Advanced System Integration Concepts

for

Future Commercial Transport Aircraft

A Thesis presented

by

A. Nadesakumar

for the degree of

Master of Philosophy

in the

University of Southampton

Department of Aeronautics and Astronautics

Faculty of Engineering and Applied Science

January 1993

UNIVERSITY OF SOUTHAMPTON

ABSTRACT

FACULTY OF ENGINEERING AND APPLIED SCIENCE
DEPARTMENT OF AERONAUTICS AND ASTRONAUTICS

Master of Philosophy

ADVANCED SYSTEM INTEGRATION CONCEPTS FOR
FUTURE COMMERCIAL TRANSPORT AIRCRAFT

By Alagesan Nadesakumar

The avionics systems in state-of-the-art commercial aircraft have become increasingly complex and sophisticated in order to meet ever increasing performance and reliability requirements. With the capability of the avionics technology improving by an order of magnitude every few years, it is envisaged that the current philosophy of one box-per-function would soon reach its limits as to what can be accommodated in terms of cost, functionality, reliability and certification.

In this thesis, the limitations of the current avionic system configuration has been addressed and the need for a new approach to avionics system integration presented. The solution that has been put forward is the integrated systems configuration. Integrated systems, with distributed processing where the resources are shared by many functions improve the reliability, availability, survivability and extensibility of a system. Furthermore they also provide the potential for reducing the acquisition, maintenance and operating costs of a system.

The thesis also addresses a number of specific issues considered necessary for the practical implementation of an integrated architecture. These include data bus requirements, electro-magnetic and radio frequency interference prevention and fault tolerance. In addition consideration was given to the implementation of novel concepts such as multi-sesnsor data integration and local actuator controllers which are compatible with distributed processing techniques.

Acknowledgement

The author wishes to express his gratitude to Professor Chris Harris, the project supervisor, for all the support and encouragement given throughout this work as well as the many helpful suggestions and comments he has made on the text of this thesis.

The author is indebted to Dr Richard Crowder of the Department of Electrical Engineering for providing the opportunity to carry out this work, and for the support and advice given throughout this project.

Thanks are due to Dr Robert Jones of Cranfield Institute of Technology for providing an insight into the origins of the CRISPS project (as detailed in the introduction) and for his initial research into secondary power systems.

Many thanks are also due to Mark Williams for his invaluable help throughout the course of this project.

Finally the author is grateful to the Department of Trade and Industry for funding this project and the CRISPS industrial consortium, especially Smiths Industries for their technical advice.

Contents

Part III. Aspects of Sensors and Actuators

List of Abbreviations

A^3P	Advance Avionics Architecture and Packaging
AC	Alternating Current
ADOCS	Advanced Digital Optical Control System
AEA	All Electric Aircraft
AEEC	Airlines Electronic Engineering Committee
AIMS	Airplane Information and Management System
AIPS	Advanced Information Processing System
AMAD	Airframe Mounted Accessary Drive
APEX	Applications/Operating System Interface
ARINC	Aeronautical Radio, INC.
ASCB	Avionics Standard Communication Bus
ATC	Air Traffic Control
ATP	Advance Turbo Propeller
ATN	Aeronautical Telecommunication Network
BAe	British Aerospace
BCD	Binary Coded Decimal
BIT	Built-In-Test
BITE	Built-In-Test Equipment
bits	binary digits
BP	Basic Protocol
CAA	Civil Aviation Authority
C^3	Command, Control and Communication systems
C^3I	Command, Control, Communication and Intelligence
CNI	Communication, Navigation and Identification
CODEC	Coder/Decoder
COEX	Core hardware and Hardware Interface System/Operating System Interface
CP	Combined-mode Protocol
CPU	Central Processing Unit
CRISPS	Collaborative Research Initiative into Secondary Power Systems
CSDB	Commercial Standard Digital Bus
CSDL	Charles Stark Draper Laboratory
CSMA-CA	Carrier Sensed for Multiple Access with Collision Avoidance
DATAC	Digital Autonomous Terminal Access
dB	decibel

DAIS	Digital Avionic Information System
DC	Direct Current
DME	Distance Measuring Equipment
DOC	Direct Operating Cost
DTI	Department of Trade and Industry
E/O	Electrical to Optical
EFIS	Electronic Flight Instrument System
EHA	Electro-Hydraulic / Hydrostatic Actuator
EICAS	Engine Indication and Crew Alerting System
EM	Electro-Magnetic
EMA	Electro-Mechanical Actuator
EMI	Electro Magnetic Interference
FAA	Federal Aviation Authority
FBL	Fly-By-Light
FBW	Fly-By-Wire
FDI	Fault Detection and Isolation
FDIA	Fault Detection, Isolation and Accommodation
FTMP	Fault Tolerant Multi-Processor
GPC	General Purpose Computer
HF	High Frequency
HIRF	High-Intensity Radio Frequency
HIS	Hardware Interface System
I/O	Input/Output
ICAS	Integrated Avionic Computer System
IDEA	Integrated Digital/Electric Aircraft
IFF	Interrogate Friend or Foe
ILS	Instrument Landing System
IMA	Integrated Modular Avionics
IQ^2	Intelligent Quantitative and Qualitative
ISO	International Standards Organisation
J	Joules
Kbits	Kilo bits
LAN	Local Area Network
LD	Laser Diode
LED	Light Emitting Diode
LRM	Line Replaceable Module

LRU	Line Replaceable Unit
LVDT	Linear Variable Differential Transducer
Mbits	Mega bits
MCE	Micro-Circuit Engineering
MHz	Mega Hertz
MIL-STD	Military Standard
mm	millimetre
MSDR	Multiple Speed Data Rate
MTBF	Mean Time Between Failure
MTBMA	Mean Time Between Maintenance Alert/Action
MUX	Multiplexed
N/A	Not Applicable
NASA	National Aeronautics and Space Administration
OBPP	Overall Benefits Pridiction Program
O/E	Optical to Electrical
O/P	Output
OPMIS	Optical Propulsion Management Interface System
OSI	Open System Interconnection
PBW	Power-By-Wire
PCB	Printed Circuit Board
PIN	Positive-Intrinsic-Negative
PROM	Programmable Read Only Memory
QSRA	Quite Short-haul Research Aircraft
RA	Research Assistant
R&D	Research and Development
RF	Radio Frequency
RFI	Radio Frequency Interference
RPA	Remote Powered Actuator
RPP	Receive Personality PROM
RSS	Relaxed Static Stability
RT	Remote Terminals
RVDT	Rotary Variable Differential Transducer
RZ	Return to Zero
SAI	System Architecture and Interface
SATCOM	Satellite Communication
SBAC	Society of British Aerospace Companies

SG	Synchronisation Gap
SIFT	Software Implemented Fault Tolerance
SIM	Serial Interface Module
Spec.	Specification
Tacan	Tactical air navigation
TDM	Time Delay Multiplexing
TG	Terminal Gap
TI	Transmit Interval
UHBPR	Ultra High By-Pass Ratio
UHF	Ultra High Frequency
VHF	Very High Frequency
VHSIC	Very High Speed Integrated Circuit
VLSI	Very Large Scale Integration
VLSIC	Very Large Scale Integrated Circuit
WDM	Wavelength Division Multiplexing
XXP	Transmit Personality PROM

1 Introduction

1.1 Background

After the fuel crisis of the 1970's and apparently ceaseless increase in aviation fuel costs that followed, interests in any technologies which promised lower fuel burn on civil aircraft were rigorously pursued. This led to major research efforts in areas such as laminar flow control and higher by-pass ratio power plants of various types.

Another area to receive consideration with regard to reducing fuel consumption was that of an aircraft's onboard system particularly those which produce, distribute and use significant amounts of power. This power is in the main extracted from an aircraft's engines and thus directly increases fuel consumption. However, systems cause further increase in fuel consumption indirectly due to their mass and, particularly if cooling air is required they may directly increase the drag.

To an operator it is not just fuel consumption of an aircraft that is important, but its overall cost of operation. This is a function not only of the fuel used but also several other costs, notably the cost of acquiring the aircraft initially and maintaining it. Unfortunately aircraft's onboard systems increase both of these costs as well. Thus, whilst fuel consumption was one major driver in research into advanced systems in the late 1970's and 1980's, other aspects were also viewed as important.

Interest in aircraft power systems at this time centred on a number of themes. These were in general not new but appeared more attractive than previously due to various technology advances. Amongst the concepts considered were evolutionary changes such as higher pressure hydraulics and improved efficiency air-conditioning, and revolutionary changes such as the All Electric Aircraft (AEA). Many of the concepts were competitive, but most could benefit from advances in onboard computing and electrical signalling which could itself show major benefits.

Although studies and various ground and flight tests of components were performed which confirmed some of the advantages of the various proposed concepts, few have been actually incorporated into production aircraft. In part this was due to the fact that possible competing systems showed similar advantages. However by the mid 1980's significant interest in the use of advanced power plant had grown and a further concern for aircraft power systems had arisen. It had been suggested that Ultra High By-Pass Ratio (UHBPR) powerplants would be unable to provide present levels of power and in particular may not be able to provide any bleed air. This spurred interest in concepts such as the AEA, which extracted power only in a shaft form from engines, but also led to a reconsideration of the idea of using independent sources of system power on aircraft. In 1985, this led NASA to commission Boeing and Lockheed Aircraft Corporation to carry out the Integrated Digital/Electric Aircraft (IDEA) concepts study, where the AEA concept was combined with digital data bussing techniques, to evaluate their benefits.

1.2 UK Interest and Studies

Partially as a result of the IDEA studies, the Society of British Aerospace Companies (SBAC) became interested in advance systems concepts such as the AEA. Doubts existed, however, as to some of the benefits claimed for the AEA approach. For this reason the UK Department of Trade and Industry (DTI) were approached to provide funding for an independent study of the concept. Although it was important that the interested parties in the airframe, engine and equipment industries should be involved, it was felt that the control of the study should rest with a body with no 'axe to grind'; the College of Aeronautics at Cranfield therefore put up a draft proposal. In its final form, the proposal was for a 'Zero Bleed Secondary Power Systems Study' where the elimination of bleed air occupied the major part of the study, due to its expected significant fuel savings - the conversion of hydraulics to electric power was also addressed, in the later part of the study.

The work done at Cranfield was largely a one-man-campaign by Dr R.Jones, who was expected to spread his resources rather thinly over a wide range of specialist areas. Despite this, the DTI were sufficiently encouraged by his findings to provide continued, not to mention increased, funding that would allow the recruitment of manpower - so was born the project which has become known as CRISPS (Collaborative Research Initiative Into Secondary Power Systems).

1.3 The CRISPS Program

CRISPS has been a two year program which began in October 1990, to evaluate the effects the adoption of a wide range of system changes would have on civil aircraft. The major work load of the study has been shared between Cranfield Institute of Technology, College of Aeronautics, and University of Southampton, Departments of Electrical Engineering and Aeronautics and Astronautics, with support and input from a number of British aerospace companies. The man-power allocation is such that at Cranfield there are two Research Assistants (RAs) concerned with the aircraft mechanical systems (mechanical and pneumatic), while a third RA, who is the computing specialist, analyses the options for change via an Overall Benefits Prediction Program (OBPP). Meanwhile at Southampton, a further two RAs have been concerned with the electrical power and systems integration aspects of the project - the author being involved in the latter.

1.4 The Need for a New Approach to Avionic Systems Integration

Current avionic systems architectures are based on the concept of 'one-box-one-function', in which the avionic systems are separated into functional areas such as flight control system, engine control system, navigation system, communication system etc. Each system has its own dedicated computer housed in the avionics bay, as line replaceable units (LRUs), where they are provided with power and conditioned air for cooling as required. Communication between the various sensors, actuators, pumps, valves etc. and the computers which control them is accomplished via dedicated point-to-point links; whilst inter-subsystem communication is achieved via data buses. To satisfy flight safety requirements, the computers are replicated and multiple paths are provided for communication.

The systems configuration described, however, suffer from a number of problems. One of the main problems is the large number of wires and interconnections found in the system. This problem is further aggravated by the data bus standard in current use on commercial aircraft (ARINC 429). Since it is a single source, multi-sink, uni-directional bus, equipment requiring data from a number of systems has scores of wires attached to them, causing physical congestion at the equipment interface. This makes the whole integration approach inefficient, unreliable and difficult to maintain.

Another problem is the use of a variety of computers to perform each function. Although LRUs have eased on-line maintenance through removal and replacement procedures, an extensive supply of spares has to be maintained [Little 1991]. In addition the variety of computers require a wide range of expensive test equipment with skilled maintenance personnel to operate them. Since aircraft are currently incapable of absorbing system faults for an extended period of time, spares and maintenance crew have to be stationed at various point on the airlines route. This imposes a huge cost burden on the operator.

A further disadvantage of the current avionic architecture is the lack of flexibility to accommodate updating and system extensions. Since the architecture is inherently coupled, i.e. there is a close relationship between all of the system elements, any change to one element of the system has an effect on the anothers - such modifications make system upgrades expensive and time consuming.

In view of this the avionics community (i.e. the airframe manufacturers, the airline operator and the equipment manufacturers) have called for a new approach to avionics system integration. This requirement forms the basis of this thesis, where the objective is to evaluate various candidate architecture concepts and address issues pertinent to systems integration in order to assess if they are capable of easing the shortcomings faced by the current avionic systems.

For a number of years the avionics community at large have argued for [AEEC 1991, Archer 1985, Morgan 1991]:

1) A reduction in overall cost of ownership through reduced acquisition cost
2) A reduction in spares requirement
3) A reduction in equipment removal rate
4) A reduction in weight and volume of wiring
5) Improved Built-In-Test (BIT) coverage to provide better maintenance diagnostics, improved fault detection and reduced unconfirmed removals
6) Maintenance free dispatch
7) Resource sharing to reduce LRU count
8) Standardisation at the functional interface to provide hardware and software interoperability.

In order to achieve these goals, it is clear that the avionic system configuration must move away from the traditional approach of point-to-point, one box-per-function philosophy. The concept which has been proposed in this thesis is the integrated, modular systems approach with distributed processing, where the computer resources are shared by different functions - the idea being that resource sharing would minimise functional replication in hardware and software providing cost savings, with the modularity allowing flexible growth.

1.5 Integrated Systems - An Overview

Integration refers to the process through which different functions are combined together to yield a higher level function [Botha 1981]. It usually implies the existence of a number of subsystems each performing a subfunction of a total system function. An integrated system is one which has been partitioned into subsystems according to some design criteria, with subsystems implemented and integrated in such a way that they meet the system requirements in the most efficient way.

The concept of functional integration refers to the process where several functions share a common resource such as computational capabilities, receivers, transmitters, etc. The idea is however not new - it was considered as early as 25 years ago, but was not implemented due to lack of enabling technologies [Janex 1988]. However with recent advances in electronics/digital technology the concept has become feasible.

Integration is more readily applicable to closely related functions, specifically those which combine to influence common outputs, actuators etc. However integration of essentially unrelated functions lead to complex failure analysis, and the cost incurred in demonstrating the adequacy of safety can outweigh the advantages gained through shared hardware or data resource [Ostgaard et. al. 1987].

Therefore during integration, consideration must be given to the level of integration, the functions to be integrated and the form that the integration should take, e.g. centralised versus distributed.

In a centralised architecture, all the processing is performed in a single, centralised computer complex in which the processors are tightly coupled and under the control of single operating system [Cronin et. al. 1985, Dolezal 1988]. By contrast, in a distributed architecture, the processing is shared by a number of separate computers. These computers are linked by a data network, and are capable of autonomous operation. The control of the system may be centralised (i.e local to one site) or distributed (i.e. control is effected through the cooperative action of all) [Cronin et. al. 1985, Lala 1984].

The centralised architecture is by far the easiest to implement, and although adequate for many applications, it is considered unsuitable for avionic application for a number of reasons. First, the centralised architecture relies on a central control unit for its operation. This makes the system vulnerable to faults in the processor which has assumed the management task. Reconfiguration strategies based on the use of back-up processor may be used to mitigate this problem, however this adds complexity to the software control configuration. Second, for flight critical application fault tolerance is paramount. In a centralised architecture, fault tolerance can be implemented through the replication of resources. This however results in functional replication in elements such as central processing units (CPUs), power supplies and software modules incurring undue weight, volume, power and cost penalties. Third, it is anticipated that the commercial avionics in the future will become even more complex as technology evolves, requiring further increase in processing. This extensibility requirement can only be efficiently met if the processing can be organised into more manageable, distributed packages instead of putting more and more capability into a centralised processor. Finally, with the trend towards the use of carbon fibre and other composite material for aircraft structure, it is envisaged that soon the aircraft data channels will become more susceptible to high intensity Electro-Magnetic Interference (EMI) and Radio Frequency Interference (RFI) effects resulting in possible corruption of data. One way of easing the EMI/RFI problem is to minimise the data flow through the system. This can only be best achieved by processing the raw data locally, from where it is gathered, and transmitting only processed data through the system. For these reasons, the preferred architecture to implement advanced avionic system is the distributed architecture. The advantages of a distributed architecture are:

1) Improved reliability, availability and survivability - the network system physically disperse functions and provides redundancy
2) Fault tolerance through dynamic reconfiguration i.e task migration from one processor to another or from one computer to another upon the detection of a fault ; graceful degradation.

3) Resource sharing - minimises the replication of hardware and software
4) Flexibility of equipment location - optimises equipment space
5) Supports equipment from mixed vendors - customer not tied to one supplier
6) Extensibility - provides growth flexibility for capability and technology upgrades

In general, for an architecture to be considered viable for commercial avionics application, it needs to demonstrate reliability, availability and extensibility. Reliability and availability requirements generally imply the application of fault tolerance. Fault tolerance usually requires redundancy and/or high integrity monitoring. The integrated system by virtue of its ability to autonomously determine its own status (i.e. self monitor) is able to detect and isolate faults, and reconfigure as required. Furthermore, since resources are shared by many functions, redundancy is inherent in integrated systems. Availability is achieved through dynamic reconfiguration. When a component (processor) in the system fails, its task is assigned to another component or distributed amongst the remaining healthy components.

Integrated systems provide extreme flexibility for future capability and technology upgrades. Furthermore they also provide the potential for significant cost savings (i.e. both first cost and operating cost) and improvements in system performance. They reduce the initial cost by minimising functional duplication of hardware and software elements, and by standardising a large portion of the system. Studies indicate that by utilising standardised equipment, the cost of equipment can be reduced by as much as 50% [Roy 1984].

An integrated system by virtue of its ability to self monitor is able to simplify test and diagnostic procedures, reducing the number of no fault found removals. Reports indicate that a dominant contributing factor to operating cost is unconfirmed (i.e no fault found) removals [AEEC 1991]. Through self monitoring the system is also able to reconfigure upon the detection of a fault. This improves the operational capability of the system, providing fail operational capability in the presence of a limited number of faults. On aircraft, this improves the dispatch rate and allows maintenance to be deferred.

Avionic systems configuration needs to be kept as simple as possible, since the reliability of the systems unquestionably suffers whenever unnecessary complexity creeps in. One of the main difficulties in designing an efficient avionic system is mastering the complexity. The integrated system, with distributed processing appears to ease this problem because it partitions the complex system into smaller, more manageable and comprehensible subsystems with simpler design problems. Since aircraft systems are already separated into functional areas, it can be readily configured into a distributed arrangement.

Distributed architectures, however, have their drawbacks. The concept does not guarantee that two devices (computer) can be used cooperatively. For example, two computers from different vendors can be connected to a network, but they may not be able to communicate to each other because of the use of different data formats. To facilitate communication some sort of format conversion software is required. Loss of information control is also a major concern. Since the architecture is distributed, it is difficult to manage their resources and to control the information available through the system. A further disadvantage of integrated system is that when a system or piece of equipment that performs multiple function is replaced, it must be tested for all functions performed, even though only one function failed. This re-testing can become veiy significant for complex systems which are safety critical. Despite these drawbacks, the potential gains derived from integrated system have been far too great to be ignored, and have provided the impetus for the development of integrated system architectures.

Through the years numerous attempts have been made at implementing the aircraft avionics system using an integrated approach. One of the first attempts at integration was to combine the CNI (Communication, Navigation, Identification) function into one band and a single unit [Botha 1981, Janex 1984]. The systems that were chosen to be integrated were the UHF communication, Tacan/DME and IFF - the idea being that the frequency band commonality, as well as similar transmitter powers would ease integration. But it proved not to be the case - feasibility studies indicated a negligible reduction in the number of modules and a significant increase in complexity; in other words, an increase in cost together with lower reliability. Therefore the project was terminated [Janex 1984].

In 1966 the Wright Patterson Avionics Laboratory initiated a program called DAIS (Digital Avionic Information System) to address issues such as distribution, processing, control and display of information in a fully integrated avionic suite, to be implemented in the laboratory [Botha 1981]. This program introduced concepts of standard interfaces, computer language and instruction sets (such as MIL-STDs 1553, 1589 and 1795) and demonstrated concepts of executive and operational software having portability across airframes and missions.

One of the best examples of recent integration concept is the Pave Pillar [Klos 1984, Janex 1988, Turk 1984]. The program initiated in 1981, was developed for the United States Air Force to provide improvements in aircraft response and capability with the aim of reducing the acquisition, maintenance and operating cost [Ostgaard et. al. 1987]. In the United Kingdom, in 1989 a similar project called Advanced Avionics Architecture and Packaging (A^3P) was initiated by the U.K Ministry of Defence to look into advanced avionic concepts using modular techniques.

To summarise, for an integrated architecture to be beneficial in future avionic systems, it should have the following attributes:

1) Resource sharing: Reduces functional duplication of hardware and software. Enables graceful degradation.
2) Elimination of discrete wiring: Reduces aircraft weight and build time.
3) Standardisation of hardware: Reduces the number of individual types of processors. Enables interchangeability of equipment between aircraft without modification. Provides the equipment manufacture with increased marketing opportunities fostering competition.
4) Fault tolerance: Enables deferred maintenance - improves aircraft dispatch rate. Enables graceful degradation.
5) BIT facility: Provides improved diagnostic capability. Reduces the number of 'no fault found' removals.
6) Growth flexibility: Enables the upgrading cost to be kept to a minimum.

1.6 Outline of the contents of the Thesis

This thesis is divided into three parts. Part 1 (chapters 1-3) addresses the issues relating to systems integration. In chapter 1, the shortcomings of the current avionics systems architecture to deal with the high level of functionality found in the state-of-the-art aircraft is presented together with the main objective of this thesis. System attributes which would make an avionic architecture attractive for use in the next generation of commercial aircraft is also presented in this chapter.

Chapter 2 provides a review of some of the candidate architectures which have been proposed as viable solutions to the current integration problem together with their suitability to implement the avionics system of the next generation commercial aircraft. In chapter 3, a detailed review of the candidate architecture considered most suitable (i.e. the integrated modular avionics) is presented.

In part 2 (chapters 4-7), specific issues considered essential for the implementation of an integrated architecture are addressed. To realise the full potential of an integrated architecture safe, efficient and practical methods of subsystems communication is required. In chapter 4 a resume of the data buses in avionic use today is presented and the qualities needed for a data bus to serve a modern commercial aircraft are identified. The chapter also provides a brief overview of the principal characteristics and features of the new, two-way, high speed data bus, the ARINC 629.

One of the concerns of using a data bus to effect communication is its susceptibility to high intensity electromagnetic interference. To contain this problem the application of fibre optics on aircraft and system level protection techniques have been considered. Chapter 5 provides a review of the primary elements of a fibre optic transmission system and highlights the limitations of the fibre optic technology when applied to avionic systems. In chapter 6 some of the aircraft electromagnetic compatibility problems are presented together with system level protection schemes which overcome some of these problems.

One of the key components in achieving high levels of integration is fault tolerance. Fault tolerance in aircraft is currently achieved through the application of redundancy which incur weight, volume and cost penalties. Modern approach to fault tolerance (i.e analytical redundancy and knowledge based concepts) provide the possibility of reducing the level of hardware redundancy whilst still maintaining the high level of reliability and availability. Chapter 7 provides a review of the current fault tolerant techniques, and addresses the issues relating to the use of analytical redundancy and knowledge based concepts for fault detection and identification, with the view of using such concepts on aircraft.

In part 3 (chapters 8-9), aspects of sensor and actuator systems are addressed. Performance of aircraft systems partially depend on the reliability of its sensors. No sensor can provide accurate measurement of the measurand all of the time, and thus information from a single sensor cannot be entirely relied upon. Integration of data from multiple sensors was therefore considered to improve the level of confidence in the data. In chapter 8 the issues relating to the use of multi-sensor data integration to enhance the validity of sensor derived information is presented. The chapter also discusses the implications of multi-sensor integration on the integrated modular avionics architecture. In chapter 9, the viability of using electric actuation for primary flight control and its associated implication on the integrated systems architecture are discussed.

Finally, chapter 10 provides a summary of the whole thesis together with the conclusions drawn from the work. Recommendations for further work is also presented in this chapter.

2 Candidate Architectures for the Next Generation of Commercial Aircraft

2.1 Introduction

For a number of years the avionics community have recognised the benefits which can be realised through effective integration. Although numerous attempts were made in implementing a fully integrated avionic suite, none were practical until recently, due to limitations of enabling technology and cost effectiveness. With recent advances in technology, integrated architectures which were once not technologically feasible and cost effective have now become realisable. To this end, a number of avionics research and development organisations have proposed a number of avionic system architectures - from a centralised architecture to a distributed one. In this chapter some of these architectures are considered together with their suitability to implement the avionics systems of next generation, commercial aircraft.

2.2 Candidate architectures

Integrated systems, especially flight critical systems, are for safety reasons inherently complex and as such require a high level of redundancy and interconnections. The large number of interconnections involved make the traditional method of interconnecting subsystems via dedicated links both unrealistic and unmanageable, and thus a data bus is required to interconnect the various subsystems. This leads to the type of architecture currently being proposed for the next generation of aircraft.

2.2.1 Advanced Information Processing System Architecture

The Advanced Information Processing System (AIPS) is a 'functional distributed' system designed by the Charles Stark Draper Laboratory (CSDL), to provide a fault tolerant and damage tolerant data processing architecture for a wide range of air vehicles [Brock 1986, Lala 1984].

AIPS Concept

The AIPS consists of a number of General Purpose Computers (GPCs) of varying capabilities, located in processing centres distributed throughout the aircraft. The computers are interconnected by a reliable and damage tolerant intercomputer bus. Communication between the computers and the various subsystems is achieved via three types of I/O buses: local buses connect a computer to the I/O devices dedicated to that computer; regional buses connect computers with I/O devices located in the vicinity of the processing sites; and global buses connect I/O devices to all computers or at least the majority of them. The devices on the global bus are usually those whose information need to be made available to all the processing centres. Fig. 2.1 shows a simplified system level diagram of the AIPS architecture. The architecture allows migration of function from one processor to another in response to some internal or external stimulus. This enables AIPS to support system reconfiguration and graceful degradation.

System Control

The control of AIPS is vested in one general purpose computer called the global computer. The global computer is responsible for the overall system management and control authority. The tasks of the global computer includes system start, resource management, redundancy management and function migration.

Communication between the global computer and the various processing centres is achieved through the network operating system, which is in charge of system level functions. The tasks of network operating system include orderly start and initialization of various buses and networks, communication between processors executing in different computers, system level resource management and system level redundancy management. Although the global computer has overall control of the AIPS, each computer operate autonomously under the supervision of its local operating system. The local operating system performs all the functions necessary to keep that processing centre operating in the desired manner. The tasks of the local operating system include order start and initialization of the GPC, scheduling and dispatching of tasks, I/O services, task synchronization and communication services and resource management. The local operating system in each computer centre is connected to the network operating system.

Reliability

It has been estimated that the probability of failure of the GPC from random faults ranges from 10^{-4} per hour for simple processors to 10^{-10} per hour for multiprocessors that use parallel redundancy [Lala 1984]. In view of this, CSDL have recommended the use of parallel redundancy for applications requiring high levels of fault tolerance and simplex or duplex processors for low criticality functions. In addition they also recommend that the redundant processors be physically dispersed to provide damage tolerant communications between theses elements.

2.2.2 The Centralised/MUX Architecture

The Centralised/MUX architecture has been proposed by Lockheed as a candidate architecture for the integrated digital/electric aircraft [Cronin et al. 1985].

Centralised/MUX Concept

As the name suggest, the architecture has a centralised fault tolerant computer system. The centralised computer system performs all the avionic functions including bus redundancy management functions. Communication between the computer and the various avionic subsystems is accomplished via quad, linear, multiplexed buses; one running longitudinally from the cockpit to the empennage, and the other running laterally from wing tip to wing tip (Fig. 2.2). The data buses are similar in structure and operation to the MIL-STD-1553B bus.

System Control

The control of the Centralised/MUX is vested in one single bus controller which has centralised control over one multiplexed bus. Command/response protocol is used to accomplish device multiplexing on each individual bus.

Reliability

Fault tolerance is achieved through hardware redundancy. Lockheed recommend the use of quad, fail operational fault tolerant processors and two independent set of quad buses. The fault tolerant processor is said to be most likely a design similar to the CSDL fault tolerant processor. Fault detection of processor is implemented in hardware with isolation and reconfiguration implemented in software.

2.2.3 Partially Distributed/MUX Architecture

This is another architecture proposed by Lockheed as a possible candidate for the integrated digital/electric aircraft [Cronin et al. 1985].

Partially Distributed/MUX Concept

The Partially Distributed/MUX architecture consists of a number of general purpose computers and 'dumb' Remote Terminals (RTs) distributed throughout the aircraft. Communication between the computers and the RTs is accomplished through a linear multiplexed data bus. The data bus uses a command/response protocol similar to a MIL-STD-1553B bus. The system is only partially distributed because a centralised bus controller is used instead of a fully distributed global bus control; the term 'distributed' refers to the physically dispersed, reassignable functional processing. To overcome the disadvantages of a centralised bus control, the architecture employs a primary bus controller (master bus controller) with two standby back-up units. The three units are interconnected by a dedicated interprocessor bus (Fig. 2.3).

System Control

The overall control of the architecture is vested in a centralised bus controller. The centralised bus controller is responsible for global redundancy and system management.

Reliability

Fault tolerance is achieved through hardware redundancy. Lockheed recommend the use of quad ultra reliable fault tolerant processors and quad buses. To offset the disadvantages of the centralised bus controller, the architecture employs two standby back-up units (computers). The back-up units are general purpose flight computers which usually perform flight control functions. In the event of failure of the primary bus controller, the control is passed to one of the standby computers via the dedicated interprocessor bus.

2.2.4 Integrated Modular Avionics

Integrated Modular Avionics (IMA) concept is a totally new approach to avionics system design in that it yields a highly integrated avionic system. The architecture takes advantage of the features found in the state-of-the-art microprocessor design, namely high throughput and fault tolerance [AEEC 1991, Large (BAe), Morgan 1991, Subra et. al. 1990].

IMA Concept

The IMA architecture consists of a number of cabinets distributed throughout the aircraft. Each cabinet contains processing modules which are capable of performing multiple avionic functions. The cabinets are interconnected by an ARINC 629 data bus which also connects various avionic hardware to form an integrated system for performing all avionic functions on the aircraft (Fig. 2.4). In areas of low data concentration, the architecture employs an ARINC 429 data bus to interface with IMA. Intracabinet communication is accomplished through the ARINC 659 backplane data bus.

The IMA concept encourages the use of network compatible devices. These devices interface directly to the ARINC 629 data bus. Data concentrators are used to interface devices which cannot be directly connected to the ARINC 629 data bus. These significantly reduce the number of discrete wires in the aircraft. The functionality of the avionic system is provided by the application software which runs in a standard environment. This decouples the hardware from the application and allows the interchangeability of hardware and software from different vendors.

System Control

One of the major features of this architecture is that it does not require a central controller for its operation. The information flow within the IMA cabinets or between the cabinets is driven by tables fixed at design time held in all modules. All modules in effect synchronously step through a copy of the same table as messages are broadcast onto the bus. Provision is made for the bus to re-synchronise if they need to do so.

Reliability

IMA supports the use of high Mean-Time Between Failure (MTBF) equipment to yield a highly reliable system. The equipment are expected to have a MTBF of 15000 hours, with no loss of function, and 99% probability of providing full function for another 200 hours [AEEC 1991]. Fault tolerance is achieved through redundancy (system level redundancy and component level redundancy), and high integrity monitoring. In system level redundancy, fault tolerance is provided by having multiple paths for the data, from its source to the processor and from the processor to the sink. In component level redundancy, fault tolerance is achieved through the use of back-ups which maintain the overall functionality of the system.

2.3 Discussion

Although the architectures considered are different in their implementation, there are a number features common to the designs; these include fault tolerance, reconfiguration and inherent flexibility for expansion.

The AIPS architecture has several advantages. The AIPS concept yields an architecture that is both fault tolerant and damage tolerant, both of which are an essential requirement for certification. The architecture supports reconfiguration and graceful degradation. Graceful degradation is a desirable attribute in that when spare resources are exhausted, remaining resources can be assigned to perform functions according to their priority. The distributed processors can be located anywhere on the aircraft where there is space, providing valuable underfloor space for payload.

The major shortcoming of this architecture is its dependence on the global computer. This makes the system vulnerable to faults in the computer. Reconfiguration schemes, based on the use of back-up computers are used in the architecture to mitigate this problem, but such schemes add complexities to the software control configuration. The CSDL fault tolerant processors used on the AIPS are SIFT (Software Implemented Fault Tolerance) and FTMP (Fault Tolerant Multi-Processor). SIFT utilises software for fault detection. This requires the software to be fault free, which complicates the validation process. In FTMP emphasis is placed on bit synchronization (i.e. must ensure that all processors are in lock step) which requires a fault tolerant master clock [Cronin et. al. 1985].

Both the Linear/MUX and the Partially Distributed/MUX architectures have inherent simplicity in that there is a single cable through which all the data propagates and to which all nodes are connected. Furthermore, the centralised control design is the easiest to design, in which a single unit is made in charge of the whole information transfer operation. The advantage of this concept is that it reduces the risks associated with reconfiguration strategies involved in a fully distributed system, since the control complexities of global redundancy and system management are contained in one unit.

A disadvantage of central bus controller approach is that whenever there is a change in the system (i.e addition of new equipment) the controller requires modification and recertification. It is not uncommon for a commercial aircraft to have modification every year for economic and regulatory reasons, and the constant recertification operation that entails can become an extremely costly operation for the operator. For this reason, data bus controllers are not preferred for commercial avionics. A further disadvantage of this approach is that a failure of the controller could result in a single point failure causing the entire system to shut down. The problem could however be overcome, at the expense of complexity, by using multiple/standby controllers which take over the control operation in the event of failure of the primary controller. Indeed, this is the approach adopted in the two architectures and in military aircraft, where the data bus is controlled by a central bus controller, with a number of back-up units. Although the scheme is acceptable for military aircraft, the stringent safety requirement imposed on commercial aircraft make the approach unacceptable even with the various back-up and control reversion schemes.

