYPE AAES	PROJECTED HVDC
Weight (Pounds)	190	76	64
Power Generation	9	3	3
Power Bus Management			
SOSTEL Power Distribution			
Equipment	177	223	163
Wiring	312	121	109
Total	688	423	339
Efficiency (%)	61	77	80
Reliability - MTBF (Hours)			
Essential Loads	63	509	625
All Loads	170	1442	1966
Maintainability (MMH/FH)	0.498	0.245	0.228
Cost			
Nonrecurring	Base	1.3B (-330)	0.5B (50)
Life Cycle	Base	0.3B (70)	0.2B (80)

NOTE: Numbers in parentheses show percentage of improvement over conventional power.

To the Airframe Manufacturer:

- Lower system weight
- Improved power quality
- Simplified system design
- Ease of parallel operation
- Higher overall efficiency
- Lower personnel hazard
- Easier change incorporation
- Elimination of constant speed drives

To the Engine Manufacturer:

- Lower overhung moment
- Lower weight

High- versus low-speed performance
Smaller size
Higher efficiency

To the Owner:

Lower life cycle cost
Improved reliability
Improved maintainability with BIT
Higher system efficiency
Lower personnel hazard
Lower weight

These benefits and economic factors in favor of HVDC power systems assure that further applications are expected in future aircraft.

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ACRONYMS AND ABBREVIATIONS

AAES	Advanced Aircraft Electrical System
ac	Alternating Current
AIAA	American Institute of Aeronautics and Astronautics
APU	Auxiliary Power Unit
ARTERI	Analytical Redundancy Technology for Engine Reliability Improvement
ASEE	American Society of Electrical Engineers
BIT	Built-In Test
CSDL	Charles Stark Draper Laboratory
dc	Direct Current
DISAC	Digital Integrated Servo Actuator Controller
EMA	Electromechanical Actuator
EMAS	Electromechanical Actuator System
EMI	Electromagnetic Interference
EPR	Engine Pressure Ratio
FADEC	Full-Authority Digital Engine Controller
FBL	Fly-By-Light
FCC	Flight Control Computer
FDFM	Fault Detection and Failure Management
FICA	Failure Indication and Corrective Action
FMECA	Failure Modes and Effects Criticality Analysis
HIRF	High-Intensity Radiated Fields
HVDC	High Voltage dc
Hz	Hertz
I/O	Input/Output
LVDT	Linear Variable Differential Transformers
N ₁	Low Rotor Speed
N ₂	High Rotor Speed
NASA	National Aeronautics and Space Administration
PBW	Power-By-Wire
PLA	Power Level Actuator
psi	pounds per square inch
PZ	Piezoelectric
RL	Resistance/Inductance
SIAM	Society for Industrial and Applied Mathematics
SSPC	Solid-State Power Controller
TRU	Transformer Rectifier Unit
UART	Universal Asynchronous Receiver Transmitter
UPS	Uninterruptible Power Supplies
VSV	Variable Stator Vane
W/P	Fuel Flow to Burner Pressure
WFM	Main fuel metering valve actuator sensor





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HANDBOOK-VOLUME II DIGITAL SYSTEMS VALIDATION

CHAPTER 16 ADVANCED VALIDATION ISSUES



PREPARED BY:

COMPUTER RESOURCE MANAGEMENT, INC.
950 HERNDON PARKWAY, SUITE 360
HERNDON, VIRGINIA 22070

PREPARED FOR:

FEDERAL AVIATION ADMINISTRATION
TECHNICAL CENTER
ATLANTIC CITY INTERNATIONAL AIRPORT, NEW JERSEY 08405

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1. INTRODUCTION

The Aeronautical Policy Review Committee was organized to formulate a plan for the advancement of United States aeronautical capabilities during the coming decades. In 1985 this committee issued a report that established three goals for America's aerospace community. The goals are comprehensive and span all areas of flight-subsonic, supersonic, and transatmospheric. The achievement of these goals will require major technological advances in all aeronautical technologies. Aerodynamic advances will include improved capabilities in computational fluid dynamics and unobtrusive instrumentation. Propulsion advances will include the development of a range of variable cycle engines and hypersonic ramjets. Structural technology advances will include the development of adaptive, thermal, damage tolerant, and propulsion structures. Material technology advances will result in new composites, alloys, and super-alloys being used throughout the aircraft. Flight systems will benefit from faster microelectronics, new processor architectures, and new software technologies. System integration will result in the interfacing of various aircraft systems under a single controlling device.

Two attributes are common to many of the advancing technologies: First, the embedding of intelligent devices into systems; and second, the aggregation of various avionic functions into higher order functions. In this tutorial these attributes will be referred to as Artificial Intelligence (AI) and System Integration, respectively. The intelligent device will virtually always be a digital microprocessor or network of processors, enriched with AI software. The direction of integration is to integrate the flight control function into the propulsion, structural, or aerodynamics system in order that a higher level system be created. The fundamental motives for these changes are decreases in operating cost, increases in service, and increases in safety.

The aforementioned attributes of advancing technology imply that digital avionics systems for coming generations of aircraft will be very different than today's systems. The certification of these new systems will require validation regulations and procedures that are different than those used in the past. This tutorial introduces the issues that will be the basis for the adaptations or changes required in certification procedures. Section 2 of this tutorial is oriented toward advancing technology. Paragraph 2.1 and 2.2 introduce the three national goals and the types of advances anticipated from their achievement. Paragraph 2.3 introduces the rotor and hypersonic aircraft. These are mentioned because they do not fall within the direct purview of the national goals, but will make significant contributions to commercial air service and/or technology. Paragraph 2.4 is an overview of the digital avionics advances anticipated in the future. Research and development efforts in AI and systems integration are highlighted. A generic "smart" engine and a heat removal technique for microchips are discussed as enabling technologies for future digital avionic systems.

Section 3 is oriented to the identification of those technological trends that will require adapting or changing some certification procedures. Paragraph 3.1 presents the Airbus 320, Boeing 7J7, Bell V-22 Osprey, and X-29 as examples of aircraft that may require immediate or near future certification. Paragraph 3.2 presents the propfan propulsion system as an example of an innovation that challenges current validation procedures; Paragraph 3.3 presents the Full Authority Digital Engine Controller (FADEC) and Fly-By-Wire (FBW) technologies in the same vein. Paragraph 3.4 discusses the problems of validating "intelligent" and multi-function systems.

Section 4 contains detailed discussions on a number of issues that are critical to certifying aircraft of the future. These issues include Fiber-Optics, AI, Advanced Controls/Displays, and Remotely Piloted Vehicles (RPV).

2. THE DIRECTION OF AMERICA'S AVIATION RESEARCH

2.1. Justification

In 1982 a report was issued by a governmental interagency group, under the direction of the White House Office of Science and Technology Policy, analyzing the state of aeronautical research and the role of the Federal Government in supporting that research. The conclusions of that report acknowledged the existence of the potential for advances in aeronautical technology that would result in monumental increases in aircraft performance. It was further recognized that the realization of these advances could only be accomplished by a coordinated effort involving both government and industry. This report resulted in the formation of the Aeronautical Policy Review Committee whose purpose was to track the implementation of the recommendations of that report. In 1985 this committee issued a report (National Aeronautical Research & Development Goals - Technology for America's Future, 1985) that established the specific goals that must be vigorously pursued if the advances are to be realized. This report also recognized that, in addition to the technological problems facing American government and industries, another factor must be considered: competition from abroad. In the interest of national economic and military security, United States aeronautics must meet these challenges and remain a decisive influence in world aviation.

2.2. The Goals

The Aeronautical Policy Review Committee established three national goals. When accomplished, these goals will be vital to America's preeminence as a world leader in aviation technology. Efforts on achieving the three goals will proceed in parallel, but with a time-phased order of attainment as depicted in figure 2.2-1 (Lupinetti, 1987). The goals are as follows:

- Subsonics goal: to revitalize aircraft technology.
- Supersonics goal: to attain long-distance efficiency.
- Transatmospherics goal: to secure future options.

The immediate emphasis will be on accomplishing the first goal before the end of the Twentieth Century. This will provide the impetus for continuing the effort and accomplishing latter goals after the year 2000.

2.2.1. The Subsonics Goal

The subsonics goal is to move a new generation of subsonic aircraft to a competitive commercial position in the world transportation system. It is important because it is a foundation for latter goals. While United States aircraft still maintain a broad base of technological advantage, the margin

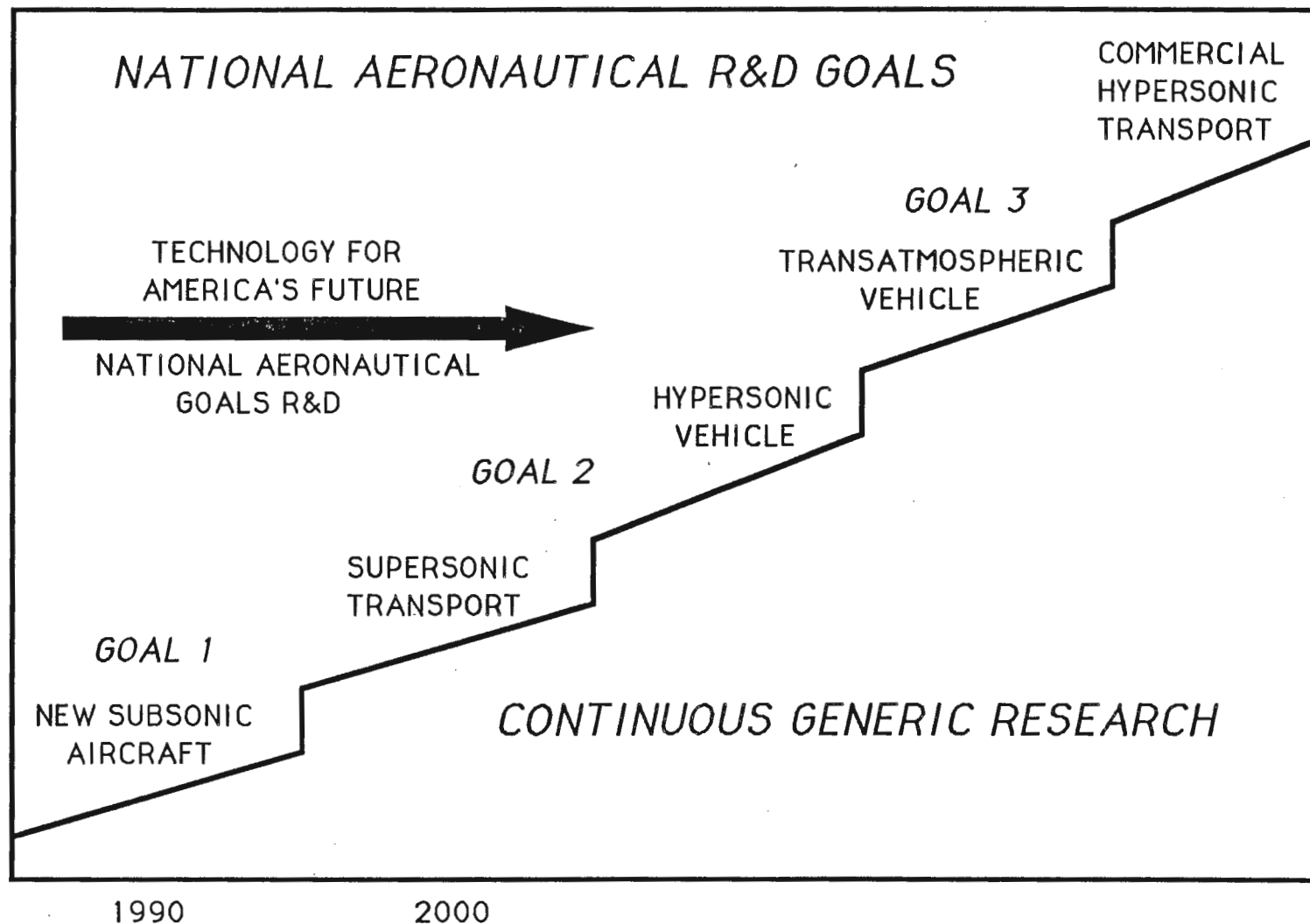


FIGURE 2.2-1. NATIONAL AERONAUTICAL RESEARCH & DEVELOPMENT GOALS
(Lupinetti, 1987)

of that advantage has been reduced significantly in recent years. In an increasing number of technological areas, foreign capabilities are now equivalent to, or superior to, that of America. During most of the last two decades, improvements in aeronautical technologies have been evolutionary. This will continue, but the new generation of subsonic aircraft must also foster revolutionary technologies. These two approaches to research and development, evolutionary and revolutionary, must produce the resources in the private sector that will enable the exploitation of the ensuing opportunities in supersonic and transatmospheric flight. This goal not only calls for the production of a safe, fuel-efficient, and technically superior aircraft but for a modernized national airspace system. This modernized airspace system is crucial if the benefits from the new aircraft are to be fully realized.

The objective is to have the new subsonic generation of aircraft readied by the mid-1990's. Consequently, the United States would enjoy advanced subsonic technology benefits before the 21st Century. Technologies that are essential to the development of the new subsonic aircraft are as follows:

- Laminar flow control advancements that substantially reduce air drag.
- A new generation of super bypass and propfan engines that will provide a more reliable and efficient operation.
- A new composite, high strain structure.
- More sophisticated, fully integrated flight controls and operating systems that will interface with the new national airspace system.

2.2.2. The Supersonic Goal

This goal becomes increasingly important because of changing transportation needs. Strategically and economically, the United States is becoming closely allied with the Pacific community of nations. This is in opposition to past development in the European community. The Pacific region's capacity for growth and development should be encouraged by America's enterprise and policy. A major constraining factor in the developing relationship between the United States and Asia is distance. Supersonic air travel will minimize the impact of distance. The development of technologies required for a new generation of supersonic transport aircraft will provide an opportunity for joint developmental efforts with the Pacific nations. Long-term political bonds may result from these cooperative efforts.

With the elimination of the United States Supersonic Transport effort in 1971, aggressive research and development was curtailed in the supersonic field. However, the Supersonic Cruise Research program, sponsored by NASA, maintained an American presence in this field through 1981; this work provides a basis for continuing the effort. Technological advances in the area of cooling methods, internal aerodynamics, coatings, and single crystal turbine blades offer a means for solving some of the problems present in the original effort. These advancements, when applied to a variable cycle engine with less noise and greater fuel efficiency, offer realistic hope for the development of a supersonic transport.

The report of the Aeronautical Policy Review Committee (National Aeronautical R&D Goals - Technology for America's Future, 1985) credits the following technological advances as being important to the supersonic effort:

- The application of powder metallurgy technology and superplastic forming techniques to load-carrying structures.
- The development of new thermoplastic, carbon-carbon, and metal matrix materials.
- The development of new fault-tolerant computers to assist in load alleviation and dynamic damping.
- The development of supersonic laminar flow that will result in significant reductions in the aircraft weight per pound of payload by providing sustained cruise speeds at much greater fuel efficiency.
- The development of computers with greater computational capabilities that will permit the computation of complex flows and optimizations extending the regions for supersonic laminar flow.

The integration of technological advances into a new supersonic transport aircraft would result in reduction of flight times to five hours or less between the most distant points in the Pacific area. For both commercial and military purposes, this is important to America's future.

2.2.3. The Transatmospherics Goal

The United States partitions flight into aeronautics and space. Some of the benefits obtained from subsonic flight can be enhanced in supersonic flight. Some of the benefits of supersonic flight can be enhanced in transatmospheric flight. For the United States to be a decisive influence in the world transportation system, it must have the following capabilities:

- Maneuvering craft into and out of the atmosphere.
- Rapidly responding to needs for low earth orbit missions.
- Attaining very rapid transport services from conventional runways between points on the earth's surface.

These capabilities will be required for America's security, both economically and militarily, in the coming century.

Much of the technology required for the convergence of the aeronautics and space flight concepts will be directly obtained from or will be extensions of the new subsonic and supersonic aircraft. The space shuttle experience will be a major source. In other cases, new technologies will be required. As scarce resources are spent in propulsion, configuration, materials, flight system, and fuel/coolant research, consideration must be given to the unique requirements of the transatmospheric goal.

2.3. The High-Speed Rotorcraft and Hypersonic Aircraft

While the three goals of the Aeronautical Policy Review Committee define the direction of aeronautical research and development, it is important to mention two types of aircraft that vaguely fit into these goals: the high-speed rotorcraft and hypersonic aircraft. Table 2.3-1 shows the accepted classes of aircraft by speed range.

TABLE 2.3-1. AIRCRAFT TYPE BY SPEED RANGE

Type of Aircraft	Speed Range
Rotorcraft/Vertical Takeoff and Landing	Hover to Mach Range 0.8
Subsonic	Up to Mach 1
Supersonic	Mach Range 2 to 4
Hypersonic	Mach Range 5 to 12
Transatmospheric	Mach Range 8 to Orbital Velocity

Strictly speaking, the rotorcraft may be considered subsonic, and the hypersonic may be considered transatmospheric. Because of their unique technologies, they are considered separately in this tutorial.

2.3.1. Rotorcraft

The short-haul commuter market is expected to grow extensively in the coming years. Vertical Takeoff and Landing (VTOL) craft, e.g., the tilt-rotorcraft should supply much of the capability for this market. These vehicles should permit commercial travel between center-cities at speeds in excess of present-day helicopters. The tilt-rotorcraft concept came into existence during World War II. When the prop-rotors are facing upward, the tilt-rotorcraft hovers like a helicopter; when they face forward, a turboprop airplane is created that can travel at speeds up to 300 miles per hour. The X-wing aircraft is another rotorcraft design that may service this same market. The X-wing is a stoppable rotor aircraft: when the rotor is active, the aircraft can hover like a helicopter; when the rotor is stopped, the aircraft becomes a conventional aircraft with cruise speeds in excess of 400 miles per hour.

The technology required for these two designs is at hand. Propulsion is by turboprop or turbo-jet; aluminum is the material used for body construction; and conventional hydrocarbon fuels/coolants are used. The major disadvantage is the greater cost per payload-mile, and this may require improvement. However, the convenience of beginning and ending flights at small, convenient heliports, rather than at large airports, offsets this cost to a degree. If this commercial market develops, the certification engineer will be required to create new or adapt existing rotocraft airworthiness rules.

2.3.2. Hypersonic Aircraft

This discussion is included for completeness. Hypersonic flight fills the gap between supersonic flight and transatmospheric flight. As noted in table 2.3-1, this spans Mach ranges 5 through 12. The potential commercial applications for transports of this class would be for transoceanic, transcontinental travel, the same as supersonic flight. The major difference is that a Los Angeles International (LAX) to Tokyo flight is estimated to require about 5 hours in a supersonic craft (Mach 2-3) or 2 hours in a hypersonic craft (Mach 5). Supersonic aircraft require conventional airports, while hypersonic aircraft would require extended runways at conventional airports.

Research in hypersonic flight started with the X-15 research program and is currently conducted by the Defense Advanced Research Projects Agency (DARPA). The current long-range cruise missile may be the first operational hypersonic vehicle. The problems of sustained hypersonic flight are significant propulsion and aerodynamic heating. The propulsion system must be able to sustain the Mach 5 and up speeds. The cruise missile uses the supersonic-combustion ramjet (scramjet) burning high-density hydrocarbon fuels. The materials required for hypersonic flight must be able to sustain temperatures of 5000 degrees Fahrenheit. Airframe research using supercold hydrogen is solving this problem. Civil aviation is unlikely to benefit from hypersonic flight or transatmospheric flight until well into the 21st Century.

2.4. Advances in Avionics-Related Technologies

Digital avionics is one of the fastest growing technologies in aerospace science. It continues to proliferate throughout aircraft and spacecraft, and finds application in subsystems, such as sensors, system controls, crew function automating equipment, displays, and communications. Research and development is currently underway for commercial FBW and Fly-By-Light (FBL) controls. The Airbus A320 is the first airliner in the world with computer-driven FBW controls which prevent the aircraft from stalling and exceeding speed/acceleration limits. While highly innovative, the Airbus Industrie plane appears already to be a commercial success: 439 booked orders and options from 15 airlines before its first scheduled flight. Certification and first entry service are planned for 1992.