IMA is a network system in which the processing capabilities are shared by many functions in the network. This significantly reduces the development cost by reducing the functional duplication of hardware and software elements. Since each cabinet is capable of providing multiple functions, a single cabinet is able to replace many individual aircraft functions. This reduces the volume and weight of avionics equipment, providing valuable space and weight for payload. The network also physically decouples processing (i.e hardware) from the peripheral devices (i.e. sensors and actuators) enabling hardware and software upgrades to be made independently. The fault tolerant architecture provides fault detection, fault isolation and automatic reconfiguration, and allows maintenance to be carried out at a central base at a time convenient for the operator. The deferred maintenance allows the airlines to reduce spares inventory, expensive test equipment and maintenance facilities. The architecture also provides flexibility to accommodate evolutionary enhancement of functions and airline-specific avionic installations.

Table 2.1 shows a brief comparison of the four candidate architectures. It is clear that the IMA architecture satisfies all the requirements and is therefore appropriate to incorporate within the avionic systems of the next generation of commercial aircraft; with this in mind IMA is the subject of further evaluation as detailed in the next section.

2.4 Conclusion

In this section four potential candidate architectures were reviewed. Of the four architectures considered, the IMA architecture appears to be the most suitable candidate for the next generation of commercial aircraft as it satisfies all the major requirements expected of an advanced avionics architecture. Furthermore it is also the architecture that has the most potential to be certifiable, as it does not rely on a central bus controller. The IMA architecture is envisaged to provide significant reduction in cost (both development cost and operational cost), weight, volume and parts count.

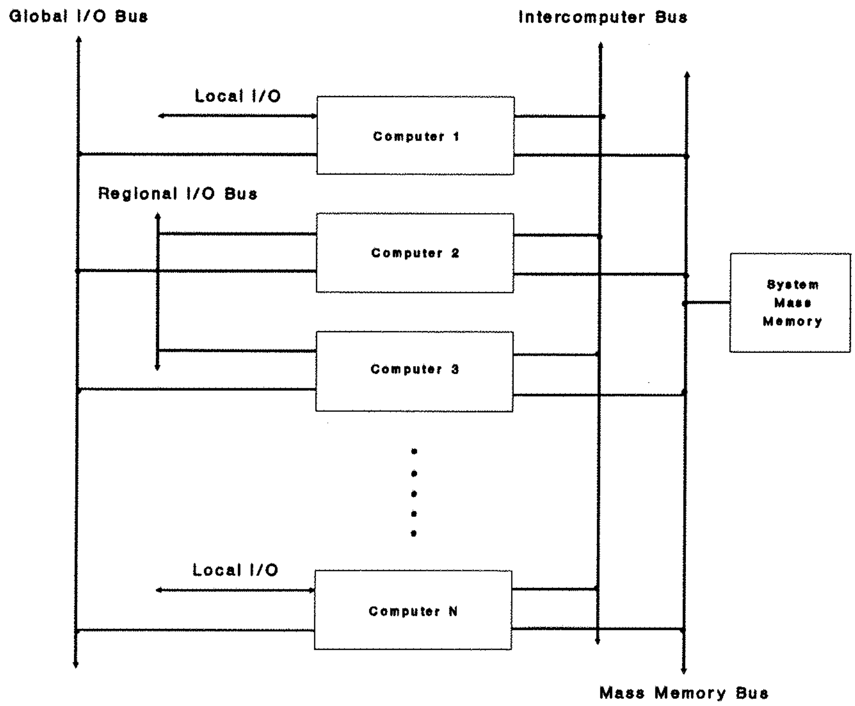


Figure 2.1 AIPS Architecture

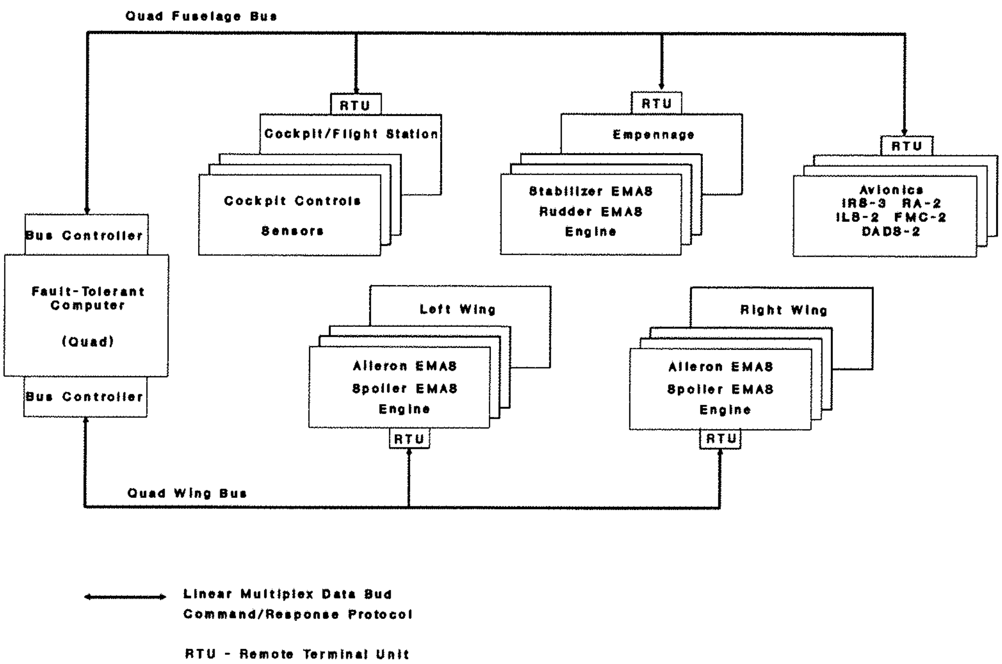


Figure 2.2 Centralised MUX/Architecture

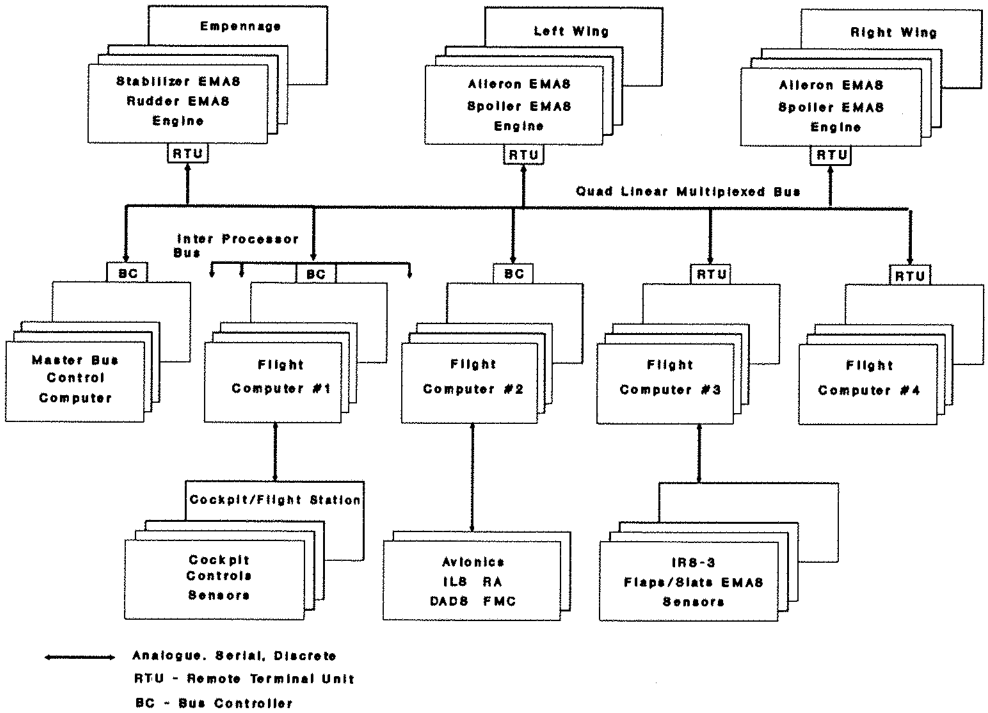


Figure 2.3 Partially Distributed/MUX Architecture

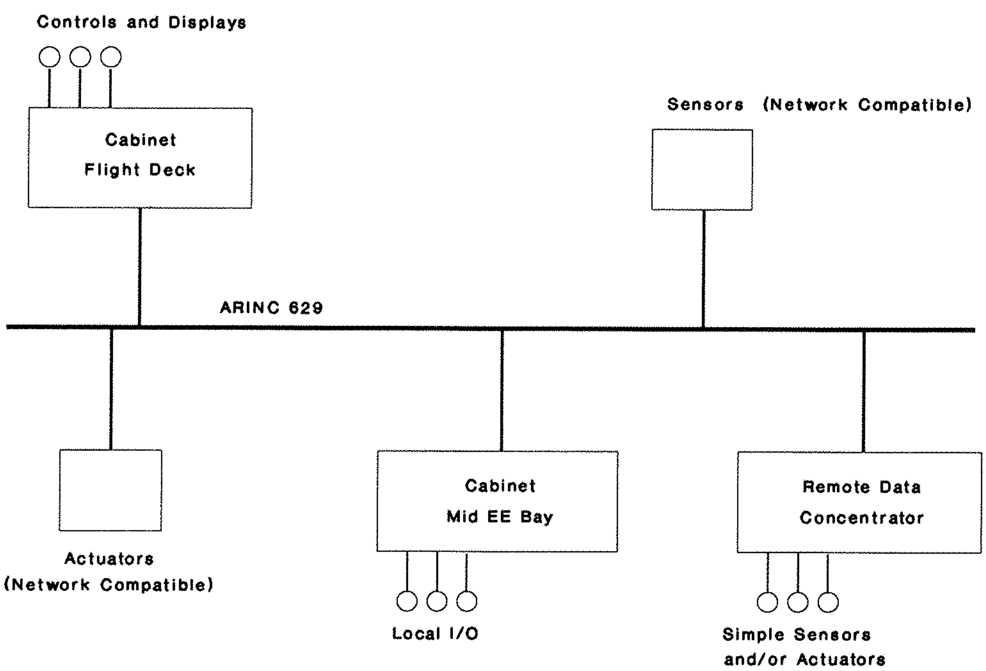


Figure 2.4 Integrated Modular Avionics Architecture

Attributes	AIPS Architecture	Centralised/MUX Architecture	Partially Distributed/ MUX Architecture	IMA Architecture
Fault tolerant	✓	✓	✓	✓
Physically distributed processing	✓	✓	✓	✓
Resource sharing	✓	X	✓	✓
Dynamic reconfiguration	✓	X	✓	✓
Graceful degradation	✓	X	✓	✓
Extensibility	✓	✓	✓	✓
Autonomous operation i.e no central bus controller	X	X	X	✓

Table 2.1. Comparison of candidate architectures

3 Integrated Modular Avionics

3.1 Introduction

Integrated Modular Avionics (IMA) represents the next generation in advanced avionics architecture. The concept is based on a number of cabinets, containing processing modules, distributed throughout the aircraft, interconnected by a high speed data bus. These processing modules are capable of performing multiple avionic functions. Thus a single cabinet is able to replace a number of individual aircraft avionic functions, and few cabinets the majority of avionic functions on a commercial aircraft. The IMA concept provides the potential for significant reductions in development costs as it affects extandability of functions, portability of product development and reduces the functional duplication of hardware and software found in today's avionic systems. Furthermore, the concept also provides the ability to differ maintenance until the aircraft reaches a central maintenance depot, avoiding costly flight delays and cancellations. This chapter describes the IMA concept and highlights the advantages of using such a concept to design the avionics system of the next generation of commercial aircraft.

3.2 The IMA Concept

The IMA concept takes advantage of recent developments in microprocessor design, namely high data throughput and inherent fault tolerance. It makes use of powerful computers for processing of applications software. These computers together with hardware modules are housed in a cabinet forming a subsystem with a common chassis design, common fault tolerant processing, redundant power supplies and flexible aircraft interfaces. Several of these cabinets are interconnected by ARINC 629 data buses which also connect avionic hardware and other peripherals (i.e. sensors, actuators, indicators etc.) outside the cabinet to form an integrated system for performing all avionic functions on the aircraft (Fig. 3.1).

A concern with this kind of system is that since transfer of data takes place in a more shared manner, a centralised failure in the architecture can bring about simultaneous loss of functions utilising shared resources. To overcome this, fault tolerance is applied in IMA to increase resource availability and integrity. In IMA fault tolerance is achieved through redundancy and high integrity monitoring.

Although fault tolerance is required to meet availability and integrity requirement goals in IMA, it also satisfies the airline goal for deferred maintenance. Furthermore, high integrity monitoring also satisfies the airlines desire for improved fault isolation, better maintenance diagnostics and reduced unconfirmed removal rates. Reports indicate that a fully integrated IMA architecture would reduce the number of unconfirmed removals by as much as 6 times over today's Line Replaceable Unit (LRU) based avionics [Morgan 1991].

The resource redundancy required to extend the Mean-Time Between Maintenance Alert/Action (MTBMA) is dependent upon the length of the extend maintenance interval and the statistical probability of successfully completing that interval before total equipment failure [AEEC 1991]. The desired extended maintenance interval is usually dictated by individual airline philosophy. However as guidance, IMA has established for a fully fault tolerant suite, a reliability requirement (or MTBMA) of 15000 hours to first failure, which must not cause loss of function, and 99% probability providing full function for another 200 hours [AEEC 1991]. These requirements are considered to be adequate to eliminate unscheduled removals of IMA common components.

Supporting Technologies

The IMA architecture is based on the application of a number of technologies. The technologies which are used as part of the IMA architecture include:

1) ARINC 659 - Backplane Data Bus

   A data bus designed for medium level data throughput. It is employed for intracabinet communication. (The data bus subcommittee is currently developing the backplane data bus standard).

2) ARINC 629 - Data Bus

   A serial, bi-directional, high speed data bus designed to carry data at 2MHz. It is employed for communication between the avionic cabinets and the sensors, displays and actuators.

3) ARINC 429 - Data Bus

   A serial, uni-directional data bus designed for medium level data transfer. It is employed wherever practical to transfer digital information to the IMA.

3.3 Fault Tolerance

Fault tolerance is the ability of a system to provide continual operation in the presence of a limited number of faults in either hardware or software. The IMA concept employs a high degree of fault tolerance to ensure high functional integrity and high functional availability, both of which are an essential requirement for certification. In an high integrity design, some form of redundancy is required to provide the ability to continue operation in the presence of a fault. This is necessary if it is a requirement to survive any single fault, and most of all if functional capability must be maintained.

Fault tolerance can be achieved by a number of hardware and/or software methods. In hardware, fault tolerance can be achieved by:

1) Input checking - where the input is subjected to a reasonable check. In the case of two inputs, the inputs are compared and if they differ by more than a preset value, the inputs are rejected and the previous input is used. For three or more inputs, the inputs are voted and the majority value is accepted.
2) Output checking - procedure similar to input checking, but applied to the output. In the case of outputs from two processors, the outputs are compared and if there are discrepancies the processors are taken off-line for diagnostic testing. For three or more processors, the outputs are voted and the majority value is accepted.
3) Partitioning - involves the physical and systematic separation of functions to prevent fault propagation.
4) Mid-value select - only applicable for systems having three or more sources of the same information. The basic technique involves the omission of the maximum and the minimum values of the input parameters and the acceptance of the remaining values.
5) Redundancy - involves the replication of hardware elements.
6) Dissimilar hardware - hardware built to the same specification by different vendors.
7) Monitoring - generally involves comparing pairs of processors, parameters relative to a fixed threshold and detection of an event.
8) Reconfiguration - the process of changing the resources which are active within the system.

Fault tolerance in software can be can be achieved by:

1) N - Version programming - two or more alternative versions of the software are executed concurrently, and the selection of a final result is accomplished using a voting system or decision algorithm.
2) Recovery blocks - one version of the software is executed. Upon the detection of an error by an acceptance test, a prior state which is known to be fault free is restored, and an alternative version of the software is loaded and the software execution is continued.
3) Consensus - where outputs from n-version software are compared. If two or more outputs agree the output is accepted.
4) Monitoring - analogous to hardware monitoring.
5) Reconfiguration - the process of transferring or copying software from one unit to a spare in the event of a failure.
6) Input checking - involves the use of 'small' programs to check the validity of input data.
7) Output checking - a procedure similar to input checking where the output is subjected to a reasonableness check to ensure that the results produced look realistic.

8) Partitioning - involves the separation or 'brick walling' of software to prevent fault propagation.

3.4 Data Network

In an highly integrated system like the IMA, when communication is desired between two heterogeneous (i.e. different vendor or same vendor different models) computers, the task of developing a software to support the communication can become very difficult. The difficulty is due to the fact that different vendors use different data formats and data connections in their machines. To avoid this pitfall, the data link protocols in the IMA are based on an OSI (Open Systems Interconnection) reference model.

The OSI model defines the standard for linking heterogeneous computers. The model is based on the principle of 'layering'. The communication functions are partitioned into layers, each layer performing a subset of the function required to communicate with another system. The OSI reference model employs a seven layer approach (Fig. 3.2). The functions of each layer are summarised in Table 3.1. This assures interoperability of communication among a variety of computing elements connected to different data buses, Local Area Networks (LANs) and other communication media.

For air-to-ground and ground-to-air communication The concept employs a communication system called the Aeronautical Telecommunication Network (ATN). Typical examples of data sources in the ATN system include Satellite (SATCOM), Mode S Transponders and Aviation VHF (Very High Frequency) Packet Communication equipment. These systems allow communication between ground and air-based communication systems that are a part of global network.

In a highly integrated architecture like the IMA some form of interconnection mechanism is required to connect the various systems with their subsystems. The interconnect devices commonly used are repeaters, bridges, routers and gateways. Repeaters are used to extend the physical boundaries of similar network. The repeaters function by receiving a message and then retransmitting them at their original strength. Bridges are used to interconnect computer networks which have different physical layers but common data link and upper layers. Routers have the capability to interconnect computer networks with different physical layer as well as data link layers, but common network and upper layers. The gateway is the most versatile device of them all. It is capable of interconnecting networks with completely different protocols and architecture, and offer maximum capability for interconnection. However in IMA only bridges and gateways are being considered for interconnecting the various system networks.

3.5 Total IMA System Level Description

The IMA system consists of a number of cabinets distributed throughout the aircraft, interconnected by ARINC 629 data buses. Each cabinet contains processing modules and local Input/Output (I/O) to sensors, actuators and indicators which cannot be directly interfaced to the data bus (Fig. 3.3). The processing modules are capable of performing multiple avionic functions simultaneously. Figure 3.4 illustrates an example of the wide mix of functions that can be handled by this architecture. It should be noted that the figure only illustrates the architectural concept and the diversity of functions that can be handled by any one cabinet or by the overall system, and does not represent any ideal or specific functional distribution.

IMA employs a high degree of fault tolerance to ensure that the system has high integrity and functional availability. Functional availability requires some form of redundancy. In IMA redundancy is provided both at functional level and component level. At functional level, redundancy is achieved by having multiple paths for the data from its source to the sinks. This reduces the number of components required for a given level of function availability. At component level, redundancy is achieved through replication of components which maintains the functionality of the system in the event of failures. However to minimise the number of redundant components, the architecture employs fault containment areas throughout the architecture to allow other components in the system to operate in the presence of a fault. As a minimum each LRM is expected to constitute a fault containment area.

3.6 Components of the System

The components of an IMA system includes: cabinets, sensors, actuators and indicators with direct data bus interface, remote data concentrators for sensors etc. which do not have a direct data bus interface, local I/O and ARINC 629 data buses to provide data paths.

The cabinet houses the functional modules and provides the computing resources for all the application that resides in the cabinet. In addition the cabinet may house the I/O for local devices. The cabinet also contains a separately enclosed compartment called the backplane. The backplane is divided into three zones. The first zone of the backplane interface the aircraft wiring to the physical backplane. The second zone is dedicated to the transfer of all intermodule traffic and the third zone is used for power distribution.

The functional modules in the cabinet are packaged as Line Replaceable Modules (LRMs). The ultimate aim of IMA is to make the aircraft interface either common to all modules or configurable such that the number of modules and their position in the cabinet need not be fixed during the aircraft design. The cabinet itself provides the basic mechanical structure and environmental control/isolation for the modules. But it does not provide any electrical services such as power transformation/filtering and bus control monitoring. The individual LRM provides these services. The power to the modules is provided by the power supply LRM. A functional view of this is shown in Figure 3.5. The function of these modules are:

1) Core processor
2) Standard I/O
3) Special I/O
4) Power supply module
5) Bus bridges
6) Gateway

The core processor provides the computing power for the cabinet. The number of core processor is dependant on each system implementation. In order to run many different functions the core processor would be partitioned so that one function will not adversely affect the other.

The number of cabinets required is determined by a variety of factors. These include the quantity of I/O, the processing throughput, the spatial location of the external interface elements, the desired grouping of processing task and the system redundancy level of certain applications. However there is nothing in the basic concept, design or implementation of the IMA architecture that restricts the system designer's choice in allocating the function except the throughput capacity and the number of I/O LRM which can be physically located in a particular cabinet. The location of the cabinet is established by trading the convenience of the location versus its proximity to the external interface elements. In general only the I/O interface of the cabinet affects the choice of its location, since this affects the possible saving in wire, connectors and manufacturing costs.

The cabinets are interconnected by standard ARINC 629 data buses. For intracabinet communication ARINC 659 data buses are used. Bus bridges are used to transfer data to and from data buses of the same type. Although the primary data bus in IMA is ARINC 629, where it is practical, ARINC 429 data buses are used to transfer information to the IMA, such as in areas of low data concentration in the aircraft.

In IMA the sensors data can either be analogue or discrete. The I/O LRM is used to transform the analogue or discrete data into digital data to be transferred to the core processor via the backplane bus. In situations where this cannot be met by I/O LRM, special LRM are used.

The power supply LRM provides power isolation and power conversion from standard aircraft power to a level that is required by the modules. The cabinet power is distributed to a number of user modules in the cabinet via individual output (O/P) line. These O/P lines have independent fault protection, so that a fault in one user module does not affect the power supplied to the other modules.

Technology permits remote electronics to be inherent in sensors, actuators and indicators to perform signal conditioning, buffering and low level control. These devices include ARINC 629 compatible actuators and sensors which can be directly interfaced to the ARINC 629 data bus. The IMA approach encourages the development of ARINC 629 compatible devices as they allow a greater degree of design and modification freedom. However not all actuators/sensors are network compatible. This is accepted impractical in all cases:

1) Interface electronics being too expensive for the particular actuator/sensor, e.g. proximity switches on doors.
2) May be constrained by physical factors: space and environment.

Non network compatible devices, also referred to as 'simple' (dumb) devices are usually connected to a data concentrator for transmission on the ARINC 629 data bus, and if this is not feasible they are connected directly to the cabinet via standard I/O modules. Data concentrators serve a number of simple devices in close proximity. They convert data into digital form which is transmitted over the ARINC 629 data bus. They can also monitor the health of sensors and actuators. The application of data concentrators has the advantage of reducing the number of discrete wires in the aircraft.

3.7 Software Architecture

Figure 3.6 shows a schematic of the overall software architecture employed in IMA. The key components of the system are:

1) Application software which performs the avionic functions
2) Core software which provides a standard and common environment in which application software can run

The core software is divided into:

1) Operating system - which manages the logical responses to application demands. Its functions include allocating processing time, communication channels and memory resources.
2) Hardware Interface System (HIS) - which manages the physical hardware resources on behalf of the operating system.

As shown in Figure 3.6, the application is interfaced to the operating system via the APEX, the Applications/Operating system interface. The APEX interface defines the common environment for application software to communicate with the operating system. The operating system is interfaced to the core hardware via the COEX, the Core hardware and Hardware Interface System/Operating system interface. The COEX interface specifies the hardware capabilities required to support the operating system.

All the data flow between the application and hardware is channelled through and controlled by the operating system via the defined interfaces. These interfaces are specifications rather than software or hardware. This in effect decouples the application from the hardware providing several benefits:

1) Allows independently produced applications to run on the same hardware.
2) Reduces the effects of software changes on hardware.
3) Improves software portability.
4) Reduces the cost of ownership of software.
5) Reduces the certification cost by separating the verification into independent tasks associated with hardware, software and the operating system.

The programming language recommended for use in IMA is Ada. Since Ada is also the preferred language of the airline community the Airlines Electronic Engineering Committee (AEEC) have adopted it as the standard high order language for all software. IMA allows software to be loaded onboard the aircraft.

3.8 Discussion

It is envisaged that the adoption of IMA architecture will increase the acquisition cost, i.e. development and production costs. However the increase in acquisition cost is expected to be offset by significant reductions in ownership cost, i.e. operating and support costs. Thus the life cycle cost (i.e acquisition cost + ownership cost) is expected to be lower for the new system when compared with the current architecture.

The acquisition cost is expected to be dominated by the development costs of the application software and the software associated with the implementation of the architecture. But it should be noted that software development costs dominate virtually all current avionics project costs. However, in IMA potential reductions in software development cost can be expected through software re-usability and portability.

The incorporation of redundancy, enabling deferred maintenance, for flight safety is also expected to add to the acquisition cost; this of course already applies to existing avionic systems. However reductions in acquisition cost can be expected through competition. The basic IMA concept allows the utilisation of common and standard flight hardware. This provides the equipment manufacture with increased marketing opportunities fostering competition - a potential factor in reducing acquisition cost. Reductions in development cost can also be expected through resource sharing. Since the IMA concept allows common resources to be shared between different avionic functions, the functional duplication of hardware and software elements found in current avionic systems is kept to a minimum (Fig. 3.7).

The standardization of the interface between the application software and the core processors decouples the application from the hardware. This decoupling of software from hardware provides the potential for reducing the verification cost by separating the verification procedure into independent operations associated with hardware, software and the operating system. The seperation of hardware from software also provides extreme flexibility for future capability upgrades and airline-specific avionic installations. This enables the upgrading cost to be kept to a minimum.

The ownership cost, in general, is dominated by the maintenance and support costs. However the adoption of IMA is expected to result in a significant reduction in ownership cost. The fault tolerant architecture provides fail operational capability through any first failure and allows maintenance to be deferred until a convenient time. The advantages derived from deferred or scheduled maintenance are many.

Deferred maintenance reduces the number of unscheduled maintenance and avoids extremely costly flight delays and cancellations. It also has implications on manning levels and provisions of spares. Deferred maintenance provides the potential for reductions in intermediate shop level maintenance providing cost savings on maintenance personnel and significant savings on test equipment. It also reduces the number of spares to be kept on the airfield reducing the cost of spares inventory.

The high integrity monitoring inherent in the architecture provides improved diagnostics capability. This should reduce the number of unconfirmed removals. A LRM found faulty can be replaced from the 'small' stock held at the airfield, lowering the flight line avionics and maintenance personnel. Unconfirmed removals refer to the 'no fault found' removals. The cost incurred in unconfirmed removals is attributed to the time taken to test and return good boxes for servicing and the cost of necessary spares to support this operation. Reports indicate that a dominant contributor to the cost of maintenance is unconfirmed removals [AEEC 1991]. Improved diagnostics would also allow swift flight maintenance action which result in higher dispatch availability.

The adoption of IMA architecture is also expected to offer significant savings in weight and volume. Weight savings stem from the use of ARINC 629 data bus and the 'avionic cabinets'. The utilization of the bidirectional ARINC 629 data bus result in a lower wire count providing substantial reduction in aircraft wiring, weight and build time, with consequent significant savings in production cost. Boeing expect a wire weight saving of about 1150 lbs on a 757 sized aircraft [Shaw 1988].

The weight of the cabinet modules (LRMs) are dependant on the manufacturer's implementation and thus are difficult to specify. Consultation with Smith Industries indicate that components and Printed Circuit Board (PCB) would be the dominant contributors to the weight of the modules. However weight savings can be expected from the use of cabinets. Since each cabinet is capable of providing multiple avionic functions, a single cabinet is able to replace a number of aircraft functions, lowering the weight and volume of the avionics on the aircraft. Initial architectural studies carried out by Boeing indicate that for a typical avionics and electrical system functions, about 50 individual LRMs can be integrated in eight Integrated Avionic Computer System (ICAS) cabinets with an estimated saving of 25% in recurring cost, 30% reduction in weight, 46% reduction in volume and 84% reduction in number of individual power supplies [Sutcliffe 1988]. (Note: The ICAS concept is very similar to the IMA concept). The weight savings resulting from the wire and the cabinets can be used to increase the payload of the aircraft, providing the operator revenue. Alternatively the savings in weight and volume can be used to increase the fuel load, giving the aircraft extended range.

For IMA to be cost effective, it depends on the application of a number of advanced technologies: high throughput, fault tolerant microprocessors, structured higher order programming language, VLSI (Very Large Scale Integration) circuit that make possible the development of standard fault tolerant I/O modules and an advanced data bus system. With advances in digital electronics, fault tolerance and software, the technologies necessary to design, build and integrate this type of architecture are already in place.

The integrated modular avionic system is dominated by software and therefore the reliability of software is paramount. From experience with a wide range of systems and equipment which are dependent upon software, it is clear that software error is one of the major causes of system unreliability. No matter how carefully the software is designed it is extremely difficult to guarantee that the software is error free. The reason being that, firstly, with increasingly complex systems the designer is unlikely to be able to foresee all the possible conditions, and secondly, program bugs do not always make themselves evident immediately once a program is installed, but may remain dormant in the program for months or more until the triggering condition occurred. In view of this software fault tolerant techniques have been developed to enhance the reliability of software. The two most commonly used approaches are the recovery block and n-version programming. Both methods make use of multiple version of independently developed software - the idea being that it is unlikely that different programmers will code the same error for the same function.

One of the drawbacks of the recovery block method is the time delay involved in reconfiguration. This makes the approach inappropriate for use in real time avionics application such as flight control systems. However there are several examples of the application of n-version programming in critical commercial avionics. Recent examples include an A310 secondary flight control system, the 737-300 AFDS and the Boeing 757 and 767 yaw damper system.

N-version programming, however, is inadequate against preventing design related failures. A number of survey into the source of errors in software indicate that the majority of errors are due to deficiencies in software specification (i.e design related) rather than coding mistakes. Furthermore studies carried out in the United States indicate that independent programmers could make the same mistakes in implementing a particular difficult function, especially if the specification is ambiguous. This however does not mean that n-version programming is unreliable, but it implies that the n-version programming is not as highly reliable as predicted.

To reduce the incidence of design related errors, structured software design methodologies (i.e. structured programming) and comprehensive verification and validation tests have been suggested. Structured programming involves a top-down design approach where the complex software is broken down into simple, understandable building-blocks, so that the structure is more manageable and visible, and hence less error prone. For example, the program may be expressed as a single statement, a representation which can be decomposed into a number of sub-tasks, each of which is independent of the others and its effects perfectly definable. The consequence is that, assuming that the sub-tasks are executed in the correct sequence the data structure will be left in a definable state between sub-tasks. By further repeated decompositions the whole program may be represented in terms of a set of low level tasks, the effects of which is totally predictable. Hence the total program execution can be predicted and any undesirable side effects eliminated. Therefore it is necessary to make use of these modern methods of software design, implementation and test to eliminate design related software errors.

A major concern of IMA is certification. Since functions with different criticality are processed in one cabinet there is always the potential for undesirable interaction between separate avionic functions. As a result questions have been raised regarding the certification of the IMA concept, particularly with respect to common mode failures, partitioning integrity, and on board software loading. Boeing have had preliminary discussions with the Federal Aviation Authority (FAA) on this matter, and envisage no formidable problems, provided that strict fault tolerance and robust hardware and software partitioning are adhered to.

In recent years the IMA concept has been extensively discussed in the Systems Architecture and Interface (SAI) subcommittee of the AEEC (Airline Electronic Engineering Committee) and their recommendations have been well documented in ARINC Project Paper 651. Although, more recently, the concept has been finally approved by the AEEC, the early IMA systems are likely to only involve limited number of systems, so as to contain the development risks associated with the new concept. This is exemplified by considering the Boeing 777 aircraft, the first to implement ARINC 651, in which the Airplane Information and Management System (AIMS) is designed based on the IMA concept. In the AIMS architecture the conventional LRUs, which typically contain a single function, are replaced with dual integrated cabinets which provide the processing, power supply, the I/O hardware and software to perform several functions [Baily 1990, Hopkins 1991, Morgan 1991] The functions performed by the AIMS include flight management, display (EFIS - Electronic Flight Instrument System / EICAS - Engine indication and Crew Alerting System), central maintenance, airplane conditioning monitoring, communication management, data conversion gateway (ARINC 429/629 conversion) and engine data interface.

In order to realise the full benefits of the IMA architecture, the concept should be applied as a whole, i.e. the application of ARINC 629 data buses, IMA cabinets and smart units (i.e. actuators and sensors). The benefits to be gained from implementing the aircraft avionics using the IMA concept are summarised in Table 3.2.

3.9 Conclusion

In this section the IMA concept was addressed in detail. The concept takes advantage of recent developments in microprocessor design to yield a distributed system which can evolve throughout the life of the aircraft. By utilising common chassis design, common fault tolerant processors, redundant power supplies, the architecture is able to provide significant cost benefits that are not realisable with current avionic sytems architecture.

The IMA concept provides significant benefits to the airlines, the airframe manufacture and the equipment supplier. The benefits to the airlines include:

- Ability to defer maintenance
- No maintenance required away from main base
- No maintenance required during turnarounds
- Improved fault isolation
- Reduction in no fault found removals
- Reduction of test equipment
- Reduced spares requirement
- Reduced cost of ownership

The benefits to the airframe manufacture includes:

- Reduced avionics cost, weight and volume
- Reduced wiring
- Reduced development cost
- Reduced manufacturing cost
- More flexible architecture to accommodate technological changes and airline- specific avionic installations
- Ability to add or modify many systems strictly through onboard software loading

The benefits to the equipment supplier includes:

- Increased marketing opportunity
- Reduced development cost
- Reduced manufacturing cost
- Easier to offer system upgrades
- Ability to add or modify many systems strictly through onboard software loading

As with all new systems, problems will be encountered with the IMA system, both in the development of the system and after introduction. However once the 'teething' problems have been eventually resolved, the concept would revolutionise avionic systems design.