2.4.1. Overview of Digital Avionics Technology Advances - Artificial Intelligence

Artificial Intelligence and expert systems are emerging in numerous digital avionics areas. Systems incorporating this technology are being designed to perform the following tasks:

- Coordinating sensor data to determine best flight path, thus avoiding terrain threats.
- Automatically initiating the execution of certain emergency procedures.
- Automatically reconfiguring aircraft systems to optimize performance or correct faults.

Artificial Intelligence and expert systems are software entities, and their emergence in digital avionics is accompanied by greater shipboard computer processing and data storage facilities.

The DARPA and United States Air Force (USAF) are cooperating in the Pilot's Associate program. While this program is directed toward combat pilots, the acceptance of this technology will most likely result in its adaptation for commercial flight. This program's emphasis is on providing the pilot with information and aids for assisting in immediate decision-making; a secondary emphasis is to assist the pilot in longer-range planning. The computer software required for this is a set of cooperating expert systems that operate at very high processing speeds; Lockheed-Georgia and McDonnell Douglas are developing this software. A new parallel-processor computer architecture is being developed to support these expert systems.

Other AI efforts involve the integration of various sensor data into a single composite display. Examples of the sensors being integrated are as follows: ground-mapping radar, stored digital map, and the location of nearby aircraft. AI is also an element in speech recognition. In this capacity, AI has advanced to the stage where a system can learn to recognize a pilot's voice in two sessions of less than 60 minutes' duration. This system makes use of the latest ITT (consultant committees for) International Telegraphy and telephony algorithms.

As mentioned earlier, the incorporation of expert systems and AI in avionic systems requires a new approach to software systems development. This is due to the greater logical complexity and the size of the software. Numerous efforts are underway with the objective of streamlining software development and maintenance in this environment; the Air Force Modular Avionic System Architecture is an example. Expert systems and AI will be covered in detail in section 4 of this tutorial. They also apply to other technological improvements, such as Advanced Controls and Displays, Voice Command Processing, Systems Integration, and Robotics. These areas will also be discussed in later sections of this tutorial.

2.4.2. Overview of Digital Avionics Technology Advances - Standardization and Integration

In addition to AI, standardization and integration of avionic modules is being emphasized. At this time, the impetus for this effort has been felt primarily in military avionics. In 1986, Congress requested that the services "prepare a joint plan for the inclusion of fully integrated, digital avionics...on all aircraft under development." DOD's (Department of Defense) response to this was the development of the Joint Integrated Avionics Plan for New Aircraft. As a result of this plan, common line replaceable modules will use Very-High-Speed Integrated Circuits (VHSIC). The common line replaceable modules would then be used to build standardized data processors, integrated systems (communications, identification, navigation), and signal processors. Standards for architecture, high-speed data bus, avionics computer, packaging, connectors, backplanes, and reusable software modules are in the process of being developed. The Navy Advanced Tactical Aircraft, the Air Force Advanced Tactical Fighter (ATF), and the Army Light Helicopter Experimental are among the initial craft destined for standardization. The effects of standardization and integration will be present in future commercial digital avionic systems.

As FBW and FBL controlled aircraft system designs gain greater attention, the development of digital Cathode Ray Tube (CRT) cockpit displays/controllers and integrated air/ground communication systems increase in importance. The V-22 Osprey, a tilt-rotorcraft with potential commercial usage, has its digital avionic systems highly integrated with other aircraft systems. All information to the pilot is provided through an operator's console consisting of four six-inch color displays. Such parameters as engine speed, engine torque, fuel quantity, fuel rate of flow, and electrical system configuration are displayed through this console. Alphanumeric keyboards with small CRT monitors are used to control the aircraft. The FBW Digital Flight Control System (DFCS) consists of three digital processors, each controlling a MIL-STD-1553 data bus. A digital system uses optical disk for the storage of maps which are used as a supplement to other navigational displays. The detection of faults and subsequent isolation of affected avionics subsystems is performed by digital software. The reliability of these digital systems is resulting in their growing acceptance into the area of flight-critical processing.

Commercial aviation is converging on an integration of air, ground, and satellite digital data links. To facilitate this, agreements are being reached on communications architecture and interface standards. With these agreements, different communications equipment will be capable of sharing data without modification to either equipment or software. An important benefit of this will be the ability to bypass faulted areas of the communications network by the process of rerouting. To some degree, the planned Federal Aviation Administration (FAA) Traffic Alert and Collision Avoidance System (TCAS) will require this technology. To a greater degree, the FAA's Automated Enroute Air Traffic Control System (AERA) will require it. AERA involves a complex modernization of all FAA on-the-ground air traffic computers. TCAS is scheduled to be operational in the near future; AERA is scheduled to begin implementation in the mid 1990's and continue implementation through the early years of the next century.

2.4.3. Overview of Digital Avionics Technology Advances - Others

In addition to the areas of AI, standardization, and integration, near-term changes are underway in other avionics-related technologies. Examples of these technological concerns and developments are as follows:

- In converting from analog to digital systems, concern is being directed to safety during the transition period. International activity is growing in the area of increasing Radio Frequency (RF) susceptibility standards and determining field strengths in the geographical areas adjacent to airports.
- Computer architectures are changing to permit greater processing speed and data handling capacity. AT&T has developed a family of signal processors that execute mathematical operations only when data is available, not at fixed time intervals. The advantages to architectures similar to this are rapid allocation of resources, high throughput, and parallel processing of multiple sets of data.
- Computer architectures based on the MIL-STD-1750A processor are being developed. These are being put on a single chip with VHSIC and Gallium Arsenide (GaAs) technologies. GaAs is also being used for cascadable arithmetic logic units, used in embedded computers requiring flexible word-lengths.
- Fiber optics technology is being developed to provide the bandwidth necessary for new digital data buses. Older MIL-STD-1553 buses are being replaced with fiber optic systems with compatible protocols and software. An experimental FBL system is being tested on a Sikorsky JUH-10-60A helicopter that utilizes an AT&T fiber optic bus; the bus appears to operate at 1 Million bytes per second (Mbps) to the user, but is actually operating at 20 Mbps.
- The Boeing 7J7, a next generation commercial aircraft, was tested with a voice recognition system. Twenty-five potential flight deck applications for voice recognition have been identified.
- Analog-oriented functions have been replaced with digital-oriented functions. This replacement has resulted in a reconsideration of the boundaries of hardware, firmware, and software. This has primarily been felt in the digital areas of process controllers with limited branches, single chips, and bit slices. Line replaceable units, earlier considered to be pure hardware, now have embedded processors which require software or firmware. DOD is attempting to standardize new software development around DOD-STD-2167 and modifications to MIL-STD-480.
- Signal processors are now being developed to process two and more functions.
- Reductions in weight and volume with accompanying increases in accuracy are being achieved with new global positioning systems.

- The "fly-by-glass" trend, the use of CRT's in place of traditional instrumentation has been under constant development. The use of CRT's places different workload patterns on the aircraft crew. Human engineering and user-friendly design factors are being approved before aircraft design is finalized.

2.4.4. An Application of Microelectronics - The "Smart" Engine

An example of the research underway that dramatically affects traditional technology is the application of microelectronics to propulsion systems. The research cited in this section was performed by A.H. Epstein, and is summarized in "'Smart' Engine Components: A Micro in Every Blade?" (Epstein, 1986).

Most of today's attention in propulsion technology is directed toward the debate over the resurrection of the old turboprop concept in the guise of the "propfan". However, other research is underway that may have major effects on the aircraft of the future. Aircraft gas turbine engines are virtually open-loop devices. They have remained so for essentially three reasons: First, relatively poor sensor technology and limited onboard computational capability have not justified a controller; second, high reliability has been achieved with simple systems; and third, high performance gains have been realized without the need for control loops. However, advances in microelectronics make possible engine control devices with major computational capacities; the result is that gas-turbine components and subsystems may be moved from open- to closed-loop operation. Mr. Epstein feels this may be desirable for four reasons:

- Current engine designs represent a constrained optimization. Improvements in the performance of components or subsystems, off-design, may not result in improved performance of the design: a high thrust-to-weight ratio conflicts with good performance retention and maintainability. A relaxation of these constraints could improve performance.
- Current design practice is expensive; self-adaptive components may allow less expensive development.
- Through dynamic, subsystem reconfiguration, adaptive components could increase damage tolerance and reliability.
- Current technology may not produce the efficient, multifunction propulsion systems that will be required for advanced aircraft.

In order that "smart" technology be incorporated in a propulsion system, a feedback control mechanism must be present that will permit a component's adaptation to local conditions. This feedback control mechanism will consist of sensors, processors, and actuators.

Applications of "smart" engines may be categorized as near-term versus long-term, low risk versus high risk. Examples of potential applications are as follows:

- Active control of turbine blade tip clearance, based on real-time measurements instead of open-loop scheduling. This is a near-term application

that has the development of highly reliable sensors as an enabling technology.

- Active exhaust-nozzle position improvement. The nozzle would be adjusted based on data from the aircraft accelerometers, thus improving vehicle performance. Today's continuously variable nozzles operate with pressure mismatches due to nonideal conditions, schedule problems, atmospheric gradients, and other factors. This is a near-term application which requires the development of variable-exhaust nozzles capable of lasting through many adjustment cycles as an enabling technology.
- Active inlet distortion control. High compressor stall margins (on the order of 10-20 percent) exist in high performance military aircraft because of inlet distortions at high angles of attack. This stall margin could be reduced if the compressor distortion tolerance could be dynamically adjusted. This adjustment could be based on local pressure distribution measurements that result in the activation of individual guide vanes. Enabling technologies consist of actuators and sensors of sufficient bandwidth and reliability, and greater understanding of compressor behavior. This is a longer-term application.
- Active noise abatement is one of many additional potential applications.

Common to virtually every potential application are the requirements for significant technological improvement in actuators, sensors, processors, data transmission, and control system architecture. In order that the requirements for withstanding large forces and for small physical size be met, actuators have traditionally used hydraulics. However, their complexity and maintenance characteristics make it unlikely that large numbers of these devices could be used throughout an engine. Large numbers would be required if the engine were closed-loop in design. Electric actuators are desirable if they can be developed to meet the torque, packaging, and bandwidth requirements. Samarium-cobalt motors and piezoelectric devices offer some promise in meeting these requirements, but each has its shortcomings. (Layers of piezoelectric material, dispersed throughout a structure, have been proposed as actuators capable of supplying distributed forces of relatively high frequencies.)

For the purpose of increasing reliability, a designer may provide additional computing elements in the system, rather than add sensors to the engine. Using a single chip to incorporate signal conditioning, transducer, data reduction, and control may provide the needed signal processing while not affecting reliability significantly. Fiber optics may also offer hope for improvement.

Microelectronics, while the catalyst for developing closed-loop engines, may also be a problem. Control elements dispersed throughout a gas turbine engine will require cooling. Current semiconductor technology is restricted to a modest temperature range. Development of high-temperature semiconductor technology is a partial alternative to a cooling system. (See paragraph 2.4.5 for a discussion of chip cooling.) Conceivably, the processing power currently available through microelectronics may be insufficient. The computational and control functions needed in adaptive applications may be beyond current technology levels. If this is the case, this technology must also be improved.

Advances in microelectronics are commercially driven. The improvements required for aircraft adaptive engines may not result from this commercial motivation. In this instance, motivation may have to be supplied by the propulsion engine community.

Advanced adaptive applications may require control mechanisms beyond the capabilities of current control system architecture. It may even be necessary to distribute the control function over multiple mechanisms. Other subtechnologies, such as nonlinear control and fault tolerance, as well as adaptive control may be utilized.

Questions arise with the issue of certification. How is an engine that contains components and subsystems that are self-adaptive certified? Does every engine require separate certification? The design, as well as the validation and performance verification techniques of the manufacturer, becomes part of the certification process.

2.4.5. An Application of Heat Removal Technology - Microelectronics

A major problem in the growing microelectronic technology is chip cooling. If digital avionic systems are going to provide the capabilities required by advanced aircraft, this problem must be solved. As discussed in paragraph 2.4.4, research in the area of high-temperature semiconductivity offers some promise for improvement. Another approach is the use of micro heat pipes. The research and development efforts mentioned in this section were obtained from an article written by Dr. George Peterson; his article (Peterson, 1987) was a basis for this discussion.

A number of recent developments in heat pipe technology offer hope for alleviating the problem of microchip sensitivity to heat. These developments are as follows:

- Large silicon wafers are being cooled by devices that are one-fourth the size and one-seventh the weight of fins, coolant loops, or cold plates, traditional cooling devices. These new devices are being produced by Radio Corporation of America (RCA).
- Transistors are being cooled by a heat pipe method that reduces junction temperature to 63° C at a power level of 40 W. Hughes Aircraft is the developer of this method.
- Cooling systems have been developed with improved thermal response times and reduced thermal resistance for thyristors. Thyristors are solid-state devices that convert alternating current to direct current.
- Before the year 2000, very small heat pipes should be commercially available in the United States. These pipes should play an important part in the development of Three-Dimensional (3-D) chips.

According to Peterson, a heat pipe is defined and operates as follows:

"A heat pipe is a passive device typically consisting of a sealed container lined with a wicking material and filled with just enough fluid to fully saturate the wick. It operates on a closed vaporization and condensation cycle. Heat added to the evaporator section of the pipe vaporizes the working fluid. The high temperature and corresponding high pressure in this region cause the vapor to flow to the cooler end of the pipe, where the vapor condenses. Capillary forces in the wicking structure generate a positive pumping pressure that forces the working fluid back to the evaporator end."

The operation of the heat pipe is shown in figure 2.4-1. Heat pipes are superior to the best solid conductors, because the pipe has a much higher effective measure of thermal conductivity; this is due primarily to the transfer of heat by the concept of phase change of the working fluid.

Heat pipes are categorized by size: large, medium, and micro. Large heat pipes are used to remove heat from component housings; medium heat pipes are used as mounts for components; micro heat pipes may be embedded within chips. Large heat pipes may be .3 m or longer; micro heat pipes may be less than 5 cm long, and 1 mm in diameter; medium heat pipes cover the ranges in between.

In avionics, a typical use of large heat pipes would be to reduce the heat in housings for various systems or subsystems. Since the housing may be sealed at the points that the heat pipes penetrate, the housing remains a closed system, free from contamination by harsh environments. Medium heat pipes are used as a substitute for single-phase coolant loops or cold plates. These are commonly used to cool semiconductor devices, integrated circuits, or other components mounted on printed circuit boards. In an avionic system, several medium heat pipes may be arrayed into a substrate with a printed circuit board mounted on the substrate. In addition to cooling the printed circuit, this substrate may serve to reduce a "local hot spot". This will increase the mean time between failures of the circuit board. The three research and development efforts alluded to earlier cooling large silicon wafers, transistors, and thyristors use medium heat pipes.

Micro heat pipes should be commercially available in the 1990's. In a commercial semiconductor chip, all chip elements lie in a plane on top of a conducting material, either silicon or GaAs. The effective gate density of a single chip would be increased if the chip contained multiple planes of chip elements; this would enable element interconnections vertically, as well as horizontally. One of the problems inherent in this architecture is the heat buildup that would result. Micro heat pipes offer a potential solution to this problem. Current heat removal systems function at the 50 W/cm² level. 3-D chip architectures would require heat removal rates of 200 W/cm². Embedded micro heat pipes could function in this range. Heat accumulations in avionic systems have the following three components: First, the resistance from semiconductor junction to the device; second, the resistance from the device to the heat sink; and third, the distance from the sink to the ambient. The micro heat pipe reduces the heat buildup in the resistance from the semiconductor junction to the device, the largest of the heat components.

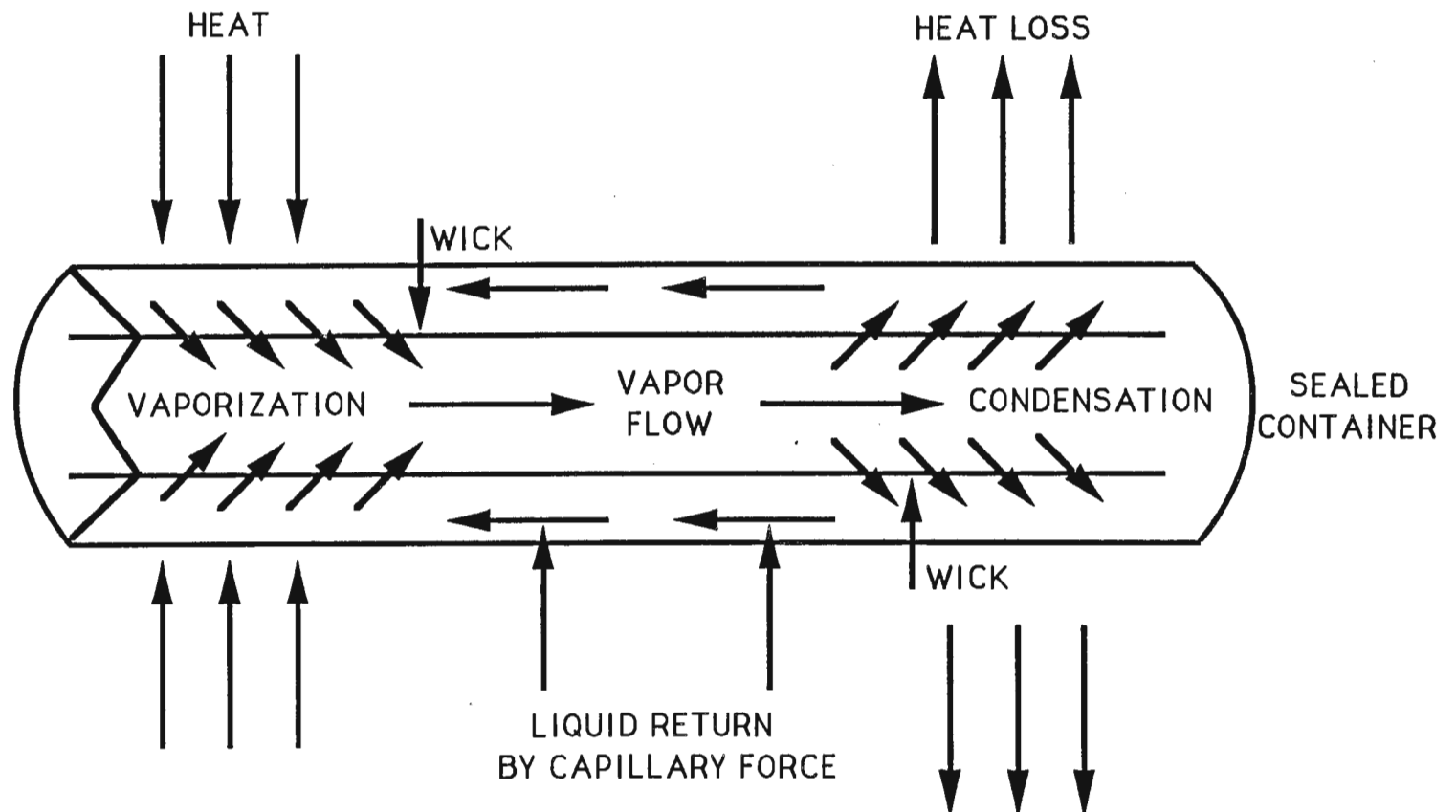


FIGURE 2.4-1. OPERATION OF HEAT PIPE IN MICROCHIP COOLING

One of the enabling technologies for advanced avionic systems, in lieu of the speed, propulsion, and friction anticipated, will be heat removal. Heat pipes are compact and light-weight. If micro heat pipes can become an integral part of 3-D chips, a major advance will have been made.



3. CERTIFICATION AND ADVANCING TECHNOLOGY

3.1. Challenges for the Certification Process and Regulations

Section 1 discussed the goals that will be driving aerospace technology for several decades. Conjecture was also given on the general technological advances that are required to support the achievement of those goals. The supersonic and transatmospheric aircraft required to meet their respective goals lie somewhat in the future; however, it is reasonably certain that there will be a new generation of commercial subsonic aircraft in use before the year 2000. The new technologies found in these aircraft pose a challenge to traditional certification procedures and regulations.

3.1.1. The New Generation of Subsonic Aircraft

Airbus Industrie's entry in the new subsonic generation utilizes today's technology; the assumption being that currently available technology is more than adequate to meet the needs of the airlines over the next decade or more. The Airbus Industrie A320 is a narrow body, short range aircraft; the A330 and A340 are the wide body, long range aircrafts. There are plans in Airbus Industrie to have their new aircraft ready for certification and service by 1992.