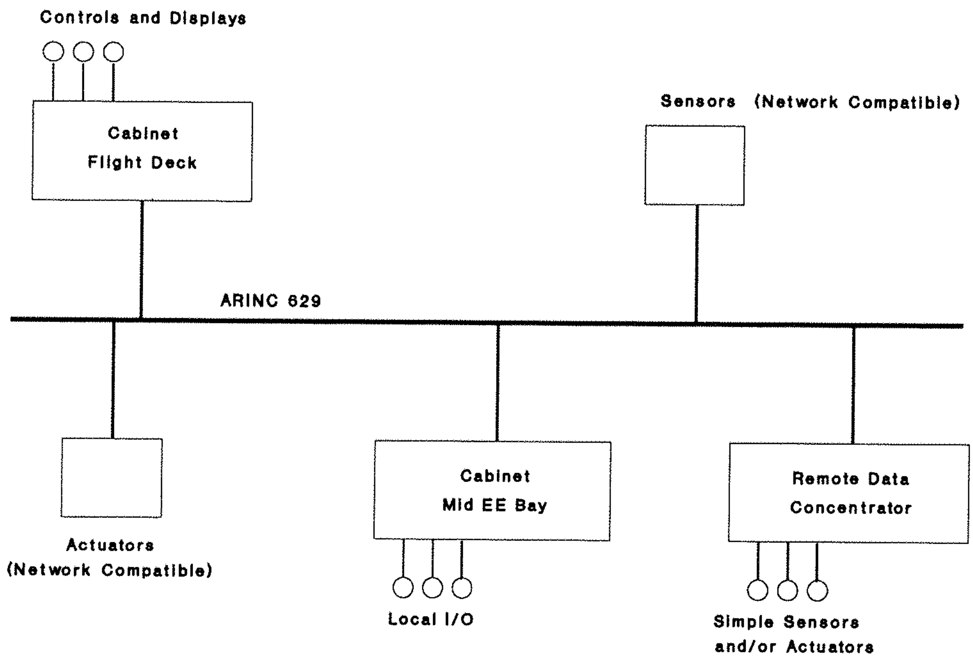


Figure 3.1 Integrated Modular Avionics - System Overview

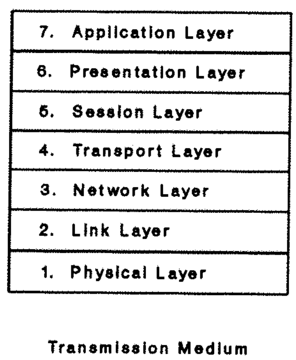


Figure 3.2 Layers of the OSI Model

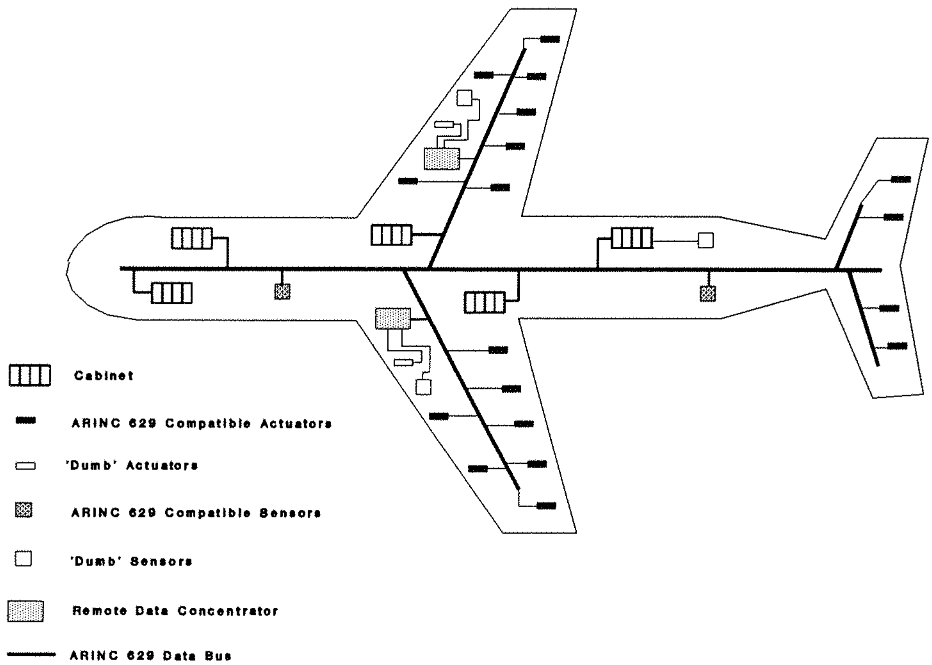


Figure 3.3 Basic Components of a typical IMA configured Aircraft

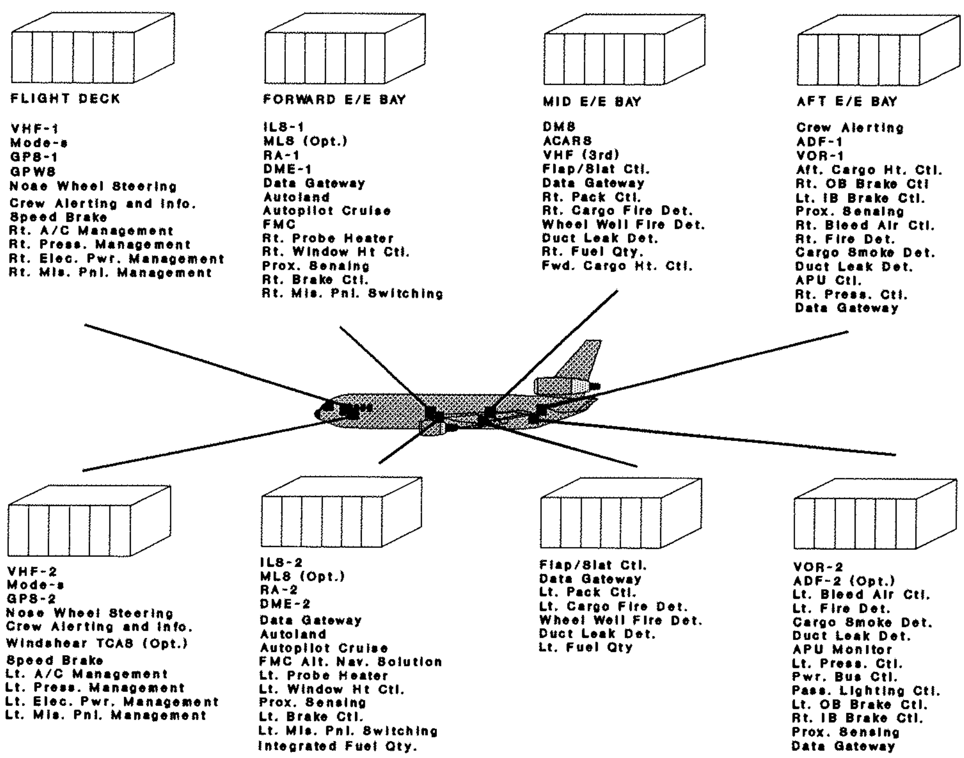


Figure 3.4 Functional Distribution – An Example

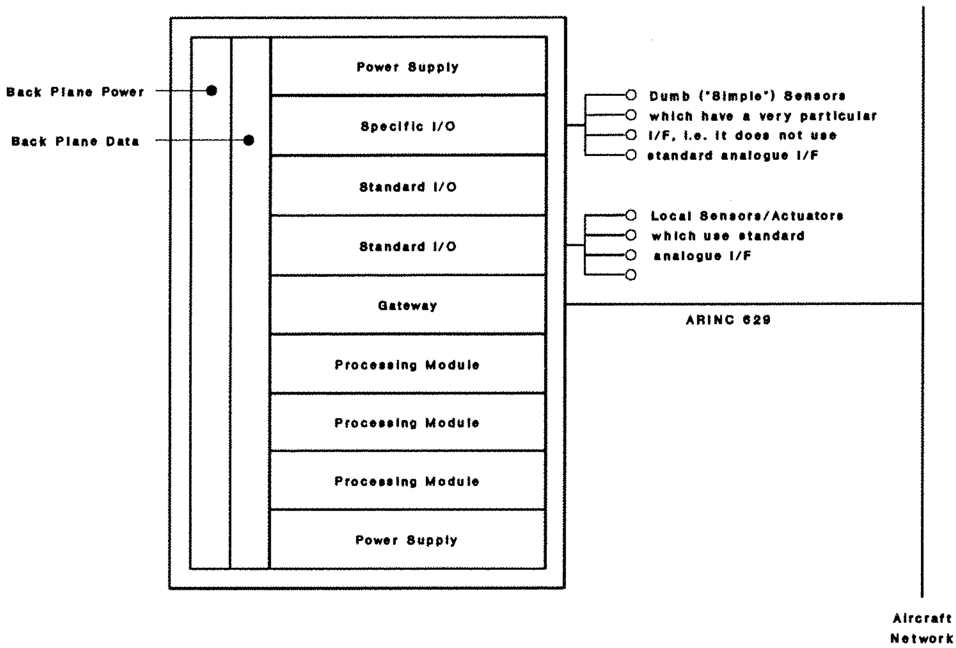


Figure 3.5 Generic Cabinet

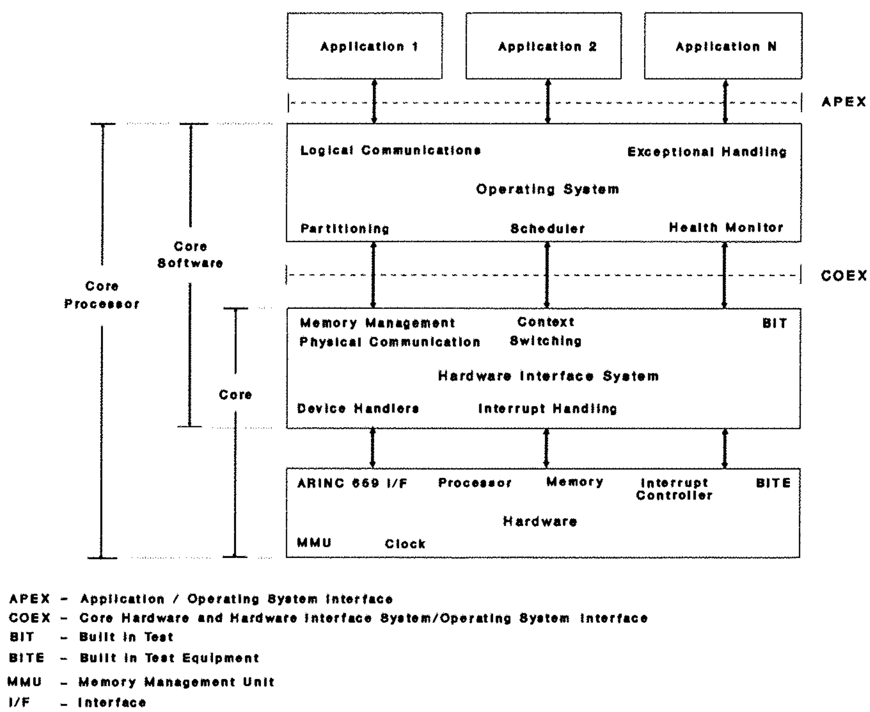


Figure 3.6 IMA Software Architecture

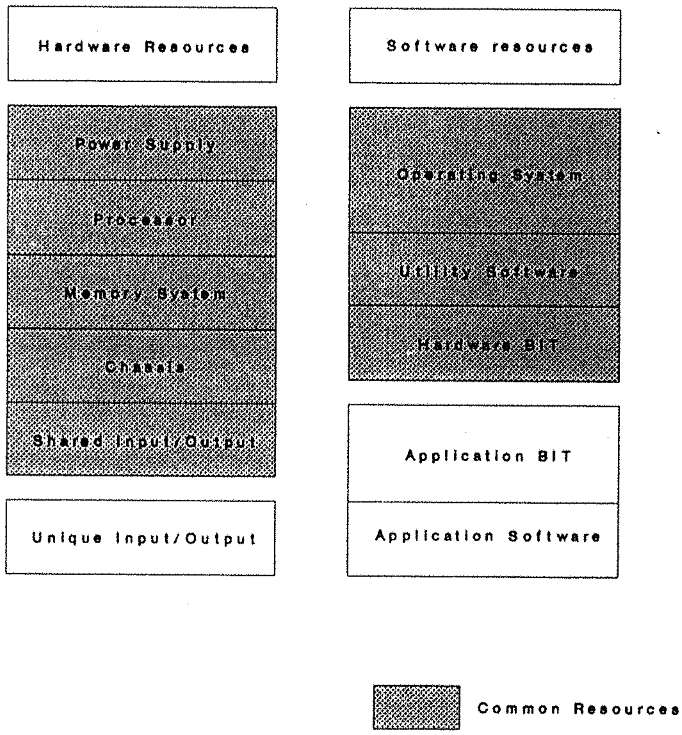


Figure 3.7 Components of a typical LRU

1.	Physical	Concerned with transmission of unstructured bit stream over physical medium; deals with the mechanical, electrical, functional and procedural characteristics to access the physical medium.
2.	Data Link	Provides for the reliable transfer of information across the physical link; sends blocks of data (frames) with necessary synchronization, error control, and flow control.
3.	Network	Provides upper layers with independence from the data transmission and switching technologies used to connect system; responsible for establishing, maintaining and terminating connections.
4.	Transport	Provides reliable, transparent transfer of data between end points; provides end-to-end error recovery and flow control.
5.	Session	Provides the control structure for communication between applications; establishes, manages, and terminates connections (sessions) between cooperating applications.
6.	Presentation	Provides independence to the application process from difference in data representation (syntax).
7.	Application	Provides access to the OSI environment for user and also provides distributed information service.

Table 3.1 Open System Interconection (OSI) Layer

Aspects	Comments
Network system with shared resources	• Reduces aircraft wiring, weight and build time • Decouples hardware from software: - Flexible to accommodate hardware and software upgrades
Cabinet with multifunctional processing capability	• Single cabinet is capable of replacing many individual aircraft functions: - Reduces functional duplication of hardware and software - Reduces weight and volume of avionic equipment
Fault tolerance	• Increased reliability and maintainability • Deferred maintenance: - Reduces number of unscheduled maintenance - Reduces intermediate shop level maintenance: - Savings on maintenance personnel - Savings on test equipment - Reduction in spares
Multivendor support	• Customer not locked to one vendor
Flexibility of equipment location	• Optimises equipment space
Compatible with fibre optics	• Provides EMI/RFI immunity

Table 3.2 Benefits of IMA

4 Data Bus Standards for Avionic Application

4.1 Introduction

In order to realise the full potential of an integrated system like the IMA architecture, an advanced data bus is required. Data buses may be uni-directional or bi-directional. However for use in an advanced integrated avionic system, the data bus must be bi-directional, high-speed, flexible, inexpensive, and most important of all, highly reliable. Furthermore the bus must not rely on a controller if it is to be considered for use on commercial aircraft. This chapter provides a review of digital data buses in avionic use today, and in doing so, identifies the qualities needed for a data bus to serve a modern commercial aircraft. The chapter also provides a brief overview of the principal characteristics and features of the new, two-way, high-speed data bus, the ARINC 629 specifically designed to transfer digital data between avionic equipment on a commercial aircraft.

4.2 Current Avionics Bus Standards

A survey of digital buses in avionics use today indicate that there are essentially four standard digital buses: the ARINC 429, the Commercial Standard Digital Bus (CSDB), the MIL-STD-1553B bus and the Avionics Standard Communication Bus (ASCB) [Card 1983, Chun 1981, Eldredge 1987, Sample 1989, Stainslaw 1984, Thomas 1983]. Both the ARINC 429 and CSDB are uni-directional broadcast type buses whilst the MIL-STD-1553B and ASCB are bi-directional 'controlled' buses.

Of the two broadcast type buses, the ARINC 429 is by far the most commonly used data bus on commercial aircraft. The ARINC 429 is a single source, multi-sink uni-directional data transmission standard [ARINC 429 Spec. 1990, Hicks 1988]. The data bus consists of a single, twisted, shielded cable pair, with the shield grounded at both ends and at all breaks. Figure 4.1 shows a simplified ARINC 429 architecture. The types of equipment employed on an ARINC 429 system are transmitter (source), receiver (sink) or transmitter and receiver. All data is transmitted in one direction only, and only one transmitter is allowed to talk on a single data bus, though many receivers (up to a maximum of 20) may receive that information.

There are two data rates associated with ARINC 429, 100 Kbit/s for high speed operation and 12-14 kbit/s for low speed operation. The two data rates, however, cannot be used on the same bus. Data is transmitted in Bi-polar RZ (Return to Zero) format. An example of Bi-polar encoding is illustrated in Figure 4.2.

In ARINC 429 all information is sent as 32 bit words (Fig. 4.3), with a parity bit included (odd parity required). In any 32 bit word, the bit numbers are allocated as follows:

- Bits 1-8 are a label to define the type of data (e.g. ILS frequency, DME distance).
- Bits 9 and 10 are source/destination identifiers.
- Bits 11-28 or 29 (i.e. 18 or 19 bits in total) are for actual data.
- Bits 29-31 or 30-31 are sign/status matrix which indicates the sign (plus, minus, north, south etc.) of the data and the status of the transmitter (failure warning, no computed data, normal operation), depending on whether the data is ISO-alphabet, BCD or binary.
- Bit 32 is the parity bit.

In ARINC 429, the data buses are in a constant broadcast mode of operation, and the data is transmitted with an associated source/destination identifier. All terminals on the bus 'listen' to the bus traffic and decode the data assigned to them. A parity bit is transmitted as part of each word to permit simple error checks to be performed by the sinks.

Although ARINC 429 is the most commonly used data bus in commercial aircraft, it is not without its limitations:

1) Uni-directional - higher wire count than a bi-directional data bus
2) Limited bandwidth (12-14 Kbit/s or 100 Kbit/s)
3) Not representative of the current state-of-the-art technology
4) Not exhaustive enough in the specification of labels needed in an integrated system

With the avionics system becoming increasingly integrated the limitations of the ARINC 429 have become more apparent. It is envisaged that the number of interlinks would soon reach its limit as to what can be accommodated in terms of weight, volume and certification. To ease this problem, it is clear that some sort of bi-directional data bus is required.

Bi-Directional Data Buses

A bi-directional data bus enables multiple equipment to be connected to a common bus and allows communication to be established between these equipment in both direction. The advantage of such an approach is significant weight reduction in the wiring of complex aircraft. One of the best examples of an early bi-directional data bus is the MIL-STD- 1553A, first defined in 1975 [Crossgrove 1980, Hall 1986]. This bus has now been superseded by MIL-STD-1553B, first issued in 1978, and is indeed the data bus which is used on military aircraft today [Bracknell 1988]. The bus consists of a single, twisted, shielded cable pair, with the shield grounded at either ends and at all breaks. The bus is terminated by resistors, and along its length equipment are connected by transformer coupled stubs (Fig. 4.4). Up to a maximum of 32 stubs can be accommodated on a single bus, with the total bus length not exceeding 100m, and the stub length not exceeding 6.1m. Information is carried at 1Mbit/s using Manchester Bi-phase pulse coded data[1] (Fig. 4.5). All information is sent as 16 bit words, with a parity bit (odd parity required) added, and preceded by two synchronization pulse which occupy a time equivalent to 3 bits, i.e. a total of 20 bits per word (Fig. 4.6).

In the MIL-STD-1553B data bus system, a single terminal is responsible for the transfer of information between the various terminals. The main concern with this approach is the architecture's dependence upon a master controller. This makes the entire system vulnerable to faults in the avionic processor which has assumed the bus management task. This could, however, be over come at the expense of complexity, by employing multiple/standby controllers which take over the control operation in the event of failure of the master controller. But due to the stringent safety requirements imposed on commercial aircraft, this approach is unlikely to be certificated for use. Furthermore any change or addition to the equipment on the bus requires a change to the software in the bus controller. Since it is common for airlines to modify the avionics on their aircraft annually for economic and regulatory reasons, the recertification involved can become an extremely costly operation. For these reasons bus controllers are considered unacceptable for use commercial aircraft.

The Avionics Standard Conmmercial Bus is similar in architecture to the MIL-STD-1553B with minor difference in data rates (0.667 Mbit/s as opposed to 1Mbit/s as in MIL-STD-1553B). Although the bus reduces the wiring over the ARINC 429 bus, like the MIL-STD-1553B it also employs a bus controller which is considered unacceptable for commercial avionic application.

It is therefore clear that of the bus standards that exist today, none appear to be attractive to serve the next generation of commercial aircraft in that they are either uni-directional, with limited data bandwidth or they employ bus controllers.

4.3 Candidate for the Next Generation of Commercial Aircraft

The data bus for use in an advanced integrated avionic systems must be bi-directional and highly reliable with sufficient data rate to accommodate the integration of the entire avionic suite on the aircraft. Furthermore it should not rely on a bus controller, and should be flexible enough to accommodate changes. In summary for a data bus to serve the next generation of commercial

---

[1]The Manchester Bi-phase code uses the phase of a square wave signal to indicate a '1' or a '0'. A 1 is encoded as a high-to-low transition and a 0 as a low-to-high transition. Each bit has a transition at its centre (Fig.4.4).

aircraft, it should have the following qualities [Card 1983, McGough 1986, Ray 1986, Stainslaw 1984]:

1) The bus must be bi-directional to achieve minimum wire count
2) Bus controllers must be avoided from the single point failure and system inflexibility point of view
3) The data rate should be sufficient (i.e. high-speed) to handle the entire cockpit integration, including radar and digitized audio
4) The data specification format, i.e. labels, should be broad enough to cover the range of equipment that are expected in the next 10-15 years due to advances in technology
5) The Bus should be flexible to accommodate future capability upgrades and airline specific avionic installation
6) The data bus technology should not be peculiar to the avionics industry and should allow the use of reasonably priced commercial hardware for input and output.

4.4 ARINC 629

The ARINC 629 data bus system is the new, bi-direction, high-speed data bus, specifically designed to transfer digital data between avionics equipment on a commercial aircraft [Herzog 1991, Shaw et. al. 1986, Shaw 1988, Stainslaw 1989, Subra 1990]. Based extensively on the Boeing's DATAC (Digital Autonomous Terminal Access Concept), the ARINC 629 bus system consists of a physical bus and stubs connecting the bus to each terminal. The terminals are connected to the stubs through a Serial Interface Module (SIM) and the stubs to the bus medium via a coupler. Up to a maximum of 120 terminals may be accommodated on a single bus, with the total bus and stub lengths not exceeding 100m and 15m respectively (Fig. 4.7). The bus media may be electrical (current mode or voltage mode) or fibre optic. However to-date only the detailed characteristics of the current mode implementation are detailed in ARINC specifications.

The ARINC 629 bus system employs a linear topology with a protocol which can be described as Carrier Sensed for Multiple Access with Collision Avoidance[2] (CSMA-CA). Information is carried at 2 Mbit/s using Manchester Bi-phase pulse coded data, and all information on the bus are sent as 20 bits word strings:

1) 3 bits to distinguish the start of the word, and whether it is a label or data

[2]The CSMA-CA approach allows a device to transmit at random. However before transmitting it 'listens' for an acknowledgement from the receiver interface unit. If an acknowledgement is not received, the transmitting interface unit assumes that a collision has occured and after a random time delay it retransmits.

2) 16 bits for label or data value/code
3) 1 parity bit

Unlike MIL-STD-1553B and ASCB, the ARINC 629 has no bus controller. Instead the bus access control is distributed among all the participating terminals which autonomously determine the transmission sequence. This enables the bus to operate continuously, irrespective of the number of failed units. Furthermore it also enables the addition and removal of units on the bus with ease, without causing an impact on the communication already existing.

Two types of bus access protocols have been defined for the ARINC 629: the Basic Protocol (BP) and the Combined-mode Protocol (CP) [ARINC 629 Spec. 1990]. Protocol BP, designed by Boeing, is the basic protocol and is the source of the ARINC 629 standard. It provides equal priority access for all terminals to transfer either periodic or aperiodic data. (Note: Periodic data is data that is transmitted on a regular basis e.g. sensor attitude data, whilst aperiodic data is that data which is transmitted only occasionally e.g. maintenance data). The more complex protocol CP, defined by Smiths Industries, is a modification of protocol BP and allows the transfer of both periodic and aperiodic data on the same bus [Moor 1991].

At present only protocol BP has been proven and validated by the DATAC program. Therefore in this section only the operational procedures of the BP protocol will be outlined briefly [ARINC 629 Spec. 1990, Moor 1991, Subra 1990]. There are three access protocol parameters associated with protocol BP:

1) Transmit Interval (TI), which is identical for all the terminals on the bus.
2) Synchronisation Gap (SG), which is also identical for all the terminals on the bus.
3) Terminal Gap (TG), which is unique for each terminal on the bus.

The operation procedure for information transfer is as follows: Once a terminal has transmitted it must satisfy three requirements before it can transmit again.

1) The transmission interval which is identical for all terminals on the bus must have elapsed
2) A quiet period, called the synchronisation gap which is identical for all terminals must have elapsed
3) A quiet period, called the terminal gap which is unique to each terminal, has existed on the bus since the synchronization gap occurred.

These three ground rules ensure orderly transmission without collision or contention, and guarantees periodic and aperiodic access to each participating terminal on the bus. The bus protocols are implemented in two separate circuits on the protocol chip, which run off a separate crystal oscillator. Both circuits agree on the compliance of the three rules before issuing a go-ahead signal to the transmitter.

The transmission of data is initiated by the protocol logic which resides in the ARINC 629 terminal i.e the DATAC terminal, and is handled by the transmitter side of the terminal through the use of a table in the Transmit Personality PROM (XPP). These tables include label assignments, memory mapping information, data string lengths and a number of user options for features such as interrupt vector and variable string lengths. All transmissions are monitored by the receiver and cross checked against corresponding tables in the Receiver Personality PROM (RPP). All terminals on the bus look at all labels and check them against their RPP. Entries on this table includes which information is required by that system and where it is to be placed in memory, including different memory mapping options for all multichannel communication [Shaw 1988].

4.5 Discussion

The new data bus, the ARINC 629 appears to have a number of desirable qualities required for a data bus to implement an advanced integrated avionic system. The ARINC 629 is a bi-directional bus, and unlike ARINC 429, requires fewer wires to connect all the equipment on the aircraft. As a result aircraft wire weight and volume are significantly reduced. The bus operates at high-speed, 2Mbit/s, compared to 12-100Kbit/s of ARINC 429 and twice the speed of MIL-STD-1553B. This speed is considered to be sufficient to handle all current data requirements and foreseeable applications for commercial aircraft flight controls, displays, communications, etc. [Shaw 1988]. Another important feature of the bus is that it does not require a bus controller, as in MIL-STD-1553B and ASCB.

ARINC 629 has been designed to maximise overall bus reliability and integrity. The coupler, stub and SIM provide separate paths between the terminal and the bus to transmit and receive signals. This allows the terminals to receive data even though the terminal is unable to transmit. Furthermore it also allows the transmitted signal to be 'wrapped around' through the bus so that it may be monitored by the terminal transmitting the data [Moor 1991]. This enables a terminal to detect failure and shut down itself. Therefore no terminal failure can affect the bus itself and cause disruption of communication between other terminals. Table 4.1 lists the essential features of an ideal data bus for commercial avionic application. It is clear that ARINC 629 satisfies all the requirements and thus seems a logical choice to replace ARINC 429 as the new standard for use in commercial aircraft.

With regard to bus access protocols, the protocol BP has already been proven and validated by Boeing under their DATAC program, and integrated circuit chips using this protocol have been made available to the avionic community [Mirza 1991]. The evaluation and proving of the protocol CP is an ongoing activity [Moore 1991, Subra 1990]. Smiths Industries and Micro-Circuit Engineering Ltd. (MCE) are currently nearing completion of prototype fabrication of the integrated circuit terminal chip capable of operating in either protocols.

With regard to electrical bus medium, the basic choice is between voltage mode and current mode of operation. The advantage of voltage mode is that it reduces signal current through the copper wire, thus reducing power losses. However it also has its disadvantages, especially RFI/EMI (Radio Frequency Interference/Electro-Magnetic Interference) problems. Cables operating in voltage mode are more prone to interference from nearby electromagnetic fields. But by shielding the conductors and by using the metal airframe as common ground, interference can be kept out of sensitive electronics. Another source of RFI/EMI problems is the bus coupler. When using voltage mode, a terminal couples to the bus via a transformer and 'safety' resistors. These resistors protect the transformers from shorting and bringing down the entire bus. But the resistors also introduce a loss in the network requiring higher drive power and increased sensitivity for the receivers. The increase in output power, however, causes problems for the other equipment by radiated RFI, while the increased sensitivity makes the terminals more prone to interference from nearby equipment (electro-magnetic compatibility).

Cables operating in the current mode, however, do not suffer from any of the aforementioned problems, and as such is the preferred option. The advantage of current mode operation is that, apart from the cables not requiring shielding, it allows terminals to be connected to the bus without having to cut or have any insulation removed from the bus [ARINC 629 Spec. 1990, Shaw 1986]. Although the concept is very attractive, it is in contrary to the requirements of the airframe manufacture who for years have called for separate harnesses throughout the aircraft to provide convenient production breaks to ease installation and repair. Since the current mode coupler obtains most of its reliability from the fact that there are no physical connections to the bus, problems are envisaged with regard to the installation of continuous cables on aircraft.

At present, ARINC 629 has been designed to operate at 2Mbit/s using an electrical (wire) medium. This speed was chosen due to practical limitations (i.e propagation delay) of wire medium [ARINC Spec. 1990]. With the application of a fibre optic medium, higher data rates should be possible improving the performance of the system.

The topologies considered for the implementation of ARINC 629 fibre optic bus include linear bus network, transmissive star network and reflective star network [See section 5]. A number of research and development establishments have put in considerable amount of effort in designing fibre optic hardware to meet ARINC 629 requirements. Litton Poly-Scientific have designed a prototype fibre optic hardware (i.e transmitter and receiver) to meet the requirement of ARINC 629 [Lewis 1991]. Although they had great difficult in meeting some of the conditions, such as maintaining adequate launch power at high temperature, receiver sensitivity and providing collision detection capability, Litton Poly-Scientific envisage no major obstacles preventing the design of fibre optic hardware to meet ARINC 629 requirements, and are currently working on their fibre optic hardware to extend its present capability to meet the environmental and reliability requirements.

One of the criticisms of ARINC 629 has been its complexity and cost. Figures of £1000 - £2000 have been quoted for an ARINC 629 terminal. Concerns have also been expressed as to its complexity and cost. Although it is suitable to transfer data between IMA cabinets, its cost and complexity are considered to make it less suitable for the control of lower level utility functions [Rogers 1991].

4.6 Conclusion

In this section a review of data buses in avionic use today has been presented, and in doing so, the qualities needed for a data bus to serve a modern commercial aircraft has been identified.

An advanced avionic architecture requires a data bus that can provide efficient method of communication with high integrity, and the ARINC 629 appears to fulfil this requirement. Arinc 629 provides an efficient, high-speed, communication system which allows equipment to be added, removed or changed (i.e. upgraded) without causing an impact on the other users of the bus. The major attributes of ARINC 629 are:

- Bi-directional
- High-speed (2Mbit/s)
- Autonomous system - i.e requires no bus controller
- Operates in both current mode and voltage mode
- Fibre optic compatible

A disadvantage is its cost. However given its ability to reduce cabling, size and weight of interface hardware, and enhance reliability makes it an appropriate system for application on commercial aircraft.

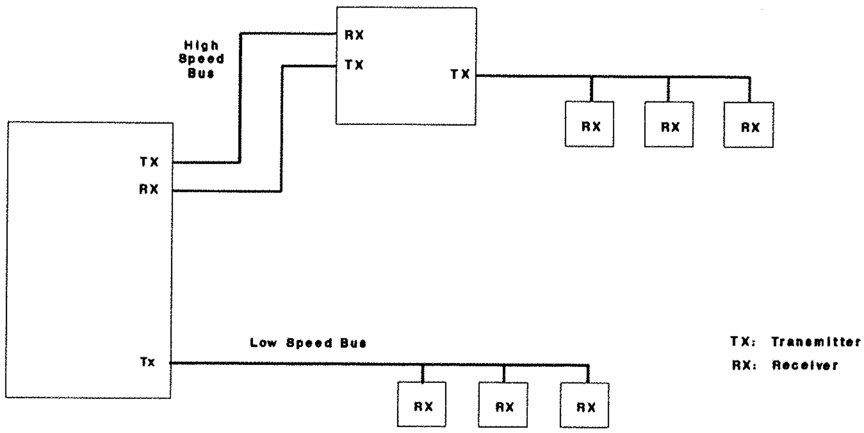


Figure 4.1 Simplified ARINC 429 Architecture

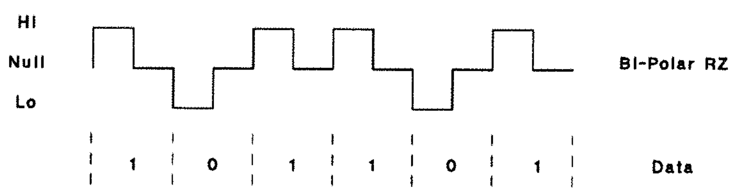


Figure 4.2 Bi–Polar RZ (Return to Zero) Format

32	31 30 29	28 27 26 25 24 23 22 21 20 19 18 17 16 15 14 13 12 11	10 9	8 7 6 5 4 3 2 1
P	SSM	MSB Data Field LSB	SDI	LSB MSB

P: Parity

SSM: Sign/Status Matrix

LSB: Least Significant Bit

MSB: Most Significant Bit

SDI: Source Destination Identifier

Figure 4.3 Basic ARINC 429 Data Word

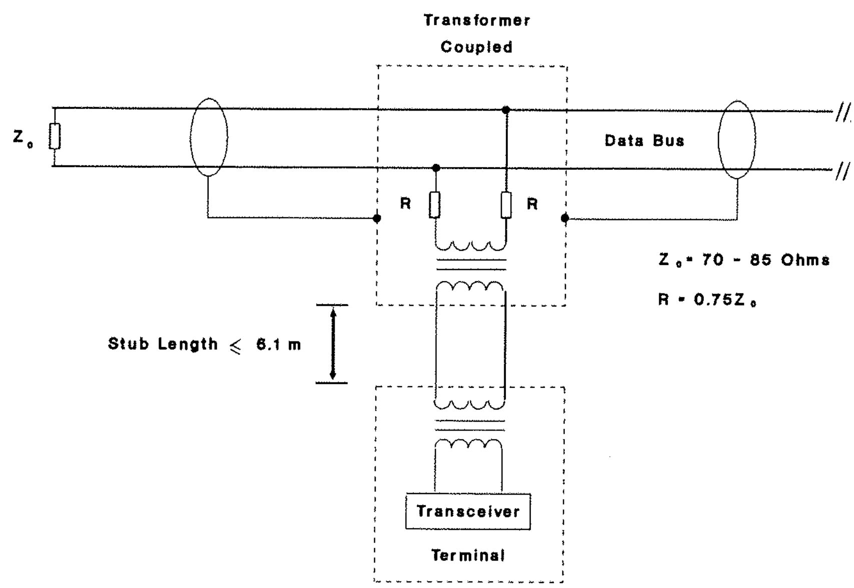


Figure 4.4 Transformer Bus Coupling

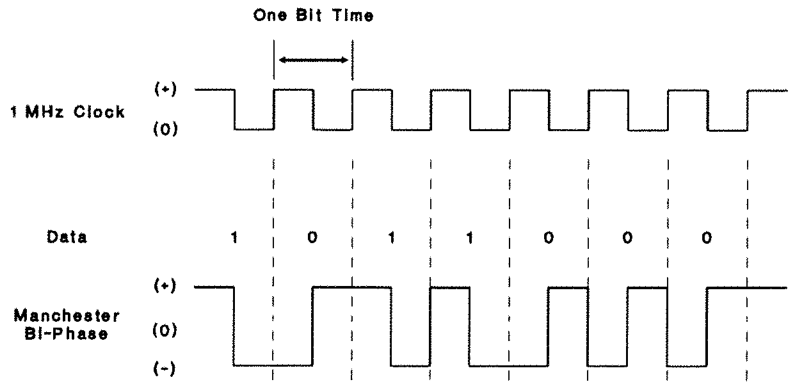


Figure 4.5 Manchester Bi-Phase Data Encoding

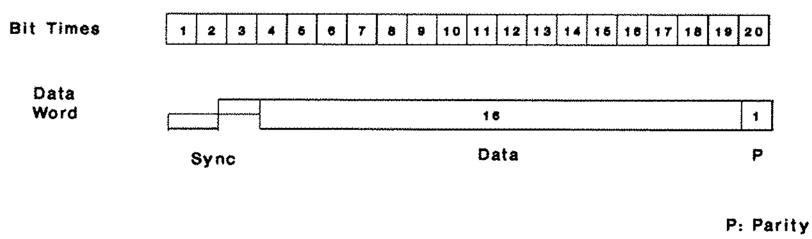


Figure 4.6 MIL-STD-1553B Data Word Format

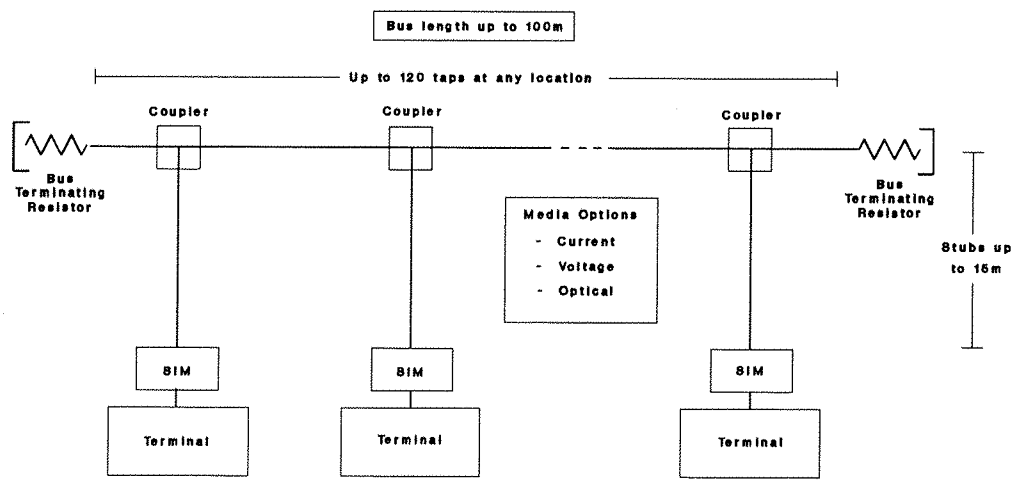


Figure 4.7 ARINC 629 Topology

Attributes	ARINC 429	MIL-STD-1553B	ASCB	ARINC 629
2 Way communication	X	✓	✓	✓
Multi-channel capability	✓	✓	✓	✓
Autonomous protocol	N/A	X	X	✓
Auto configuration	X	✓	✓	✓
Immunity to single point failure	N/A	X	X	✓
OSI compatible	X	✓	?	✓
Mixed vendors on the bus	✓	✓	✓	✓
Terminal failure isolation	N/A	✓	✓	✓
Self monitoring terminals	✓	✓	✓	✓
Bulk data capability	✓	✓	✓	✓
Minimum impact reconfiguration	X	X	X	✓
Fibre optic compatible	X	✓	?	✓

Table 4.1 - Data Bus Requirements

5 Fibre Optic transmission System and its Limitation when Applied to Avionic Systems

5.1 Introduction

For a number of years it has been recognised that Electro-Magnetic Interference (EMI) and high intensity Radio-Frequency Interference (RFI) can severely compromise the flight safety of Fly-By-Wire (FBW) systems. However recently, with the growing trend towards the increase in use of composite material for aircraft skin and primary structures, these concerns have been heightened.