The FBW technology incorporated in the A320 will tax certification regulations. With FBW technology, heavy mechanical and hydraulic aircraft systems are being replaced by electronic wiring; the result is an "all-electric" airplane. In any Flight Control System (FCS) using FBW there is a set of sensors which produce electrical inputs into a flight control computer. There are also a number of actuators on the end of the communications link which drive control surfaces. The flight computers take signals from all sensors, cross monitor them, produce consolidated signals, and identify them as the best information. The computers then process the aircraft's control laws, determining what the control surfaces should be doing. Control signals are either forwarded directly to the control surfaces or to actuator drive units.

On the A320, FBW will be used on all surfaces for the rudder and tailplane trimming reversionary mode. Hydraulic activation is used on all surfaces. The FBW system incorporates two elevator aileron computers and four spoiler elevator computers. These computers drive left/right combinations of aileron, ground spoiler/speed brakes, and elevator left/right surface segments. The pitch axis control includes an automatically trimmed horizontal stabilizer. The ailerons and spoilers also receive inputs from a load alleviation system. For pilot control, side-sticks are used in lieu of control columns. The side-stick includes the autopilot disconnect feature, the radio communication/interphone switches, and a datum adjust switch for varying selected headings and vertical speeds.

Boeing plans to introduce its next generation aircraft, currently called the 7J7, by the mid-1990's. It has been announced that the 7J7 will take advantage of emerging technologies in propulsion, aerodynamics, structures, materials, aircraft systems, and cabin design. It will be powered by ultra-high bypass engines.

3.1.2. A New Application for Vertical Takeoff and Landing Aircraft

In addition to the new subsonic aircraft, the year 2000 may also see VTOL craft playing a major role in commercial aviation. This craft will most likely utilize either a tilt-rotorcraft or X-wing design. The military success of the Bell XV-15 and its larger version, the V-22 Osprey, give the tilt-rotorcraft design the advantage. Bell has issued a design proposal for the Bell D326 Clipper, a commercial version of the V-22. Numerous turboprop aircraft are now used by "third level" commercial carriers. Such aircraft as DHC-7's (DeHavilland Twin Otters) and Shorts 330's dominate this market. A civilian version of the V-22 would probably carry 44 passengers. The United States Marine Corps version carries 24 fully armed assault troops.

The potential advantages of VTOL craft replacing turboprop craft are significant. As mentioned earlier, travel between cities serviced by third level carriers would be between center-cities, not between major airports. To the traveller, this may mean greater cost since VTOL craft are less fuel efficient; however, the time saving may be great. VTOL travel in this marketplace has been estimated to be three times faster door to door than any other combination of vehicles. When the cost of ground transportation is considered, the cost differential may lessen, or even disappear. To the air travel community as a whole, the reduction in airport and airway congestion may be major; air travel may become considerably safer. Airline deregulation has increased the problem of congestion. Conventional jet aircraft now fly on many of the well travelled routes much more frequently with deregulation. Third level carriers have grown in number to service the routes deserted by the trunk lines; these carriers usually fly lower performance aircraft. The third level carrier aircraft use the same runways as the high performance aircraft flown by the trunk carriers. If the third level carrier used VTOL craft, simple airport facilities could be used that were remote from major airports and convenient to center-cities. An example of this is a feasibility study conducted by the Port Authority of New York and New Jersey on using the Manhattan Battery Park heliport for XV-15 landings and takeoffs.

Another potential market for tilt-rotorcraft is the corporate air travel market. This market is currently serviced by planes such as the Learjet. The advantages to the traveller and the air travel community are similar to those previously mentioned.

3.1.3. The Experimental Aircraft - X-29

The new generation commercial subsonic plane(s) and the tilt-rotorcraft are two new aircraft fostering new technologies that will impact certification. Another aircraft that is an indicator of even more advanced technological change is the X-29. The A320 incorporated new technology but at a moderate pace. Performance improvements of 5 to 10 percent over existing aircraft may be expected in

aerodynamics, structures, propulsion, navigation, communication, and FCS's. The A320 will utilize the CFM56-5 or IAE V2500 turbofan engines. The 7J7 is a bit more radical. The initial use of polymer material in the airframe, a flight management system with stability augmentation, and an ultrabypass engine is estimated to improve performance another 10 percent beyond the latest turbofan aircraft. A generic subsonic aircraft using laminar flow technology and ultra high bypass propfan engines, as well as active stability augmentation and very high aspect ratio supercritical wings, is estimated to improve overall performance by 60 percent over today's aircraft. The X-29 is considered most radical.

The X-29 is an advanced technology experimental aircraft developed under the guidance of the DARPA and the National Aeronautics and Space Administration (NASA). The X-29 incorporates the following new technologies:

- Forward Swept Wing: provides aero efficiency.
- Close Coupled Canard: low-speed control.
- Thin Supercritical Airfoil: control at high angle of attack.
- Discrete Variable Camber: enhanced maneuverability.
- Relaxed Static Stability: favorable volume distribution.
- Three-surface Control: increased agility.
- Aeroelastic Tailoring: solves aero divergence.
- Composite Wing Covers: reduces drag and weight.
- Triplex Digital Design: flexible/multiple modes.
- Fly-By-Wire: tailor handling qualities.

The X-29 possesses an inherent 35 percent negative static stability margin: a design feature enabling extensive maneuvering abilities, yet offering a severe control system problem. The FBW DFCS stabilizes the aircraft, permitting normal, acceptable handling. The control laws for the NASA's Space Shuttle, NASA's F-8, and the Air Force AFTI/F-16 could not stabilize this degree of negative static margin. The pitch instability of the X-29 is four times that of the AFTI/F-16. Therefore, the development of the X-29's control laws and a DFCS capable of stabilizing the aircraft is a major accomplishment.

The DFCS was designed to exploit the aerodynamic advantages of the aircraft, e.g., the forward swept wing and the close coupled canard. The DFCS is multi-mode and has three channels. The system uses seven F-16 National Waterlift actuators to drive the two canards, six segmented-trailing-edge flaperons, and a single rudder. A newly designed actuator drives each of the two strake flaps. Flight control computers, air-data sensors, and an attitude reference system are the remaining components of the DFCS. Each of the three digital computers has an analog backup in parallel. Exercise of the several modes provides a graceful

degradation feature when components begin to fail. Normal operations use the full digital mode and the associated automatic camber control. A normal approach mode controls takeoff and landing. There are two reversion modes:

- A digital reversion mode using different software than that used in the normal mode. This mode does not utilize noncritical sensors.
- An analog reversion mode. This mode is only activated when there is a generic digital control fault.

The up and away, power approach, and ground contact for weight on any wheel operational regions are available with each mode. There is also a precision approach control mode which enables the pilot to directly control flight path angle by means of a pitch stick. When switching between modes, precautions have been taken to minimize transients.

While the immediate beneficiary of X-29 technology is the ATF program, it is a matter of time before advanced civil aircraft will likewise reap benefits. A successful experimental aircraft is expected to increase performance by five to ten times that of existing aircraft. Experimental aircraft are being studied jointly by NASA and the Navy for an oblique wing design, and by DARPA and NASA for an X-wing rotorcraft design. The future holds promise for an experimental aerospace plane. The technologies fostered in these experimental craft extend beyond subsonic flight and into supersonic and hypersonic flight.

3.1.4. New Aircraft - Summary Comments

The requirement for change is the same whether the certification domain is transport, rotorcraft, or small general aircraft. New systems oriented to today's technological categories will affect changes in certification regulations and procedures. Examples of these are Upper Surface Blowing (USB) propulsive lift and the propfan engine. FAA certification procedures, as of 1984, did not allow for the advantages of powered lift technology as incorporated in USB designs. NASA has found the USB design to be fuel-efficient for low-speed, Short Takeoff and Landing (STOL) flight; current research is testing its performance in high-speed flight. If the technology is adapted to civil use, FAA certification must follow. The propfan propulsion design is a resurrection of the turboprop engine. The propfan, a result of NASA research, incorporates new propeller technology in an energy efficient engine. Paragraph 2.4.4 of this tutorial introduced microelectronics to a propulsion system on a theoretical basis. Most propfan designs utilize microelectronic digital devices for propeller control functions, devices not yet FAA certified for that purpose. The electronic devices will be located in the vicinity of the corrosive exhaust system of the engine, a high temperature environment. While the microelectronics will be used for propeller control, e.g., pitch change control, it will not be used for adaptive control of the engine elements. The propfan is an example of revolutionary engine/propeller design, augmented by digital avionics.

3.2. A Specific Problem for Certification - The Propfan

Current air-worthiness regulations do not directly cover the design features present in the latest technological advances. Propfan or ducted/unducted

variable pitch fan propulsion system designs are examples. As a response to this, the FAA is taking steps to assure that certification programs will accommodate the latest technologies. "New Propulsion Concepts and FAA Certification" (Aerospace Engineering, 1988) describes the difficulties in certifying the propfan.

The intention of the Federal Acquisition Regulation (FAR) is to classify aeronautical products into types and to define the minimum requirements for certification. Each product type is covered by a unique regulation called a Part. The product type is evaluated against its respective regulation. As examples, Part 25 contains the criteria against which transport category aircraft are evaluated; Part 33 contains the criteria for aircraft engines; and Part 35 covers aircraft propellers. In the past, engines and propellers with the propfan design have been classified as two distinct product types. This classification must be reconsidered for accuracy. FARs permit the stipulation of special conditions when the existing regulation does not seem to cover the particular product type. The intention of the FAR is to promote public safety, not define limitations of aircraft design. Special conditions are required for the propfan. These conditions must guarantee that the aircraft containing the propfan propulsion system achieves the safety standards demanded by Part 25, while the propfan satisfies the standards established in Parts 33 and 35.

One of the problems with propfan certification is the requirement for containment of the propeller blade. Part 33, the engine regulation, stipulates that any failed or released fan blade must be contained within the engine housing. This requirement is based on traditional turbofan engine design. In a turboprop-based design, the blades are not shrouded and do not have any means of containment. With safety as the underlying motive, it is then necessary to provide the design measure that minimizes the chance that a blade will fail or be released. Part 25, the transport aircraft regulation, stipulates that the aircraft must be able to withstand the impact of a failed or released blade with no structural damage that would pose a hazard to the safety of the aircraft. Applicants for certification must demonstrate that the aircraft can sustain the structural damage caused by a failed or released blade, and that the aircraft's other critical systems will continue to operate correctly during the period of blade loss. The propfan applicant should demonstrate that in his aircraft the probability of blade loss is acceptably low, and that the structure of his aircraft is such that blade loss would not jeopardize the safe continuance of the flight.

A second problem is that the propeller hub found in a propfan design may differ significantly from that found in traditional propeller aircraft. The differences are in the areas of size, number of blades mounted, and temperature environment. Propfan propeller hubs are large, have several blades mounted, and operate in a high temperature environment. Blade retention strength is the prime safety consideration for the propfan hubs; the FAA requirement is that the hub's blade retention strength be sufficient to ensure that the loss of a blade is a virtual impossibility. If the hub is part of the turbine rotor, then its shattering may be a greater threat to the structure of the aircraft than blade loss. Adequate margin between load and material strength should be a design attribute to eliminate this safety concern. All propfan components, the hub included, must demonstrate that they do not deteriorate measurably when exposed

to the exhaust stream of the engine (high temperature vulnerability). To summarize, the propeller hub (blade retention facility) cannot be sufficiently validated by the precise criteria contained in Part 35. The criteria must be supplemented with special conditions to show that the same safety objective is achieved.

A third problem is the propfan pitch control system. It is a digital electronic control system. Digital avionics have now been incorporated into the propulsion system. Digital electronic control devices have not yet been certified by the FAA for propeller control. The pitch control mechanism for the propfan also includes mechanical or hydraulic linkages and a two-stage counterrotating rotor. All of these are not traditionally found in the pitch control system. The safety goals are as follows:

- No failures that result in excessive propeller speeds that may result in fragmentation.
- No failures that will result in a high drag condition.
- No control system failure that results in an undesirable command to autofeather (automatically move the propeller to zero drag position).

Current industry design practices have resulted in pitch control mechanisms that satisfy these safety requirements. The problem of certification is to assure that the corresponding propfan mechanism demonstrates the same degree of airworthiness.

A fourth problem is the design of the propfan propeller blade. These blades may be more susceptible to time and usage than traditional blades. Propfan blades are characterized by swept platforms, wide chords, and thin airfoils. They are made of layers of composite material, enclosing a light filler material. As such, they may be more vulnerable to chemicals, foreign object damage, heat, and humidity. Because of this, it will be necessary to track each blade and hub to determine its performance history. Periodic inspections for durability, flutter, vibration, and stress will also be required. This is not required for normal propeller blades and hubs. These are usually certified for an unlimited life, given that certain maintenance levels are sustained. As a result, detailed records of time-in-service are not maintained. Another safety concern over the propfan blade is that composite material does not conduct electricity as well as metal; therefore, the effect of lightning strikes must be determined.

Other potential problems with propfan certification lie in the areas of potential gearbox failure due to the high horsepower ratings, noise, and safety requirements for aft-fuselage mounted engines. As examples of the latter, minimum clearance standards are different for aircraft with engines mounted behind the fuselage and ingestion of foreign objects is of greater concern when propellers are located at the rear of the craft.

The propfan is an example of a product of a growing technology that has novel and unusual design features. These design features are not addressed in current

certification regulations. However, the issuance of and compliance with special conditions will assure that proper controls are in place for assuring public safety.

3.3. A Specific Problem for Certification - The Rotorcraft

Certification rules for helicopters have evolved as helicopter technology has advanced. As avionic systems utilizing microprocessor technology are increasingly used in helicopters, these certification procedures will be challenged. There is a major concern with respect to the impact of digital electronic controls on civil helicopters. Technologies such as mechanical systems, pneumatic systems, and hydraulic systems are mature and are covered by the "specific rules", of air-worthiness regulations which state specific requirements. As electromagnetic and electromechanical technologies were increasingly used in helicopter systems, certification procedures migrated from specific rules to "general rules". With a general rule, more emphasis is placed on achieving a desired safety level and less on how that safety level is achieved. One of the purposes in doing this was to maintain safety standards, but not constrain design. Some of the difficulties of certifying helicopters using advanced technologies are described in "Certification Issues in Civil Helicopters" (Brahney, 1988). The information contained in this section is based on "Certification Issues in Civil Helicopters" (Brahney, 1988) and information obtained from John D. Swihart, Jr., and Richard Vaughn of the FAA.

3.3.1. The Full Authority Digital Engine Controller

The FADEC was one of the first systems introduced in a helicopter that utilized advanced control technology. The FADEC uses microelectronic technology instead of the traditional hydromechanical technology; the former offers increased accuracy, improved control modes, better maintenance, and substantially reduced life cycle costs. The FADEC must operate in a hostile environment for thousands of hours without failure or maintenance. The components of the FADEC use lead-less chip carriers and include advanced 16-bit microprocessors with error correcting memory, analog/digital converters, pressure/thermal sensor signal converters, and digital data bus interfaces. The Sikorsky S76B is equipped with the FADEC.

There are two general rules in FAA regulations that are applicable to the certification of helicopters which use systems based on advanced control technologies. FAR 29.1301 covers the operation of the systems in the aircraft; FAR 29.1309 specifies the interaction between the system and the helicopter environment, and the acceptable ramifications of system failures. Software validation for the FADEC requires demonstrating through testing in the helicopter environment that the system is not only logically correct but that the user's requirements are satisfied. Errors may exist that result from software deviations from requirements and from timing considerations during program execution. The software must be evaluated by testing that begins with a sequence of stand alone tests, proceeds through simulated engine tests, continues with testing on an engine test stand, and culminates with flight test. A difficulty with software testing is that general rules in certification require that the testing procedures and results be verified. This means that the FAA must review documented test results, as well as software development

documentation. In the absence of a standard, this documentation will be unique to each manufacturer. Until recently, there was no movement for development of standards for software documentation.

3.3.2. Fly-By-Wire Technology

The incorporation of microprocessor technology in helicopter FCS's is more recent than its use in power plants. This innovation has come in the form of FBW. The V-22 Osprey tilt-rotorcraft is an example of a tilt-rotorcraft that utilizes FBW technology. Since FCS's are more relevant to flight safety than FADEC, this innovation is viewed differently by the FAA. In FBW technology, flight safety will be almost completely in the hands of microprocessor controls. The certification of these FBW subsystems for safety is of critical importance. The specific and general rules that are applicable to these subsystems must be sufficient for assuring public safety. There are three major concerns in this area:

- The reliability of the electrical power supply system.
- The security of the FBW subsystems against electromagnetic hazards.
- The reliability of embedded software.

While these concerns are mentioned in the context of certifying FBW technology in rotorcraft, they are equally applicable to certifying FBW in fixed wing aircraft, e.g., Airbus A320.

3.3.2.1. Electrical Power System Reliability

With the introduction of digital electronic control systems in FCS's, the importance of the electrical power supply systems has been elevated. Since the FBW subsystems are critical to flight safety, the electrical power systems supplying energy to them must always operate at a level sufficient for sustaining the FBW subsystems' functioning. Consequently, the validation and certification of these electrical power systems is of major importance to the FAA. The power systems must be tested thoroughly under both normal conditions and conditions in which system errors occur.

3.3.2.2. Security Against Electromagnetic Hazards

Prior to FBW, a helicopter's avionic systems did not perform flight-critical functions. Usually the avionic systems controlled only instrumentation not essential for the safe landing of the helicopter. In this situation, electromagnetic hazards were of little concern. With the proliferation of microprocessors throughout flight-critical systems, this is no longer the case. Since High-Energy RF (HERF) fields (generated by commercial radio broadcast transmissions and radar installations) and lightning strikes occur with some frequency, their effect on critical electronic rotorcraft subsystems must be considered. For the purpose of certification, although there are some similarities in the protection against HERF fields and lightning strikes, the major differences to be considered include:

- Amperage: Amperage from RF fields is in the milli- to microampere range while amperage from lightning strikes is in the kiloampere range.
- Frequency: Frequencies from HERF are mainly greater than one megahertz. Frequencies from lightning are mainly less than one megahertz.
- Interaction: Lightning attaches directly to the aircraft in isolated spots. A HERF field envelopes the entire aircraft.
- Effects: A major difference in effects is that lightning can cause burnout or destruction while HERF can only upset or cause glitches in systems.

Because of the use of electronic devices in flight-critical subsystems, the FAA is issuing voluntary testing standards for certifying that the electronics of an aircraft are adequately shielded against electromagnetic hazards. The need is acute because not only are flight-critical subsystems becoming more susceptible to electromagnetic hazards, but protective metal airframe skins are being replaced by nonconductive composites. Since graphite composites are not good conductors, they cannot shield interior circuits. Currents resulting from high-energy fields now move directly along interior conductors. Advances in circuitry also aggravate the problem for the following reasons:

- Increases in microcircuit bandwidths result in increases in the bandwidth against which they must be protected.
- Miniaturization of circuitry decreases the operating voltage which increases the amount of protection required.

Protection from HERF fields can be achieved by either shielding the desired component or subsystem with a conducting material or by isolating the component or subsystem electrically from induced charges. A combination of the two methods may be most desirable for reasons of weight and cost.

The FAA standards initially only covered structural damage to aircraft; but the latest standards, modeled after those developed by the Society of Automotive Engineers (SAE), are more comprehensive. These standards are documented in the SAE Committee Report (SAE Committee AE4L Report No. AE4L-87-3). They were the basis for early approvals of FADEC systems. An aircraft meeting the new standards is expected to be able to withstand lightning strikes with currents in the 200 kiloamp (kA) range. These strikes usually occur only once every few hundred aircraft-years of flying, but the standards should be adapted to protect aircraft from the maximum currents and maximum rates of rise of lightning.

While the FAA standards may be in place, the avionics technology required to meet them will not be easily achieved.

3.3.2.3. Embedded Software Reliability

The standard accepted by the FAA for certifying software in flight-critical systems is Radio Technical Commission for Aeronautics (RTCA) document DO-178A. However, this document states that the procedures contained therein are not sufficient for assuring an acceptable level of safety for flight-critical

systems. The FAA has recommended that manufacturers providing systems with embedded software conform to DQ-178A. Exceptions have been made when redundant software is present. In this instance, the FAA has stipulated that:

- There must be at least two different software systems.
- There may be no shared software modules.
- Each software system must be capable of performing all critical functions independently of another software system.

3.3.2.4. A Backup for FBW - Fluidic Flight Control

Some of the concerns expressed in paragraphs 3.3.2.1 and 3.3.2.2 would be alleviated if there were redundancy to the FBW system. Test results from experimentation with a fluidic FCS serving as backup to FBW is discussed in "Fluidic Flight Control: Early Test Results" (Aerospace Engineering, 1988).

In hostile environments, such as high temperature, vibration, and Electromagnetic Interference (EMI), fluidic flight control is being considered as a functionally redundant, nonelectric backup alternative for FBW. The motivation behind this research is the failure probability that exists in FBW systems that is non-existent in traditional hydromechanical FCS's. The composite FCS is a full authority, two-axis, lateral-directional control system for a Navy T-2C aircraft; the tests are being conducted at the Naval Air Development Center.

The selection of either FBW or the fluidic flight control modes is under pilot control. There are no transients when the modes are switched because the idle system is always in active standby status. Early test results indicate that using fluidic technology as a backup for FBW in aircraft subject to high temperature buildups, vibration, and EMI is a viable alternative.

3.4. The General Problem for Certification

In paragraph 3.2 the propfan propulsion system design was critiqued from the certification point-of-view. In paragraph 3.3, the difficulty of certifying new rotorcraft because of its use of recent advances in microprocessor technology was detailed. In this section, difficulties of a broader nature will be discussed.

3.4.1. Intelligent Components, Subsystems, and Systems

Intelligent components, subsystems, and systems that are applied to many systems involving microprocessor elements are emerging as individual technologies. Their emergence as technologies has paralleled the technological growth made in digital computers and storage devices on the microelectronic level.

3.4.1.1. Adaptive Components and Subsystems

Adaptive components and subsystems are intelligent components that are characterized by digital controls containing logic that adapts to meet the needs

of different inputs or environments. These adaptive components and subsystems create problems for type certification. Questions arise including:

- How are requirements defined?
- How is a system containing one or more self-adapting subsystems type certified?
- Does each system require independent certification?

These questions must be answered if certification regulations and procedures are to accommodate advancing technology.

The cost of fuel is a major factor in acceptance of an aircraft design. For this reason, such measures as seat-statute-miles-per-gallon have gained acceptance as figures of merit for judging propulsion system design. Figure 3.4-1 (Lupinetti, 1987) demonstrates an estimated tripling in performance between the introduction of the first generation turbofan engines through the anticipated propfan engine. One element in the current trend is to gain further engine performance improvement through "smart engines". At this point, digital avionics impacts propulsion systems. Paragraph 2.4.4 of this tutorial discusses a "smart" engine in detail. Aircraft gas turbine engines have historically consisted of an integrated set of open-loop subsystems with a feedback fuel control. Only a few sensors are used to provide inputs, through a feedback process, to a simplified engine model; the controller is based on this model. Restricted onboard computing capacity, insufficient sensor reliability, and insufficient sensor adaptability to harsh environments have dictated this simple control scheme. Advances and anticipated advances in microcomputer technology remove much of this restriction. It has been proven that the use of real-time intelligence (available today with the processing power of microprocessors and data stores) in closed-loop control can be used to improve performance and operability of propulsion system components and subsystems. Propulsion system components and subsystems include, but are not limited to, inlets, fans, compressors, turbines, and nozzles. The theory is that advances in onboard processing capability and sensor technology will enable the incorporation of feedback control within engine components, thus optimizing engine performance and life expectancy.

At this time, research activities are underway identifying the ways to use real-time intelligence for improving propulsion system controls. The specific areas expected to benefit from this identification are compressor stall alleviation, active clearance control, secondary airflow modulation, and active pattern factor control based on blade stresses and temperatures. The enabling technologies involve improving sensor systems, computer systems, actuator systems, and component models. The continuing revolution in microelectronics suggests that control over factors determining fuel consumption will continue to grow. Among the technological elements of this control are adaptive control, nonlinear models, model fidelity, and fault tolerance. This improvement should alter fundamentally the nature of gas turbine components and subsystems from an open-loop to a closed-loop operation, resulting in optimal performance.

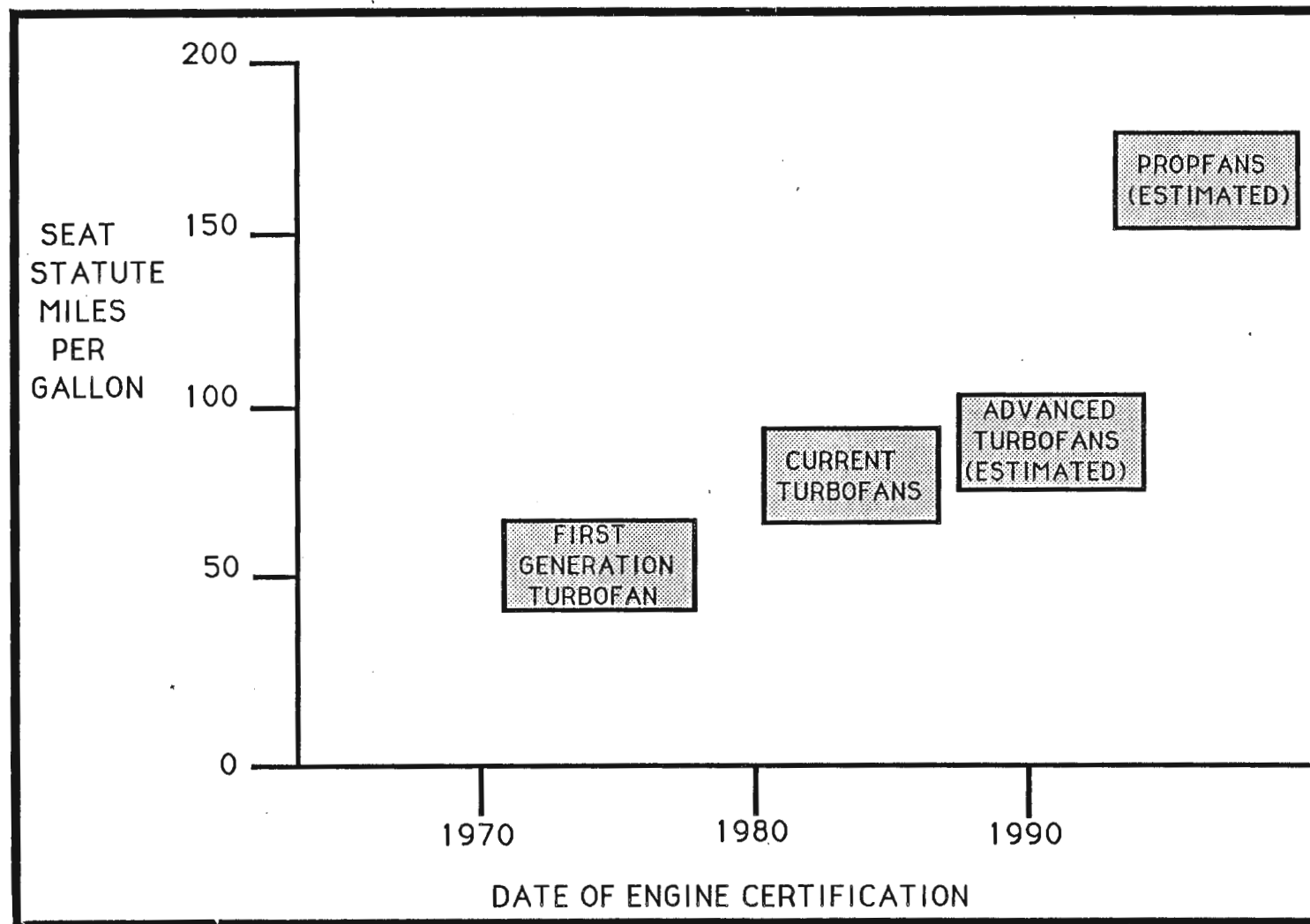


FIGURE 3.4-1. AIRCRAFT PERFORMANCE BY PROPULSION SYSTEM
(Lupinetti, 1987)

3.4.1.2. Artificial Intelligence and Expert Systems

Artificial Intelligence and expert systems were introduced in paragraph 2.4.1 as being an elemental discipline for emerging avionic technologies. AI is a branch of computer science that creates "intelligent" computer systems that understand, learn, reason, and solve problems. Expert systems are systems that use knowledge and construct inferences regarding problems that would otherwise require human expertise; expert systems result from an application of AI. Rapid advances in AI have only been possible because of the increases in computing power and data manipulation enabled by today's digital processors and data stores. Information processing technology has been advancing approximately by a factor of ten every five years; this pace has been maintained for the past thirty-five years. The problem posed for certification of a digital avionic system utilizing AI is similar to that discussed in paragraph 3.4.1.1; the system does not conform to a rigid set of specifications. Its response to a given stimulus is dynamically determined and time dependent. At a different point in the learning process, the system may respond to the same set of stimuli differently than before. Adaptive control, as discussed in paragraph 3.4.4.1, may incorporate learning, but is more likely not to. For this reason AI-based systems' certification rationale must be based more on a demonstration of the manufacturer's design and the integrity of the manufacturer's performance verification techniques, than on conformance to rigid specifications.

Artificial Intelligence, while having been an academic subject for many years, is still in its infancy. The technology for implementing quasi-intelligent subsystems should be in place by the year 2000. At this time it should be mature enough to support its application in aeronautical system products and processes. An expert system is an example of a quasi-intelligent system. Using a set of facts and rules, the expert system is able to reason. Figure 3.4-2 is a schematic depicting the structure of an expert system. Facts represent inputs to a situation interpreter and actor, the "inference engine". The set of facts is referred to as a "situation base". The inference engine also processes rules from a "knowledge base". The result of processing the facts against the rules is an action or conclusion. Facts are inputs derived from particular sensing mechanisms. Rules are complex data structures that define what relationship of inputs (facts) leads to a certain conclusion (action). The inference engine is an interpretive computer program that applies inputs to rules, and initiates an action when a rule is satisfied.

The major difficulty with expert systems is the need for complete specification. In the design of a large system, a designer must provide a complete definition of all permissible states together with a number of state transition diagrams which specify all transitions between states. An expert system uses a complete set of rules that explain how to find all permissible states, and how to determine which transitions are permissible between states. An expert system requires that all knowledge on a system's functioning be gathered and encoded into a knowledge base. The operation of the system is modified by changing the rules that constitute this knowledge base.

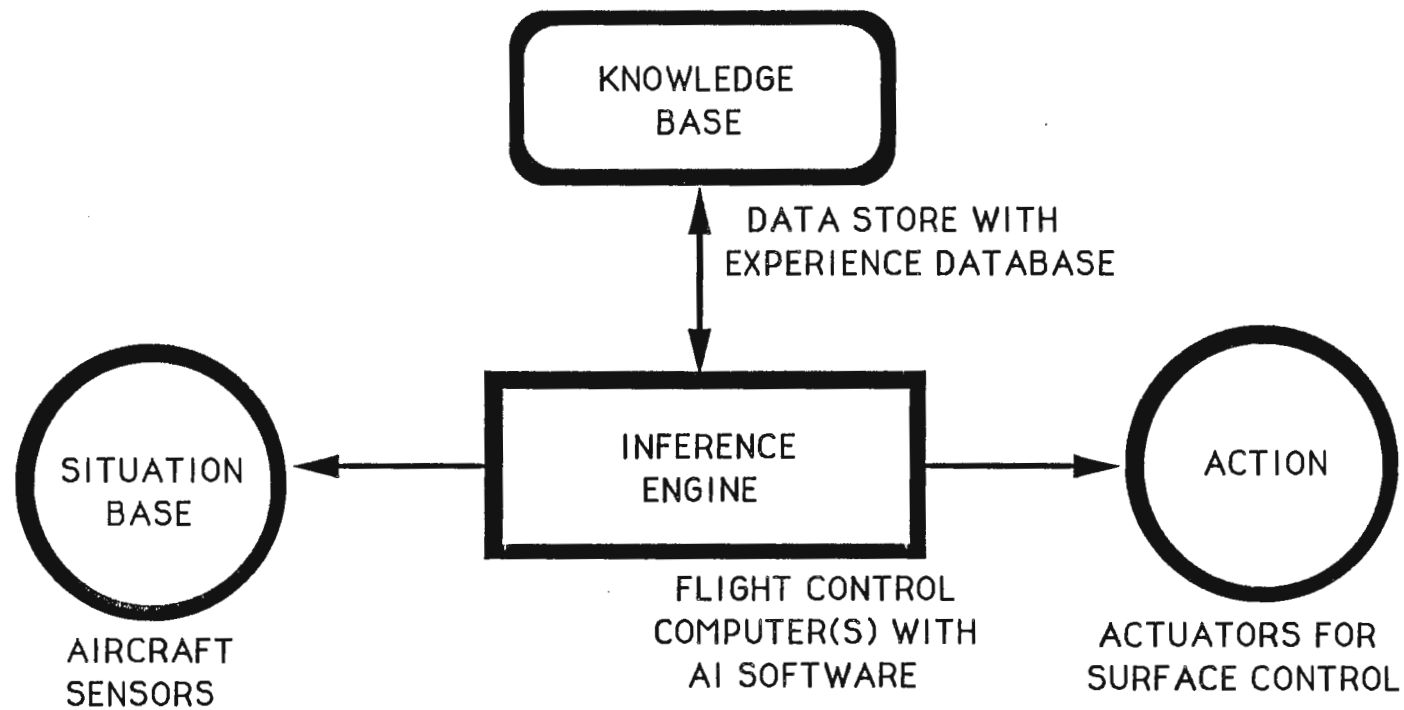


FIGURE 3.4-2. EXPERT SYSTEM

Improvements in this technology are not tied to the inference engine, but to the acquisition and organization of knowledge in the domains that are relevant to the system. The manual acquisition organization and the structuring of the knowledge base are the major bottlenecks in this process. A knowledge base of 1000 rules may require many man-years of effort to develop. By the year 2000 it is anticipated that there will be specialized expert subsystems in common use that will utilize knowledge bases containing several thousand rules. These subsystems will be limited in scope and applied within a distributed, multifunctional airborne environment. While a limited number of such systems exist today, they are restricted to non-time-critical applications. By the year 2000, expert systems with moderately sized knowledge bases will be used in real-time applications.

As with any system, an expert system is only capable of accomplishing what its designer plans. When we say a system "learns", we mean a knowledge base is improved. The difficulty of translating a discipline into a knowledge base is not to be underestimated. In doing this, structure and quantification are being applied to subjects unaccustomed to it. Frequently, these subjects behave in the planned manner for reasons other than those anticipated by the designer.

In addition to expert systems, AI is also an elemental discipline for Theorem Provers, Robotics, Goal Seeking Systems, Vision, Speech Recognition, and Natural Language Understanding. These are expected to result in new technologies that will be beneficial to the aeronautics community.

3.4.2. System Integration

System Integration was introduced in paragraph 2.4.2 as being an elemental discipline for emerging avionics technology. Traditionally, avionic systems have consisted of a collection of individual components, each performing a function independent of other aircraft systems. With the advent of powerful microprocessors, avionic system designers were able to fully integrate navigation, display, and control systems. Integration implies sharing of the data produced by several different sensors and computers. This sharing has the cooperative effect of improving the quality of decisions that can be made by either human or electronic intelligence. When Automatic Control Systems (ACS) are used to stabilize the aircraft, conventional aerodynamic stability requirements can be relaxed. Modern tactical aircraft have low, and even negative, levels of longitudinal static stability; thus, maneuvering capability and cruise performance are enhanced. It is only possible to fly an aircraft with this level of longitudinal static stability because of a full-time, full-authority automatic stability augmentation system that stabilizes the aircraft. Integration also implies the physical consolidation of system functions, which offers its own advantages.

Guidance, Navigation, and Control (GNC) systems will be a forcing function in aircraft design during the coming decades. With today's microprocessors, the designer can exploit the interaction and integration of aerodynamic, structural, and propulsion system controls to provide better, more efficient aircraft.

The advantages of integrated design have been demonstrated in experimental aircraft. The following are two examples:

- Integrated digital control of engine inlet and autopilot systems have resulted in a significant range increase.
- Integration of wing extensions with an active control system have resulted in significant cruise performance improvements.

These are considered to be first generation improvements resulting from system integration. More significant improvements should be realized by the year 2000. The Aeronautical Policy Review Committee (Review of National Aeronautics Policy) states that:

"Systems integration will play an increasingly important, if not dominant role in the development of advanced technology aircraft... advances in the traditional aeronautical disciplines will no longer ensure a superior product."

While hardware advances in the next two decades will provide greater computational capability, improved reliability, and reduced component cost, these will not be the pacing technology developments. The application of a multidisciplinary design approach to these developments is the challenge of research and development.

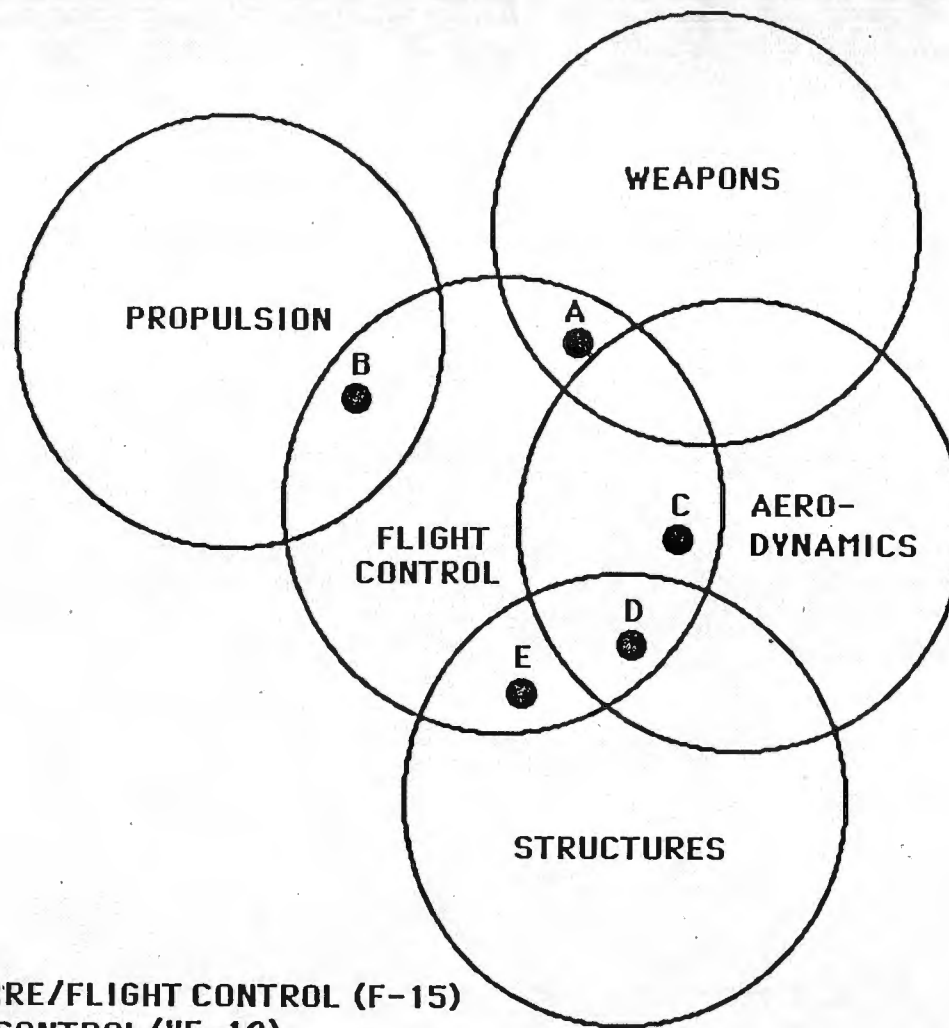
The following is an example of the demands that will be placed on a tactical aerospace vehicle by the year 2010 (Aeronautics Technology Possibilities for 2000: Report of a Workshop, 1984).

"... it will have to deal with massive amounts of information such as: targets, ground and airborne threats, navigation data, route options, weather, weapons, aircraft system status, and commands. This information will have to be collected, stored, processed, correlated, integrated, analyzed, and acted upon in a very short period of time."

The aircraft computer systems must be integrated to accurately process this magnitude of data. Equivalent integration requirements will exist for civil aircraft.