At present aircraft avionic systems and their associated data channels are protected from external EMI and RFI sources by the Faraday shielding inherently provided by the aircraft skin. Composite materials, however, do not offer shielding to external EMI and RFI sources, and thus additional shielding is required for the data channels, which incur undue weight and cost penalty. Although it may be relatively easy to provide adequate shielding, the verification procedures to test their effectiveness becomes an expensive and time consuming operation.

Fibre optics provides inherent immunity to EMI, RFI, lightning and other related disturbances and thus appears to provide an ideal solution to the above problem, by eliminating the need for heavy electromagnetic shielding, and the time consuming and costly verification procedures. Fibre optics has been extensively used by the telecommunication industry for a number of years, however, to date they have not been used to any extent on a production commercial aircraft. The main reason for this appears to be the cost effectiveness of the overall system and the reliability of the fibre optic components in the aircraft environment. This chapter reviews the primary elements of a fibre optic transmission system and highlights the limitations of the fibre optic technology when applied to avionic systems. The section also provides a review of the work that has been carried out to develop a high speed fibre optic data transmission system for avionic application.

5.2 Fibre Optic Transmission System

The primary elements of a fibre optic transmission system include transmitters, transmission medium, receivers, optical couplers and connectors. The design of these elements dictate the performance of the system.

The transmitter consists of an optical source, driver circuit, support electronics for the optical source and a power supply. The optical source in a fibre optic system is an Electrical to Optical (E/O) converter. The E/O converter receives electrical signals and converts them into a series of light pulses.

There are two types of E/O converters currently available, namely the Light Emitting Diode (LED) and the Laser Diode (LD). Both devices provide a mechanism for converting electrically encoded information into optically encoded information. However their application depends on a number of factors, which include response times, temperature sensitivity, power levels, system life, expected failure rate and cost [Bermingham et. al. 1987, Held 1989]. Each should be carefully considered before selecting a source for system design.

LEDs emit light over a relatively broad spectrum and they also disperse the emitted light over a rather large angle [Friend et. al. 1984]. As a result LEDs couple much less optical power (in the order of μW) into an optical fibre compared to laser diodes. However the operation of the LED is much simpler than the laser diode, and it is also less costly and more reliable.

Laser diodes, due to their narrow spectrum of emission are able to couple a high level of optical power (in the order of mW) into an optical fibre, achieving greater transmission distance than possible with a LED [Couch 1985, Cruickshank 1989]. Laser diodes also offer a fast response time in converting electrically encoded signals into light encoded signals. As a result they are suited for very high data transfer application. Laser diodes, however, are more prone to temperature changes and ageing, requiring a complex driver circuit to ensure that the driver current is compensated properly for both temperature and ageing of the source [Pierce, Uhlhorn 1987]. This results in the device having a higher overall cost than LEDs. Laser diodes also take a longer time to turn on from a totally off (unbiased) state.

The transmission medium is an optical fibre cable, which can be made out of either plastic or glass material. There are two types of optical fibres available for use in cables: step index and graded index. In a step index fibre, there is an abrupt change in the refractive index between the core material and the cladding. By contrast in a graded index fibre, there is a gradual change in the refractive index between the core material and the cladding. The gradual change in the refractive index serves to minimise the optical signal dispersion as light traverses the fibre. The minimisation of signal dispersion allows greater bandwidth to be achieved, which enables transmission to occur over a greater distance at higher data rates than possible with a step index fibre [Uhlhorn AGARD].

The capacity of a fibre optic link of a given distance depends on a number of factors: attenuation and light dispersion characteristics of the fibre, core size, and the numerical aperture[1] of the optical

[1]The numerical aperture value of an optical fibre is a measure of its light gathering ability and indicates the potential efficiency in coupling of the light source to the fibre cable

fibre. Attenuation in optical fibres result from scattering loss, absorption loss, connector and connection loss, and bending loss. Scattering arises from microscopic imperfections in the core material. However there is a limit below which scattering cannot be reduced no matter how perfectly the core is made.

Absorption refers to the conversion of power in the transmitted light into heat in some material. Absorption losses can be minimised by transmitting light at 0.8, 1.3 and 1.5 μm, where there are reductions in the absorption curve for light [Friend et. al. 1984]. Both scattering and absorption losses are generally very low (a few dB/km) for modern fibre optic cables [Uhlhorn AGARD].

Connector losses depend on the quality of the connectors used, whilst connection losses depend on the precision alignment of optical fibres at the splice. Unlike conventional wire connected system, the performance of a fibre optic system is sensitive to the kinds of connection and the number of them. A large source of losses in a fibre optic system are due to connector and connection losses.

Bending losses come from the effects of microbending and macrobending. Microbending refers to the bending which occurs during the fabrication process, i.e. cabling and jacketing process, and can be controlled by careful design. A larger diameter fibre resists microbending better. Macrobends refer to the bends introduced into the cable during installation.

Optical fibres not only attenuate signals but they also distort them [Uhlhorn 1987]. There are two effects which contribute to signal distortion in multimode fibres: modal dispersion and material dispersion. Modal dispersion is caused by different modes (rays) taking different path inside the fibre. In general, a larger diameter fibre will have a larger modal dispersion, for a given numerical aperture, because optical path differences are greater. The effect on a large diameter fibre however can be overcome by lowering the numerical aperture of the fibre. Material dispersion is caused by different wavelengths (colours) travelling through the fibre at different speeds. Thus broader the spectrum of the emitted source, the greater the dispersion.

The optical receiver consists of an optical detector, i.e. a photodiode, an amplifier with low noise, wide dynamic range and high bandwidth, and support electronics. The receiver employed in the system is an Optical to Electrical (O/E) converter. The O/E converter changes the received light into its equivalent electrical signal. Ideally the receivers must have very high sensitivity and low noise.

There are a number of O/E converter devices currently available: PIN (Positive-Intrinsic-Negative) photodiodes, avalanche photodiodes, phototransistor and photomultipliers. Of these devices, PIN photodiodes and avalanche photodiodes are most commonly used for fibre optic transmission systems due to their efficiency, light signal reception capability and cost. Avalanche photodiode, however offer greater receiver sensitivity than the PIN photodiode, but it is more susceptible to temperature changes and as such there operational environment requires careful examination [Couch 1985, Pierce, Uhlhorn 1991] . In addition the avalanche photodiode requires an auxiliary power supply which introduces noise into the system. Therefore additional circuitry is needed to limit the noise which makes it more costly compared to a PIN photodiode.

There are two types of optical couplers available: the 'T' coupler and the star coupler. Of the two, the most commonly used coupler is the star coupler. The star coupler can be implemented as either transmissive or reflective network (Fig. 5.1). In a transmissive network, two fibre optic cables are used: light is transmitted to a terminal along one and received via the other. In a reflective system a single fibre optic cable is used for both transmitting and receiving data. As a result the reflective system requires a three port coupler (e.g. a 'Y' coupler) at each terminal to separate the signal into its transmitted and received components. A commonly used multipoint optical coupler is the transmissive star (also referred to as fused biconical-taper star) [Couch 1985, Stallings 1984]. The star coupler is fabricated by fusing together a number of optical fibres. The couplers can be made with any number of inputs and outputs. The light signal on any input is distributed by the device more or less equally to all outputs, with a splitting loss of 10 log N dB, where N is the number of outputs. Star couplers with up to 512 ports have been reported, however, due to limits on the energy available after the power divisions, excess loss and cost, only about 16 ports are considered practical [Glista 1991, Uhlhorn et. al. 1990].

So far only the primary elements of the optical transmission system has been described. In order for the information to be transferred within the system, some form of standard is required. In recent years several standards have been proposed, they include: the SAE AE-9B High Speed Ring Bus standard, the ANSI X3T9.5 Fibre Data Distribution Interface standard, the SAE AE-9B Linear Token Ring Passing Multiplex Data Bus standard and the MIL-STD-1773 standard, the fibre optic equivalent of MIL-STD-1553B [Bermingham et. al. 1987]. Topologically the first two standards are implemented as a ring network whilst the last two are suitable for a star network.

To summarise an optical transmission system requires:

1) A light source (i.e. a transmitter) of high intensity to couple a high level of optical power into the fibre
2) A fibre of low loss and wide bandwidth to minimise transmission loss and accommodate large quantities of data to be transferred

3) A photodetector (i.e. a receiver) of high sensitivity and low noise to provide reliable reception after transmission losses and
4) Connectors (ideally low loss) to connect the various fibre optic transmission system elements.

There are several benefits to be gained from replacing the conventional metallic cables with an optical medium. They include:

1) Electromagnetic non susceptibility - Since optical energy is not affected by electromagnetic (EM) radiation, a fibre optic medium:
   a) Does not require any special shielding formally required for metallic cables - reduces assembly cost and weight, removes EMI verification test from the certification program and eliminates the difficulty associated with evaluating the continued effectiveness of the shielding.
   b) Eases cable routing, since the rerouting required for metallic cables around devices emitting EM radiation does not cause a problem when routing fibre optic cables.
   c) Can operate in an electrically noisy environment. Optical fibres do not produce any EM radiation, as a result they do not generate cross talk. This property allows multiple fibres to be routed in one common cable, simplifying the system design process.
2) Increased bandwidth - With potential information capacity directly proportional to the frequency of the transmitted light, fibre optics yields a transmission medium capable of providing very high data rates. This allows for mixed voice, video and data on one line.
3) Immunity from electrical hazards - Light energy eliminates the potential of short circuits and shock hazards.
4) Reduced weight and size - Optical fibres are smaller and lighter compared to metallic cables of the same transmission capacity.

5.3 Discussion

Although an optical transmission medium is an attractive alternative to conventional metallic cables offering several benefits, it has a number of limitations when used in aircraft systems. These include topology, connections/connectors and receiver design. Topology refers to the overall physical arrangement of a system in terms of its elements and the way they are interconnected. At present there are essentially three network topologies under consideration, namely the linear topology, the ring topology and the star topology (Fig. 5.2).

The linear topology employs passive 'T'-couplers or taps to obtain access to the bus from each node. From the physical structure of the aircraft and its equipment layout, it is clear that a linear topology would be the simplest and the most effective way of distributing the data throughout the aircraft. This is also true from both installation and maintenance points of view. But the number of nodes in the linear topology is limited by the optical power that can be launched into the fibre, as well as the fibre size and the total attenuation which include connector splice and T-coupler losses. At present, T-coupled linear optical buses supporting only about 10 terminals have been realisable due to losses in the couplers [Lewis 1991].

The ring topology employs an optical source and detector at each terminal in the ring. These devices act as repeaters by detecting and regenerating the optical signals for data passing through the terminal. As a result the topology allows the use of low sensitivity and low dynamic range receivers, since only the next adjacent terminal needs to be accessed, where the optical signal is regenerated [Glista 1991]. Thus the arrangement is ideal for supporting a large number of terminals. One of the major disadvantages of this topology is the potential for single point failure. A failure in either the optical, electronic or power supply element can bring down thewhole ring (Fig. 5.3). For this reason ring topologies are not considered suitable for avionic application.

A way of improving the reliability of the ring topology would be to use a counter rotating implementation as shown in Figure 5.4. These make use of by-pass switches to perform local loop backs to recover the ring [Cohn 1988]. But this incurs undue complexity and cost. Spring loaded electromechanical by-pass switches are slow, as a result significant amounts of data is lost during loop back. Furthermore they are also costly, and are inadequate to operate in the harsh environment of the aircraft (i.e. vibration and shock). An alternative to the electromechanical by-pass switch is the electro-optic by-pass switch. But these are lossy (typically several dB/cm), and are not readily available [Glista 1991]. Studies conducted by Rosen et. al. at the Naval Air Development Centre, Pennsylvania, indicate that in the case of avionic systems where connectors often represent a significant failure mode, counter rotating ring topologies are less reliable compared to star topologies [Rosen et. al. 1988]. They also found that the techniques for by-passing large number of failed stations did not significantly improve the ring reliability.

In the star topology, optical power is divided equally among a number of terminals through the use of a transmissive or reflective passive coupler. A major benefit of a passive star topology is that since there are no active components in the star, a failure of power cannot cause the star to fail itself. However a major disadvantage of this topology is that all transmitters and receiver modules must run via the star. In an aircraft this increases the initial installation cost due to the large number of bulkhead penetrations required. The topology also employs a large number of fibres than that required for a T-coupled bus implementation. This can be significant if the buses are to be implemented with redundancy. The number of fibres could however be reduced by operating in a reflective mode i.e. a reflective star topology. But this configuration suffers from back reflections from the transceiver connectors [Lewis 1991]. A further disadvantage of the passive star topology is its limitation when implementing a large number of terminals. Since the optical power from any one transmitter is divided more or less equally amongst the terminals connected to the star, the problem becomes one of providing sufficient optical power. This can be achieved by making the passive star active (i.e as a repeater), but this, however, is a source for single point failure. In general multiport active stars are complex and expensive. Nevertheless, they have been used as a practical alternative for aircraft with large number of terminals - the active star is currently being used as part of the avionic data bus on the Navy A-12 aircraft for non flight critical applications [Glista 1991].

Another major limiting factors of fibre optic system is cable splicing. Optical fibres may be interconnected by welding, gluing or through the use of mechanical connectors. All three techniques incur some degree of signal loss between the spliced cables. Welding or fusion of fibres provide the lowest loss of transmission between the splice elements. The welding operation involves cleaning each end of the fibre and aligning them with great precision before fusing them with an electric arc. The operation is a time consuming task and in the confined space on board an aircraft, it is a technique much to be desired. In addition welding result in permanent splices, whereas for avionic systems most connections need to be demountable for ease of maintenance and repair. The gluing method requires a bonding material that matches the refractive index of the core material and incurs a larger transmission loss than with the welding process. Mechanical connectors reduce the splicing time considerably and by far the easiest technique to employ. However they incur a path loss which can vary with connector contamination and with axial, radial and angular misalignment of the mating fibres [Miller 1990]. Pistoning of fibres is also a common problem encountered in connectors [Figueroa et. al. 1991, Lewis 1991]. Fibre pistoning refers to the movement of fibre behind or beyond the end of termination due to failure of an adhesion bond. Possible solution to this problem are being investigated and include gluing of the joints. But in some cases this has caused sever and unacceptable sensitivity problems [Lewis 1991].

Optical bus configurations have been found to have considerable and undefined transmission losses between the source and detector due to the coupler and the connectors [Couch 1985]. These effects when combined with the source power and detector sensitivity give rise to an uncertain power level. Therefore to provide a reliable reception, a high gain, wide dynamic range receiver is required. It is possible to design receivers with wide dynamic range, but these receivers take a long period to adjust to changes in signal level, which is not ideal for the instantaneous requirements of a high speed data bus receiver. This is a major concern of optical data bus receiver designs [Couch 1985].

In addition to the problems mentioned above, test equipment also appear to be a problem area. It has been reported that current test equipment are difficult to use on the flight line. Another problem area is the maintainability of the fibre optic system. One of the common complaints has been the susceptibility of the cable to maintenance induced failures [Little 88]. Concerns have also been expressed as to the ability of fibre optic hardware to operate reliably in the severe aircraft environment. (i.e. shock, vibration, humidity and wide temperature excursions).

Recent Developments in Fibre Optic Transmission Systems

In recent years, a number of designs have been put forward by a number of Research and Development centres as viable fibre optic data transmission systems for avionic application. However, it must be said that most of the schemes that have been put forward are only proposals and neither have they been developed nor implemented.

Bermingham and co-workers from Naval Air Development Centre, Pennsylvania, have designed and constructed a 50Mb/s, 64 station fibre optic bus [Bermingham et. al. 1987]. The bus implements a star topology and uses a token passing protocol based on the SAE AE-9B Linear Token Passing Protocol. Data is transferred over the bus using Manchester II coding. The system uses LEDs and PIN photodiodes as active optical components for transmitters and receivers, respectively. One of the concerns of implementing a large star based system is the ability to provide sufficient power, since optical power is more or less equally divided among the stations. But the system gets around this problem by using a novel automatic gain circuit which provides greatly increased sensitivity in the receiver.

Uhlhorn and co-workers from Harris Corporation have carried out a design study to implement a 50Mb/s, 64 station passive star-coupled fibre optic data bus for military application [Uhlhorn AGARD, Uhlhorn 1987]. Data is transferred over the bus using Manchester coding. The design employs LEDs and avalanche photodiodes as active optical components for transmitters and receivers, respectively. The feasibility of the design has been supported by detailed performance calculations; the details of which could be found in reference [Uhlhorn 1987].

Husband from Mitre Corporation and Katz from Naval Avionics Centre, have proposed a technique called the Multiple Speed Data Rate (MSDR) transmission concept [Husbands 1983]. The technique allows a number of avionic equipment operating at different data rates to coexist on a single command/response multiple data bus. To support such a concept, the design requires a transmission medium that is not band limited, and as such the design employs MIL-STD-1773, the fibre optic version of MIL-STD-1553B. The optical data bus implements a transmissive star topology and consists of standard MIL-STD-1553B terminals and special high speed terminals called the MSDR remote terminals (Fig. 5.5). In this design low speed (1 Mbit/s) data transfer between remote terminals is conducted, as specified in MIL-STD-1553B. To perform data transfer between terminal with large data bases, at high data rates, the bus controller establishes remote terminal to remote terminal contact as per MIL-STD-1553B. However, once a point to point contact is made between the terminals, a high speed modem pair in the terminals is activated, and the data base information is transferred at a higher data rate (20 Mb/s). At the end of the transfer the terminals respond at the low data rate to convey the completion of the transfer action.

Chown and co-workers from Standard Telecommunication Laboratories Ltd have also designed a high speed fibre optic MIL-STD-1553B data bus [Chown 1986]. However their design implements a passive local star topology. The configuration consists of, as many as four fully interconnected multipoint transmissive star couplers, with each star connected to the other via an interstellar link. Data is transferred over the bus using Manchester coding. The design concept is said to have been made possible, mainly due to the development of a fibre optic transceiver which encompasses novel coding and decoding circuits and a special coding technique which is designated 'FSK'. The design has been proven by the implementation of a full dual-redundant rig, based on the local star topology.

Westland Helicopters have developed and flight demonstrated an optical MIL-STD-1553B data bus [Cruickshank 1989, Kennett 1986]. The data bus is implemented as a local star topology, with three transmissive couplers interconnected to each other via interstellar links. The optical data bus has been installed on a Lynx helicopter and has been successfully flown. The optical bus incorporates a MIL-STD-1553B chipset which consists of a transmitter, a receiver and a coder/decoder (CODEC). As of 1989, Westland's helicopter reported no problems associated with the data bus [Cruickshank 1989].

At this point, however, it must be said that all the research and development work carried out during the past few years into fibre optic technology for avionic application have been extensively for use in military aircraft. This, however does not mean that fibre optics has not been applied to commercial aircraft.

Douglas Aircraft Company have implemented three flight test systems as at 1988. They include a fibre optic passenger entertainment system, installed on the DC-10; a fibre optic aileron position sensing system, installed on the DC-10 as a non interfering redundant system; and a fibre optic wheel speed sensing system, installed on the MD-87 as a non interfering redundant system [Todd 1988]. Also, as of 1988 the passenger entertainment system was flying and operational whilst the other two systems were under evaluation. As of 1989, Douglas Aircraft Company have also been working on a fibre optic Fly-By-Light (FBL) engine control system called the Optical Propulsion Management Interface System (OPMIS) [Todd 1989].

From the various fibre optic transmission systems put forward it is clear that the preferred topology for avionic application is the star topology, either central star or local star. A central star topology may be well suited for a small fighter type aircraft, where all the avionics are grouped together in a central area, with generally a short distance between them. However in a larger commercial aircraft where the distances are much longer, the central star coupler approach is considered to be unattractive since:

1) Running optical cables from each remote terminal to the central star require a large number of bulkhead penetrations, which can escalate the initial cost of installation.
2) A single point failure can disable the entire bus.
3) Limits the flexibility for adding more terminals at arbitrary locations.
4) Makes maintenance difficult due to the large number, and length of the optical cables

The local star topology, however, is well suited for larger aircrafts. Since in an aircraft the remote terminals (equipment) are clustered together in zones, for example in the cockpit, the avionics bay and the engine bay, these zones can be served more efficiently by local star couplers. This would considerably reduce the amount of cabling and the number of bulkhead penetration, and therefore the cost. The topology also eliminates single point failure from disabling the entire bus system.

5.4 Conclusion

In this section a review of the various elements of an optical transmission system and the limitations of the fibre optic technology applied to avionics systems has been presented. Topologically, the T-coupled linear bus provides the greatest flexibility in installation. However at present, linear optical buses supporting more than 10 terminals are not realisable due to losses in connectors, and as such the star topology in particular the local passive star appears to be the topology of choice.

Fibre optics offer increased bandwidth, immunity to EMI and RFI, electrical isolation and weight reduction. As a result there are several benefits to be gained from the application of fibre optic systems on aircraft. At present the main factors hindering the near term acceptance of fibre optic systems on aircraft appear to be the cost effectiveness of the overall system and the instability, maintainability and in-service reliability of cables and connectors in the aircraft environment. In order for fibre optics to become truly competitive with wire and for FBL to become a reality:

- Better connectors are needed which are simple to terminate and are either self cleaning or inherently protected from contamination
- Ways must be found to reduce the back reflection of the transceiver connector
- Ways need to be found to locate cable faults more easily
- Simple, easy to use, test equipment and procedures are needed
- Sensitive optical receivers with sufficient dynamic rang are required

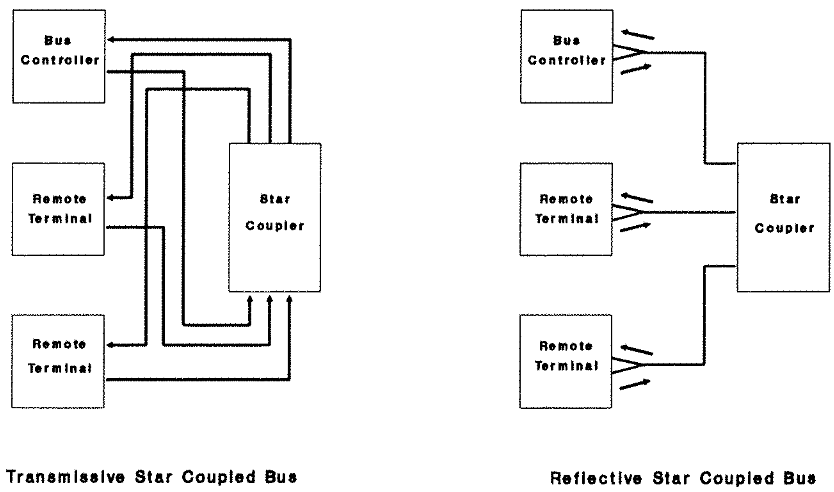


Figure 5.1 Transmissive and Reflective Star

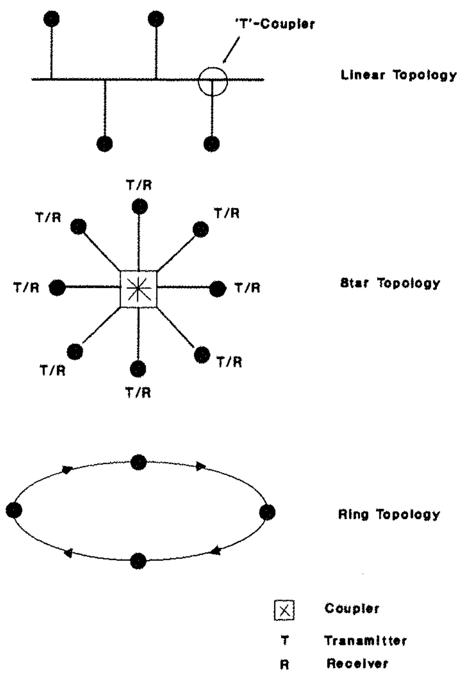
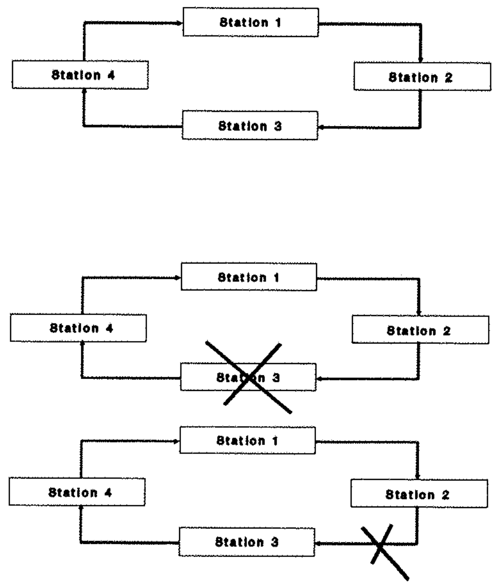
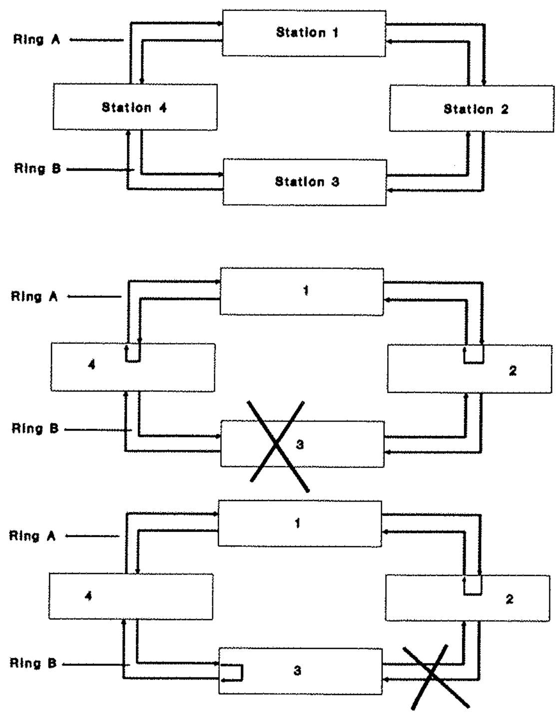


Figure 5.2 Commonly used Fibre Optic Topologies



A Single Station or Link failure Causes the Ring to go Down

Figure 5.3 Ring Topology



Loopback Recovery in a Dual, Counter-Rotating Ring

Figure 5.4 Counter Rotating Ring Topology

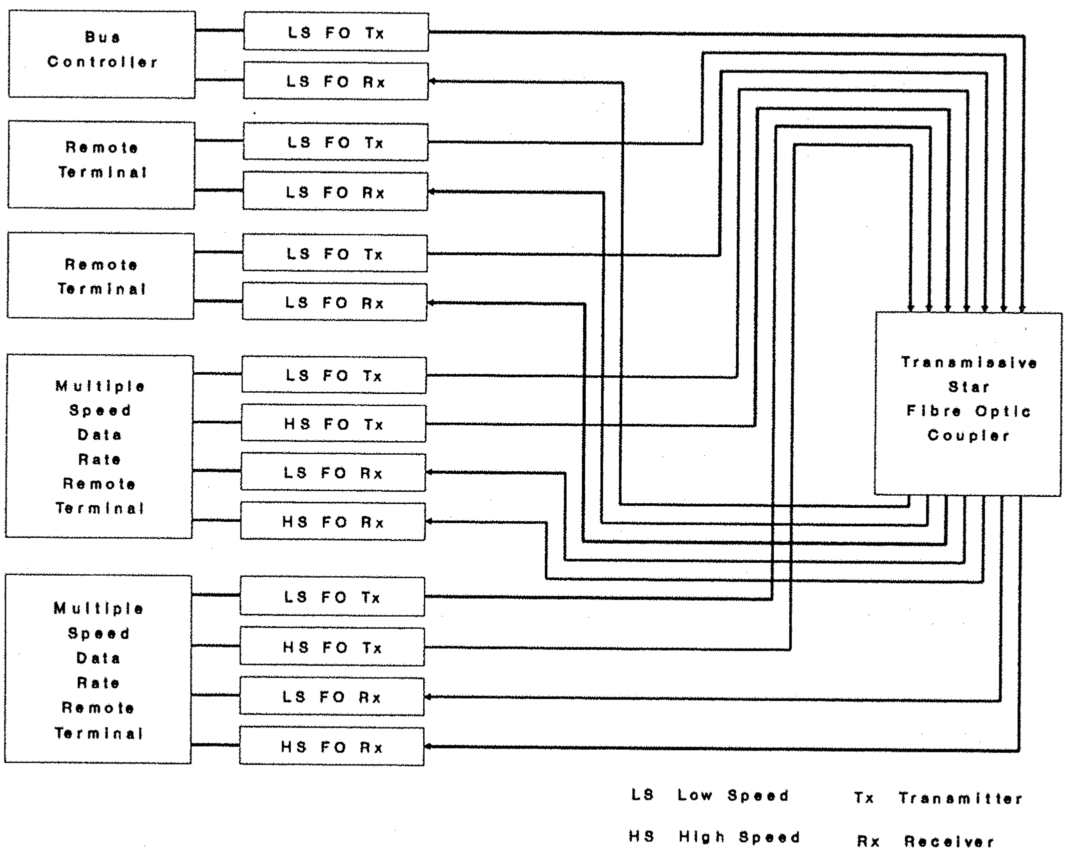


Figure 5.5 Optical Data Bus Network

6 Effects of High Energy Electromagnetic Fields on Aircraft Avionic System

6.1 Introduction

With the advent of digital avionics, almost every function associated with aircraft operation, from monitoring to processing, has been performed by digital systems. Although significant improvements have been gained by the introduction of digital systems, they have also introduced some problems. One of the concerns has been digital system susceptibility to high-level momentary Electro-Magnetic (EM) environment (e.g lightning and radio frequency). Various protection schemes such as filter pin connectors, metallic overbraids, and conduit or shielding enclosures have been successfully used to screen the system from EM fields. However these schemes incur a weight and cost penalty, and reduce reliability and maintenance accessibility. As a result aircraft system design engineers have looked to architectural techniques to overcome some of the problems associated with traditional protection schemes. This chapter provides a review of some of the aircraft electromagnetic compatibility problems and examines system level protection schemes which overcome some of the problems.

6.2 The Threat

It is envisaged that the next generation of commercial aircraft, like the all electric aircraft, will use a greater percentage of composite material for the vehicle skin and primary structures. Since the conductivity of composite material is significantly lower than aluminium, the shielding provided by the vehicle surface and internal structure membranes will be significantly reduced. As a result the aircraft electrical/electronic systems will be 'exposed' to external EM fields to a greater degree. The main threat to the system are from high-level external EM fields radiated from very high power broadcast, communication and radar facilities.

The next generation of aircraft is also expected to incorporate modern avionic systems which use VHSIC/VLSIC (Very High Speed Integrtaed Circuit/Very Large Scale Integrated Circuit) technology. These system operate at lower power levels and higher speeds. The susceptibility of these devices is usually the response that is caused by a rogue transient voltage that creates an unwanted logic state which shows up at the system output [Hess et.al. 1988].

Lightning and high-energy Radio Frequency (RF) are the two most severe EM threats which affect the operational characteristics of an aircraft. Of the two, lightning produces the most intense electromagnetic environment. During a lightning strike, relatively large currents flow along the structure of the aircraft, which in turn induces large voltages and currents in the aircraft wiring. These voltages and currents appear at equipment interface circuits. Induced voltages are usually less than 200V for an all metal aircraft, although 1000V or more have been projected for some aircraft when subjected to severe EM radiation [Hess et. al. 1988]. For aircraft which are constructed with large percentage of composite material, the lightning induced voltages can be substantially higher.

Although lightning produces the most intense electromagnetic levels, it is the high-level RF environment which exposes the electrical/electronic systems to the effects of EM environment [Hess et. al. 1988]. When a frequency from an external RF spectrum corresponds to an electrical/electronic system resonance, the effect is the multiplication of any induced voltages or currents.

In addition to upsets from external EM fields, aircraft avionics are also susceptible to EM fields emanating from the aircraft, e.g transmitters, receivers and power supply. In a recent survey involving seventeen types of military aircrafts, eighty system level problems were identified due to EM fields arising from the aircraft. The observed problems fall into six coupling categories [Zentner 1983]:

a) Antenna-to-antenna - Many inter antenna problems are caused by harmonics of the transmission frequency being in-range with the receiver frequency, the random noise generated by the output stage of the communication unit being in-range with the receiver resulting in desensitization, and arcing in the antenna coupler causing interference at a receiver.
b) Antenna-to-wire/box - These are caused by EM fields from aircraft HF, VHF and UHF transmission coupling into aircraft wiring.
c) Wire-to-antenna - Most problems are due to narrow band harmonics of digital clocks radiating from aircraft wiring coupling in communication antennas.
d) Wire-to-wire: Power frequency - Caused by aircraft 400 Hz power frequency coupling into signal cabling.
e) Wire-to-wire: Transients - Most transient wire to wire coupling are caused by fast rise time high spikes that result when unsuppressed airframe relays and solenoids are de-energized.
f) Precipitation static - Caused by discharges occurring at the antenna.

Most of these problems have been alleviated through traditional protection techniques, ranging from the use of capacitors, high/low pass filters to improved shielding.

6.3 System Level Methodologies

The goal of system level methodology is to provide a mechanism which enables the system to tolerate disruption of either input/output or internal computation. This has been achieved by providing the system with the ability to detect disturbances and rapidly clear faults caused by the disruption, and the capability for rapid 'transparent' computational recovery.

There are two basic approaches to transparent recovery [Hess et. al. 1988]:

a) Cross-lane recovery, where state variable data are transmitted from valid channels to the channel which has been committed faulty, and from which a recovery is to be attempted (Fig. 6.1).
b) In-lane recovery, where a valid set of state variables are stored away prior to disruption in memory which is protected from the effects of disruption to a level far greater than the rest of the system (Fig. 6.2).

The cross-lane recovery strategy is inefficient against disruptions which affect all channels simultaneously. Since reliability and in particular the avoidance of single point failure is important in avionics application the cross-lane strategy is considered not suitable for avionic application. However, the in-lane recovery strategy, due to its 'hardened' memory location is much more suited for avionics application.

Figure 6.3 shows a schematic of a digital processor architecture including the transparent recovery elements [Hess et. al. 1988]. The system consists of a process monitor, recovery control and a storage region hardened against EM field. The detection of processing irregularities is accomplished through data reasonableness tests and through reasonableness of processor behaviour. Dynamic data is continually read from a hardened storage region, updated by the digital processor and then written back into the storage region. In the event of a temporary disruption of the digital processor the state data (i.e the data which contains the information that has been gathered by the processing activity over a period of time and which is not reproducible) and the sensor data are used to quickly recover the processing activity in a transparent manner. (Note: The set of state data and the set of sensor data together establish the state of the system at each instant of time). When a 'soft fault' (i.e digital circuit upset) is detected, the supervisory hardware control cycle the digital processor through a recovery sequence. This process involves reinitialization of the system states using data from the protected memory region, resetting the program counter to restart program execution and the actual restarting of program execution.

6.4 Discussion

Avionics system, either digital or analogue, are susceptible to high-level EM fields. Although both respond to the same threats, the susceptibility for upsets for a digital systems to a momentary high-level EM fields is potentially far greater than its analogue counter part, because considerably less externally coupled pulse energy is required to cause a state change. This has been accentuated by the use of VHSIC/VLSIC technology: In older digital systems the transient energy required to cause a system upset was in the order of 10^{-5} J, but with the introduction of VHSIC/VLSIC technology system upsets occur at 10^{-9} J [Hess 1985, Hess et. al. 1988]. Furthermore these digital systems can be upset by transient disturbances lasting less than 2ns. In contrast, previously the disturbance had to last a few milliseconds to cause upset.

Commercial aircraft avionics are susceptible to RF threats particularly in the frequency range of 1.5 to 300MHz, and is of major concern since aircraft wiring at these frequencies acts as a highly efficient antenna [Larsen 1988]. Various traditional protection schemes have been used to screen the aircraft avionics from EM fields. However they incur a weight and cost penalty and are usually sensitive to maintenance procedures. In addition the verification procedure to assess their effectiveness becomes an extremely difficult and costly operation.

System level methodologies provide an effective way of protecting sensitive avionics from high-level, momentary EM fields. The major advantage of the methodology is that it provides a means to tolerate temporary disruptions without allowing the fault to propagate outside the affected system. As a result the system continues to provide its function in the face of temporary disruption of the processor. Since for safety-critical systems, fault tolerance is of the essence, the scheme can be used to provide the additional margin of safety required for flight critical functions.

6.5 Conclusion

In this section the effects of high energy EM fields on avionics systems was presented together with system level protection schemes which can alleviate some of the problems. System level protection is achieved through detection of processing irregularities, retention of state variables in protected storage and rapid recovery of processing activity. A digital system design based on such a concept is able to be fully operational even when there is a temporary disruption of the processor. The concept provides an effective way of protecting sensitive avionics from high-level EM fields, and can be used in conjunction with traditional protection techniques to provide the margin of safety required for flight-critical function.

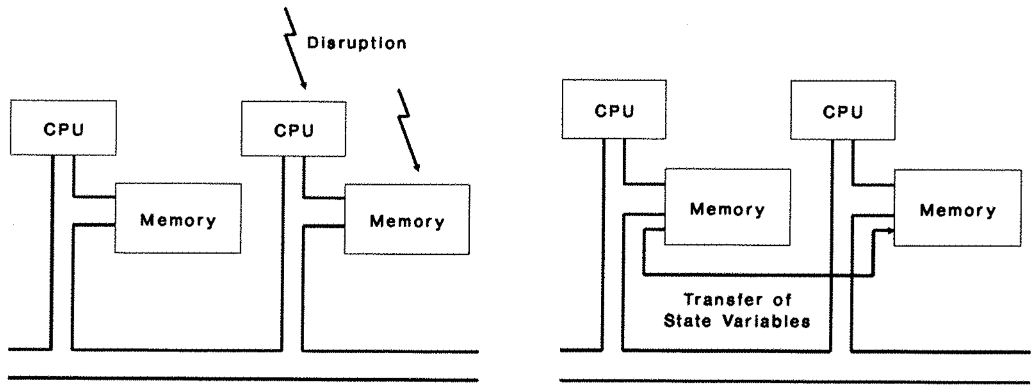


Figure 6.1 Cross–Lane Recovery

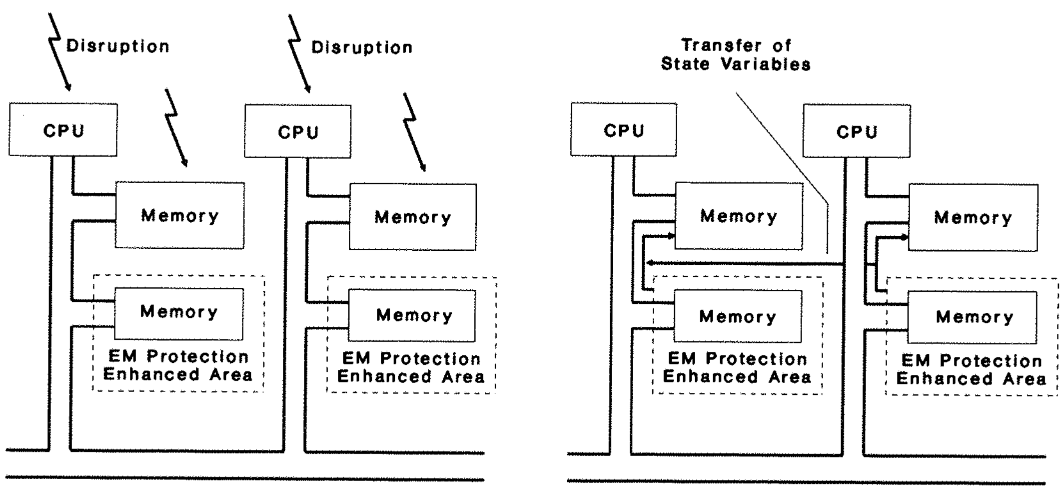


Figure 6.2 In–Lane Recovery

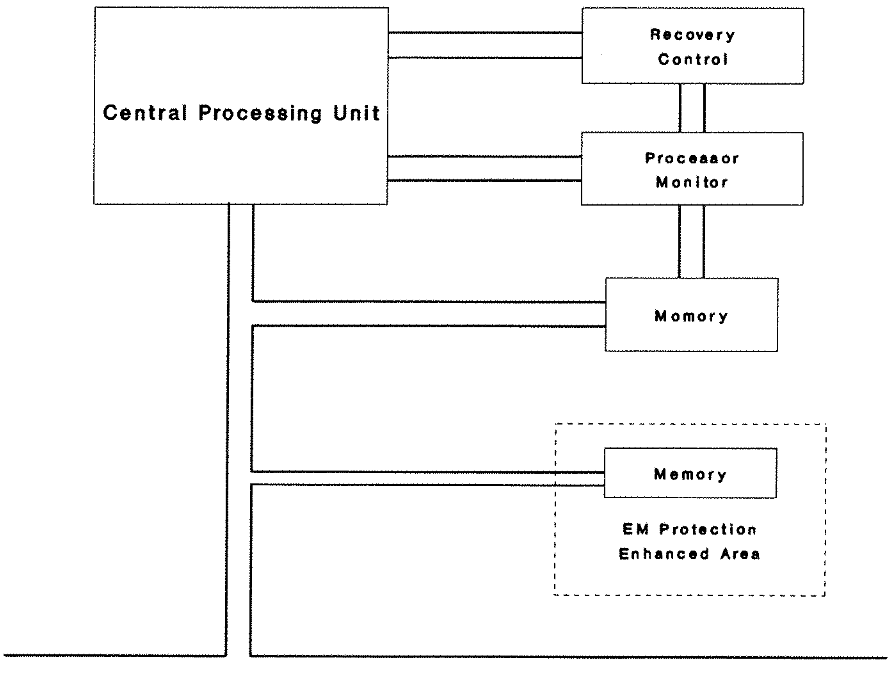


Figure 6.3 Digital Processing with Elements for Transparent Recovery

7 Fault Tolerance in Aircraft using Analytical Redundancy and Knowledge-based Concepts

7.1 Introduction

Fault tolerance has always been an essential requirement of aircraft design. The technique currently adopted in aircraft to provide fault tolerance is based on the concept of hardware redundancy, i.e. the use of multiple processors, communication paths, sensors and actuators accompanied by a voting system. Although the concept has provided useful information for fault detection, isolation and accommodation the benefits have been achieved at a price: increase in weight and cost. In recent years, however, a number of novel Fault Detection and Isolation (FDI) techniques have emerged, which appear to provide the possibility of reducing the level of hardware redundancy, while still maintaining the high level of reliability and availability through the use of analytical redundancy and knowledge based concepts. This chapter reviews the current fault tolerance techniques, and addresses the issues relating to the use of analytical redundancy and knowledge based concepts for FDI with the view of using such concepts on aircraft.

7.2 The Concept of Fault Tolerance

A fault is considered as an unexpected change in the dynamics of a system, which causes the system to deviate from its normal characteristic behaviour. These changes may be due to plant variations, actuator and sensor failure, computer failure or power supply/transmission failure. Fault tolerance is the ability of a system to provide continued operation in the presence of a limited number of hardware or software faults. The advantages of a fault tolerant system are obvious; they provide increased reliability, maintainability and survivability.

Fault tolerance can be achieved in two ways: firstly, by improving the reliability of the individual functional units and secondly, by employing an efficient Fault Detection, Isolation and Accommodation (FDIA) strategy [Frank 1990]. Fault tolerance in aircraft is currently achieved through the use of hardware redundancy accompanied by a voting system. Although the approach is straightforward and provides useful information for FDIA, it suffers from a number of practical problems. Reported problems associated with the implementation of redundancy include selection logic, nuisance trips, generic failures and reliability of voting system/unit [Pau 1986]. Furthermore the approach also involves substantial penalties in weight, volume, cost and maintenance. One way to mitigate this problem, at least partially, is to use analytical redundancy and knowledge based concepts.

Analytical redundancy concepts, also referred to a as model based approach, make use of mathematical models of the system. They require advanced information processing techniques such as state estimation, parameter estimation, adaptive filtering, variable threshold logic, statistical decision theory and various logical operation [Frank 1990, Patton et. al. 1989, Pau 1986]. FDI using analytical redundancy assume that a model of the system being monitored is available or that it is possible to evaluate certain parameters by using on-line identifications techniques to the input and output signals of the monitored system. By using these models the FDI system generates signals to be used for logical processing to detect a fault and to identify the faulty component. One of the main advantages of this concept is that it does not require installation of any additional equipment, and since an analytical model is used it can be implemented on a computer.

Knowledge based systems or expert systems are computer programs which use knowledge (i.e both structural and functional knowledge) and heuristic reasoning to perform complex tasks at a level of performance usually associated with an expert. The advantage of using knowledge based systems for FDI is that it is able to find a solution to the problem even if the system being monitored is too complex for modelling. Furthermore the diagnosis procedure is made efficient, since the length of reasoning process required to reach a conclusion tends to be small.

7.3 Type of Faults

There are essentially two types of faults: random (physical) faults and generic faults, and they can be grouped into the following general categories [Yount 1984, Yount 1985]:

1) Random hardware faults
2) Generic system faults
3) Generic software faults
4) Generic hardware faults

These general fault types can be broken down into further two categories:

1) Abrupt (sudden) faults
2) Incipient (slowly developing) faults

Of the two, abrupt faults causes the greatest concern in flight critical systems as they are most likely to cause a catastrophic effect on the system [Frank 1990]. Therefore it is vital that these faults can be detected or anticipated early enough so that their effects can be avoided by early system reconfiguration. Incipient faults, although not as serious as abrupt faults can also cause catastrophic failure if left unattended for an extended period of time. The detection of incipient faults play an important role in problems related to maintenance where early detection of worn out components is required.

7.4 General procedures involved in fault tolerance

The task of fault tolerance essentially involves four stages [Iserman 1984, Patton 1990, Xu 1991]:

1) Fault detection
2) Fault isolation
3) Fault identification/evaluation
4) Fault accommodation

In the first stage, i.e. fault detection, the appearance of a system fault is detected by checking if certain measurable or unmeasurable estimated parameters are within an expected range of values. A system failing this check is considered to be at fault. In the second stage, i.e. isolation, the location of the fault is determined. The first two stages of the operation are collectively known as fault detection and isolation. In the third stage, i.e. fault identification/evaluation, an assessment is made on how the fault will affect the system. If the fault is evaluated to be tolerable, the operation of the system is allowed to continue. However if the fault is intolerable, the operation of the system is stopped, the fault eliminated and the system reconfigured before the system is restarted. The reconfiguration operation involves the final stage of the fault tolerance operation, i.e. fault accommodation. Figure 7.1 shows a schematic of the fault tolerance operation [Iserman 1984].

7.5 Fault detection techniques - Traditional

The traditional approaches to fault detection include [Patton 1990]:

1) Limit checking - Outputs from the system units are checked against preset values or limits. Any system output exceeding its corresponding limit represents a fault condition.
2) Installation of special sensors - Usually these are limit sensors, which carry out performance limit checks in hardware (e.g. limit temperature or pressure) or sensors which measure some special parameters such as sound, vibration etc.
3) Frequency spectrum analysis - Some system measurements are characterised by typical frequency spectrum under normal operating condition. Any deviation from this typical pattern indicates a fault or malfunction.

7.6 Fault tolerance techniques - Traditional [Yount 1984, Yount 1985]

Random hardware faults

Random hardware faults are mainly due to weakness in the components. The problem is however overcome by using highly reliable components and hardware redundancy i.e. multiple components.

Figure 7.2 shows a classification of hardware redundancy scheme in the context of processor fault tolerance. In the uni-resource system there are no redundant hardware modules, but there may be more than one unit of the same type in the system. However their individual functions cannot be reallocated to another unit in the event of a failure.

Multi-resource systems have some level of hardware redundancy and are usually implemented as either dedicated or pooled resources. In the dedicated multi-resource system, there are more than one spare unit dedicated to a function. These units are used in two ways:

1) To execute functions in the event of the failure of the active unit
2) To execute functions in parallel with the active unit, so that their outputs may be compared for the purposes of failure detection.

In the pooled multi-resource system, the spare units are allowed to perform the functions of any other unit in the pool, and they are usually implemented as either cold spares or hot spares. In the cold spare scheme, the spare unit is only activated in the event of a failure of one of the active units. By contrast in the hot spare scheme, all the spares execute functions in parallel with the active unit in preparation to immediately take over the task of the active unit in the event of its failure.

Generic system faults

The main cause of generic system faults are system specification errors. These faults are minimised by using extensive validation tests such as flight test verification and simulation.

Generic software fault

Generic software faults are caused by errors introduced unintentionally during the development of a software. The two primary techniques that have been developed to tackle this problem are n-version programming or multi-version software and recovery blocks. In the multi-version software technique, two or more alternative versions of the software are executed concurrently and the selection of a final result is accomplished using a voting system or decision algorithm (Fig. 7.3). The concept of recovery block is analogous to the hardware concept of cold spares. Only one version of the software is executed. Upon the detection of an error by an acceptance test, a prior state which is known to be fault free is restored, and an alternative version of the software is loaded and software execution is continued (Fig. 7.4).

Generic hardware faults

There are two types of generic hardware faults: component generic fault and sequential machine generic fault. Component generic faults, although relatively rare, are usually caused by errors introduced during manufacture and by undetected design flaws. Sequential machine generic faults are caused by rare external environmental states which make the machine to enter a rare state thereby causing identical processors to fail simultaneously. Both the problems are however overcome by using dissimilar hardware.

7.7 Modern approaches to fault detection and isolation

Analytical redundancy

The general procedure of fault tolerance using analytical redundancy essentially consists of three steps [Frank 1989, Frank 1990]:

1) Generation of residuals i.e. signals that carry information about the faults
2) Detection of the fault and the isolation of the faulty element
3) Accommodation of the faulty system to normal operation

Figure 7.5 shows a schematic of a conceptual structure of a fault detection scheme using analytical redundancy. The analytical redundancy approach makes use of a residual generator whose function is to generate an output (i.e. a residual) based on the difference between the outputs of the actual system and that of a system model which simulates the normal behaviour of the system. When there are no faults in the system, the residual generator produces no output. However when a fault occurs a residual is generated which is used to form an appropriate decision function. These are evaluated in the fault decision logic to monitor both time of occurrence and location of the faults.

Figure 7.6 shows a detailed structural diagram of a FDI system. The system makes use of three types of models: normal model, observed model and faulty system model. To achieve a high degree of failure detection the model of the normal system must be known and tracked with precision. The model of the faulty system shows the effects the faults have on the parameters being analyzed.

There are a number of ways of generating the residuals using analytical redundancy, some of the more common schemes being the dedicated observer approach, the fault detection filter approach and the parameter identification approach [Iserman 1984, Labarrer, Patton et. al. 1989, Patton 1990].

The general idea behind the observer approach is to predict the system output states from the model of the system, given the input, and then to use this estimation to generate an error for the detection and isolation of faults. Figure 7.7 shows a basic configuration of a linear full order state estimator.

The simplest configuration (Fig. 7.8) that can be used to detect instrument failure is the single estimator (observer or Kalman filter). In this configuration, the estimator is driven by the output of a single sensor to yield a full reconstructed output. The actual output, **y**, is then compared with the estimated output, $\hat{\mathbf{y}}$, to generate a residual $\mathbf{r} = \mathbf{y} - \hat{\mathbf{y}}$, which with the aid of threshold allows the detection of single faults.

An improvement on the above approach, which provides more flexibility in the isolation of sensors and actuator faults is achieved by using a bank of estimators (Fig. 7.9). In this configuration, the estimators are driven by the actual output of the system accompanied by multiple hypothesis testing to detect failures. In this case each estimator would be designed for a different fault hypothesis, e.g. H_1: no faults, H_2: bias in sensor 1; H_3: zero output in sensor 1, etc. [Frank 1990, Patton et. al. 1989].

The fault detection filter (or failure sensitive filter) is a full order state estimator with a special choice of feedback gain H (see Fig. 7.7). By proper choice of H, the filters are made to be sensitive to specific failures and thus provide output signals that are characteristic of the failure [Patton 1990].

The parameter estimation approach is an alternative to state estimation approach, where the scheme makes use of the fact that faults in dynamic systems are generally reflected by changes in physical parameters, as for example friction, mass, resistance etc. The basic idea of parameter estimation approach is to detect faults via the estimation of the parameters of the mathematical model. This involves the following procedures [Frank 1990, Patton 1990]:

1) The development of the system equation from the measurable input and output variables
2) Determination of the relationship between the model parameter θ_i and the physical parameter p_i

$$\theta = f(p) \qquad (1)$$

3) Estimation of the model parameter θ_i based on the input and output of the actual system
4) Calculation of the process coefficients

$$p = f^{-1}(\theta) \qquad (2)$$

5) Determination of the deviation, Δp, from its normal value
6) Decision on a fault by using the relationship between faults and changes in physical parameters Δp_i.

Figure 7.10 shows a schematic of a parameter identification approach.

7.8 Knowledge based Systems or Expert Systems

A knowledge based system consists of:

a) A knowledge base

b) A data base

c) An inference engine

d) A user interface

The first component of the knowledge based system is the knowledge base. This contains facts, rules and domain knowledge elicited from the experts, and are more or less in declarative form i.e. independent of how it is to be used to solve problems. The data base is the second component of the knowledge base system and it contains information about the present state of the system. The third component, and the important part of the knowledge based system is the inference engine. It consists of reasoning or problem solving strategies on how to use the knowledge in the knowledge base and the data base to make decisions (conclusions). Arriving at a decision basically involves searching for a solution from a search space i.e from a set of possible solutions. A number of search schemes have been developed to aid with this process, the common ones being [Capon 1990, Jackson 1990, Pau 1986, Xu 1991]:

1) Heuristic or ordered search: Uses domain specific knowledge acquired through experiential knowledge

2) Generate/Test: In this case the search space is not constructed a priori, but as when required. Possible solutions are generated as the system proceeds and are evaluated shortly after.

3) Forward chaining (data driven) / Backward chaining (event driven): In the former chaining occurs from conditions that are known to be true towards a problem state which those conditions allow. In the latter chaining backwards occurs from a goal state towards the condition necessary for its establishment.

The final component of the knowledge based system is the user interface whose function is to request relevant questions of the user, and to convey the message to other parts of the system, to provide advice and answers from the system to the user and to provide the explanation required by the user. Figure 7.11 shows a schematic of a fault diagnosis system.

One of the factors which determine whether or not a knowledge based FDI system is successful, or not, is in the way in which knowledge is represented. Knowledge representation refers to the task of modelling real world knowledge in terms of computer data structure, and is usually closely related to the problem of how the knowledge is used. Since knowledge is extensively used to solve problems in knowledge based system, the knowledge should be efficiently useable and efficiently expandable. Some common knowledge representation schemes include production rules, semantic networks, frames and object oriented representation /programming.

Of all the schemes, production rule is the most popular encoding scheme used to represent knowledge. Production rules have two components which are usually represented as two lists. The first component is a list of one or more conditions and the second, a list of one or more actions which may be performed if and when the conditions are satisfied [Capon 1990, Jackson 1990].

Semantic networks (also referred as semantic nets) is a graph which consists of a number of nodes linked together by arcs. The nodes represent concepts and the arcs represent the relationships between them. Figure 7.12 shows a simple example of a semantic net. This form of representation is also referred to as object-attribute-value representation [Capon 1990, Jackson 1990].

A frame is a data structure which consists of a special slot which contains the name of the object it stands for and a number of slots which contain values of various common attributes associated with the object. Frames are usually arranged in a loose hierarchy in which frames 'lower down' the network can inherit values for slots from frames 'higher up' [Capon 1990, Jackson 1990].

In object orientated knowledge representation or programming, the programs consist of a number of definitions of objects, and these objects would have associated with them attributes which store values local to the objects and methods which allow operations to be performed on them.

7.9 FDI using Knowledge based Systems

Fault detection in knowledge based system is accomplished using two kinds of knowledge: behavioural information about the system, and knowledge about the normal behaviour of the system. Behavioural knowledge is usually obtained directly from the sensors whilst knowledge about the normal behaviour of the system is derived from the simulation of a model of the system. The presence or absence of a fault is determined by comparing the two data - any discrepancies between the data indicating a fault condition.

Once a fault is detected, the next step involves the identification of the fault i.e. fault diagnosis. Fault diagnosis in knowledge based system is usually accomplished using either shallow knowledge models or deep knowledge models. In the shallow knowledge based approach an inference rule exists to explain all the possible faults or malfunctions in the system. In the deep knowledge based approach the information about what kind of observation to expect for each fault is generated by running a simulation of the model under various fault conditions. The two typical models used in deep model based diagnostic systems are structural or connectivity models and functional models. Structural models describe the system structure or connectivity and the way in which they influence the system behaviour, whilst functional models describe how the system and its components work (i.e. interact) during actual operation.

7.10 Discussion

The required level of fault tolerance is currently accomplished using redundancy accompanied by a voting system. Although theoretically, the use of replicated components to improve system reliability, maintainability and survivability may be a good idea, practically, the concept suffers form a number of problems. These include selection logic, nuisance trips, generic failures and reliability of voting unit [Pau 1986]. Furthermore the concept also involves substantial penalties in weight, volume, cost and maintenance.

Analytical redundancy provides the potential for eliminating hardware redundancy. On aircraft, it may be implemented on the on-board computers providing reductions in weight and volume while providing increase in system redundancy and reliability [Patton 1990]. Hardware redundancy, however, cannot be completely eliminated because they are essential for reconfiguration. Therefore in aircraft application, the role of analytical redundancy is seen as reducing the level of hardware redundancy (e.g. from triplex to duplex system).

Fault tolerance using analytical redundancy is based on the generation of residuals, and there are a number of techniques which can be used to generate this signal. To use the parameter estimation approach a necessary requirement is that an inverse relationship equation (2) exist.

One of the problems associated with the use of analytical redundancy for FDI application has been the sensitivity of the detection system to modelling errors [Frank 1990, Xu 1991]. Since the system model on which the redundancy is based is not exactly known, there is a possibility that the actual system outputs may not match the model outputs, even when there are no faults. Thus the residuals will not be zero in general and some form of threshold would have to be used to distinguish faults. But the problem with using thresholds is that they reduce the sensitivity of the detection system - choosing a threshold too low increases the rate of false alarms and choosing it too high reduces the net effect of fault detection. This problem has however been recognised, and several schemes have been proposed in recent years to increase the robustness of the fault detection system. These include robust observers schemes, proper choice of the threshold and adaptive threshold [Emami-Naeini et. al. 1986, Emami-Naeini et. al. 1988, Patton et. al. 1989].

Recently, Merrill and co-workers from NASA Lewis Research Centre carried out a real time evaluation study to detect sensor failure using analytical redundancy. Their studies indicate that software based failure detection algorithms do indeed work and work quite well. Furthermore, their studies indicate that the algorithms can be implemented in a realistic computer environment with an update rate consistent with real time operation [Merrill 1988].

FDI systems using analytical redundancy concepts are designed based on the assumption that a good mathematical model of the system being monitored is available. As a result the achievable quality of the system depends on the quality of the of the model. For large complex systems it may not always be possible to obtain a mathematical model of the system and this is one of the limitations of this approach.

When analytical models are not available for the task of FDI, knowledge based models (i.e. expert systems) can be used instead (Fig. 7.13). This is one of the advantages of knowledge based systems. A further advantage is that they provide the facility to represent experiential knowledge which is hard to come by and difficult to capture numerically. Expert systems can not only be used to complement analytical redundancy and provide failure diagnosis, but they also provide the capability to predict faults (i.e prognostics) before they occur. The implementation of failure prediction on aircraft diagnostic system would provide several benefits: Rapid fault isolation - additional information is made available for the identification of faulty elements; ability to plan maintenance action; reduces the probability of 'knock on' faults; and enhances safety.

A number of expert systems currently exist which are capable of providing FDI. A comprehensive list of various knowledge based systems together with a review of their capabilities can be found in references Gilmore 1987, Pau 1986 and Tzafestas 1989.

The selection of an appropriate knowledge base is an important issue in knowledge based systems, since the problem solving strategy is heavily dependent upon their representation. IF_THEN type production rules provide an effective way of representing heuristic knowledge especially if the knowledge is plentiful but relatively unstructured. However this type of representation is not easy to program, for example, knowledge about facts (X is the name of the aircraft), knowledge about system structure (the system consists of actuators and sensors) and knowledge about causal relationships (failure of computer will cause the following malfunctions) are difficult to represent in a simple IF_THEN type representation [Pau 1986]. Knowledge representations based on frames and objects, on the other hand have proven useful in applications where detailed knowledge was required [Iverson, Jackson 1990].

The main advantage of rule based (i.e. shallow) expert systems is that it provides a single uniform method of knowledge representation. Furthermore it allows for the incremental growth of knowledge through the addition of rules. However there are a number of problems associated with this approach such as completeness of knowledge base not guaranteed, excessive number of rules and knowledge base being highly specialised to the individual process [Capon 1990, Patton et. al. 1989, Xu 1991].

These disadvantages can however be overcome, to an extent, by decomposing the problem into smaller problems either in a hierarchial manner or according to unit operations using either a fault dictionary (i.e. a list of cause and effects) or the diagnostic tree (i.e. a way of restricting the search to go along different diagnosis paths) [Tzafestas 1989]. But these schemes also do have a number of drawbacks. Since they use look up tables, all the potential faults have to be computed in advance. Furthermore, for a large system the look up tables can have a large number of entries and as a result the diagnostic procedure becomes difficult and time consuming.

The deep knowledge based diagnostic system however does not suffer from any of the drawbacks of the shallow based approach since it uses deep models (structural and functional models). Deep knowledge based expert systems are capable of computing the underlying principles of the system explicitly and therefore do not need to predict every fault condition or need to use precomputed rules. As a result the approach is able to handle a wide range of problem types and large problem domain.

One of the essential requirements of an onboard fault diagnosis system is the ability to respond to faults quickly and to provide safe and reliable system reconfiguration. Efficient diagnosis systems can be implemented by combining quantitative models (i.e analytical redundancy) with qualitative reasoning (i.e. knowledge based concepts). These hybrid diagnosis systems are generally referred to as Intelligent Quantitative and Qualitative (IQ^2) systems since they are capable of reasoning, perceiving relationships and analogies, storing and retrieving information, and adjusting to new situations. For example, analytical redundancy can be used to detect irregular behaviour and qualitative reasoning in the form of forward/backward chaining can then be applied to this knowledge to reach a diagnostic conclusion. If however, the symptoms fail to correspond to a known fault, deep reasoning can then be used to identify and confirm the fault location. The idea behind using shallow reasoning first is to narrow the number of possible faults or fault locations thereby reducing the computational burden of deep reasoning. Alternatively the two concepts can be used to complement each other (Fig. 7.14). The advantage of this approach is that it provides redundant information for fault diagnosis. Furthermore, the diagnosis system is able to provide a wide fault coverage as it can make use of a variety of sources of knowledge and also base its diagnostic knowledge not only on numerical/analytical evaluation techniques but also on qualitative and symbolic logical inference. The IQ^2 schemes however incur a significant increase in computational expenditure and as such it would not be practical to implement such schemes other than for flight critical applications such as flight control and engine control.

The outputs from the IQ^2 diagnosis system can also be used to perform reconfiguration. However to perform system reconfiguration some level of redundancy is required. Redundancy can be provided through the replication of system components i.e. parallel channels for measurement and control, and from flexible logic that synthesise missing measurements or control forces using operable sensors and actuators. As mentioned earlier, faults in a 'system' can develop due to a variety of reasons: sensor failure, controller failure and plant failure. Of these failures, plant failure is the most difficult to contain, and as yet there have been no techniques reported to accommodate this kind of fault. Sensor and controller bias failures can be accommodated through artificial unbiasing of the suspect sensor or control signal, whilst stuck sensors and 'runaway' controllers can be accommodated by retrieving from memory, gains that were calculated off-line assuming that the faulty element never existed.

One of the key concepts in achieving high levels of integration in architectures like the IMA (Integrated Modular Avionics) is fault tolerance. By incorporating intelligent quantitative and qualitative schemes in the IMA architecture the fault tolerance capability of the system can be further enhanced to provide the margin of safety required for flight critical functions.

Combination of analytical redundancy and knowledge based concepts has been applied to the problem of aircraft fault tolerance by Handleman and Stengel. However the system is only capable of tolerating a limited number of faults [Handleman 1985, Handleman 1986]. More recently Xu has implemented a fault diagnosis system using the aforementioned concepts to monitor faults in flight control [Xu 1991].

Knowledge based systems, especially large complex systems can be prone to uncertainties. The uncertainty usually arises from three main sources: lack of complete data, incomplete pertinent knowledge and uncertainty inherent in the process. As a result the ability to handle uncertainties has been an essential requirement in knowledge base systems. There are a number of techniques which have been proposed to handle uncertainties, these include the use of probability (i.e. Bayes theorem), Dempster Shafer belief theory, certainty factor and fuzzy logic [Capon 1990, Pau 1986].

One of the concerns of using analytical redundancy and knowledge based concepts is the large increase in computational expenditure. Although large increases in processing speed are becoming available with the development of high throughput 32 bit processors, it is envisaged that the computational requirement needed to implement an expert diagnosis system for all of the aircraft subsystems are beyond current avionic computational capability [Palmer et. al. 1987, Schutte 1989]. A solution to this problem would be to use parallel computer architecture i.e. parallel processing. With the advent of transputers, recently, it has been shown that fault tolerant systems can be built based on parallel processing. Systems designed based on this concept is said to offer not only high reliability and high performance but also lower cost, graceful degradation and enhanced system maintainability [Beton 1991]. But the application of parallel processing (transputers) on civil aircraft raises certification issues and therefore needs further investigation. Current research at University of Southampton Parallel Applications Centre on engine digital fuel pump controls, indicate that the above methodology is both feasible and certifiable. Another major concern is the software reliability. Since both the concepts are software based, reliability of software is paramount. N-version programming has been suggested as solution to the software reliability problem.

7.11 Conclusion

In this section an attempt has been made to address the issue of fault detection and identification using analytical redundancy and knowledge based concepts. The main advantage of these approaches is that it can be implemented without the need for any additional hardware in the system, providing savings in weight, volume and maintenance.

Analytical redundancy makes use of mathematical models, as a result the achieved quality of the FDI system depends on the quality of these models. Knowledge based systems make use of heuristic reasoning to perform FDI. The advantage of this approach is that it is able to find a solution to the problem even if the system being monitored is too complex for mathematical modelling.

Both analytical redundancy and knowledge based concepts have their limitations however. But the limitations can be overcome by combining the two schemes together to yield a flexible and efficient diagnosis system - the IQ^2 system.

The cost for these benefits are a substantial increase in computational expenditure. However with the development of 32 bit high-throughput computers it is envisaged that the increase in processing requirement can be accommodated. Furthermore transputers can be called upon to ease the computational burden.

To-date, very few researchers have considered the possibility of combining quantitative models with qualitative reasoning for fault diagnosis. In view of the many advantages that can be accrued from such a scheme, the concept merits further investigation.