3.4.2.1. Current Efforts in System Integration

As mentioned earlier, system integration is now at the first generation level. Intermediate level integration involves the linking of selected functions or subsystems for the purpose of achieving improved performance and has been accomplished. An example of this is the coupling of the throttle, flight control, and navigation systems to provide automatic flight path control. Figure 3.4-3 depicts the state-of-the-art schematic for system integration above the intermediate level. The five circles represent the five major disciplinary areas of aeronautics. The intersections represent functions that have been integrated as a result of a design decision. The solid circles within intersections represent those integrated functions that have been accomplished in production or research aircraft.



- A INTEGRATED FIRE/FLIGHT CONTROL (F-15)
- B COOPERATIVE CONTROL (YF-12)
- C SUPERMANEUVERABILITY (HIMAT)
- D FORWARD-SWEPT WING (X-29)
- E STRUCTURAL MODE CONTROL (B-1)

FIGURE 3.4-3. SYSTEMS INTEGRATION

The potential advantages of these integration efforts are enormous. The following results are expected:

- An integrated GNC will permit engineers to trade inherent stability for weight and drag reductions.
- An integrated control and structural system will permit engineers to design structures to achieve minimum weight by load control at key points, maximizing load factor capability over the complete dynamic range of the aircraft.

An example of technological advance for the purpose of enabling system integration is the Advanced Fuel Management (AFM) program. This program is under Navy sponsorship; its purpose is to improve aircraft performance by a full integration of the propulsion control and FCS's. A thorough discussion of this program is given in "Moving Closer to Fully Integrated Control" (Brahney, 1986). Engine control technology must be improved to make this system integration possible. The unit that controls the propulsion system must be as reliable as the FCS's. The design guidelines ensure that the control unit continues to operate the engine, without performance degradation, following the failure of any single mechanical or electrical component. Selective redundancy was included on the control unit to achieve reliability. The inclusion of selective redundancy in the control unit was the method by which reliability was to be achieved.

The decision to use selective redundancy as a means of achieving desired reliability levels eliminated "single string" electronics architecture as a possible approach to control unit design. The FADEC and Full Authority Fault Tolerant Electronic Engine Control (FAFTEEC) established that dual redundant system architecture is a viable approach to providing high reliability in controlled propulsion systems. However, neither of these designs was adequate for tactical aircraft of the 21st century. For this reason the AFM program was initiated. According to the design specification, the AFM system must integrate fuel, hydraulic, electrical, and mechanical components into a dual channel, full authority, fault tolerant, engine control system. It is responsible for monitoring engine status, reading appropriate data, and scheduling selected parameters for the purpose of achieving a desired engine performance level. Power Level Angle (PLA), certain speeds, temperatures, and pressures are the primary inputs. The outputs are scheduled parameters that relate to engine efficiency, e.g., gas generation and augmenter fuel flow parameters. Figure 3.4-4 is a schematic of the AFM.

The AFM system is composed of three subsystems: the fuel subsystem, the airflow subsystem, and the electronics subsystem. The major components of the fuel subsystem are flow sensors, fuel pumps, and valving. This subsystem is responsible for pumping and metering the fuel to the gas generator and augmenter. The major components of the airflow subsystem are vane and nozzle actuators. This subsystem is responsible for controlling the variables that affect engine airflow. The major components of the electronics subsystem are engine harnesses, generators, and controllers. This subsystem monitors the fuel and airflow subsystems. Figure 3.4-5 demonstrates how the AFM system interfaces with the FCS and the propulsion system (engine).

3.4.2.2. Barriers to Implementing System Integration

Experience with integration efforts, like the AFM, shows that the most significant barrier to integrated control is the difficulty of incorporating integrated systems in aircraft. Incorporating systems is difficult and very risky due to the uncertainty of the performance of the aerodynamic, structural, propulsion, and environmental systems under normal and failed flight conditions. Additional problems to be considered are implementing safety precautions to ensure reliable and safe operation and operating without concise measures for rating performance levels.

Current technology is not designed to facilitate system integration. There is emphasis on the Input-Output concept of system operation. The system is not designed to process all the different combinations of inputs and outputs, and there is not enough understanding about how the different technologies interface. Systems engineering is attempting to solve this interface problem, but is handicapped by the absence of higher-level design tools, accurate model definitions, and experience in applying tools in real-world situations..

Some specific design needs are as follows:

- Development of measurements for successful/unsuccessful flight systems performance in aircraft that incorporate state-of-the-art technology, in particular, high-authority DFCS.
- Development of design methodology and techniques for designing in an environment of distributed, parallel processor architectures.
- Development of system architectures that provide high degrees of reliability and dynamic reconfigurations for flight-crucial systems. There is a supporting need for the development of test and assessment aids for ensuring that systems developed satisfy the requirements.
- Development of practical optimization algorithms to accurately extract data from large databases. This processing will be in near real-time mode to assist the pilot in flight path prediction and flight decision making. Distributed and parallel computer architectures, and supporting software may be required.
- Development of software validation and verification methods that are less subjective than the methods currently in use and that maximize existing automation technology.
- Development of techniques for task allocation among crew and vehicle systems based on mission requirements and human error theory.
- Development of models for use in multidisciplinary design for highly interactive systems. Specific examples are as follows:
 - Dynamic response of aeroelastically tailored structures.

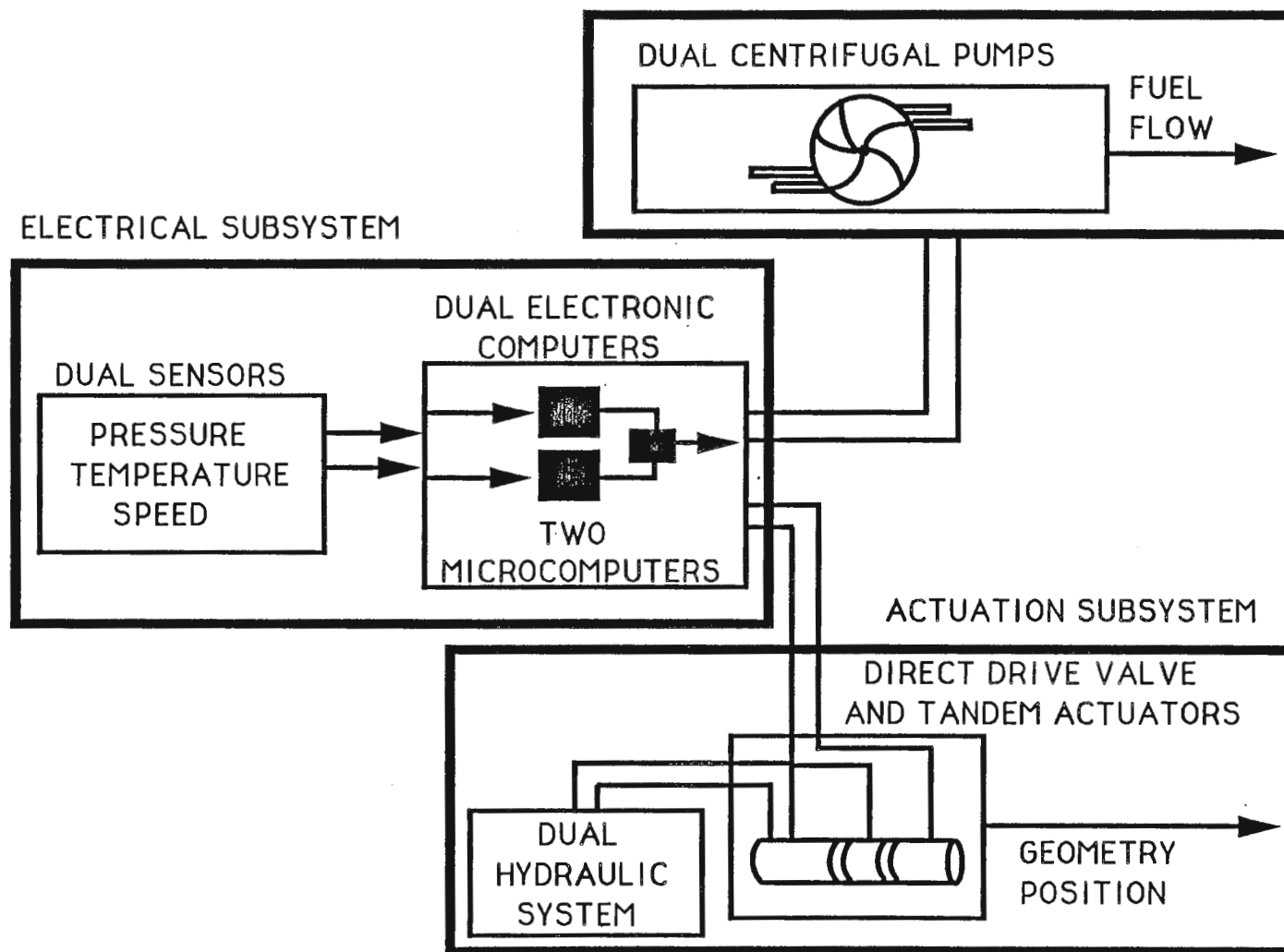


FIGURE 3.4-4. ADVANCED FUEL MANAGEMENT SYSTEM (TOTALLY REDUNDANT)

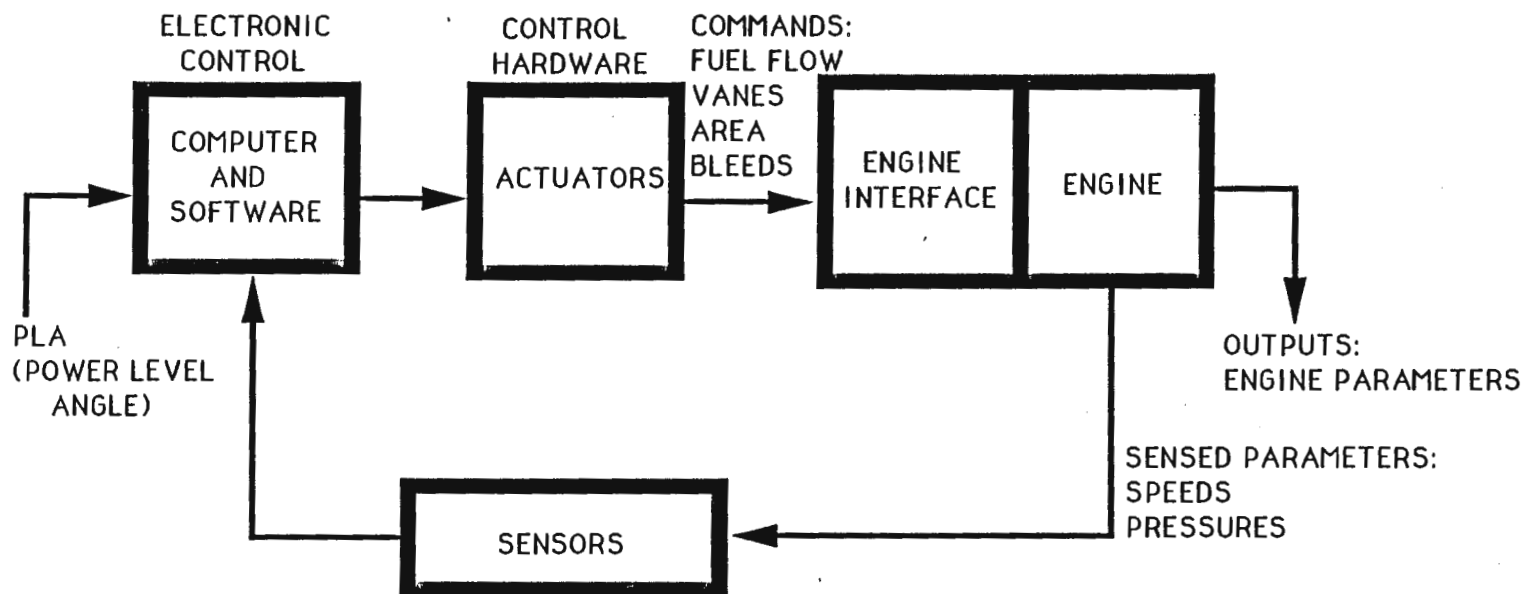


FIGURE 3.4-5. ADVANCED FUEL MANAGEMENT SYSTEM INTERFACES

- Unsteady dynamics at high angle of attack.
- Interaction of engine, inlets, and nozzles.
- Pilot-command/external-disturbance environment for advanced aircraft with state-of-the-art systems.
- Development of computer-assisted design techniques for efficient multi-input and multi-output control for a large number of control parameters. (Practical optimization algorithms for simplifying complex feedback mechanisms are also required.)
- Development of better hardware simulation methodology for building simulator interfaces that permit a more realistic test situation.

3.4.2.3. Advantages to be Gained from System Integration

System integration at the highest level will not be achieved without the development of efficient design and analysis tools. The GNC system will be the impetus for aircraft design integration. The key technology will be the multidisciplinary design of interactive aircraft systems that will meet both civil and military performance requirements for speed, range, maneuverability, navigation, efficiency, and safety. In the case of military aircraft, observability and threat avoidance will also be important. Inherent in this methodology will be the ability to resolve serious design conflicts across supporting technologies. When these multidisciplinary designs are fact, the following benefits are possible:

- Reductions in weight and drag by permitting compensation for unstable aerodynamic, structural, and propulsion conditions using a highly authoritative GNC system. Such factors as maneuver load control, gust load alleviation, active flutter suppression, and relaxed static stability would be involved.
- Providing new mission capability (e.g., low observability or super-maneuverability) by permitting compensation for unstable aerodynamic, structural, and propulsion conditions using a highly authoritative GNC system.
- Increasing capabilities for maneuvering, trimming, and STOL by integrating highly interactive subsystems such as vectored thrust and aerodynamic control.
- Maximizing capacity for military aircraft to conduct complex attack/defense missions by integrating fire, flight, and propulsion system controls.

Aircraft development will benefit from a reduction of costly, time-consuming iterations resulting from adverse interactions between systems and unrealized performance goals. Resultant aircraft systems should require less redesign and refinement over the life of the aircraft. These systems should result in less risk, cost, and time for developing new aircraft.

The multidisciplinary design is viewed as an enabling technology for advanced combat aircraft (fighters, multimission rotorcraft, and VTOL), advanced supersonic transports, hypersonic aircraft, and transatmospheric aircraft. The multidisciplinary design will result in improvements in subsonic aircraft.



4. RELATED VALIDATION ISSUES

4.1. Fiber Optic Technology

Fiber optics is a branch of optics concerned with the propagation of light along thin fibers. Each fiber consists of a core and sheath; the core and sheath are different transparent materials. In practice, more than 100,000 fibers that are a few microns in diameter have been grouped in a single bundle. Light entering one end of a fiber is transmitted the length of the fiber by internal reflections.

Ground-based telephone and data transmission systems, using twisted-pair and coaxial cable technologies, are now being replaced with fiber optic systems. The primary advantages of the fiber optic technology are the greater band-width (approximately 1,000 times higher), and smaller size. The application of fiber optics to aeronautics offers the advantages of lower cable weight, less space required by conduits, and less vulnerability to the nearness of power cables. In aircraft using composite structures, instead of traditional metallic, the advantage is even greater: less vulnerability to high-level RF interference from power cables, lightning, and other electromagnetic noise.

4.1.1. Fly-By-Light

Fly-By-Light is an outgrowth of FBW. In an FBW system, FCCs receive inputs in the form of electronic signals from numerous sources: sensors, avionic systems, pilot control, and maintenance input. In the case of pilot control or maintenance inputs, the FCC may initiate an adjustment to a control surface directly. The control laws for the aircraft exist as a portion of the software for the FCC. Normally, sensor and avionic system inputs are processed through the control law software, and a decision is made that some control surface adjustment is required. The FCC will then send an electronic signal to a control surface or to an actuator controlling a control surface resulting in an adjustment. Usually, heavily shielded coaxial cable is used for the data communications between the FCC, its input sources, and its output destinations. In an FBL system, an optical data bus or bundles of light-sensitive fibers would replace the coaxial cable. Figure 4.1-1 is a schematic of a typical FBL system.

Figure 4.1-1 is a simplified schematic of a system utilizing a single multi-processor control computer with a single channel. In reality there would be multiple systems and channels configured in a redundant architecture. The Primary Control System (PCS) consists of pilot controls, a processor, and actuators. The processor has an electrical power source. The communication between the processor and the pilot controls, and the processor and the actuators is by optical fiber bundles. The PCS is a flight-critical system

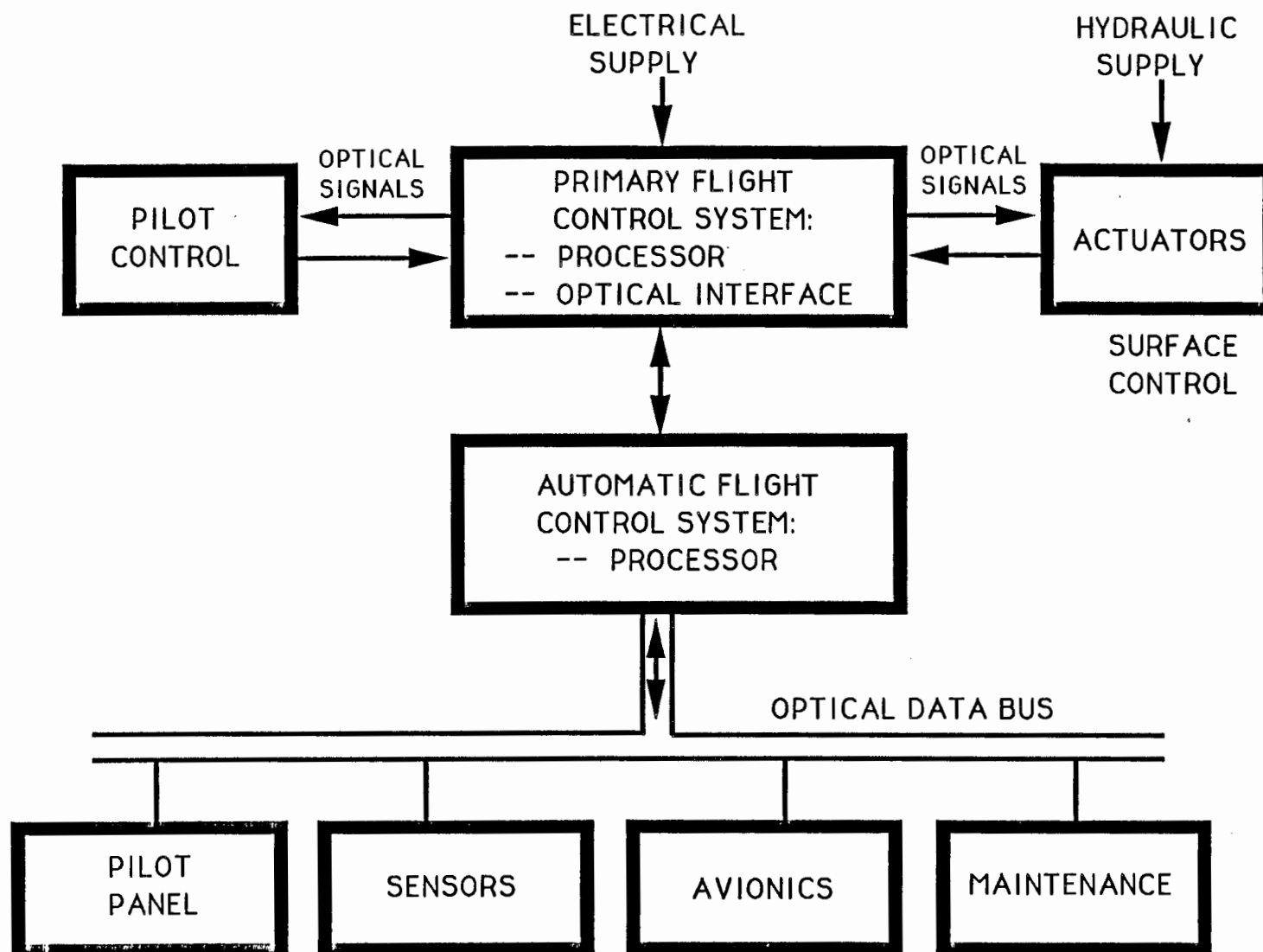


FIGURE 4.1-1. FLY-BY-LIGHT SCHEMATIC

and would therefore be heavily redundant. Flight safety reliability is required of the PCS. The actuators drive control surfaces, so they would most likely require a hydraulic supply.