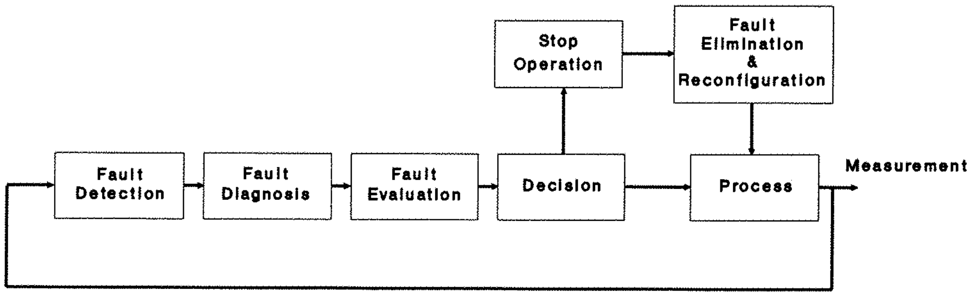


Figure 7.1 Schematic of Fault Tolerance

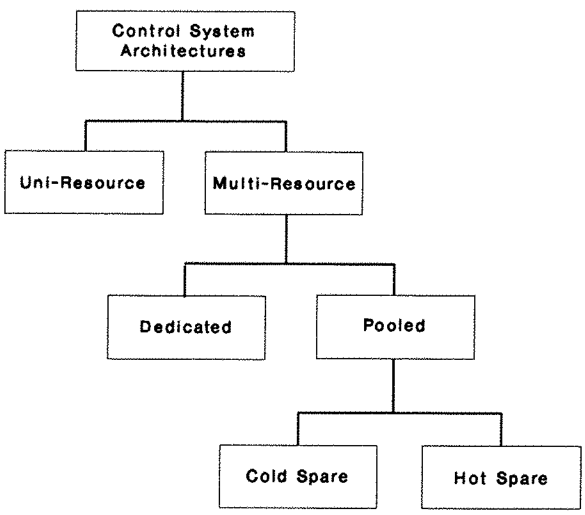


Figure 7.2 Hardware Redundancy Scheme

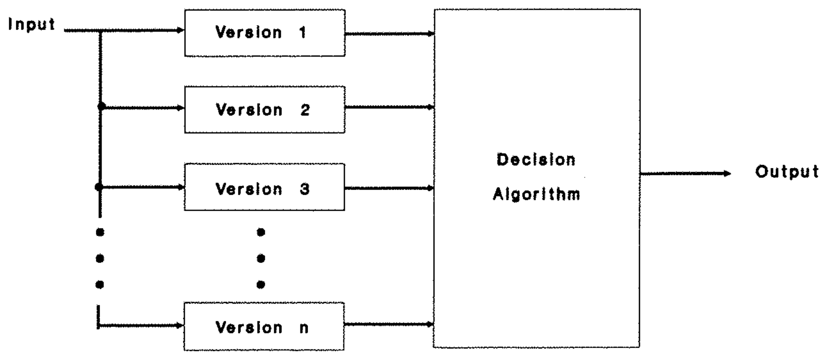


Figure 7.3 Multi−version Software

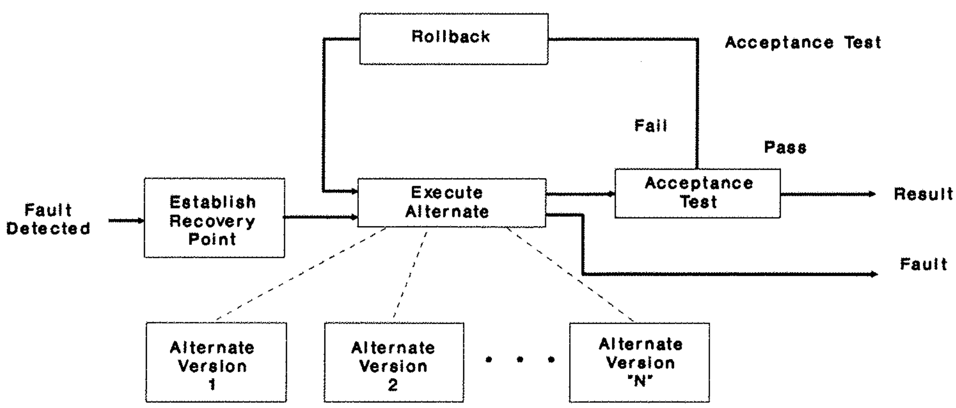


Figure 7.4 Recovery Block

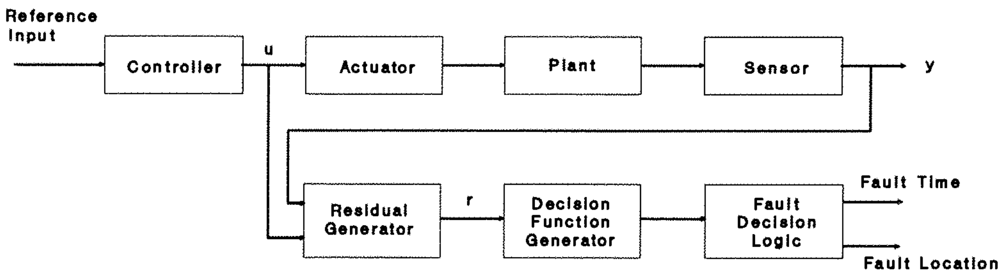


Figure 7.5 Conceptual Structure of Fault Detection and Isolation using Analytical Redundancy

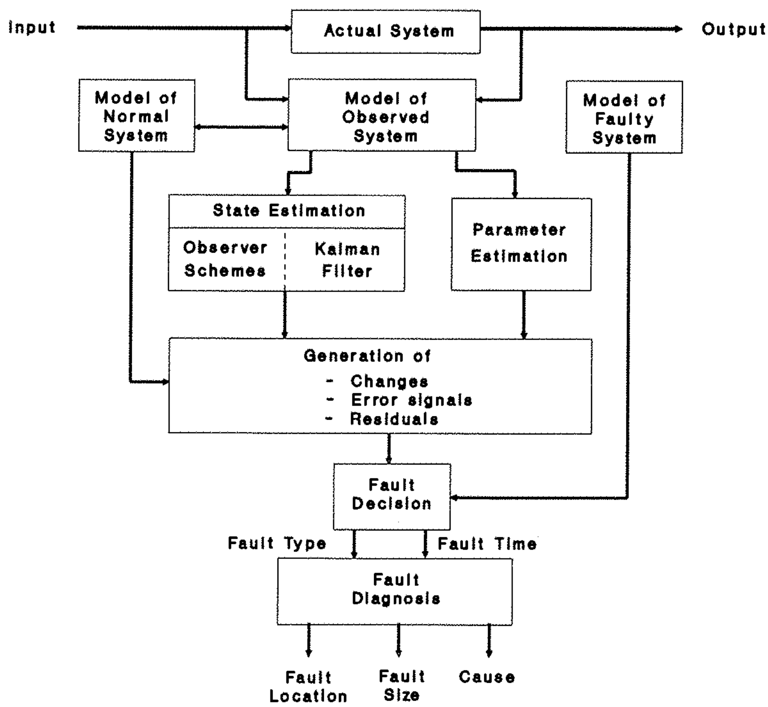


Figure 7.6 General Architecture of FDI based on Analytical Redundancy

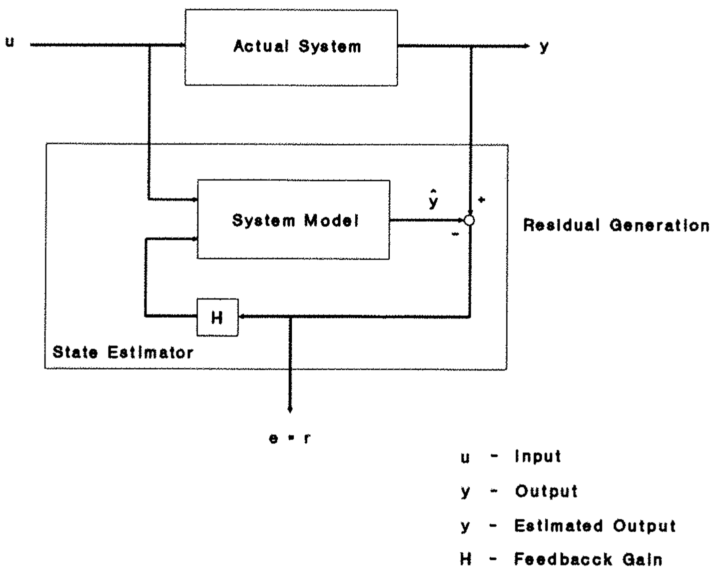


Figure 7.7 Basic Configuration of Residual Generation through State Estimation

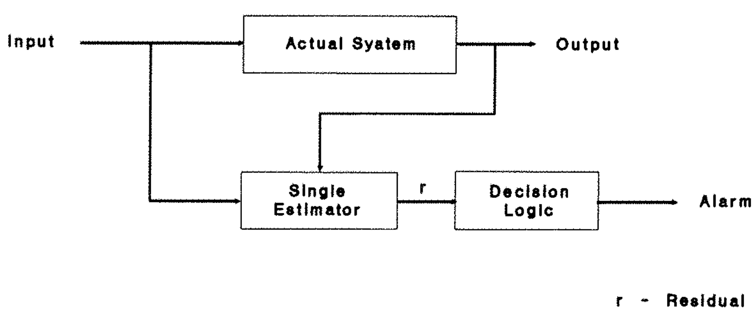


Figure 7.8 Fault Detection using Single Estimator

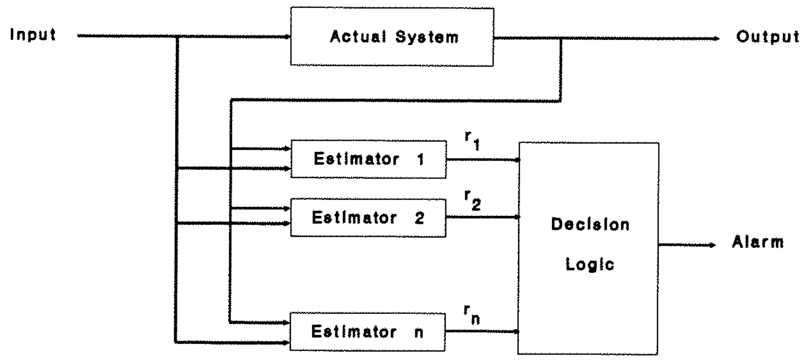


Figure 7.9 Fault Detection using a Bank of Estimators

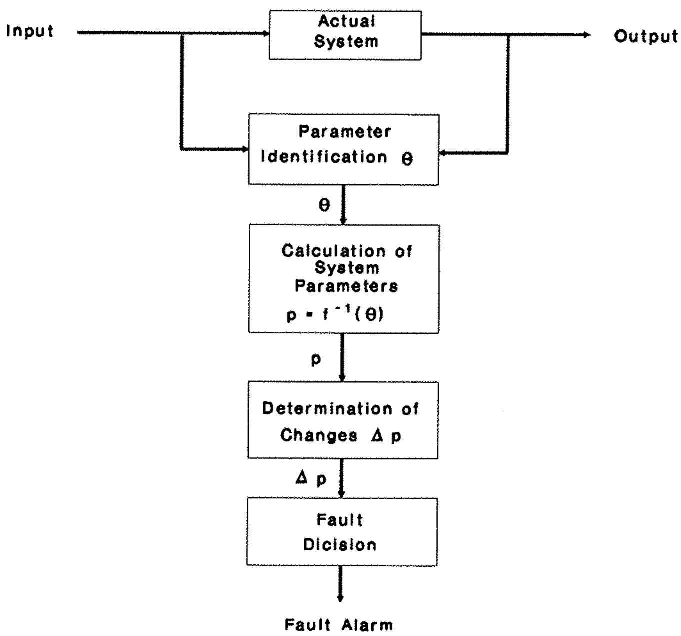


Figure 7.10 Parameter Identification Approach to Fault Detection

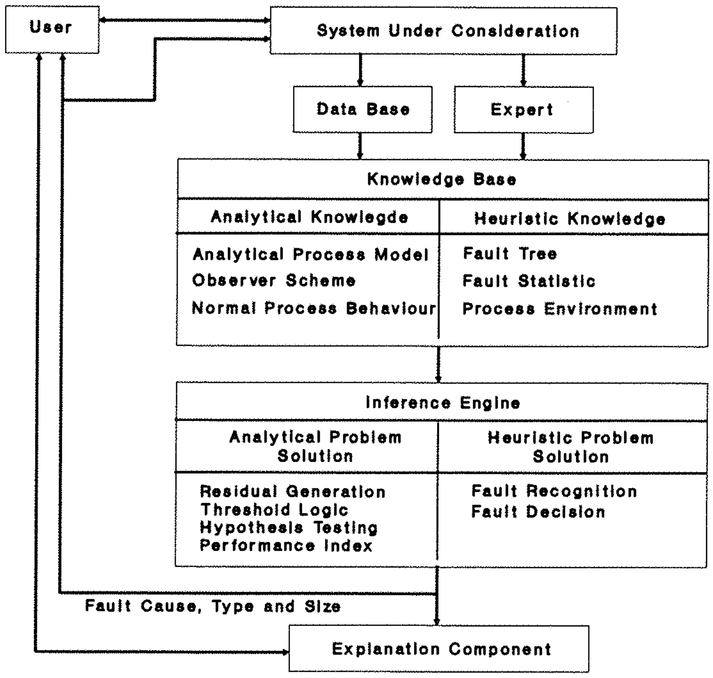


Figure 7.11 Schematic of a Model and Knowledge Based Diagnostic System

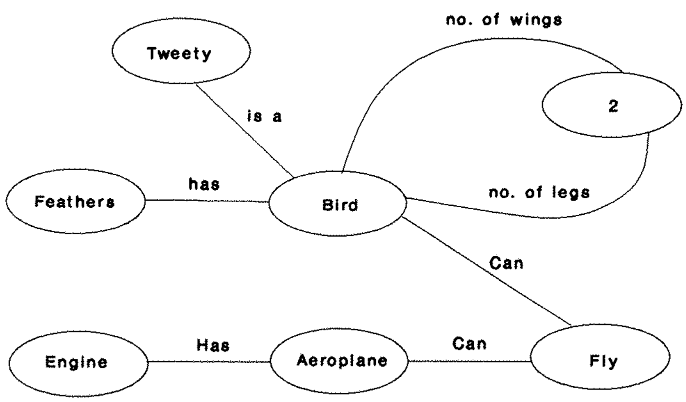


Figure 7.12 Semantic Network

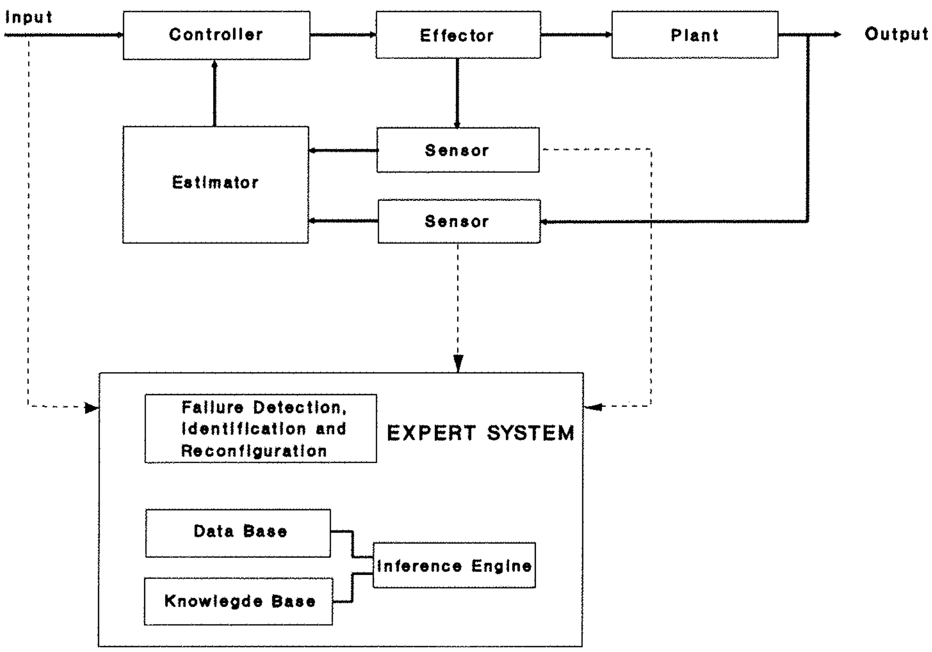


Figure 7.13 Expert System approach to Analytical Redundancy

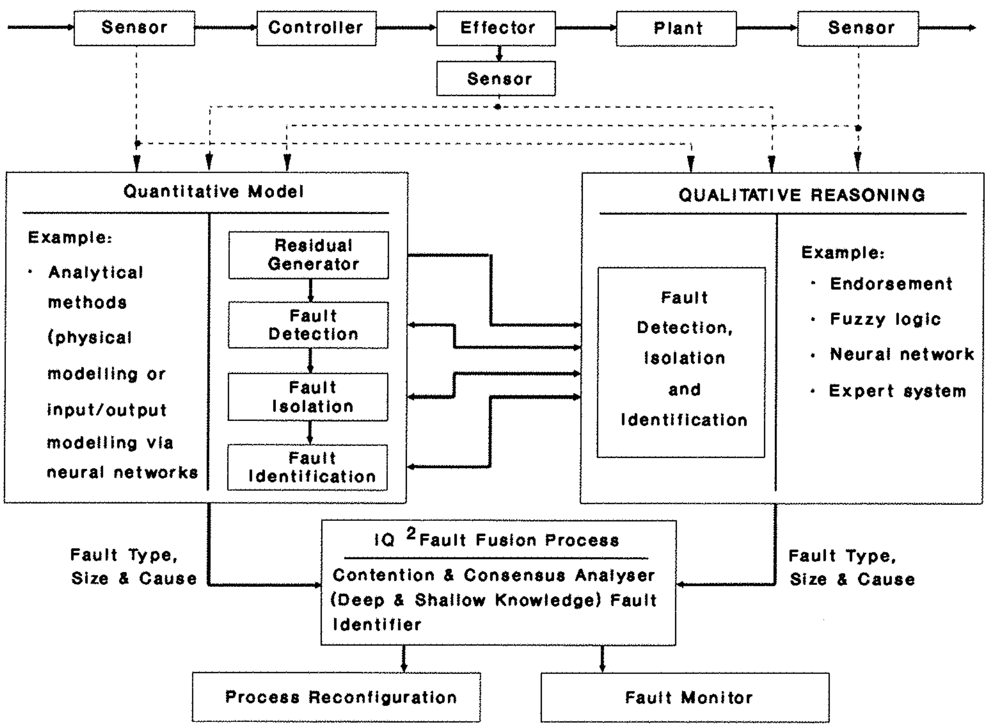


Figure 7.14 The IQ^2 System

8 Multi-sensor Data Fusion to Enhance the validity of Sensor Derived Information

8.1 Introduction

No sensor can faithfully provide accurate measurement of the measurand all of the time. Sensors generally introduce uncertainty in the quantity they measure. The uncertainties may be due to sensor noise, incorrect measurements or even due to actual failure of the sensor. As a result data from a single sensor cannot be entirely relied upon and ways need to be found to reduce the uncertainty, and thus improve the level of confidence in the information provided by the sensor. One way of reducing uncertainty is to use multiple sensors. In order to use multiple sensors, it is necessary to perform local estimation of the parameter under observation, combine the local estimates in a central processor and to detect, isolate and accommodate sensor failure. This process of combining information from a number of identical sensors or a range of disparate sensors to produce a single, best estimate of the system state being observed is known as sensor integration or multi-sensor data fusion.

This chapter addresses the issues relating to the use of multi-sensor data integration to enhance the validity of sensor derived information. A suitable data integration algorithm is described which uses the information from the individual sensors effectively, to yield a single best estimate of the system state being observed. The objective is to examine the advantages and drawbacks of various multi-sensor architectures, and in doing so identify an architecture, which could be used on a flight critical system to enhance the performance and reliability of the system. The section also discusses the implications of multi-sensor integration on the integrated modular avionics (IMA) architecture.

8.2 Overview

It may appear obvious that the use of multiple sensors (of the same type or of different type) would give a better or more accurate measurement, however, this is not necessarily true. For instance, simple averaging of sensors can be subjected to distortion by individual rogue measurements (outliers), and even when there are a few outliers the averaging makes the overall variance of measurement worse. However it has been shown that through proper choice of fusion algorithms the accuracy of the resultant data arising from the integration of multiple sensors can be greatly improved.

There are a number of advantages in integrating information from a multiplicity of identical or disparate sensors [Harris 89]:

1) Robust operation
2) Increased confidence

3) Reduced ambiguity
4) Improved detection
5) Improved system reliability

However, multi-sensor data fusion is not without its problems. Many of the problems associated with sensor data fusion centres around technique for dealing with uncertainty and error. Most data fusion techniques require that the uncertainty be symmetrical and independent [Richardson 1988]. But the uncertainty in information from different sensors may not be symmetrical or independent. Additional problems can also result from errors during sensor operation (e.g. calibration), sensor failure and the control necessary to co-ordinate the operation of different sensors [Luo[a] 1988, Luo[b] 1988, Rao 1991, Rothman 1989].

Multi-sensor data integration is a complex process and covers many specific issues. Therefore in order to simplify the study, only one class of problem is addressed, the one most relevant to commercial aircraft, i.e how to integrate data from a number of sensors of the same modality measuring the same entity.

8.3 The Categorisation of Data Fusion

The problem of data fusion can be categorised into three problem domains [Miles 1988, Harris 1988]:

1) Designed (or well structured) world:

   In this domain, the normal states are known, and the data sources are reliable and accurate (e.g. industrial processors, ATC).

2) Benign real world:

   Here the normal states are only partially known, the data sources are reliable but have poor coverage, the data rates are partially variable and the system is usually unaffected by sensing (e.g. patient monitoring, weather forecasting).

3) Hostile real world:

   In this class of problem the normal states are not easily defined, and the data rates are highly variable. Furthermore the data sources can be inaccurate, unreliable and subjected to interference, and sensing can significantly affect the system, e.g. military C^3 (Command, Control and Communication systems).

These worlds can be further categorised into spatial descriptions:

1) Small world:

   This category covers a world where sensors provide complete coverage and in which real time sensor data is utilised. Typical examples of sensing include velocity, temperature, colour and shape size.

2) Large world:

   This category describes a world where the coverage is incomplete, imprecise and requires other world knowledge to obtain the best results. Typical examples include C^3I systems and long range targeting/tracking.

8.4 The Concept of Multi-sensor Data Fusion

The data fusion process essentially involves three stages [Durrant-Whyte 1988, Harris 1989]:

1) Estimation of data
2) Consensus of data
3) Assimilation of data

Since the class of problem under consideration fall under the category of 'small' and 'designed' worlds, the consensus issue becomes unnecessary since the system being observed has been 'man-designed' to operate with known parameters. Therefore the two issues that need to be addressed in data fusion problems involving similar sensors measuring the same entity are estimation of data and assimilation of data. However, if consensus is an issue it can be implemented through hypothesis testing.

Estimation of Sensor Data [Isermann 1981, Rao 1991]

Estimation of data requires some knowledge of the data source (i.e. the sensor), this is obtained by modelling the capabilities of each sensor. Since estimation of data is essentially a prediction process, it may be represented through first order estimators such as a Kalman filter in which the sensors can be represented in state-space form.

Consider a linear system described by the following state and output equations:

$$\mathbf{x}(k+1) = \mathbf{A}\mathbf{x}(k) + \mathbf{F}\mathbf{v}(k) \qquad (1)$$

and

$$\mathbf{y}(k) = \mathbf{C}\mathbf{x}(k) + \mathbf{n}(k) \qquad (2)$$

where $\mathbf{x}(k)$ - State vector

$\mathbf{v}(k)$ - Input noise vector with known statistical property and co-variance matrix $\mathbf{V}$

$\mathbf{y}(k)$	-	Output vector
$\mathbf{n}(k)$	-	Output noise vector with known statistical property and co-variance matrix $\mathbf{N}$
$\mathbf{A}$	-	System matrix
$\mathbf{F}$	-	Input matrix
$\mathbf{C}$	-	Output matrix

A, **F**, and **C** are assumed to be time invariant. The objective is to estimate the state vector $\mathbf{x}(k)$ based on measurements of the output $\mathbf{y}(k)$ which are contaminated by Gaussian white noise $\mathbf{n}(k)$. The following are assumed known a priori:

A, **C**, and **F**

$E\{\mathbf{v}(k)\} = 0$

$\mathrm{cov}[\mathbf{v}(k), \tau=i-j] = E\{\mathbf{v}(i)\mathbf{v}(j)^T\} = \mathbf{V}\,\delta_{i,j}$

$E\{\mathbf{n}(k)\} = 0$

$\mathrm{cov}[\mathbf{n}(k), \tau=i-j] = E\{\mathbf{n}(i)\mathbf{n}(j)^T\} = \mathbf{N}\,\delta_{i,j}$

$$\textit{where}\quad \delta_{ij} = \begin{cases} 1 \; \textit{for} \; i = j \\ 0 \; \textit{for} \; i \neq j \end{cases}$$ is the Kronecker delta function.

As the estimates are time varying in most applications a recursive estimation is preferred, in which the states $\mathbf{x}(k)$ are calculated after the measurement of $\mathbf{y}(k)$.

For a system described by state and output equations (1) and (2), the Kalman filter provides a recursive solution for the estimate $\hat{\mathbf{x}}(k|k)$ of the state $\mathbf{x}(k)$ in terms of the previous estimate $\hat{\mathbf{x}}(k-1|k-1)$ and the new measurement $\mathbf{y}(k)$ given by

$$\hat{\mathbf{x}}(k|k) = \hat{\mathbf{x}}(k|k-1) + \mathbf{W}(k)[\mathbf{y}(k) - \hat{\mathbf{x}}(k|k-1)] \tag{3}$$

with an associated covariance

$$\mathbf{P}^{-1}(k|k) = \mathbf{P}^{-1}(k|k-1) + \mathbf{C}^T(k)\mathbf{N}^{-1}(k)\mathbf{C}(k) \tag{4}$$

where

$\mathbf{W}(k) = \mathbf{P}(k|k)\mathbf{C}^T\mathbf{N}^{-1}(k)$ is the Kalman gain matrix

$\hat{\mathbf{x}}(k|k-1) = \mathbf{A}(k)\hat{\mathbf{x}}(k-1|k-1)$

$\hat{\mathbf{x}}(k|k-1) = \mathbf{C}(k)\hat{\mathbf{x}}(k-1|k-1)$

$\mathbf{P}(k|k-1) = \mathbf{A}(k)\mathbf{P}(k-1|k-1)\mathbf{A}^T(k) + \mathbf{B}(k)\mathbf{V}(k)\mathbf{F}^T(k)$

Note: The estimates derived above are for a linear system. The Kalman filter can be applied to non-linear processes as the extended Kalman filter by linearising non-linear functions about various operating points [Rao 1991].

Assimilation of Sensor Data

Having obtained several state estimates $\hat{x}(k|k)$ relating to the same entity, the next step is to effectively utilise this information to yield a single, accurate estimate of the entity being observed.

A number of numerical methods exist for achieving this goal, the major ones being Bayesian, Dempster-Shafer and Fuzzy logic [Blackman 1988, Harris 1988, Harris 1989, Luo 1988]. For the problem in hand, a fusion algorithm based on Bayesian formalisation is used, since it leads directly to a nice interpretation of results.

Consider a state vector $\mathbf{x}(k)$ (say position, velocity etc.) observed by n sensors. Assuming that the measurements are Gaussian distributed, the overall (or combined) estimate of the sensors can be expressed as follows:

$$\hat{x}(k|k) = \frac{\sum_{i=1}^{n} \hat{x}_i(k|k-1) P_i^{-1}(k|k)}{\sum_{i=1}^{n} P_i^{-1}(k|k)} \tag{5}$$

Eq.(5) is not just a raw average of estimates from n sensors. The estimates are weighted by the estimator covariance matrices. This helps in biasing the sensors with large covariance (mean squared error), there by reducing their contribution to the overall estimate. In general, the covariance matrix of each sensor will be different due to their individual characteristics. However if we consider the special case of identical sensors measuring the same entity, and if it is assumed that the covariance of all the sensors are equal, then Eq.(5) reduces to

$$\hat{x}(k|k) = \frac{1}{n} \sum_{i=1}^{n} \hat{x}_i(k|k-1) \tag{6}$$

which is simple averaging of the estimates from individual sensors.

8.5 Multi-sensor System Architectures

The key to developing an efficient multi-sensor system lies in the organisation of the system, which depends on the architecture (i.e., the interconnections of all the sensors) and on the co-ordination and control of the system (i.e., how information is to collected and distributed around the system). There are a number of multi-sensor architectures, each with varying degree of merit. They can be categorised as either centralised or decentralised or as a hybrid - a 'halfway house' between the two [Hashemipour et. al. 1988, Rao 1991]. Figure 8.1 shows a schematic of a centralised multi-sensor architecture which is the simplest architecture for a multi-sensor system. The system consists of a number of 'dumb' sensors connected to a central processing unit. Raw data from each sensor is sent to the central unit where they are combined to yield a unified result.

Figure 8.2 illustrate a hierarchically centralised multi-sensor system architecture. The structure of the system is similar to the centralised architecture but with one major difference: the sensors are made 'semi - intelligent' by providing each sensor with limited processing capability. In this architecture the data acquisition is performed locally, but the fusion of sensor data is still performed at the central unit.

A schematic of a blackboard based decentralised architecture is shown in Figure 8.3. The system consists of a number of sensors, each with its own processing unit, connected to a common communication medium called the blackboard, which is essentially an area of shared memory. Each sensing node has sufficient processing power which enables it to control sensors, perform data acquisition and validation, and also to fuse received data. Each node sends its local information after some processing, to the blackboard where it is made available to all the other nodes. When a node is ready to receive information, it simply reads off the data from the blackboard. Since all the nodes communicate with each other via the blackboard a global result is realised at each node.

Figure 8.4. shows a schematic of a fully decentralised multi-sensor architecture. The system consists of a number of sensors each with its own local processing unit with the capability to control sensors, perform data acquisition and validation, and also to fuse received data. Although the sensor nodes are similar to that of the blackboard based architecture, they are not however connected to a common communication medium, but instead connected directly to each other. The architecture has no central controller, instead the responsibility for the control of the node is left to the node itself. Furthermore the co-ordination of information transfer between the connected nodes take place without the need for a supervisory facility. The nodes perform sensing operation at will and share their local information with the nodes connected to it in order to distribute the information throughout the system.

In the decentralised multi-sensor architecture, each sensor computes its local estimate and its variance in accordance with Eqs. (3) and (4), and communicates this information to every sensor node that is connected to it. At each node the fusion algorithm described by Eq. (5) is used to combine the local estimate together with the estimates from other sensors to yield an overall global estimate.

8.6 Failure Detection

One of the major difficulties associated with the use of multiple sensors is the failure to detect erroneous measurements [Bullock et. al. 1988]. It is essential that these measurements be omitted from the weighted average as they would otherwise increase the uncertainty in the estimate. A sensor may register an erroneous measurement due to a number of reasons, the most common being changes in sensor characteristics (i.e. sensor degradation), sensor noise and actual failure of the sensor (i.e open circuit or short circuit).

In general, as the sensor degrades its mean distribution (i.e covariance) becomes large. But as the integration algorithm is weighted by the estimator covariance matrix, the contribution made by the degraded sensor to the overall estimate would be small. However it should be noted that as the covariance becomes large so does the Kalman gain matrix $\mathbf{W}(k)$. The consequence is that the quality of the estimate may be poor. The reason for this is discussed in the following paragraphs.

When a sensor is described by the following state and output equations:

$$\mathbf{x}(k+1) = \mathbf{A}\mathbf{x}(k) + \mathbf{B}\mathbf{u}(k) + \mathbf{v}(k) \tag{7}$$

$$\mathbf{y}(k) = \mathbf{C}\mathbf{x}(k) + \mathbf{w}(k) \tag{8}$$

changes in sensor characteristics are usually reflected by changes in sensor parameters, i.e. **A**, **B** and **C** matrices. Therefore Eqs. (7) and (8) can be written to account for parameter changes as follows:

$$\mathbf{x}(k+1) = (\mathbf{A}+\delta\mathbf{A})\mathbf{x}(k) + (\mathbf{B}+\delta\mathbf{B})\mathbf{u}(k) + \mathrm{v}(k) \tag{9}$$

$$\mathbf{y}(k) = (\mathbf{C}+\delta\mathbf{C})\mathbf{x}(k) + \mathbf{w}(k) \tag{10}$$

where $\delta\mathbf{A}$, $\delta\mathbf{B}$ and $\delta\mathbf{C}$ are changes in sensor parameters.

This raises the question as to whether or not the same Kalman filter can be used for the sensor described by Eqs. (7) and (8), as well as Eqs. (9) and (10). It may well be possible to use the same Kalman filter due to the simple fact that the Kalman filter is robust to parameter variations. However it would be more appropriate to state that the applicability of the same Kalman filter depended upon $\delta\mathbf{A}$, $\delta\mathbf{B}$ and $\delta\mathbf{C}$, and their characteristics (i.e. probability distribution).

Suppose it is assumed that $\delta\mathbf{A}$, $\delta\mathbf{B}$ and $\delta\mathbf{C}$ are not known, however their distribution is known, i.e:

$$\delta\mathbf{A} = \overline{\mathbf{A}} + \delta\omega_A$$

$$\delta\mathbf{B} = \overline{\mathbf{B}} + \delta\omega_B$$

$$\delta\mathbf{C} = \overline{\mathbf{C}} + \delta\omega_C$$

where $\overline{\mathbf{A}}$ is a constant and known

$\delta\omega_A$, $\delta\omega_B$ and $\delta\omega_C$ are noise like terms with zero mean and known distribution.

Then substituting for $\delta\mathbf{A}$, $\delta\mathbf{B}$ and $\delta\mathbf{C}$ in Eqs. (9) and (10) yields:

$$\mathbf{x}(k+1) = (\mathbf{A}+\overline{\mathbf{A}}+\delta\omega_A)\mathbf{x}(k) + (\mathbf{B}+\overline{\mathbf{B}}+\delta\omega_B)\mathbf{u}(k) + \mathbf{v}(k) \tag{11}$$

$$\mathbf{y}(k) = (\mathbf{C}+\overline{\mathbf{C}}+\delta\omega_C)\mathbf{x}(k) + \mathbf{w}(k) \tag{12}$$

By considering the state and control noise as pure additive term Eqs.(10) and (11) can be written as

$$\mathbf{x}(k+1) = (\mathbf{A}+\overline{\mathbf{A}})\mathbf{x}(k) + (\mathbf{B}+\overline{\mathbf{B}})\mathbf{u}(k) + \mathbf{v}(k) + \delta\mathbf{v}_p \tag{13}$$

$$\mathbf{y}(k) = (\mathbf{C}+\overline{\mathbf{C}})\mathbf{x}(x) + \mathbf{w}(k) + \delta\mathbf{w}_p \tag{14}$$

where $\delta\mathbf{v}_p$ and $\delta\mathbf{w}_p$ account for additional state and control noise terms due to parametric variations.

Eqs (13) and (14) can now be applied directly to the Kalman filter. Not surprisingly mapping of parameter changes to inputs, increases the state and measurement noise covariance matrices, and reduces the quality of state estimation through the Kalman filter as measured through the error covariance matrix **P**. This is due to increased uncertainty in sensor parameters.

8.7 Discussion

Of the four multi-sensor architectures considered, the centralised architecture is the simplest to implement. The coordination of the sensors is simple as they are under the direct control of a single central control unit. This has a number of implications on the system. Since all the computation is carried out in the central unit and as the data acquisition and assimilation is governed by the speed of the central unit's processor, the architecture is susceptible to computational and communication bottle-necks. Furthermore any addition of sensors to the system is made at the expense of a loss of speed in operation. Another concern with this architecture is its dependence on the single central processing (control) unit. This makes the system vulnerable to faults in that unit, and for flight critical applications this is not acceptable.

The implementation of the hierarchically centralised architecture is similar to the centralised architecture, but it removes some of the computational chores from the central unit. Nevertheless the architecture is still prone to computational and communication bottle-necks since all the fusion occurs at the central unit. The architecture is also vulnerable to faults in the central unit and thus not suitable for flight critical applications.

The blackboard based architecture yields a modular system in that each node can operate autonomously i.e., independently of all other nodes. One of the major difficulties encountered in this architecture is that of designing efficient control laws for the blackboard. An essential requirement of the architecture is that only relevant data be temporarily posted on the blackboard at any time, which necessitates that old data be removed [Rao 1991]. This in general is said to be extremely difficult to accomplish. Although the control of the system is performed at each node, co-ordination is still essentially centralised. Therefore the architecture is vulnerable to faults in the blackboard. A failure of the blackboard will bring the entire system down and render it useless.

The decentralised multi-sensor architecture is a network architecture in which the sensor information is shared by all the sensors connected to it. The architecture has no central processing facility and no centralised communication medium, and thus has no potential source for single point failure. Each sensor node has its own processing and communication facilities minimising computational and communication bottle-necks. Each sensor node is capable of operating independently of the other nodes, and thus the system is capable of absorbing failures in one or more of its sensors, providing graceful degradation. The modularity of the system enables the addition of sensors with relative ease.

It is clear that the decentralised multi-sensor architecture provides an efficient approach for alleviating the shortcomings of the centralised architecture. Furthermore it yields a sensing system that is fast, robust, modular and fault tolerant, and thus is ideal for enhancing the performance and reliability of a flight critical system.

One of the desirable attributes of the decentralised architecture is that it lends itself to IMA system implementation. The IMA system consists of a number of cabinets (housing processing units) distributed throughout the aircraft interconnected by a high speed data bus. The concept also encourages the use of network compatible devices which interface directly to the data bus. This has a number of implications on the implementation of the decentralised multi-sensor architecture.

First, the decentralised architecture calls for 'smart' sensors, which usually implies the incorporation of microprocessors or intelligent electronics in or on the sensor. However when implemented via the IMA system, the sensors do not need to be 'smart', they only need to be network compatible since the computing resources for the individual sensors can be provided by any one of the processing modules in the cabinet. Furthermore, since IMA is a network system, the sensor data can be readily shared amongst the group of multi-sensors in order that a common estimate can be evaluated.

Second, it may not always be feasible to incorporate microprocessors/electronics in or on the sensor due to physical constraints: space and environment. In such situations the processor would have to be divorced from the sensor and ways need to be found to connect the sensors with their associated processors. By implementing the decentralised multi-sensor system using the IMA system approach, the above problem can be alleviated, since the sensors can obtain access to processing resources via the data bus. But at this point it must however be said that not all sensors are network compatible. This is accepted impractical in all cases due to the interface electronics being too expensive for the particular sensor or constraints by physical factors as mentioned earlier. In this case the sensors are connected to the data bus via a data concentrator.

Third, in a decentralised multi-sensor architecture each sensor has a dedicated processor which performs data estimation and fusion operations. In the event of the failure of one of the processors, the sensor associated with the failed processor is rendered useless. Since IMA system supports reconfiguration, in the event of processor failure spare resources can be reallocated to the sensor thereby keeping the sensor in operation thus enhancing the reliability of the multi-sensor system.

It is clear that IMA system provides the necessary facilities required to implement a decentralised multi-sensor architecture. Therefore multi-sensor integration can be used in conjunction with IMA to enhance the validity of sensor derived information.

8.8 Conclusion

In this section, the issues relating to the use of multi-sensor data integration to enhance the valifdity of sensor derived information was addressed. It is clear that the use of multiple sensors, together with the proper choice of integration algorithm can enhance the validity of sensor derived information.

With regard to the choice of multi-sensor architecture, the decentralised architecture provides an efficient configuration in that it yields a system that is fast, modular and fault tolerant.

Without a doubt, the processing involved in the multi-sensor system is computationally more intensive than in a single sensor system. Furthermore the concept is heavily dependant on software and therefore the reliability of software is of prime importance, especially in flight critical applications. But with the development of high throughput 32 bit processors and software fault tolerant techniques, such as N-version software and recovery blocks, it is envisaged that the necessary components to accommodate the increase in processing requirement, and the mechanism to tolerate software generic faults are in place.

The implications of using a decentralised multi-sensor system in conjunction with the integrated modular architecture was discussed and it is clear that the two concepts can be used in combination to enhance the performance and reliability of a flight critical system.

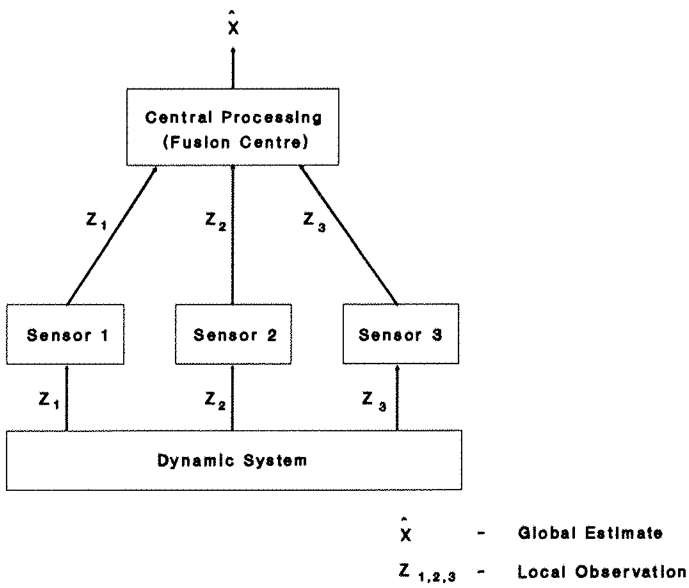


Figure 8.1 Centralised Multi-sensor Architecture

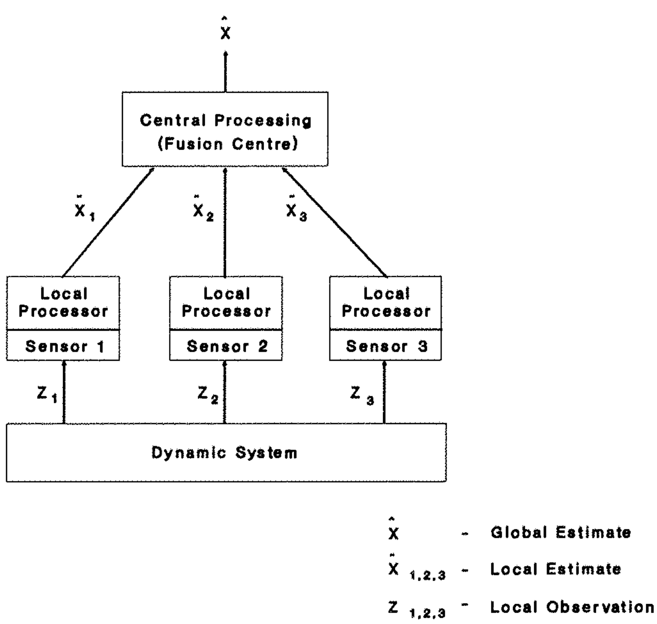
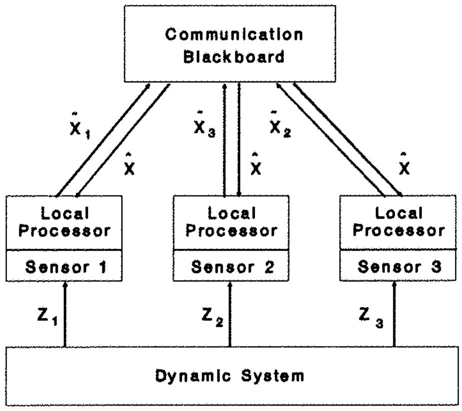


Figure 8.2 Hierarchically Centralised Multi-sensor Architecture

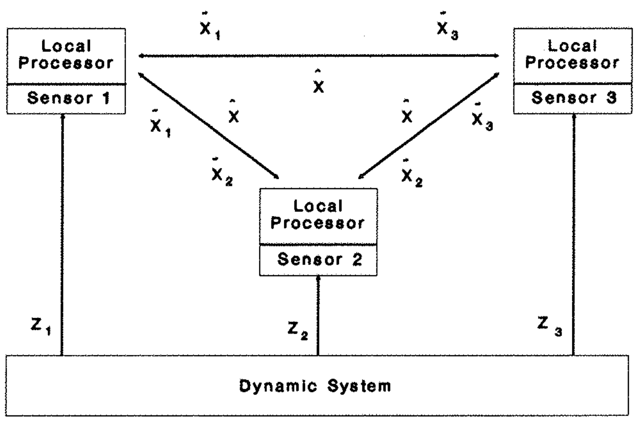


$\hat{x}$ - Global Estimate

$\tilde{x}_{1,2,3}$ - Local Estimate

$z_{1,2,3}$ - Local Observation

Figure 8.3 Blackboard Based Decentralised Multi-sensor Architecture



$\hat{x}$ - Global Estimate

$\tilde{x}_{1,2,3}$ - Local Estimate

$z_{1,2,3}$ - Local Observation

Figure 8.4 Fully Decentralised Multi-sensor Architecture

9 Electric Actuators

9.1 Introduction

The majority of aircraft in service today use hydraulically powered servo actuators for surface control. These actuators are controlled directly by the pilot through mechanical linkages, pulleys and cables. However, with recent advances in electronic and computer technology the aircraft industry has been slowly moving toward an increase use of Fly-By-Wire (FBW) control. These system consist of two parts: the fly by wire part which deal with the logical elements and the transmission of commands, and the actuation part which provide the muscle. As a result the system was dependent on two secondary power sources: electrical power for the operation of logic elements and hydraulic power for the actuator operation. Furthermore to satisfy flight safety requirements both the power sources were replicated.

As FBW evolved, however, concerns were expressed as to the systems dependence on two sources of secondary power and also the practicality of such an approach where it was necessary to develop two separate, replicated secondary power systems. In view of this the aircraft industry have been looking at the concept of electric actuation for surface actuation, the idea being that it would provide the potential for significant savings in weight and maintenance requirements - a single system to service instead of two. This chapter provides a review of remote powered actuators (RPA), i.e. electric actuators, and considers the direct application of the said concept to secondary and primary flying surfaces. The chapter also discusses the command and control aspects of the actuators.

9.2 Flight Control Actuation - An Overview

Early primary flight control systems were configured with the pilot providing all the power to move the control surfaces via mechanical linkages, pulleys and cables (Fig. 9.1). With the continued increases in aircraft speeds, aerodynamic surface loads became greater and power assistance in the form of hydraulics was required to reduce the demands on the pilot. Since loss of operation of flight critical actuators, even for short periods of time, will typically result in loss of the aircraft, the result is an airplane with two or three hydraulic systems, together with some form of mechanical back-up.

With the maturity of hydraulic power, military aircraft designers began to look at reduced static stability in the quest for even greater performance [Leonard 1983, Leonard 1985, Thompson 1985, Thomson 1983, Thomson 1985]. This led to the concepts of stability and command augmentation systems, which basically generate electrical control surface demands based on aircraft dynamics (rates and acceleration), and eventually the complete elimination of the mechanical portion of the flight control systm. This has become known as Fly-By-Wire (FBW) (Fig. 9.2).

In the development of the all-electric-aircraft concept, the next logical step in evolution appears to be the elimination of the vast network of hydraulic piping and the implementation of the electric actuator - Power-By-Wire (PBW) [Hair 1985, Helsley 1983, Ward 1983]. An exception to this rule is the landing gear; despite suggestions such as the use of local energy storage devices, the short-term, peak-power requirements of such an actuator are still best provided by traditional hydraulic techniques.

9.3. Electric Actuators

Electric actuators using high speed servo motors have been developed in two principal configurations using hydraulic fluid gearing in Electro-Hydraulic/Hydrostatic Actuators (EHA) or mechanical gearing for torque amplification in Electro-Mechanical Actuators (EMA) - the greater emphasis having been placed on the development of the latter. In both cases, power is distributed electrically to the actuator and the energy conversion is provided by a rotating electrical machine.

The EHA utilises an electric motor powered reversible pump to transfer fluid from one side of the actuator to the other in response to position commands while the EMA uses electric motor driving through a ball and/or screw to achieve the desired actuation.

These actuators have been around for many years in aircraft utilities and missile flight control systems but have suffered from being heavy, cumbersome and notoriously unreliable. However, the applicable technologies have advanced to a stage enabling EMAs and EHAs to be considered viable candidates for primary flight control activities. More directly, the use of electric actuators for aerospace applications has been made possible by the availability of high energy magnetic materials and the development of microprocessors and high power switching transistors.

Continuing with the concern for weight reduction, even with state-of-the-art technology, an EMA for primary flight control will always be heavier than an hydraulic actuator of the same output force and rate capability, due to the techniques used to by-pass motor failures (mechanical clutches and brakes); the weight savings are realised on a total system basis - the aircraft minus the significant weight of hydraulic generation and transmission systems [Leonard 1985, Leonard 1983]. EHAs, on the other hand, are considered to occupy approximately the same volume and weight as a conventional hydraulic actuator of the same output.

Although the two electric actuation concepts are viable candidates for surface control actuation, it appears that the airframe community favour the EMA system; the reason being that they provide the potential for safe flight after loss or deactivation of control surfaces.

There are two variations of the EMA: the velocity summed and the torque summed actuators - the reason for their existence is that compact electric motors are likely to require consolidation of their torque/velocity outputs to be compatible with flight control drive demands and in so doing dual load path redundancy is built-in [White 1987].

In the case of the torque summed actuator (utilising a gearbox), a clutch would decouple a jammed motor and allow the remaining channel to continue operating at half the original torque [Leonard 1985, Leonard 1983, Barnes 1983]. Conversely, a velocity summed actuator (utilising a differential) is not affected by a jam. Instead backdriving of the failed motor occurs and there is no actual output ('open-circuit'); the failed channel is actively restrained by a brake and the surviving channel continues operating at half original velocity. A possible application of this brake is to maintain a static surface position, in the presence of an opposing load, without burning out a stalled motor [Hair 1985].

Torque summing is less complex than velocity summation, due primarily to the fewer gears required, e.g. 33 versus 57, and to a less sophisticated servo loop requirement [Leornard 1985, Leonard 1983]. Due to the relatively large time period between failure and the active restraining of the surface, velocity summing cannot be applied to flutter prone/critical surfaces whereas torque summed actuators tend to lock at the point of failure; the AEA is therefore likely to be a mixture of velocity and torque summed actuators.

9.4 Reliability and Redundancy Issues

A number of questions have been raised regards the use of electric systems for primary flight control:

1) can EMA/EHA flight control actuators perform satisfactorily i.e. can the performance match that of hydraulic actuators;
2) can EMA/EHA systems be designed to equal the flight safety reliability of dual tandem hydraulic actuators;
3) can solutions be found to dissipate the heat generated by the actuators and power controllers.

The answer to question 1 is that studies show that EMAs can out-perform hydraulic actuators, especially under load [Leonard 1985]. The answer to question 2 is not all that clear since no indepth study has been carried out on this area. A major concern has been the probability of a jam or structural failure in the EMA gear box resulting in the probable loss of aircraft. The advocates of EMA argue that since Airframe Mounted Accessary Drive (AMAD), which is essentially a gear-box, is currently used on state-of-the-art aircraft to drive hydraulic pumps, safe flight is now already dependent upon gears [Leonard 1985]. Nevertheless it is a critical issue which needs to be addressed. The answer to question 3 is that electronics have been demonstrated to operate in the actuator environment, and the heat dissipated can be dealt with [Leonard 1985] - the unknown quantity is the thermal management of the actuators.

The heat dissipation issue assumes that there is a place to radiate or convey the heat generated by the actuators. Due to space constraints fins cannot be used to dissipate the heat, thus fuel appears to be the obvious coolant. But whether or not it has the capacity to absorb the amount of heat, especially when the fuel levels are low is questionable. One way to get around this problem would be to carry extra fuel aboard the aircraft for the sole purpose of cooling. But this approach may not be popular with the airline community due to the weight penalty and from the safety point of view; perhaps a more acceptable method is the transmission of heat to the non-composite structure of the aircraft - the large surface area of the wings has potential to be a significant heatsink. However, it would seem that heat dissipation is not so much of a problem with electrical actuators; following a test program involving a Sunstrand EMA fitted to the aileron of a USAF C-141, it was concluded that the actuator bay was in fact cooler with the EMA than the hydraulic actuator [Norton 1986, Norton 1987].

Redundancy can and should be applied to an actuation system. Some of the techniques available include 2 or more actuators per surface with isolation capability and split control surfaces - in-built load path redundancy is of course inherent to torque and velocity summed EMA actuators.

9.5 Flight Control Actuation and the Interface with Digital Commands

Communication between the Fly-By-Wire (FBW) flight control system and the control surface actuators is currently accomplished via dedicated data links. This is the approach adopted by Airbus for their A320 aircraft. In this FBW flight control system, the pilot's commands are sent in the form of electrical signals to a number of Flight Control Computers (FCCs). These computers also receive information from other aircraft subsystems such as the air data computer, inertial reference system etc., which are processed by the FCC using control laws to generate commands for the control surface actuators. These commands are sent to the actuators via dedicated links [Tucker 1988, Potocki de Montalk 1988].

One of the common complaints about the current actuator design has been the number of wires connections; on a typical FBW aircraft there can be as many as 60 to 70 wires per actuator [Eardly 1991, Leonard 1985]. These wires have to be routed from the FCC, located at the electrical and electronics bay, to the wings and the empennage; these very long wire runs makes the system vulnerable to Electro-Magnetic and Radio Frequency Interference (EMI/RFI).

By making the actuators 'smart[1]' the interface to the FCC can be simplified, requiring only serial data which reduces the wire count significantly (Fig. 9.3). The smart fly-by-wire flight control system is said to reduce the wire count by as much as 75% compared to the conventional fly-by-wire system [Eardly 1991].

In current FBW flight control systems, the actuator loops are closed by processors within the multi-processor FCC. However by combining FBW control with distributed processing, microprocessor based controllers (i.e local actuator controllers) can be implemented which can be divorced from the centralised FCC and located in the close proximity of the actuators. These controllers would effect FBW, communicating with the rest of the flight control system via the data bus. The advantage of this approach is that the interface to the FCC and the actuator is simplified, resulting in a lower wire count. The flight controls of the Boeing 777 is envisaged to employ a similar strategy. In this particular FBW flight control system, the pilot's commands together with the rudder pedal commands are sent to a actuator control unit known as the ACE (Actuator Control Electronics), where the signals are digitised and sent on to the Primary Flight Control Computer System (PFCS) via the aircraft data bus (ARINC 629); the PFCS also receives information from other aircraft subsystems. This information is then processed using control laws to generate the command to control the surface actuators, which is returned to the ACE via the data bus (Fig. 9.4). The ACE controls a number of primary and secondary control surfaces [Aerospace 1991, Hopkins 1991].

There are many advantages to be gained from using local actuator controllers. These include [Krogh 1983]:

1) Sophisticated Test and Monitoring Capability - By dedicating the computational power of the microprocessor to the control and management of an actuator, or group of actuators, it will be possible to implement pre-flight and maintenance tests which are more comprehensive than what would be practical with a conventional centralised system. It will also allow in-flight monitoring schemes to be implemented which provide rapid and accurate failure detection and isolation.
2) Tuneable Actuator Characteristics - The controllers will allow the actuator loop to be closed digitally. Since microprocessors are included in the control loop, the closed loop response

---

[1]Smart generally implies the incorporation of microprocessors/electronics in or on the actuator.

of the actuator can be actively varied. This will allow actuator characteristics to be optimised for different flight phases.

3) Performance and Degradation Detection/Comparison - The improved monitoring capability offered by the controller will not only provide failure detection but also allow sub-optimal actuator performance to be observed; such controllers can be preprogrammed to detect degradation in performance and compensate for them in flight.

4) System Architecture Flexibility - The controllers provide the designer with a number of possibilities in system architecture. The number of controllers for a given surface is selected according to the criticality of the control surface. Furthermore with the proper interface, the controllers allow any type of actuator, including the integrated actuator package to be controlled.

By applying reliable active control FBW to the aircraft pitch control, Relaxed Static Stability (RSS) may also be utilised in the aircraft design. This generally leads to a substantial reduction in both weight and drag, thereby enhancing manevurability and range. Although the concept may be more suited for a combat aircraft, studies have shown that large gains in fuel economy can be achieved for commercial type transport aircraft [Krogh 1983].

9.6 Discussion

In concern for the reliability of critical flight control actuators, the prospect of actuator jams has been cited as more likely in the EMA because of the planetary gear reduction and concerns for the problems of a broken tooth or a mechanical seizure that could stall a surface and any actuators connected to it; jamming of an EMA in the multiple actuation of a surface represents single point failure and the annulment of the concept of redundant actuation. The obvious solution is to de-clutch the failed gearbox but this effectively represents the introduction of a further 'weak link' to a flight critical airfoil. Alternatively, split surfaces could be used in which the remaining actuator could deploy the redundant surface in an asymmetric sense to the failed actuator/surface, thus allowing continued flight - as to whether a 'fail-safe' operating regime for essential surfaces is acceptable to the CAA/FAA is questionable.

With regard to smart actuators, although significant benefits can be achieved through the application of smart actuators, their use depends on a number of factors: the temperature of the operating environment, vibration and the location. Of the three, the most important factor is the environmental temperature. High temperature electronics for smart actuators not only have to survive in the harsh actuator environment, but they also need to operate reliably in this environment. This is the main reason as to why smart actuators currently do not exist. Although high levels of vibration exist in the actuator environment, it is considered not to be a significant problem [McLaughlin 1989].

However the location of the flight control actuators may cause a problem, in particular the aileron actuators. Since they are located at the extremities of the wing and in an unprotected environment, the electronics would face temperature extremes which may affect the reliability and thus the performance of the electronics.

Lucas Aerospace have carried out a trade study to evaluate various smart actuation architectures [Eardly 1991]. The study showed that significant reductions in weight and interference susceptibility can be achieved by adopting a smart Fly-By-Light (FBL) solution. However due to limitations in optical technology, i.e. connectors and encoders, many of the FBL solutions considered are currently not realisable. As a result Lucas have opted for a FBL solution which employs an optical harness with electrical terminations - the electro-optic conversion taking place within the connector using active components. The system is currently under evaluation.

An aircraft configured based on RSS is unstable without augmented control and as such its primary control systems need to be extremely reliable for adequate safety of flight. Although the concept provides the potential for significant gains in fuel economy, concerns over safety and reliability have hindered their application on commercial transport aircraft.

9.7 Conclusion

In this section the viability of using electric actuation for surface control has been presented. Current technology permits the use of electric actuation for flight control. Studies show that the electric actuation for flight control application will undoubtedly be heavier than their hydraulic counterparts, however on a total system basis weight savings can be expected. The weight reductions are mainly due to the elimination of hydraulic piping and fluid.

Although the two electric actuation concepts are viable for flight control applications, it appears that the airframe industry favour the EMA, or more precisely the torque summing variant of this system; the theory being that it provides the potential for safe flight after loss or deactivation of a control surface.

On the avionics front, smart actuators would appear to simplify the interface to the FCC resulting in a lower wire count in the harness whilst still providing a sophisticated test and monitoring capability. It has been demonstrated that electronics can be made to operate in the actuator environment but, again, their reliability in these conditions has yet to be proven.

An aircraft configured based on RSS provides the potential for significant gains in fuel economy, however, concerns over safety and reliability have hindered their application on commercial passenger aircraft.

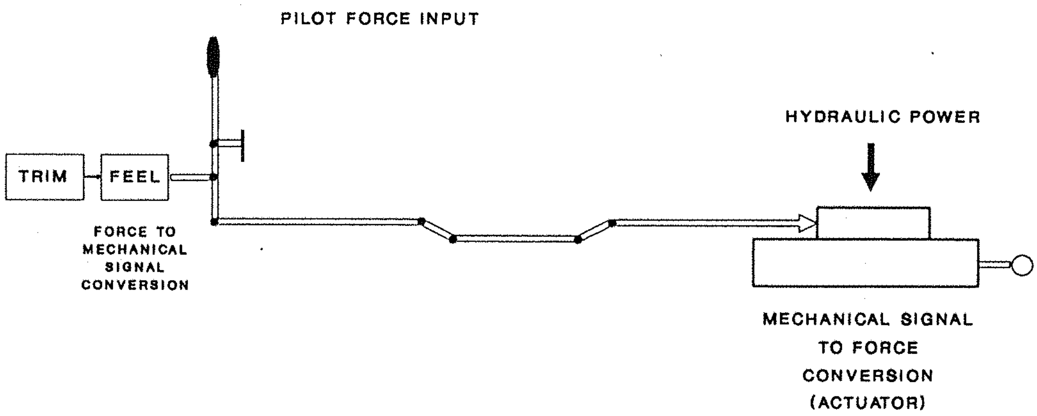


Figure 9.1 Schematic of Pilot Flight Control with Fully Powered Actuator

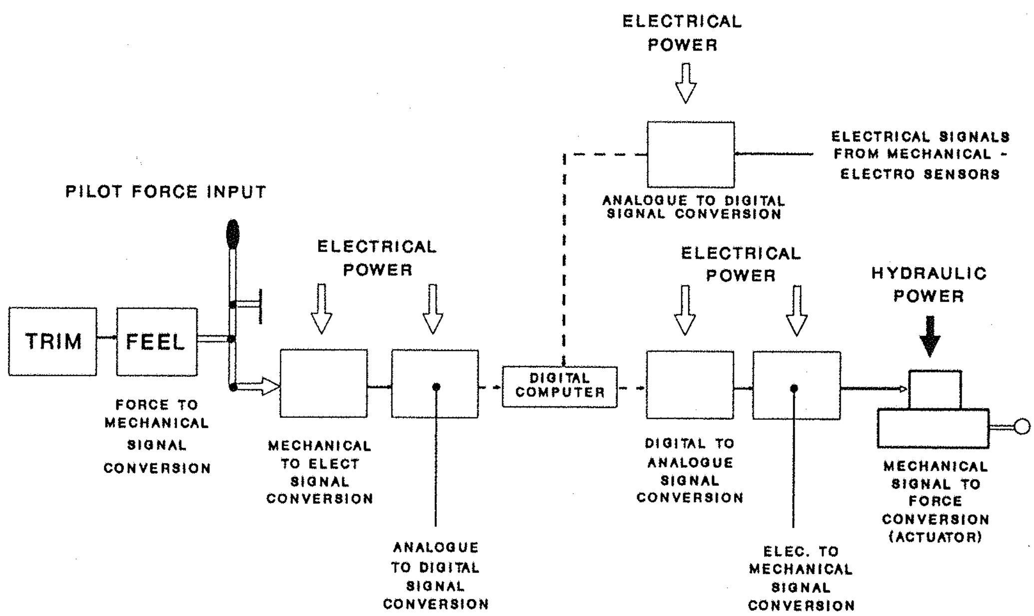


Figure 9.2 Schematic of Digital Fly-By-Wire Flight Control System

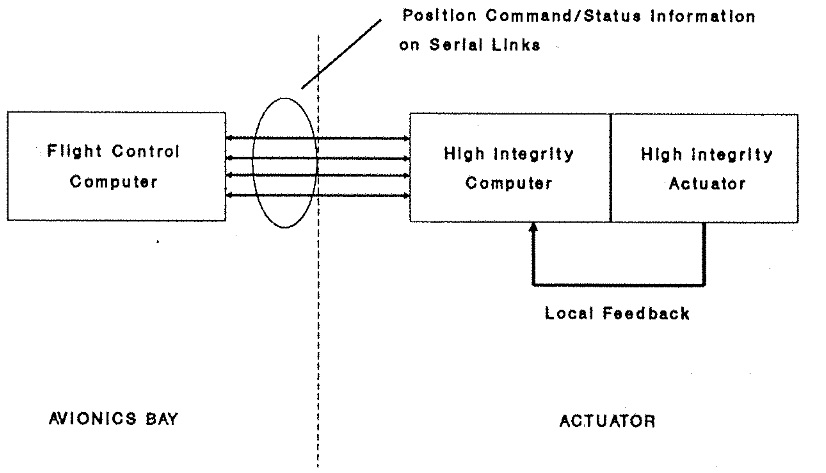


Figure 9.3 Smart Actuator

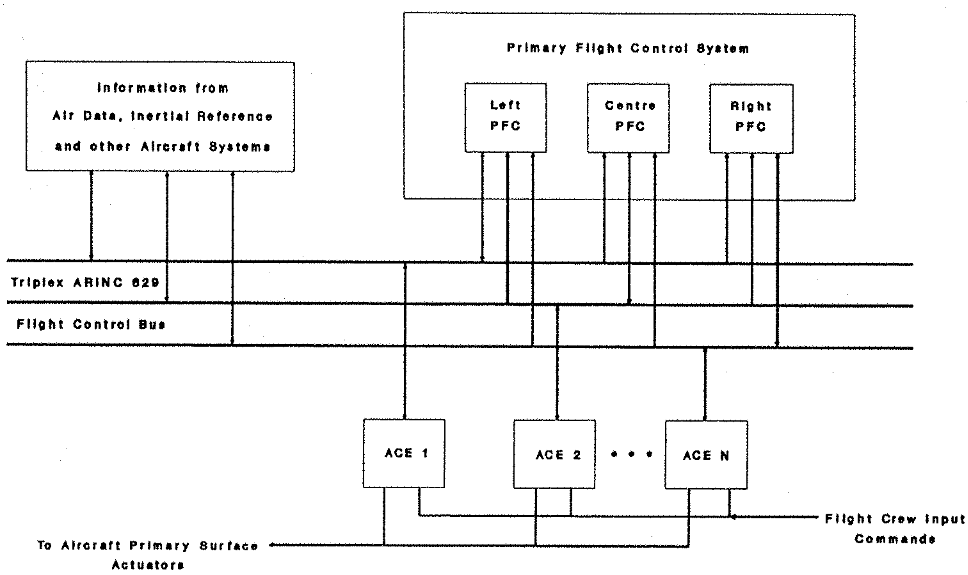


Figure 9.4 Fly-by-wire Flight Controls for the Boeing 777

10 Summary and Conclusion

The avionics systems in state-of-the-art commercial aircraft have become increasingly complex and sophisticated in order to meet ever increasing performance and reliability requirements. With the capability of the avionics technology improving by an order of magnitude every few years, it is envisaged that the current philosophy of one box-per-function would soon reach its limits as to what can be accommodated in terms of cost, functionality, reliability and certification.

In this thesis, the limitations of the current avionic system configuration has been addressed and the need for a new approach to avionics system integration presented. The solution that has been put forward is the integrated systems configuration. Integrated systems, with distributed processing where the resources are shared by many functions improve the reliability, availability, survivability and extensibility of a system. Furthermore they also provide the potential for reducing the acquisition, maintenance and operating costs of a system.

In recent years a number of architectures - from a fully centralised architecture to a distributed architecteur, have been proposed as viable solutions to the current avionics integration problem. In this thesis four potential candidate architectures have been evaluated to assess their capability in easing the shortcoming faced by the current avionics systems architecture. The findings indicate that the IMA architecture is the most suitable for commercial avionics application as it satisfied all the major requirements expected of an advanced integrated avionic architecture. Furthermore it was also the architecture that had the most potential to be certifiable for use on commercial aircraft, as it did not rely on a central bus controller.

The IMA systems architecture is a distributed system, which provides extreme flexibility to accommodate evolutionary enhancement of functions and airline-specific avionic installation. By utilising a common chassis design, common fault tolerant processor and redundant power supplies, the concept is able to provide significant cost savings that are not realisable with current avionic systems architecture. However to quantify these benefits would be extremely difficult, or even impractical since they depend on numerous factors including the manufacturer's implementation.

The IMA architecture is envisaged to provide benefits to the airframe manufacture, the airline operator and the equipment manufacture. The benefits to the airframe manufacture include: 1) reduced weight and volume; 2) reduced manufacturing and development cost; and 3) more flexible architecture to accommodate technological upgrades and airline-specific avionic installation. For the operators, a significant reduction in operating cost is envisaged through better maintenance. Other benefits to the operator include: 1) the ability to defer maintenance; 2) no maintenance required during turnrounds; 3) improved fault isolation; 4) reduction in number of no fault found removals; and 5) reduced spares requirement. For the equipment manufacture, the architecture allows: 1) increased marketing opportunity; 2) easier to offer system upgrades; and 3) easier to offer additional systems

IMA architecture however has its problems. A major concern is certification. Since functions with different criticality are processed in a single cabinet there is always the potential for undesirable interaction between different avionic functions. As a result questions have been raised regarding the certification of the IMA concept, particularly with respect to common mode failures, partitioning integrity, and on board software loading. Boeing however have had preliminary discussions with the FAA on the aforementioned problems, and envisage no insurmountable problems provided that strict fault tolerance and robust hardware and software partitions are adhered to.

The advantages of an integrated system, like the IMA, can only be realised if efficient, safe and practical methods of subsystem intercommunication can be achieved. The ARINC 429 data bus, although suitable for use in simple aircraft is considered not efficient for use in today's complex avionic systems. Comparison of data buses in commercial avionics use today would indicate that in order for a data bus to serve a modern commercial aircraft it must be able to meet the following requirements: 1) should not rely on a bus controller; 2) should be flexible to accommodate future capability upgrades and airline-specific avionic installation; 3) bi-directional; 4) the bus should have substantial room for label expansion to accommodate new avionic systems that are expected in the next 15-20 years due to advances in technology; 5) allows the use of reasonably priced commercial hardware for input and output interface; and 6) able to handle both periodic and aperiodic data transfer on the same bus.

A review of the ARINC 629 data bus would indicate that it appears to satisfy all the major requirements expected of an advanced data bus. It also yields a data distribution system that is extremely flexible for capability upgrades. In view of the fact that it is able to reduce cabling, size and weight of interface hardware and improve reliability, this makes it a very desirable data bus for commercial avionic application.

One of the concerns of using a data bus to effect communication is its susceptibility to high intensity electromagnetic interference. With the application of VHIC/VLSI technology is aircraft and the growing trend towards the increase use of composite material for both aircraft skin and primary structures, the threat is envisaged to become even more sevre. To contain this problem the application of fibre optic technology and system level protection schemes have been considered in this thesis.