The ACS consists of a processor, optical data bus, and the following interfacing systems: sensors, other avionic systems, pilot malfunction, and maintenance. The criticality of this system varies. If the aircraft has a high pilot workload or has a high degree of inherent instability, this system may be quite critical. If it is critical, it would involve a high degree of fault tolerance and redundancy. The ACS receives all necessary sensor data through an optical interface with the mission equipment bus in highly augmented/automated/integrated craft. The ACS software is mission-oriented and performs such functions as control law processing and selection of automatic modes. The ACS processor does not communicate with control surface actuators but accomplishes that purpose through the PCS processor.

A major function in a system similar to that defined in figure 4.1-1 is the conversion between optical, electrical, and hydraulic energy forms. An Electronic Interface Unit (EIU) would most likely be used to (1) interface with the processor through a Transistor-Transistor Logic (TTL) compatible buffer, (2) house a transmitter that performs electrical-to-optical conversions, (3) house a receiver that performs optical-to-electrical conversions, and (4) display diagnostic data. A fiber interconnect cable would connect the EIU to a transducer. A digital optical rotary position transducer may be used for pilot input. A digital optical linear position transducer may be used for actuator feedback. Digital optical differential hydraulic pressure transducers may be used to feedback differential pressure between cylinders of a hydraulic actuator or to prevent the force flight between cylinders with different pressures. Controlling an actuator with an optical command may require the conversion of the incoming command to an electrical signal; this signal may then control a conventional electrohydraulic servovalve. A solution to this problem may involve a hydraulic-oriented conversion or a photoelectric conversion. The photoelectric conversion, while seeming obvious, would probably require too much input power. A hydraulic-oriented conversion is probably most practical. This approach does not introduce the need for a new energy form since hydraulic power is already available in the actuator. A turbine-driven alternator or a piezoelectric crystal stack using hydraulic pressure for pulsing the stack ends are state-of-the-art approaches to this conversion.

4.1.2. Fiber Optic Sensors

Figure 4.1-2 is an application of fiber optic technology to a wing structure. This wing may contain a large number of sensors that would be subgrouped. Each subgroup would contain a number of binary coded sensors such as code plates, switches, and shaft speed sensors. The sensors of each subgroup would be multiplexed via a singlemode optical fiber to an optical receiver. The receiver would be a component of an EIU which would convert the binary encoded data to an electronic signal and buffer the data to a processor.

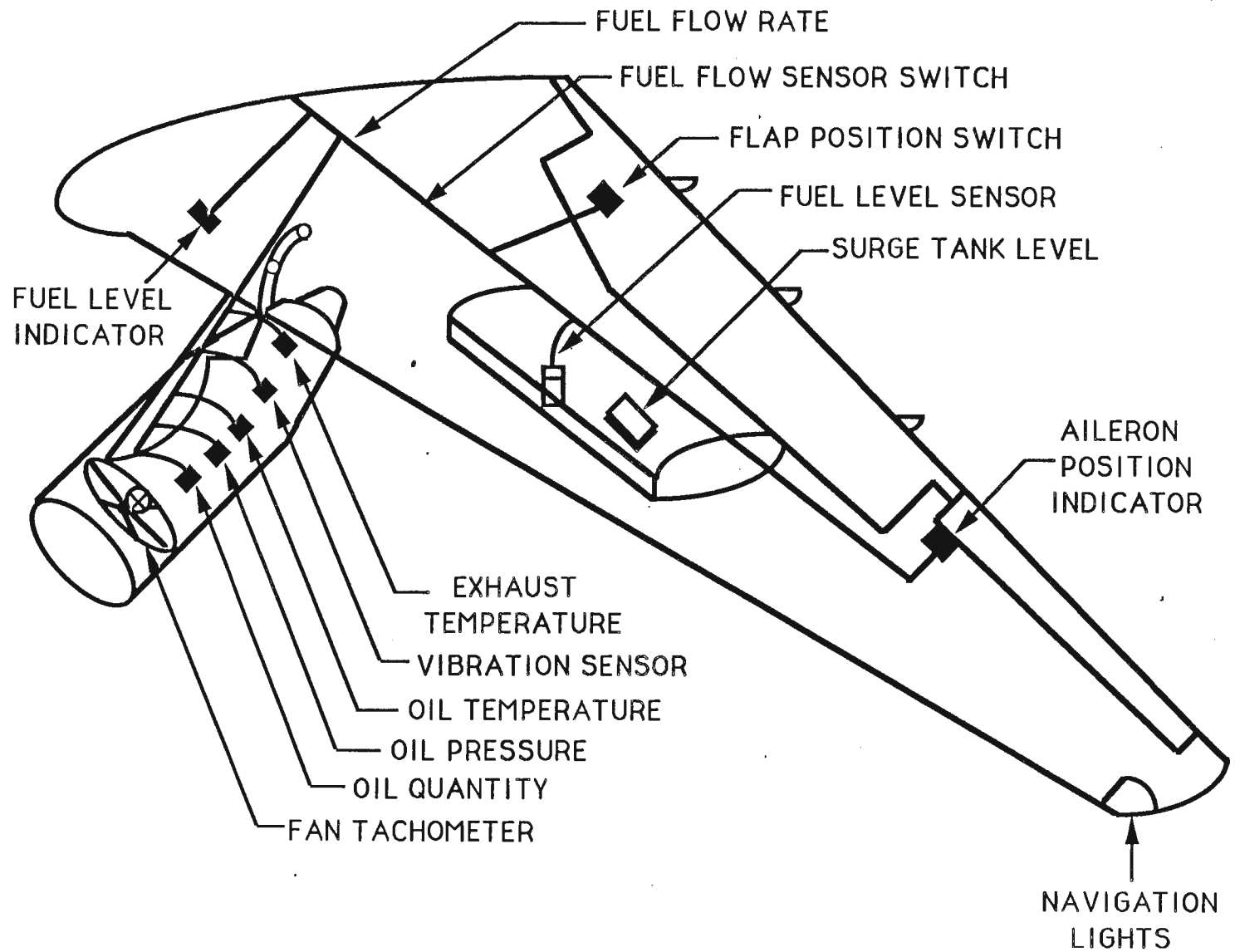


FIGURE 4.1-2. FIBER OPTICS IN WING ASSEMBLY

Due to the relatively large size of fiber optic connectors and the large number of sensors, it is necessary to multiplex data gathered from several sensors over a single fiber. A transducer using multimode fiber optic delay lines with a pulsed laser diode is frequently used as a multiplexor. This multiplexing scheme represents a traditional time division multiplex system. An advantage of this system is the use of a serial binary readout. Disadvantages are the system data losses and delay intervals inherent in time based multiplexing. Research on this type of multiplexor is documented (Farina, Hubbard, and Lefkowitz, 1983). Proposals have been made for a multiplexor utilizing coherence multiplexing. With this approach an encoder is used to read the position of an optical code plate or the switches in a parallel binary format. Each channel is delayed with respect to the others by use of fiber optic delay lines and the data combined onto a single fiber. A phase-carrier, coherent detection scheme with a limited coherence laser is used to multiplex the information onto a single fiber. The advantages of the coherence approach are shorter delay lines and reduced mode system losses. Research on coherence multiplexing is documented (Glomb, 1986).

Fiber optic sensors are increasingly being considered in aeronautical design. Fiber optic length monitors can measure strains (elongations and compression) from 0.1 micrometers to 1.0 millimeters. Fiber optic acoustic area monitors can detect structural damage. These developments compensate for some of the difficulties anticipated as composite materials are increasingly used in aircraft structures. Some of the research and development that justifies the use of fiber optic sensors in aeronautical structures is documented (Otaguro, Michal, and Watanabe, 1986).

Quality control for brittle materials has used proof testing methods and analytical formalism as techniques for validation. These techniques involve subjecting the structure to applied loads much larger than expected service loads for short time periods. Measurements of flaw size can be used to estimate structural life under related conditions. These techniques assume that in-service stress distribution conditions are identical to proof testing conditions. No allowance is made for in-service wear, damage, or age. Safety considerations are factored in by the process of structural over-design. Fiber optic sensors are essentially glass or plastic fiber waveguides with small encircling spools. These sensors can be used to detect low amplitude acoustical signals. While several techniques exist for detecting the occurrence of acoustical signals in the structures containing the sensors, the techniques rely on two basic concepts:

- Measuring the resulting cyclic change in velocity of the optical signals in the fibers of the sensor (interferometric method).
- Measuring the change in propagation intensity in the fiber due to acoustic signal mode mixing.

The interferometric method is used in passive sonar device technology, i.e., submarine detection. Similarly, the interferometric method can be adapted for detection of acoustic emission from strained composite structures.

The acoustic sensor system would be designed as a network of sensors and built into the aircraft's composite structure. The glass fibers would be glued to the internal surfaces of the composite structures. The networks would be connected by fiber communication links to a processor. The networks would be the vehicle for sending the sensor signals to the processor. Time delay spectroscopy could identify the source of the signal to the processor; acoustic emission analysis would determine the development of flaws. A real-time evaluation of structural integrity would result from this system. The composite structure could be integrated with this "listening blanket" during proof-testing, and the proof-testing period could be extensive. The purpose of this would be to determine the suitability of the composite structure before major assembly. In addition to stress determination, fiber optic sensors can also be used, within certain constraints, as temperature and chemical detectors.

This basic technology is at hand. Fiber optic Sagnac and Mach-Zehnder interferometric sensors are being used to monitor strain. The integration of sensor development, acoustic emission analysis, and computerized evaluation of brittle material integrity is missing. With proper emphasis, an integration of control and structural technologies within the above guidelines could be accomplished early in the coming century.

4.2. Artificial Intelligence

Artificial Intelligence has been discussed earlier in this tutorial. In paragraph 2.4.1, AI was identified as one of the two elemental disciplines for developing avionic technologies. In paragraph 3.4.1.2 it was discussed as a catalyst for review of current aircraft certification procedures and regulations. AI is that branch of computer science which attempts to create intelligent computer systems that understand, reason, learn, and solve problems (Buckanin, 1984). AI can casually be categorized into three areas: expert systems, image processing and interpretation, and natural language communication. In paragraph 3.4.1.2, the expert system category was defined and examples were given. In this section emphasis will be on the other two categories. Voice Command Processing (VCP) is an example of the natural language communication aspect of AI. Machine Vision (MV) (See paragraph 4.2.2.1 for definition) is an example of the image processing and interpretation category. Both of these examples have potential application in tomorrow's aircraft. VCP is considered to have a wide range of application; MV is more restricted.

4.2.1. Voice Command Processing

VCP will have a major impact on how the pilot of the future commands and controls his aircraft. However, the integration of this technology with a future cockpit poses many difficulties, both in the hardware and system interface areas.

4.2.1.1. The Need for Voice Command Processing

As the complexity of avionic systems increases, so does the workload on the pilot. This is a concern to both the civilian and military aspects of aerospace. The concern is most acute for today's high performance military aircraft that must perform single-seat tactical missions. The hands busy/eyes busy

mission pushes the limits for both pilot and aircraft. VCP is part of the solution to this problem; it should reduce the button and switch effort significantly. Developments in recognizer/synthesizer hardware and in recognition algorithms have made this technology one that should provide near-term benefits.

Boeing's entry in the next generation of subsonic transports, the Boeing 7J7, will utilize a voice recognition system. The USAF is testing voice activated electronics for its C-135C transport and its advanced F-16 fighter. These systems will respond to inquiries regarding fuel level, weapons status, and navigation; they will be responsive to numerous commands, e.g., changing radio channels and target proximity. There are over two-hundred basic information arrangements (pages) in the F-16C; each of these has numerous permutations and combinations. ATF will have hundreds of pages. If a conventional display approach is taken in the ATF, the pilot-aircraft interface will be impossible. Speech and hearing are two highly efficient forms of communicating. Systems oriented to their use must be developed for command and control.

4.2.1.2. The Problems for Voice Command Processing

Speech recognition is the process of correctly translating human speech into digital data. Speech synthesis is the reverse procedure. The latter is a mature technology: digital images of known responses can be stored and used when needed. Speech recognition is less developed. The sound of a given word may vary for the following reasons:

- Voice quality and attributes of the speaker.
- Context in which word is used.
- Inconsistent separators between words.

For a pilot, the sound of a given word may also vary for these reasons:

- Stressful conditions.
- Noise of aircraft.
- Presence of an oxygen mask.
- Altered g forces.

This difficulty is minimized by (1) requiring that speech be in isolated words, (2) limiting the acceptable vocabulary, and (3) programming the computer to recognize a single user. The imposition of the constraint that only isolated words be used will be accomplished with the definition of the command language. The pilot must be thoroughly trained in this language, and some type of video prompt should be present in the cockpit. The acceptable vocabulary is being limited. It is estimated that a vocabulary of between 200 and 500 words will suffice for even the most sophisticated systems. Programming the computer to recognize the pilot's pronunciations will be discussed later.

4.2.1.3. A Procedure for Voice Recognition

Figure 4.2-1 is a schematic for the voice recognition process. The audio input drives this process. There are three steps in voice recognition: extracting words out of the audio input, pattern recognition, and context analysis. The first step is that of extracting words out of the audio input. This consists of (1) converting the audio input to a spectrum and (2) extracting the spectral data. The conversion is normally handled by a Texas Instruments special purpose chip TMS320. This chip is a very fast 32-bit microprocessor that develops an energy distribution of the spoken word(s) by time. The extraction may be handled by such techniques as linear predictive coding, fast Fourier transforms, or digital spectral analysis. Linear predictive coding performs a polynomial fit on the sound spectrum; fast Fourier transforms develop frequency spectrums; and digital spectral analysis results in a series of attributes that characterize the primary frequencies of speech during each segment of time.

The second step is pattern recognition. This step utilizes the output of the first step and a library of relevant words. Depending on the technique used in the first step, one of the following will be produced:

- A set of coefficients representing the waveform.
- A frequency distribution of the waveform.
- A set of attributes of the waveform.

The library of relevant words is encoded in the same format as the output of step one. Assuming that digital spectral analysis is the extraction technique, the output of the first step may consist of attributes such as, the frequencies of each major frequency band or peak in the spectrum. These frequencies are referred to as formants. The library would contain the formants for different relevant words; there would probably be multiple formants per relevant word. Each formant is used as a "template" and is passed over the incoming sound message (output from the first step). The analysis continues until as much as possible of the incoming sound message is identified.

The third step is context analysis. Once the identification of individual words or phrases, is complete, the intent of the entire message is determined. This may be accomplished by graph analysis or encoded tables. The context analysis may be very efficient and permit only known words, or maybe very user-friendly and permit unknown words. Some known words may even be optional. The output of this step is the recognized pilot command. While not a function of voice recognition, the next step is interpretive logic that will obey the command.

The library of relevant words is developed during the programming process. This is a special mode of operation to the VCP. In this mode the pilot will identify a word to the VCP. The pilot will then pronounce the word under different circumstances. The system will then be tested under actual flight conditions. If there are pilot messages that the system does not recognize, the training mode will be reentered in an attempt to reproduce these conditions. Usually, a number of iterations through the training and actual flight cycle will be

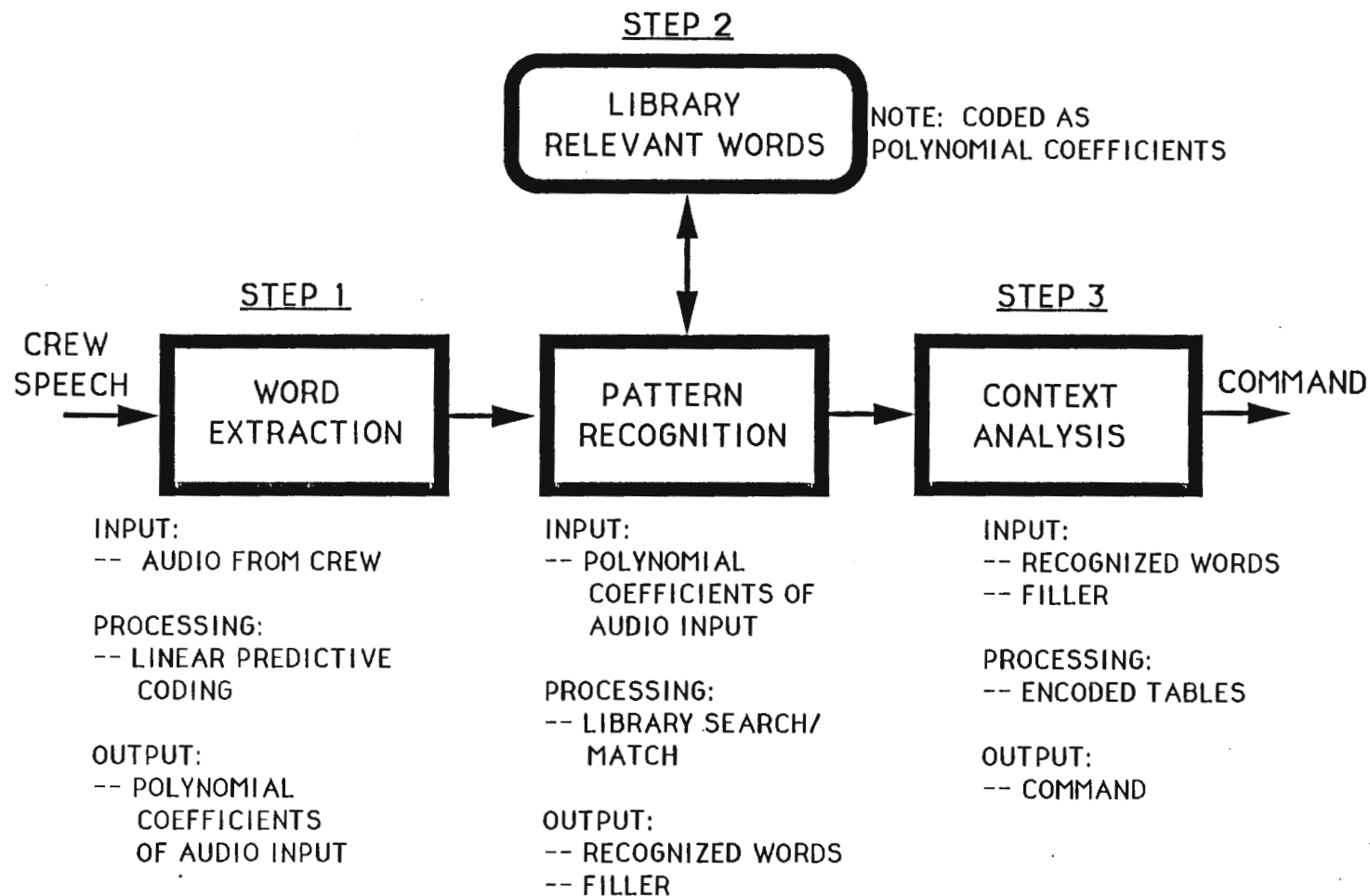


FIGURE 4.2-1. VOICE RECOGNITION SYSTEM

required. It is possible to have multiple libraries for a given aircraft, one for each potential pilot.

4.2.2. How Machine Vision is Accomplished

Machine Vision is used in the early stage (analysis stage) of research and development. It commences with the digitizing of the subject's image. An area is scanned by conventional or structured light beams. A charge-coupled camera or TV camera converts the image of the area being scanned into an array of picture elements (pixels). The intensity of light at each pixel is encoded as a binary number; this number represents a tone on a gray scale. Following the scanning and encoding, the array of pixels must be analyzed; it is at this point that most procedures for vision processing differ.

If the application is simple, such as the inspection of pieces, the pixel array is organized into dark and light segments. This process is called "thresholding". From this process, edges will be determined and shapes outlined. A library of standard shapes is used to determine the identity of the subject scanned. If microcircuits are being inspected, the library of shapes will be oriented to connections between circuit lines. If the scanned subject has edges that produce another topology, it may have flaws or breaks.

The analysis stage may be very simple; repetitive calculations on pixels or groups of pixels. Computers with simple Von Neuman architectures may find this an exhaustive job when the pixel density is high and/or the area scanned is large. This is an ideal application for parallel processors. Vision processing will encourage the development of parallel processing computer architectures.