Fibre optics provides immunity from EMI/RFI effects, sparks or fire hazards, short circuit, cross talk between cables and lightning surge currents. Furthermore it also provides increased channel capacity. Although fibre optics is an attractive alternative offering several benefits over metallic cables, there are several factors which limit the application of fibre optics on aircraft. At present the main factors hindering the near time acceptance of fibre optic systems on aircraft appear to be the cost effectiveness of the overall system and the instability, maintainability and in-service reliability of the connectors. This does not mean, however, that fibre optics has not been used on commercial aircraft. Early examples of fibre optics on commercial aircraft for non-flight critical applications include the Boeing 747 in-flight entertainment (IFE) system; the reliability of these systems was poor, though efforts are being made to improve on it.

Although some fibre optic component technology are mature, such as optical fibres; others like connectors and couplers still need further development before the full performance potential of the system can be realised. In order for fibre optics to become truly competitive with metallic cables and be readily accepted for use on aircraft: 1) better connectors are needed; 2) ways need to be found to reduce the back reflection of the transceiver connector; 3) ways need to be found to locate fault more easily; and 4) simple, easy to use, test equipment and procedures are necessary.

To an extent protection from high intensity electromagnetic interference can also be achieved through system level methodologies. The goal of system level concepts is to provide a mechanism which enables the system to tolerate disruptions of either input/output or internal computation. This is achieved through detection of processing irregularities, retention of state variables in 'hardened' storage and rapid recovery of processing activity. The concept provides an effective way of protecting sensitive avionics from the effects of EMI/RFI, and can be used in conjunction with traditional protection schemes to provide the margin of safety required for flight critical functions. A digital system design based on such a concept is able to be fully operational even in the presence of a temporary disruption of the processor.

One of the key components in achieving high level of integration is fault tolerance. Fault tolerance in aircraft is currently achieved through the application of redundancy which incur weight, volume and cost penalty. Analytical redundancy and knowledge based concepts provide the potential for eliminating hardware redundancy, while still maintaining the high level of reliability and availability. These concepts make use of efficient fault detection, identification and accommodation (FDIA) strategies. On aircraft, the concepts can be implemented on the on-board computers providing savings in weight, volume and maintenance. Hardware redundancy however, cannot be completely eliminated because it is essential for reconfiguration. Therefore the application of the said concepts on aircraft is seen as reducing the level of hardware redundancy (e.g from a triplex system to a duplex system).

Analytical redundancy makes use of mathematical models, as a result the achieved quality of the fault detection and identification system depends on the quality of the models. Knowledge based concepts make use of heuristic reasoning to perform fault detection and identification. The advantage of this approach is that it is able to find a solution to the problem even if the system being monitored is too complex for mathematical modelling. Both the concepts have their limitations however. But the limitations can be overcome by combining the two schemes together to yield a flexible and efficient diagnosis system -the Intelligent Quantitative and Qualitative (IQ^2) system.

One of the concerns of using analytical redundancy and knowledge based concepts is the large increase in computational expenditure. Although large increase in processing speed are becoming available with the development of 32 bit processors, it is envisaged that the computational requirement needed to implement a knowledge based system for all of the aircraft subsystems are beyond current avionic computational capability. A solution to this problem would be to use parallel processing. However the application of parallel processors (transputers) on civil aircraft raises certification issues which need to be addressed.

A major area of complaint of many aircraft operators is the reliability of present sensors. Often an indication of a failure are due to failure of a sensor rather than the actual system being monitored. Accurate and reliable sensors are essential to the operation of all aircraft systems, but in recent years they have also become increasingly important as sources of information for computerised control, e.g. active control of aircraft. Therefore the application of multi-sensor data integration to enhance the validity of sensor derived information was considered in this thesis.

The key to developing an efficient multi-sensor system lies in the organisation of the system, which depends on the architecture and on the co-ordination and control of the system. To this end, various architecture were considered. Of the architectures considered, the decentralised architecture was found to be the most efficient configuration in that it yielded a system that was fast, modular and fault tolerant. The implication of using a decentralised multi-sensor system in conjunction with the IMA architecture was also considered, and it appears that the two concepts can be used in combination to enhance the performance and the reliability of a flight critical system.

Without a doubt, the processing involved in the multi-sensor system is computationally more intensive than in a single sensor system. Furthermore the concept is heavily dependant on software, and therefore the reliability of software is of prime importance, especially in flight critical applications. However with the development of high throughput 32 bit processors and software fault tolerant techniques, such as N-version software and recovery blocks, it is envisaged that the necessary components to accommodate the increase in processing requirement, and the mechanism to tolerate software generic faults are in place.

With recent advances in electronic and computer technology the aircraft industry has been slowly moving toward an increase use of FBW control. The current FBW control rely on two secondary power sources: electrical power for the operation of logic elements and hydraulic power to provide the muscle for moving the control surfaces. If however electric actuation were used instead, then only one (replicated), secondary power source would be required. Current technology permits the use of electric actuation for flight control. Studies show that electric actuation for flight control application will undoubtedly be heavier than their hydraulic counterparts, however on a total system basis weight savings can be expected. The weight reductions are mainly due to the elimination of hydraulic piping and fluid. Although the Electro-Mechanical Actuator (EMA) and the Electro-Hydraulic/Hydrostatic Actuator (EHA) are viable candidates for surface control actuation, the airframe community favour the EMA system; the reason being that they provide the potential for safe flight after loss or deactivation of a control surface.

In current FBW flight control systems, the actuator loops are closed by the processor within the multiprocessor FCC. However by combining the FBW control with distributed processing local actuator controllers may be implemented. These would allow: 1) sophisticated test and monitoring; 2) tunable actuator characteristics; 3) performance and degradation detection/comparison; and 4) system architecture flexibility.

In general an architecture that integrates several functions each having access to a set of shared common components provides the potential for significant reduction in cost, weight, volume and parts count. With this in mind, it is envisaged that the next generation of civil avionics architectures in all probability will be based on an integrated system centred on common modules (e.g. arithmetic computational modules, data and signal processing modules, power supply modules and I/O modules). These modules would be plugged into a cabinet, and a number of these cabinets would be distributed throughout the aircraft, interconnected by a high speed data bus. Furthermore the cabinets will be capable of supporting a number of avionic functions. In essence the whole system will be seperated into pieces such that each piece can be designed, bought and integrated. An avionics architecture designed based on this concept is envisaged to provide increased supportability, reduced life cycle cost and increased upgradability. However in order to reap these benefits, the modules described above need to be interchangeable and plug-in compatible, and as such require some form of standardization.

The concept of standardization, however, has had mixed views from the avionics community at large. On the one hand, standards make the task of designing and integrating systems easier, by creating logical building blocks and defining standards. On the other hand, the designers have complained that standards do indeed restrict the freedom of design, thus preventing technological progress. Nevertheless it is widely believed that standardization independent of internal hardware implementation, i.e., standardization for technology transparent implementation is a good thing. This method of standardizing hardware involves defining the functional, interface and physical features of a module, but not the internal hardware architecture or component technology.

One of the most common standardization concept is the Form, Fit and Function (F^3) standardization. In this concept the requirements for the electrical interfaces between the avionics, the aircraft and other aircraft subsystems are established by the equipment acquiring organisation. Furthermore provisions for cooling, connector, pin assignment, automatic test equipment and other parameters have also been defined in enough detail, that it allows a manufacture to produce an equipment that is interchangeable with another vendor's equipment, despite probable differences in the internal design of the equipment.

Standardization is not a practice uncommon to the avionics industry. F^3 standards for avionics system have been widely used by the airlines for decades. Commercial airlines, through the AEEC and ARINC have also been successful in defining standard ARINC characteristics at the LRU level. However to realise the benefits of the IMA approach, standardization is required at the module level, and this needs to be arrived at through industry consensus. This will have benefits reaching all aspects of system design and support, including interoperability and interchangeability. The degree of compatibility will depend on the support of the industry participants. Standardization at the module level, (i.e., definition of functional modules and their associated interfaces) is, however, not sufficient alone to yield the benefits of the IMA concept. An overall software architecture is also required into which the building blocks of avionic applications can be integrated.

In the IMA approach, the functionality of the avionics system is provided by the applications software which is distributed among the network of processing centres - this requires the integration of many software applications. In order to allow the integration of theses applications with each other as well as the particular hardware, clear specification are also required. These specifications must define the interfaces between each application and the hardware resources, and define how hardware resources are made available.

The IMA architecture will be dominated by software based systems. Therefore the reliability of software is paramount. From experience with a wide range of systems and equipment which are dependent upon software, it is clear that software error is one of the major causes of system unreliability. No matter how carefully the software is designed it is extremely difficult to guarantee that the software is error free. A number of surveys into the source of errors in software indicate that the majority of errors are due to deficiencies in software specification (i.e design related) rather than coding mistakes. To reduce the incidence of design related errors, structured software design methodologies (i.e. structured programming) and comprehensive verification and validation tests have been suggested. Therefore it is necessary to make use of these modern methods of software design, implementation and test to eliminate design related software errors.

To adopt the hundreds of aircraft currently flying today to the IMA approach will be an extremely costly operation. Furthermore it will be a long time before there is sufficient module level definition which will allow the designing of an entire avionic suite. In view of this and the fact that as with all new systems, problems will be encountered with the IMA concept, (both in the development stage and after introduction), the early IMA systems are likely to involve only a limited number of systems.

10.1 Recommendations for Further Work

Fault tolerance has always been an essential requirement of aircraft design. In aircraft, fault tolerance is currently achieved through the use of hardware redundancy accompanied by a voting system. However, fault tolerance can also be achieved through the application of an efficient Fault Detection, Isolation and Accommodation (FDIA) strategy as mentioned in Section 7. One of the important issues associated with the FDIA strategy is fault diagnosis. The anticipation or early detection of potential faults can help in avoiding catastrophic failures. For example, the characteristic of a gearbox rotor on a helicopter can be monitored using signal processing techniques and based on the results, predictions can be made on any potential problems.

Over the years considerable effort has been made in developing a fault diagnosis system based on analytical (quantitative) methods and knowledge based (qualitative) methods. However, to-date, only a few researchers have considered the possibility of combining the two concepts together into one diagnostic system; the idea being that the two schemes can complement each other. A potential application for such a sophisticated system on an aircraft would be as an onboard monitoring system. The system could monitor the health of various subsystems and diagnose potential faults and report back to the ground crew (via the Aeronautical Telecommunication Network) the status of the fault with a solution; the ground crew can then be ready with the required spares and necessary equipment when the aircraft lands - this would significantly reduce the aircraft turnaround times. Furthermore the information on the health of the various subsystems can be used to schedule maintenance at a predetermined time convenient to the operator. In view of these benefits the concept merits further investigation. Consideration must be given to the theoretical development of the fault fusion process (i.e. the combination of quantitative knowledge and qualitative knowledge) algorithm. For instance, if one technique indicates a fault and the other does not, then a strategy needs to be formulated to come to some form of a consensus. This calls for some form of decision making, and as such research into AI is recommended as it provides the reasoning required for the decision making involved.

In this study the application of multi-sensor data integration to enhance the validity of sensor derived information was considered. However, the study only focused on one class of problem, i.e., how to integrate data from a number of sensors of the same modality measuring the same entity. The scheme can equally well be applied to dissimilar sensors measuring different entities. This information can be combined, using appropriate algorithms, to provide a better view of the overall system. For example, information from the radio altimeter, terrain data and global positioning system can be used to build a three dimensional picture of the outside world. Although the concept is particularly attractive for the military, it also has a useful application on civil aircraft, for example, the resulting information may have been useful in preventing the recent spate of air crashes in mountainous areas. Another example of possible application is in air traffic control. In view of this, it is recommended that the fusion of data from dissimilar sensors measuring different entities be considered for civil aircraft applications.

Novel concepts such as multi-sensor data integration, analytical redundancy and knowledge based concepts all require a significant increase in computational expenditure i.e. increase in processing power and data throughput. Furthermore the adoption of full authority digital engine control will also require an increase in processing power. The solution to these problems is the application of parallel processing. However the application of parallel processors on commercial aircraft raises certification issues and therefore merits further investigation.

Another area for consideration is the integration of Air Traffic Control (ATC) with the aircraft flight control computers to ease Air Traffic Management problems. It is anticipated that in the future, as airways get congested and airports reach their take-off and landing capacity, new procedures would be required to handle the increase in the additional traffic. In fact the problem already exists in some of the major international airports where it is not uncommon for an aircraft to be put on a holding pattern, from anywhere between 5 minutes to half an hour or more, until a suitable landing slot becomes available. If measures are not take to combat this problem, the savings in direct operating cost that can be achieved as a direct result of the application of 'new' technologies will be erroded.

The IMA architecture allows communication between ground and air-based communication system that are part of a global network. The advantage of this approach is that it allows air traffic management function to be integrated with the aircraft avionics providing reduction in ATC workload with less use of voice communication, and higher traffic density (i.e increase in landing capacity). A further benefit is that it provides the capability to implement a global 'on route' air traffic management system to aid with the scheduling of landing slots. The global network also provides the operator with a better overall picture of the movements of their fleet. Therefore it is recommended that consideration be given to the implementation issues of these concepts.

With the application of VHSIC/VLSI technology in aircraft electronics and the growing trend toward the use of composite material for aircraft skin and primary structures, the aircraft electronic systems have become increasingly vulnerable to external EMI/RFI effects. Therefore it is recommended that the extensive use of fibre optics should be considered in the design of new aircraft instrument and control systems. In order for the application of fibre optics on aircraft to become a reality efforts must be made to develop: 1) fibre optic cables with optical characteristics that remain constant over the entire aircraft temperature range; 2) fibre optic connectors with low insertion losses that remains constant even after repeated disconnects and reconnects; and 3) optical sources which can launch high power into optical fibres and can operate reliably over the entire avionic temperature range.

References

1. [AEEC 1991]
AEEC 1991, 'Design Guidance for Integrated Modular Avionics', ARINC Report 651, Published November, 1991.

2. [Aerospace 1991]
Aerospace. 'Europe on board the Triple 7 and the F-22, Aerospace, June 1991.

3. [Archer 1985]
Archer H. S. 'Digital Avionics for Modern Aircraft - A Case Study into the Problems and Promise of Aircraft Electronics', Proc. of IEEE, 1985.

4. [ARINC 429 Spec. 1990]
ARINC 429 Spec. 'ARINC Specification 429-12', Mark 33 Digital Information Transfer System (DITS), Published 1 July, 1990.

5. [ARINC 629 Spec. 1990]
ARINC 629 Spec. 'ARINC Specification 629', Multi-Transmitter Data Bus, Part 1, Technical Description, Published 7 March, 1990.

6. [Bailey 1990]
Bailey J. 'Honeywell Joins 777', Flight International, 12-18 December, 1990.

7 [Barnes 1983]
Barnes N. 'Study of Flight Control Actuators for Future Combat Aircraft', BAe Final Report, October 1983

8. [Bermingham et. al. 1987]
Bermingham W. J, Alfonsi E. A and Rosen W. A. 'A High Speed Fibre Optic Data Bus for Avionic Applications', NAECON '87, 18-22 May, 1987.

9. [Beton et. al. 1991]
Beton R, Kingdon J and Upstill C. 'High Availability Transputing Systems', Proc. of Transputing '91.

10. [Blackman 1990]
Blackman S. S and Broida T. J. 'Multiple Sensor Data Association and Fusion in Aerospace Applications', Journal of Robotic Systems, John Wiley and Sons Inc., 1990.

11. [Blackman 1988]
[Blackman S. S. 'Theoretical Approaches to Data Association and Fusion', SPIE, Vol. 931, Sensor Fusion, 1988.

12. [Botha 1981]
Botha D. G. 'Integrated Avionics: Concepts and Concerns', AIAA Paper No. 81-2211.

13. [Bracknell 1988]
Bracknell D. R. 'Introduction to the MIL-STD-1553B Serial Multiplex Data Bus', Royal Aerospace Establishment, Farnborough, Hants., U.K., September 1988.

14. [Brock 1986]
Brock L. D and Lala J. 'Advanced Information Processing System: Status Report', IEEE Conference Paper, NAECON '86.

15. [Bullock et. al. 1988]
Bullock T. E, Sangsuk-iam S, Pietsch R and Boudreau E. J. 'Sensor Fusion applied to System Performance under Sensor Failures', SPIE Vol. 931, 1988.

16. [Capon 1990]
Capon J. T. M. B. 'Knowledge Representation: An Approach to Artificial Intelligence', Academic Press Ltd., 1990.

17. [Card 1983]
Card M. A. 'Evolution of the Digital Avionics Bus', Proc. of IEEE, 1983.

18. [Chown 1986]
Chown D. P. M and Farrington J. G. 'A High Performance Fibre Optic MIL-STD-1553B Data Bus', SAE Paper 860841.

19. [Chun 1981]

Chun R. K. 'ARINC 429 Digital Data Communication of the Boeing 757 and 767 Commercial Airliners', Digital Avionics Systems Conference, 4th St. Louis, MO, 17-19 November, 1981.

20. [Cohn 1988]

Cohn M. 'Approaches to Maintaining a Fault Tolerant Ring in a Tactical Military System', MFOC '88, the Second International Military and Government Exposition, Arlington, V.A., 7-10 March, 1988.

21. [Couch 1985]

Couch P. R and Ferris K. D. 'High Speed Fibre Optic Data Bus for Avionics Integration', NAECON '85.

22. [Cronin et. al. 1985]

Cronin M. J, Hays A. P, Green F. B, Radovcich N. A, Helsley C. W and Rutchik W. L. 'Integrated Digital/Electric Aircraft Concept Study', NASA Contractor Report 3841, 1985.

23. [Crossgrove 1980]

Crossgrove A. 'Development and Application of MIL-STD-1553', Aerospace Congress and Exposition, Los Angeles Convention Centre, 13-16 October, 1980.

24. [Cruickshank 1989]

Cruickshank A and Kennett M. J. 'The Development and Flight Demonstration of an Optical MIL-STD-1553B Data Bus', Conference and Exhibition, MIL-STD-1553B and The Next Generation, Heathrow Penta Hotel, U.K., 29-30 November 1989.

25. [Dolezal 1988]

Dolezal R. M and Valentino G. J. 'Resource Sharing in Complex Fault-Tolerant System', Vol 1, NAECON '88.

26. [Durrant-Whyte 1988]

Durrant-Whyte H. F. 'Sensor Models and Multisensor Integration', The Internal Journal of Robotics Research, Vol. 7, No. 6, December 1988.

27. [Eardly 1991]

Eardly J. 'The Development of Smart Actuation Systems', IEE Professional Group C8 (Control and System Theory) Colloquium on Integrating Issues on Aerospace Control, Savoy Place, London, U.K, 4 April, 1991.

28. [Eldredge 1987]

Eldredge D and Hitt E. F. 'Digital System Bus Integrity', NASA Contractor Report DOT/FAA/CT-86-44, March 1987.

29. [Emami-Naeini et. al. 1986]

Emami-Naeini A, Akhter M. M and Rock S. M. 'Robust Detection, Isolation and Accommodation for Sensor Failures', NASA Contract Report 174825, July 1986.

30. [Emami-Naeini et. al. 1988]

Emami-Naeini A, Akhter M. M and Rock S. M. 'Effect of Model Uncertainty of Failure Detection: The Threshold Selector', IEEE Trans. on Automatic Control, Vol. 33, No. 12, Dec. 1988.

31. [Figueroa et. al. 1991]

Figueroa L, Hong C. S, Huggins R. W, Miller G. M, Popoff A. A, Porter C. R, Smith D. K and Van Deventer B. 'Fibre Optics for Military Aircraft Flight Systems', IEEE LCS, February 1991.

32. [Frank 1989]

Frank P. M and Wunnenbey J. 'Robust Fault Diagnosis using Unknown Input Observer Schemes', Fault Diagnosis in Dynamic Systems: Theory and Application, Edited by Patton R, Frank P and Clark R, Prentice Hall, 1989.

33. [Frank 1990]

Frank P. M. 'Fault Diagnosis in Dynamic Systems Using Analytical Redundancy and Knowledge-based Redundancy - A Survey and Some New Results', Automatica, Vol. 26, No. 3, 1990.

34. [Friend et. al. 1984]

Friend G. E, Fike J. L, Baker H. C and Bellamy J. C. 'Understanding Data Communication', Howard W. Sams and Co., 1984.

35. [Gardiner 1987]

Gardiner P. and Edwards R. 'Fibre Optic Sensors for Aircraft Flight Control', RAeS, 22 January 1987.

36. [Gayer 1986]

Gayer A. 'MIL-STD-1553 and Avionic System Design', Aerospace Design and Components, July 1986.

37. [Gilmore 1987]

Gilmore J. F. and Gingher K. 'A Survey of Diagnostic Expert Systems', SPIE, Vol. 786, Application of Artificial Intelligence V, 1987.

38. [Glista 1991]

Glista A. S. 'Fault Tolerant Topologies for Fibre Optic Networks and Computer Interconnects Operating in Severe Avionics Environment', IEEE LCS, February 1991.

39. [Hair 1985]

Hair K. 'Electromechanical Actuation Reliability and Survivability', NAECON 1985.

40. [Hall 1986]

Hall W. H. 'Communication Techniques for MIL-STD-1553 Data Transfers', MIL-STD-1553B: Applications, Developments and Components, Seminar Proceedings, 25-16 November 1986, London.

41. [Handleman 1985]

Handleman D. A and Stengel R. F. 'Combining Quantitative and Qualitative Reasoning in Aircraft Failure Diagnosis', AIAA Paper No. 85-45917.

42. [Handleman 1986]

Handleman D. A and Stengel R. F. 'A Theory for Fault-Tolerant Flight Control Combining Expert System and Analytical Redundancy Concepts', AIAA paper No. 86-2092.

43. [Harris 1988]

Harris C. J. 'Reducing Uncertainty and Ignorance in C^2 Systems, Application Of Artificial Intelligence to Command and Control Systems, Peter Peregrinus Ltd. 1988.

44. [Harris 1989]

Harris C. J and Jackson J. V. 'Techniques for Intelligent Multi-Sensor Data Fusion for Application to Autonomously Guided Vehicles', IMA Conference on Robotics: Loughborough 12-14 July 1989.

45. [Harris 1989]

Harris C. J. 'Distributed Estimation, Inferencing and Multi-Sensor Data Fusion for Real Time Supervisory Control, Department of Aeronautics and Astronautics, University of Southampton,1989.

46. [Hashemipour et. al. 1988]

Hashemipour H. R, Roy S and Laub A J. 'Decentralized Structures for Parallel Kalman Filtering, IEEE Trans. on Automatic Control, Vol. 33, No. 1, January 1988.

47. [Held 1989]

Held G. 'Data Communications and Networking Devices', John Wiley and Sons, 1989.

48. [Helsley 1983]

Helsley W. 'Airplane Actuation Trade Study', NAECON 1983.

49. [Herzog 1991]

Herzog H. K. 'The Requirements - From DATAC to ARINC 629', ARINC 629 Databus, One Day Conference, RAeS, London, U.K., 24 September, 1991.

50. [Hess 1985]

Hess R. F. 'Implications Associated with the Operation of Digital Data Processing in the Presence of the Relatively Harsh EMP Environment Produced by Lightning', 10th International Aerospace and Ground Conference on Lightning and Static Electricity, Paris, France, June 1985.

51. [Hess et. al. 1988]

Hess R. F, Yount L. J, Knoller H, Masson G. M and Larsen W. E. 'Sharing the Protection of Aircraft Electronic Systems Against the Effects of High-Level Electromagnetic Environments between Traditional Protection and System Architecture', AIAA Paper 88-3917-CP.

52. [Hicks 1988]

Hicks P. 'ARINC 429 An Introduction', Technitron Systems Ltd, Technitron House, Redfield Park, Church Crookham, Aldershot, Hampshire, England, May 1988.

53. [Hightech Components[a]]

Hightech Components[a]. Sales literature: Review of Currently available Rotary Velocity and Position Transducers - Hightech Components Ltd, U.K.

54. [Hightech Components[b]]

Hightech Components[b]. Sales literature: Feedback transducers for High Technology Closed Loop Systems - Hightech Components Ltd, U.K.

55. [Hopkins 1991]

Hopkins H. 'Delivered with Feeling', Flight International, 20-26 November, 1991.

56. [Husbands 1983]

Husbands C and Katz R. 'The Application of Multiple Speed Data Bus Structure', 5th Digital Avionics System Conference, Seattle, W.A., 31 October - 3 November, 1983.

57. [Iserman 1984]

Iserman R. 'Process Fault Detection Based on Modelling and Estimation Methods - A Survey', Automatica, Vol. 20, No. 4, 1984.

58. [Iserman 1981]

Isermann R. Digital Control Systems, Springer-Verlag, Berlin 1981.

59. [Iverson]

Iverson D. L and Patterson-Hine F. A. 'A Diagnosis System using Objected Oriented Fault Tree Models', NASA Conference Publication 3073.

60. [Jackson 1990]

Jackson P. 'Introduction to Expert System', Addison Wesley Publishing Company, 1990.

61. [Janex 1988]

Janex A. 'Reflections on Avionic Integration', AIAA Paper No. 88-3950-CP.

62. [Kennett 1986]

Kennett M. J. 'The Development of a Fibre Optic Data Bus for Helicopters', ICAs '86.

63. [Klien 1975]

Klien A. 'Refinements in Position Measuring Methods Provide Higher Accuracies, Resolutions', Control Engineering September, 1975.

64. [Klos 1984]

Klos L. 'Pave Pillar Avionics - An Architecture for the Future', NAECON '84.

65. [Krogh 1983]

Krogh S. B and Chenoweth C. C. 'Application of Microprocessors to Aircraft Fly-by-Wire Hydraulic Actuation Systems', SAE Committee A-6, SAE paper No. 831489.

66. [Labarrere]

Labarrere M. 'Aircraft Sensor Failure: Detection by Analytical Redundancy', System and Control Encyclopedia, Vol. 1, Pergamon Press, Oxford.

67. [Lala 1984]

Lala J. H. 'Advanced Information Processing System', AIAA Paper No. 84-2644.

68. [Large BAe]

Large A. J. 'Avionics Architecture for Application to Future Civil Aircraft', British Aerospace.

69. [Larsen 1988]

Larsen W. E. 'Digital Avionics Susceptibility to High Energy Radio Frequency Fields, National Aerospace and Electronic Conference Dayton, Ohio, 1988.

70. [Leonard 1983]

Leonard J. 'A System look at Electromechanical Actuation for Primary Flight Control', NAECON 1983.

71. [Leonard 1985]

Leonard J. 'A System look at Actuation Concepts and Alternatives for Primary Flight Control', Proc. of Advanced Actuation Control and Integration for Space Vehicles, Sandiego, California, 9 October, 1985.

72. [Lewis 1988]

Lewis N. E. and Miller M. B. 'Wavelength Division Multiplexed Fibre Optic Sensors for Aircraft Application', SPIE, Vol. 989 Fibre Optic Systems for Mobile Platforms II, 1988.

73. [Lewis 1989]

Lewis N. E. and Miller M. B. 'Wavelength Division Multiplexed Fibre Optic Sensors for Aircraft Application', SPIE, Vol. 1173, Fibre Optic System for Mobile Platforms III, 1989.

74. [Lewis 1991]

Lewis N. E. 'ARINC 629 Optical Bus', ARINC 629 Databus, One Day Conference, RAeS, London, U.K., 24 September, 1991.

75. [Little 1988]

Little W. R. 'A Review of Fibre Optic Flight Experience; Past problems, Future Direction', SPIE., Vol. 989, Fibre Optic Systems for Mobile Platforms II, 1988.

76. [Little 1991]

Little R. 'Advanced Avionics for Military Needs', Computing and Control Engineering Journal, January 1991.

77. [Lucas Schaevitz[a]]

Lucas Schaevitz[a]. Sales literature: Linear and Angular Displacement Transducers - Lucas Schaevitz.

78. [Lucas Schaevitz[b]]

Lucas Schaevitz[b]. Sales literature: Magnetostrictive Linear Displacement Transducer - Lucas Schaevitz.

79. [Luo[a] 1988]

Luo R. C and Kay M. G. 'Multisensor Integration and Fusion: Issues and Approaches', SPIE, Vol. 931, Sensor Fusion, 1988.

80. [Luo[b] 1988]

Luo R. C and Lin M. H 'Robot Multisensor Fusion and Integration: Optimum Estimation of Fused Senor Data', Proc. of IEEE, 1988.

81. [McGough 1986]

McGough J. 'Evaluation of Data Busses for Flight Critical Control Applications', Proc. of IEEE, 1986.

82. [McLaughlin 1989]

McLaughlin R and Shaw E. 'F/A-18 Aileron Smart Servoactuator', Proc. of IEEE, 1989.

83. [Merrill et. al. 1988]

Merrill W. C, De Latt J. C and Bruton W. M. 'Advanced Detection, Isolation, and Accommodation of Sensor failure - Real Time Evaluation', Journal of Guidance, Vol. 11, No. 6, Nov-Dec. 1988.

84. [Miles 1988]

Miles J. A. H. 'Techniques for Knowledge-Based Data Fusion', Application Of Artificial Intelligence to Command and Control Systems, Peter Peregrinus Ltd. 1988.

85. [Miller 1990]

Miller G. E. 'Application of Analogue Fibre Optic Position Sensors to Flight Control Systems', SPIE, Vol 1367, Fibre Optic and Laser Sensors VIII, 1990.

86. [Miller 1988]

Miller G. E. 'Fibre Optic Sensors for Aircraft', SPIE Vol. 985, Fibre Optic and Laser Sensors VI, 1988.

87. [Mirza 1991]

Mirza N. A and Earl M. P. 'DATAC Experience on the Boeing 7J7', ARINC 629 Databus, One Day Conference, RAeS, London, U.K., 24 September, 1991.

88. [Montalk 1988]

Montalk P. P. D. 'A320 - The Next Generation', Civil Avionics - The Future International Scene, RAeS Proc. 17 March, 1988.

89. [Moor 1991]

Moor J and Shorland M. 'ARINC 629 Status and Terminal Technology', ARINC 629 Databus, One Day Conference, RAeS, London, U.K., 24 September, 1991.

90. [Morgan 1991]

Morgan M. J. 'Integrated Modular Avionics for Next-Generation Commercial Airplanes', NAECON '91, Dayton, OH, May 1991.

91. [Norton 1986]

Norton W. 'Advanced Electromechanical Actuation System Flight Test, Final Report'; Rep No AD-A176148, 4950-FTR-86-4, June 1986.

92. [Norton 1987]

Norton W. J. 'Flight Test of the Advanced Electromechanical Actauation System' 18th Annual Symp. of the Soc. of Flight Test Engineers, Lanchester, California, September 28-October 2, 1987

93. [Osder 1986]

Osder S. S. 'Helicopter Avionics Architecture for Integrating Flight Critical Functions', Proc. of IEEE, 1986.

94. [Ostgaard et. al. 1987]

Ostgaard J, Szkody R, Bennung S, Purdy K, Maueraberger G and Soraya M. 'Architecture Spec for Pave Pillar', AFWAL-TR-87-1114, January 1987.

95. [Palmer et. al. 1987]

Palmer M. T, Abbott K H, Schutte P. C, and Ricks W. R. 'Implementation of a Research Prototype Onboard Fault Monitoring System', AIAA Paper No. 87-2777.

96. [Patton 1989]

Patton R. 'Fault Diagnosis in Dynamic Systems: Theory and Application', Edited By Patton R, Frank P and Clark R, Prentice Hall, 1989.

97. [Patton 1990]

Patton R. J. 'Fault Detection and Diagnosis in Aerospace System using Analytical Redundancy', IEE Colloquium on Condition Monitoring and Fault Tolerance, IEE Inst. Savoy Place, London, November, 1990.

98. [Pau 1986]

Pau L. F. 'Survey of Expert Systems for Fault Detection, Test Generation and Maintenance', Expert Systems, Vol. 3, No. 2, April 1986.

99. [Penny and Giles]

Penny and Giles. Sales literature: Position Sensors.

100. [Pierce]

Pierce J. 'A Multiplexed System using a Fibre Optic Serial Data Bus', Sperry Flight Systems, Phoenix, Arizona.

101. [Rao 1991]

Rao B. S. Y. 'Data Fusion Methods in Decentralised Sensing Systems', PhD Thesis, Department of Engineering Science, University of Oxford, 1991.

102. [Ray 1986]

Ray A. 'Selection of Media Access Protocol for Distributed Digital Avionics', 5th A.M. Control Conference, Seattle, W.A., 18-20 June, 1986.

103. [Richardson 1988]

Richardson J. M and Marsh K. A. 'Fusion of Multisensor Data', The International Journal of Robotics Research, Vol. 7, No. 6, December 1988.

104. [Rogers 1991]

Rogers B. P. Private Communication, GEC Aerospace Ltd., Abbey Works, Titchfield, Fareham, Hampshire, U.K., 17 July, 1991.

105. [Rosen et. al. 1988]

Rosen W. A, Schrager C and Smith S. M. 'An Analysis of Reliability in Fibre Optic Ring and Star Networks', SPIE, Vol. 989, Fibre Optic Systems for Mobile Platforms II, 1988.

106. [Rossenberg 1984]

Rossenberg K. W. 'Trends in Digital Flight Control systems', Design and Advanced Concepts of Avionics/Weapons System Integration, RAeS Symposium, 3-4 April, 1984.

107. [Rothman 1989]

Rothman P. L and Bier G. 'Evaluation of Sensor Management Systems', NAECON '89.

108. [Roy 1984]

Roy S. D. 'The Impact of Integrated Systems - The Helicopter Designer's Viewpoint', Design and Advanced Concepts of Avionics/Weapons System Integration, RAeS Symposium, 3-4 April, 1984.

109. [Roy 1989]

Sample J. C. 'Three Civil Aviation Databusses', Avionics, March 1989.

110. [Schutte 1989]

Schutte P. C. 'Real-Time Fault Monitoring for Aircraft Application Using Quantitative Simulation and Expert Systems', AIAA Paper No. 89-3103-CP.

111. [Shaw et. al. 1986]

Shaw J. L, Herzog H. K and Okubo K. 'Digital Autonomous Terminal Access Communication (DATAC)', 7th Digital Avionics Systems Conference, Fort Worth, Texas, 13-16 October, 1986.

112. [Shaw 1988]

Shaw J. L and Sutcliffe P. L. 'ARINC 629 Data Bus System', Civil Avionics - The Future International Scene, RAeS Proc., 17 March 1988.

113. [Sherif 1985]

Sherif F. 'Advanced Digital Optical Control Actuation for the ADOCS', SAE, 1985.

114. [Sydenham 1985]

Sydenham P. H. 'Transducers in Measurement and Control', Third Edition, Adam Higler Ltd, 1985.

115. [Stainslaw 1984]

Stainslaw D. L. 'General Aviation Data Bus Update', AIAA Paper No. 84-2637.

116. [Stainslaw 1989]

Stainslaw D. L. 'ARINC 629: The New Airline Data Bus - Part 1', Avionics, Vol 13, No. 3, March 1989.

117. [Stallings 1984]

Stallings W. 'Local Networks - An Introduction', McMillan Publishing Company, 1984.

118. [Subra et. al. 1990]
Subra H, Paquier M and Graves D. 'New Avionics Architecture Concept for Commercial Aircraft', 17th ICAS Congress, Stockholm, Sweden, 1990.

119. [Sutcliffe 1988]
Sutcliffe P. L. 'Integrated Avionics- Future Developments', Civil