Inspection is a simple example of MV. The hardware would consist of the camera system and a processor. Robotic vision, while having more hardware, is not necessarily more complex. A common application is locating an object that the robot will act on. For example, the vision system may find a particular spot on a circuit board where the robot is to insert a component. In this case, simple silhouettes may be sufficient for a program to identify a part by area, length, number of holes, or other features.

Surface curvature poses a problem to vision systems. Curvature may cause variances in reflection that are normal; this makes identification more difficult. An algorithm has been developed to handle this problem. The variations in light intensity are determined with scans moving in different directions. Reflection intensities resulting from surface curvature will vary from scan to scan; reflection intensities from flaws will not vary. When robotic vision requires great precision, 3-D vision may be required. A procedure used for 3-D vision is to utilize a single camera for illumination of the scanning beam and an additional camera(s) for a triangulation at the scanning spot. Robots have drilled holes in parts to a precision of 0.005 inches using this technique.

4.2.2.1. Machine Vision in Aircraft

Because the flight environment for civil aircraft is reasonably structured, these aircraft have ways of "seeing" that do not conform to traditional concepts

of vision. For tactical military aircraft, the flight environment is not so structured. Most of the aeronautical research and development in MV is for military aircraft. The DOD is currently studying the problem of applying MV to target recognition and navigation. One project calls for the recognition of a target by an aircraft or incoming missile; another project calls for the cross-country navigation of a high speed vehicle. Success in research as ambitious as this may be more than a decade away. On a simpler level, MV is successfully being used in factories to ensure the quality of manufactured electronics components. It is also being utilized to improve the quality of robotic welding, drilling, and cutting.

4.2.2.2. Two Complex Applications of Machine Vision

Target recognition is an advanced application for MV. The major problem is that of distinguishing between a tank and a truck when either vehicle may be viewed from any perspective (angle). The fact that the background for the subject being identified is undefined complicates the problem. Pattern recognition algorithms are being applied to this problem. In general, pattern recognition is to have a predefined image of the subject(s) for which identification is desired. This predefined image is then superimposed throughout the scanned area to determine whether or not it is present. For a single item, it would be necessary to superimpose the images from all perspectives throughout the scanned area. Other difficulties result from the following:

- The predefined images will vary as the elevation and distance of the scanning camera changes.
- The basic subject may not be fixed; e.g., the tank may have its gun pointed in any direction or the truck may be loaded/unloaded.

Synthetic discriminant algorithms are also being considered for target recognition. With this approach, a subject is defined to be a combination of a number of abstract geometric objects. The tank may be identified by the treads. This alleviates the problem of concern for orientation, but intensifies the problem of isolating the object from the background.

Another complex application is ground vehicle navigation. In this research, a vehicle with MV is to be driven over a winding road. The processing commences with a gray scale image of the road. Edges in the image that were parallel and approximately the same distance apart as the preceding image were taken as the road edges. Shadows and trees pose major problems. Color cameras reduce this problem as light intensity is used as a parameter for isolating the road from the remainder of the scanned area.

Obstacle avoidance is another facet of the navigation problem. Since this involves a distance dimension, a 3-D approach using lasers, sonar, or camera triangulation is being attempted. The success of this effort varies as to the ability of the obstacle to reflect sound. In addition to using a single detector (laser, sonar, or camera), research is underway to utilize all three sensors and have a master program compare the data submitted by the processor controlling each sensor (vote-taking). This data is coordinated with background data (images supplied by a satellite) to create the final computer image.

Progress in both target recognition and navigation is slow; however, MV is being used in manufacturing and domestic applications.

4.3. Computer Technology

Aircraft have become increasingly dependent on computers for both performance and safety improvements. This technology has grown by a factor of 10 every 5 years for the past 35 years; continued growth is anticipated. This rapid growth has primarily been the result of mammoth breakthroughs in device technology: vacuum tube to transistors to integrated circuits. Numerical software has also been greatly improved.

4.3.1. Advances in Microelectronic and Component Technologies

Since advances in these technologies have been a root cause of the evolution of today's digital avionics hardware, this subject has been alluded to in earlier sections of this tutorial. By the early 1990's it is anticipated that microelectronics will achieve today's recognized limits for Very-Large-Scale Integrated Circuits (VLSIC). Currently, Field Effect Transistor (FET) geometry cannot be scaled below approximately 0.25 micron. The reduction in the feature size of microelectronic devices will level off before reaching this limit. New techniques must be developed. As mentioned in section 2.4.5, 3-D architectures are a potential solution. These architectures would permit vertical as well as horizontal element interconnections. If problems such as heat removal can be solved, this technology may help sustain the reduction in feature size. An alternative to vertical interconnections is a reduction in the number of interconnections. This can be accomplished by developing modular digital function devices that accomplish more complex functions. More complex function devices is a second potential solution. A third possible solution is a new digital integrated circuit technology that uses submicron structures and new physical principles to solve the interconnect problem.

On the component level, rapid advances in processing and storage device technology are expected to continue. Logic and memory component density is expected to continue to increase by a factor of ten every five years for the next decade. As feature size converges to the 0.25 micron range, manufacturing chip cost will level off and, perhaps, increase. The cost per unit capability should continue to decline because of the increased functionality of the chip. Mass storage density has increased by a factor of four every five years. Current optical storage technology increases storage by a factor of 10 over current magnetic storage per unit volume. The utilization of video storage is also being studied for its adaptability to digital systems.

The "splitting" problem of optical bus systems is being researched. Optical bus systems that support in excess of 64 taps per kilometer are envisioned. In paragraph 4.1.2 of this tutorial, frequency division multiplexing techniques for optical fibers (permitting up to 50 channels per optical fiber) were discussed. Each channel has a bandwidth of 0.5 to 1 gigabit per second.

Single devices housing solid-state sensors, processors, and memory are anticipated within the next decade. As mentioned earlier, new chip-level and

wafer-level interconnection technologies will be used to alleviate the component interconnection problem. Holography is an example of a new interconnection technology.

4.3.2. Advances in Parallel Processor Organization

The traditional processor architecture, the von Neumann architecture, is characterized by sequential operation. The processor's cycle time is the limiting factor on performance. The cycle time is bounded by the propagation time of signals. The propagation time of signals inhibits clock periods shorter than one nanosecond. Theoretically, the maximum achievable performance is one billion (1.0^{+9}) instructions per second. A parallel architecture is required if this limit is to be significantly passed. By the year 2000, effective parallel architectures consisting of thousands of processors, high bandwidth communications, and parallel structured software is anticipated.

Parallel processing is simple. In sequential processing, all data is accessed from a central data store. A processor that is working on several applications will spend most of its time retrieving data instead of calculating. In paragraph 4.2.2.1 of this tutorial, MV was seen to be an application that consisted of many disjoint tasks. This application is very conducive to implementation in a parallel environment. Each task would be performed on a subprocessor with its own memory.

If a microprocessor can perform thousands or tens of thousands of operations per second, a parallel machine with thousands of microprocessors could perform millions or tens of millions of operations per second. If the microprocessor costs \$50, a parallel architecture of 1000 microprocessors would cost \$50,000. A supercomputer with similar capacity would cost \$10 million.

There are three major difficult steps to implementing parallel processor architectures. The first step is to develop the capability for a meaningful, efficient communication between many processors. This includes the responsibility of controlling several processors working on different tasks on the same problem. The second step is to simplify programming in a parallel environment. The third step is to develop parallel-oriented solutions to problems. The following two approaches for resolving the difficulties of these steps are being studied:

- Utilizing a single instruction/multiple data stream architecture (one sequence of instructions carried out in parallel on data).
- Utilizing a multiple instruction/multiple data stream architecture (many sequences of instructions carried out in parallel on data).

The multiple instruction/multiple data stream architecture is the purest form of parallel processing but offers the greatest challenge. Within these two broad groupings, parallel architectures can be subgrouped by grain size, degree of connection, shared memory, and the technique for decomposing a problem into subproblems that may be solved in parallel. Grain size is defined to be the degree of parallelism. An architecture with a few large processors (up to one hundred) has a large grain. An architecture with thousands of small processors

has a small grain. The degree of connection relates to the interconnectivity of the processors. An architecture that has all processors communicating with all other processors has a high degree of connection. If processors can only communicate with a small, restricted set of processors, a low degree of connection exists. The shared memory attribute relates to whether all processors share a single memory store or whether each processor has its own memory store. The problem decomposition attribute relates to whether the machine decomposes the problem into tasks that may be executed in parallel or whether a programmer must do this.

NASA's Massively Parallel Processor (MPP) is an example of the first generation of parallel architectures. The MPP was developed by Goodyear Aero-space. The MPP is a small grain machine: 16,268 processors arranged in a 128 X 128 square. It has a low degree of connectivity. Each processor is connected to its four nearest neighbors. Every processor has its own 1,024 bits of memory. Problem decomposition is the responsibility of a programmer. A processor consists of a logic unit, an arithmetic unit, 35 shift registers, and the memory. A processor works on a single bit of data per executed instruction. All processors simultaneously could process 16,000 bits of data per cycle; a conventional mainframe processes 64 bits per cycle. Since each processor can execute 10 million instructions per second, the MPP maximum computing rate is 200 Million floating-point operations per second (Mflops). This rate is comparable to the Cray supercomputer.

The MPP is primarily performing signal and image processing. These applications are conducive to the nearest-neighbor connection scheme. The MPP outperformed the Cray by a factor of five. In the area of physics, plasma and gravitational field simulations are being attempted. These simulations are not as conducive to the nearest-neighbor connection scheme as are aerodynamic simulations such as image and signal processing.

NASA-Ames and Yale University have received two iPSC parallel machines. The iPSC was developed by Intel. The iPSC contains 128 processors, and would, therefore, be classified as a large grain machine. Each processor has over 512 kilobytes of memory. A hypercube design was used that interconnects each processor to seven neighbors. Thirty-two and sixty-four processor models of the iPSC permit each processor to connect with five and six neighbors, respectively. Problem decomposition is again the responsibility of a programmer.

Programming a parallel processing computer is very difficult. The four approaches being studied as potential solutions to this problem are as follows:

- Developing functional programming languages that automatically translate into a parallel algorithm.
- Developing process definition languages that permit a machine to dynamically allocate resources.
- Retrofitting parallel-oriented statements into a traditional algorithmic language like FORTRAN.

- Developing tools that permit a programmer to allocate resources in a simple manner.

A dataflow approach is the partitioning of a problem into a sequence of serial and parallel tasks which may be performed by separate processors. Dataflow processors are dedicated processors that allocate these tasks to their subprocessors. A program incorporates certain statements that define the route that data must follow and the transformations that occur to that data enroute. Using this program, the dataflow processor then assigns work to sets of processors. A nondataflow processor will only perform the work assigned to it.

4.3.3. Advances in Distributed Processor Organization

Distributed processor organization involves the arrangement of multiple processors into a network interconnected by some telecommunications medium. The growth of distributed computer system architectures has paralleled the growth in microelectronic technology. The result of microelectronic technological advances increased the importance of small computers. A network of small computers with a distributed workload could perform certain applications more reliably, efficiently, and economically than a single large computer with point-to-point communication to terminal devices. FCS's can be developed using distributed architectures. Paragraph 4.1.1 of this tutorial discusses the components of a FCS. The following discussion uses the FCS as an example of a distributed computer system. The architecture presented here is based upon the Bendix Multicomputer Architecture for Fault Tolerance (MAFT). "Fault-Tolerance in Distributed Digital FBW FCSs" (Gluch and Paul, 1986) covers this subject in great detail.

Figure 4.3-1 is a schematic of a FCS. From this schematic it is obvious that multiple processors are interconnected by a telecommunications medium, a trunk, or bus. The Primary Flight Computer, Secondary Flight Computer, Actuator Control Electronics, Data Sensor Units, and Autoflight Computer are all intelligent devices. They are all microprocessor-based units including memory, data store, and software. The bus may be electrical or fiber optic. Competition for the bus between intelligent units must be regulated by a protocol. A Carrier Sensed Multiple Access (CSMA) or token passing protocol may be used. With the CSMA, each of the nodes on the bus has a guaranteed window within a predetermined Access Time Interval (ATI). A node is a connection from an intelligent unit to the bus. In order for the FCS to perform adequately, the ATI must be small, perhaps on the order of 10 milliseconds. A token passing protocol requires that a token constantly circulate on the bus; only the node with the token may transmit.

Three types of data sensor units represent nodes on the bus. The pilot interface unit receives analog inputs from pilot controls, converts these to digital signals, constructs the appropriate communication layers, and places the resultant message on the bus during the allotted ATI. The appropriate actuator control electronics unit will take the message off the bus and act on it. Other nodes, such as the primary and secondary flight control computers, may also receive copies of this message, but for other purposes. An appropriate response may be returned to the originator and/or other interested parties by

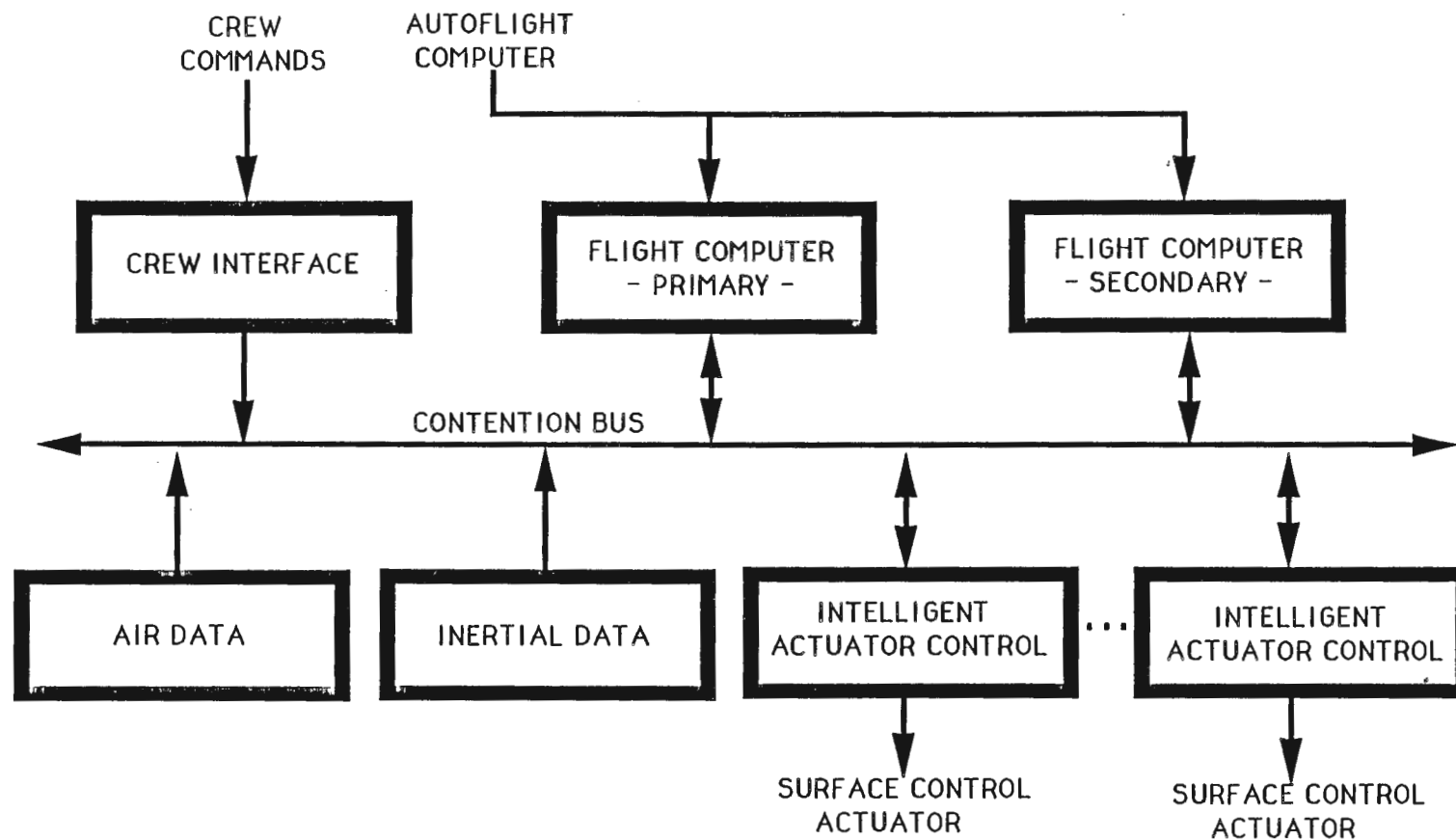


FIGURE 4.3-1. DISTRIBUTED PROCESSOR FLIGHT CONTROL SYSTEM

any recipient. In a similar manner, air data and inertial data sensors will broadcast data across the bus during their ATI. This data will be received by the autoflight computer, the primary flight computer, and the actuator control electronics units. The autoflight computer is responsible for determining whether flight sensors are recording flight parameters that meet projections. If not, surface control commands may be issued to correct the situation.

Actuation control of the surfaces and interface to the bus is accomplished by the actuator control electronics units. These intelligent units may interface to either semi-electromechanical or hydraulic actuation systems. These units perform closed loop control, low frequency equalization, fault detection, and reconfiguration services for each of the control surfaces.

The architecture depicted here represents a highly distributed approach to the implementation of a digital FBW system. Intelligent processing units located throughout the aircraft work collectively through communications provided by a multi-access bidirectional bus. While the overall system functions are distributed, the hardware and software of the primary flight computer is the central ultra-reliable LRU in the system. Reliability concerns dominate the system. The probability of system failure must be virtually zero. Multiple hardware redundancy is employed. This redundancy involves multiple communication busses, sensors, flight control processors, and actuators. Additionally, there will be redundancy in all software modules. All intelligent systems will incorporate fault-tolerant logic.

4.4. Advances in Control and Display Technologies

As the number and complexity of avionic systems grow, it becomes increasingly difficult for the aircraft crew to assume all past responsibilities, as well as the workload resulting from the new systems. With computer-driven surface controls providing stable flight in inherently aerodynamically unstable aircraft, the pilot is no longer the operator of the aircraft, but is rapidly becoming a flight monitor/manager who will intervene in the event of system failure. There are two directions that future systems may take. First, systems may be designed that would augment the operator's information acquisition and processing capabilities. Second, the systems may be designed to eliminate the need for operator intervention on the flight deck.

4.4.1. Improved Man-Machine Interface

The explosion in numbers of avionic systems has resulted in extremely large volumes of data being present on the aircraft computers. The crew may not view all of this data; however, sufficient data to permit the crew's full awareness of the aircraft's situation is important. Information from onboard sensors, stored knowledge, and observation/interaction of the crew must be digested and presented in a manner that is not overwhelming. The crew must focus on the decision process, strategy development, and strategy implementation.

4.4.1.1. Short Range Cockpit Improvement

The A320 cockpit bears limited resemblance to its predecessors, but one that may be a future trend setter. Roll and pitch controls are allocated to "side

sticks" that are installed on lateral consoles. The conventional control column is no longer present. In conventional aircraft the control column function required large mechanical assemblies that occupied much of the subfloor compartment. The sidestick controller uses analog displacement sensors to permit large grip motion, instead of force sensors that would restrict motion. The sidestick controller includes roll and pitch sensor boxes and centering "feels". A solenoid-operated detent locks the control lever at neutral when in autopilot mode. The stick does not respond to autopilot commands. The canted hand grip has maximum angular deflections of 16 degrees in pitch and 20 degrees in roll. The sensors detect displacements of the hand grip; this results in the transmission of analog signals to the FCS computers. All roll and pitch control is under the guidance of the fully electronic FCS.

Minimal mechanical backup is provided by a mechanically controlled rudder and standby tailplane trim. The basic yaw control uses a single rudder controlled from the cockpit pedals. A trim switch is located at the rear of the central pedestal and controls rudder trim.

Another recent innovation in today's cockpit is the presence of CRT displays. Usually, one set of CRTs displays traditional flight control mode information, as well as consolidations of artificial horizon, horizontal situation indicator, and airdata instrumentation. Another CRT displays integrated plan and profile views of the projected flight plan.

Smart flat panel displays are being considered as potential replacements for CRTs. Examples of the technologies being considered are gas plasma, liquid crystal, and thin film electroluminescence. The Liquid Crystal Display (LCD) is getting particular attention because of its low power and low voltage requirements. It has the disadvantages of poor temperature performance, low dynamic range, and limited viewing angle; however, these problems are solvable. Research with thin-film transistor LCD matrix devices has established the LCD technology as the most likely successor to the CRT.

4.4.1.2. Long Range Cockpit Improvement

Whenever a new avionic device is placed in the cockpit, more visual workload is placed on the pilot. Researchers working on the virtual panorama display, a precursor of the USAF Supercockpit Project, are attempting to estimate the workload produced by a new display before the display appears in a cockpit. This involves creating semantic networks from the information content in an image. A path is created in the network connecting a certain piece of information in the image with information known to the pilot. The number of semantic nodes that must be traversed are used as a measure of the workload placed on the pilot by the image. This technique has been tried in simulated missions, and the simulation confirmed the technique. The more complex the semantic network rating, the greater the deterioration in pilot performance in the simulated mission.

In the supercockpit the pilot will wear a helmet weighing approximately three and one-half pounds. Two tiny video screens will be mounted in front of the pilot's eyes. These screens will enable the pilot to view avionics displays, as well as magnified and illuminated real-world scenes. The data to be viewed

will be selected by the pilot, or the FCS will display data on an exception basis.

In the supercockpit, the screens will be translucent, so the images will be projected on the real world. A flight simulator helmet from which the supercockpit helmet will evolve utilizes two tiny CRTs; each CRT projects a scene one thousand pixels wide and one thousand pixels high, each pixel has a width of approximately twelve microns. The concept of the translucent display, instead of the CRT, is being tried in the Army's developmental LHX helicopter. There is concern over pilot confusion resulting from superimposing the display images on the real world. Research has shown that simulated images cause little confusion; whereas, actual images (such as magnifications) may cause considerable confusion. A 14-bit precision system is used to detect the orientation of the pilot's head. If the pilot turns his head, the display image will rotate with his head in relatively the same position. The system uses a set of three orthogonal radio-frequency coils mounted in the cockpit and three tiny magnetic detectors mounted in the helmet. Signals from the detectors are used to determine the exact position of the helmet. Special purpose electronics were developed to compensate for distortions in the optical system. These electronics result in (1) the blending of edges when there are two overlapping optical fields, and (2) the creating of the effect of tunnel vision.

If the scene becomes overly complicated, some of the data may be fed into the pilot's ear. The speakers would be located in the pilot's helmet. Locations of nearby aircraft may be indicated by sounds located in virtual or apparent space around the pilot. Techniques for localizing sound are primitive. Stereo indicates the lateral location of sound (left or right), but doesn't localize it in the two remaining dimensions, up/down or front/back. The cues used in stereo (timing, phase shift, and loudness difference between the ears) are important, but are incomplete. Paragraph 4.2.1 of this tutorial discussed VCP. Audio sensors for picking up the pilots' speech would also be present in the helmet.

Pilots can be overloaded with sounds, as well as sights. With VCP the voice can be used to control the aircraft. The sense of touch can also be used. In the cockpit of the future, the pilot may be able to reach out and touch a "virtual switch" to activate a sensor. The switch may be an image from his helmet. This may be accomplished through fiber optic technology or by having a pneumatic device built into a special glove. Far more sophisticated pneumatic arrays using piezoelectric materials may be able to simulate the touch of any object.

4.4.1.3. Pilot Associate Program

Developments in AI will have a tremendous impact on future cockpits. AI Implications for Advanced Pilot/Vehicle Interface (PVI) Design (Maxwell and Davis, 1985) is an excellent source on the methods and extents to which AI will effect the PVI in future generation aircraft. Avionic systems are proliferating in today's aircraft. This has been particularly a problem in military tactical aircraft with flight crews of one or two persons. In this case, flight crews are facing rapidly increasing workloads which must be accommodated under stressful conditions. As avionic systems increase in commercial and general

aviation, the same effect will be felt for aircraft in these classes. The amount of data to be analyzed and pressures to reduce crew costs will increasingly reduce the role of the human crew in data analysis and decision-making. The PVI is that portion of the cockpit that presents information to the flight crew for their analysis, and receives control information from the crew. As the amount of aircraft data and control requirements have increased, PVIs have been designed to compensate in several ways.

First, multifunctional displays have been designed that do not display all data but only the data that is functionally relevant. Second, integrated displays are designed to provide cross-system data that is functionally related. Third, automated systems are designed that operate quasi-autonomously from the crew; by doing this, the crew needs only to monitor systems intermittently instead of continuously controlling them. Fourth, multifunctional systems are designed that permit crew controls at a high level; they do not have to give attention to subsystem details. The effect of each of these should be to reduce the workload on the crew.

Artificial Intelligence is a technology that offers the possibility of massive improvements in this area. The DOD Pilot Associate program is studying the feasibility of having a computer responsible for the routine tasks of flying, and acting as a copilot. Additionally, the pilot's brain waves and heart muscle electrical rhythms are monitored to determine "black out". If a pilot blacks out, the computer will completely take over flight of the aircraft. Figure 4.4-1 depicts a PVI utilizing an AI computer system. There are three relationships that may exist between a pilot and an AI system. AI Implications for Advanced PVI Design (Maxwell and Davis, 1985) discusses these relationships in detail.

The first relationship is the Pilot Manager/AI Associate model. In this model the pilot is responsible for those decisions at the outer control loop level. The pilot performs a systems manager function; the AI system responds to the directions of the pilot. Decisions in the inner control loop are the responsibility of the AI Associate. This model is acceptable in operational situations where tasks may be partitioned vertically. The pilot performs tasks at the upper end, while the AI Associate performs those at the lower end.

The second relationship is the Pilot/AI Colleague model. In this model the pilot works with the AI Colleague on the basis of equality and shared responsibility. Each oversees the actions of the other and contributes to the performance of virtually every task. This model is best suited to operational situations in which a vertical partitioning is either impossible or inefficient.

The third relationship is the Autonomous Assistant model. In this model the computer may actually assume control of the aircraft, and the pilot will be subordinate to the AI system. This relationship is desirable if the pilot becomes incapacitated to the extent that the pilot cannot function in one of the other two relationships.

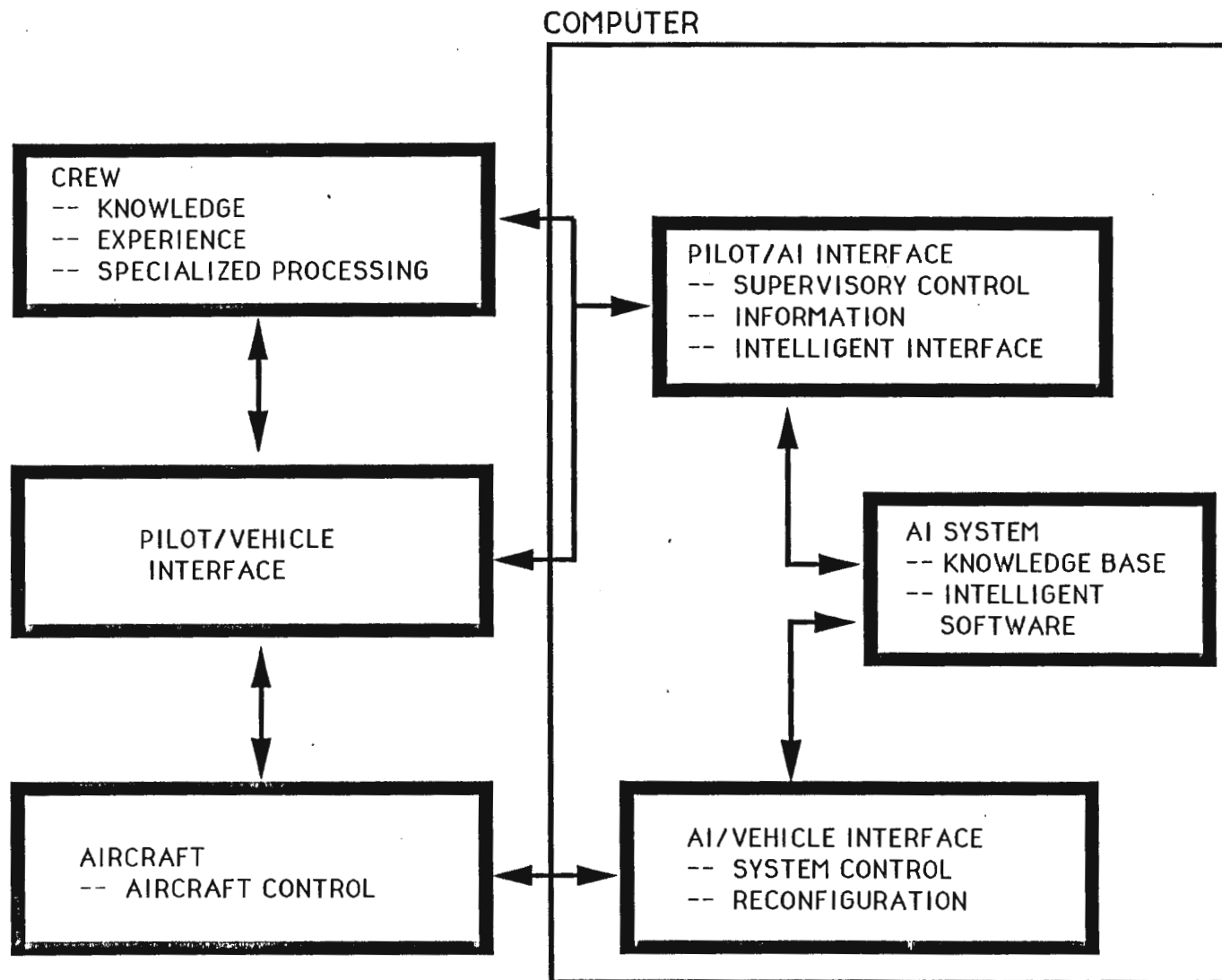


FIGURE 4.4-1. PILOT/VEHICLE INTERFACE WITH ARTIFICIAL INTELLIGENCE SYSTEM

These three relationships are not mutually exclusive. A single PVI may support all three relationships. During a single mission, each of the models may be used when operational situations dictate. The particular blend of the above three models into the design of a single PVI is determined by the characteristics of the tasks being performed and the problems being solved.

The technology for supporting such a PVI is far in the future. In addition to the hardware and software enabling technologies, design algorithms must be developed for resolving disagreement between the pilot and AI system, determining when the AI system should interrupt the pilot from his current task and enabling the PVI to change from one relationship to another.

4.4.2. Remotely Piloted Vehicles

With the exception of highly restricted applications, there appears to be no movement to eliminate humans from the flight deck. However, improved computer technology does enable autonomous and semiautonomous aircraft to be remotely piloted. This may be desirable when the presence of a human pilot is undesirable. Examples include vehicle missions that require very long endurance (e.g., space probes, communication relays, and fire controls) and military missions that are too dangerous for manned aircraft (e.g., defense suppression). When communication links may be sustained, the RPV may operate under the direction of a control station. In missions subject to the severance of communication links, the unmanned vehicle may have to operate autonomously. Autonomous operation is far in the future. The technology advances needed to support RPV development are intelligent sensor control, intelligent sensor processing, AI, communications, and remote control related devices.

4.4.2.1. Control of a Remotely Piloted Vehicle

Semiautonomous RPV must be controlled from a "mother" craft. The next generation of spacecraft will include several RPVs. Examples of these are the Orbital Maneuvering Vehicle (OMV) and the Orbital Transfer Vehicle (OTV). The NASA Space Station will be the "mother" craft. These vehicles will serve as supply carriers, service satellites, and propel missions into deep space.

The latest technology is being applied to controlling the OMV and OTV from the space station. Pilot video monitors provide navigational displays to the pilot. The primary video monitor will display the main video camera signal received from the RPV. The secondary video monitor provides overlays to the primary monitor; the selection of a particular overlay is at the discretion of the pilot. Information displays will provide the pilot with work aids and secondary displays. These include set-up programs, checklists, operating procedures, real-time flight trajectory graphs, and alarms. The information displays will most likely be a color microcomputer display. A control panel will also be present in the form of a microcomputer display; touchscreen technology will provide graphic representation of push-button panels and associated status information.

Hand Controllers provide the facility for manually maneuvering the RPV. Usually two controllers will be present: one for control of the X-, Y-, and Z-axis maneuvering and one for roll, pitch, and yaw maneuvering. These two controllers

permit a six degree of freedom vehicle control. Touchscreen is the primary mode of control for function selection, but a keyboard and mouse may be used as a backup. An alternate mode of control is speech recognition and synthesis.

4.4.2.2. Avionics for a Remotely Piloted Vehicle

The preceding section discussed RPV control for a vehicle in a space application. Other RPVs may have to navigate the near-earth atmosphere at varying degrees of altitude. The avionics for such a vehicle would include flight control, inertial attitude reference, air data terminal, and payload/datalink systems. These are shown in schematic form in figure 4.4-2. At present, these RPVs are primarily used in military applications. As a result, the payload may consist of sophisticated weaponry with digital components or electronic countermeasures. The datalink is a secure communications system that supports the RPVs remaining at a safe distance from enemy defenses and maneuvering in a highly evasive fashion. The total cost of the RPV avionics would be concentrated in these two systems.

The FCS consists of a microprocessor with Random Access Memory (RAM) and/or Erasable Programmable Read-Only Memory (EPROM) memory. This processor handles the software functions associated with attitude stabilization, guidance, navigation, command execution, payload interface, datalink interface, and miscellaneous computational tasks. This microprocessor receives inputs from the flight sensor instrumentation and generates outputs to various servos and actuators. The inertial attitude reference is a sensor that provides accurate data on instantaneous vehicle rates. The air data terminal is the end of the data link on the RPV; it communicates with the control station.

Avionic systems represent approximately 80 percent of the cost of an RPV; the payload, itself, may represent 60 percent of the cost. Research and development effort is directed toward reducing the payload and ground control station costs. Since a major application for this type of RPV is target recognition, MV is an enabling technology for future RPV advances. (See paragraph 4.2.2.)

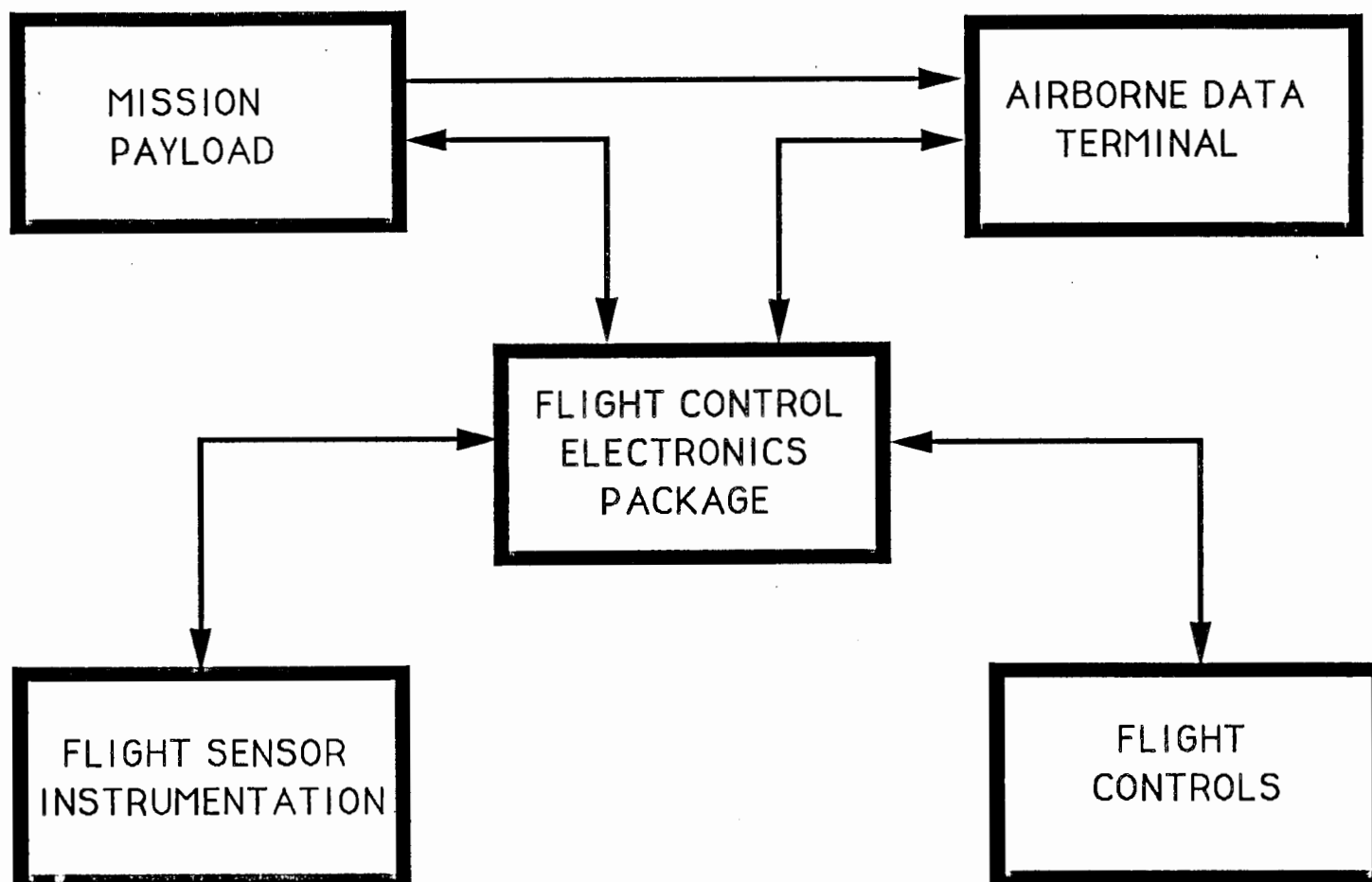


FIGURE 4.4-2. REMOTELY PILOTED VEHICLE AVIONICS

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GLOSSARY

Ambient. The substance which absorbs heat from the heat sink.

Artificial Intelligence. The characteristics of a machine programmed to imitate human intelligence functions.

Autofeather. To automatically and swiftly feather the propeller when the engine fails to drive it.

Canard. A tail-first aerodyne, usually with auxiliary horizontal surface at the front and a vertical surface at the back.

Fly-By-Glass. Flight control system where fiber optics carry the signal.

Fly-By-Light. Flight control system where fiber optics carry the signal.

Fly-By-Wire. Flight control system with electric signaling.

Gigabit. One billion bits.

Kilobyte. One thousand bytes.

Micron. One-millionth of a meter.

Nanosecond. One-billionth of a second.

Thyristors. Solid-state devices that convert alternating current to direct current.

ACRONYMS

3-D	Three-Dimensional
ACS	Automatic Control System
AERA	Automated Enroute Air Traffic Control System
AFM	Advanced Fuel Management
AI	Artificial Intelligence
ATF	Advanced Tactical Fighter
ATI	Access Time Interval
CRT	Cathode Ray Tube
CSMA	Carrier Sensed Multiple Access
DARPA	Defense Advanced Research Projects Agency
DFCS	Digital Flight Control System
DOD	Department of Defense
EIU	Electronic Interface Unit
EMI	Electromagnetic Interference
EPROM	Erasable Programmable Read-Only Memory
FAA	Federal Aviation Administration
FADEC	Full Authority Digital Engine Controller
FAFTEEC	Full Authority Fault Tolerant Electronic Engine Control
FAR	Federal Acquisition Regulation
FBL	Fly-By-Light
FBW	Fly-By-Wire
FCS	Flight Control System
FET	Field Effect Transistor
GaAs	Gallium Arsenide
GNC	Guidance, Navigation, and Control
HERF	High-Energy Radio Frequency
ITT	(Consultative Committee for) International Telegraphy and Telephony
ka	kiloamp
LCD	Liquid Crystal Display
MAFT	Multicomputer Architecture for Fault Tolerance
Mbps	Million bytes per second
Mflops	Million floating-point operations per second
MPP	Massively Parallel Processor
OMV	Orbital Maneuvering Vehicle
OTV	Orbital Transfer Vehicle
PCS	Primary Control System
PLA	Power Level Angle
PVI	Pilot/Vehicle Interface
RAM	Random Access Memory
RCA	Radio Corporation of America
RF	Radio Frequency
RPV	Remotely Piloted Vehicle
RTCA	Radio Technical Commission for Aeronautics
SAE	Society of Automotive Engineers
STOL	Short Takeoff and Landing

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 23. Police Services
 24. Emergency Services
 25. Other

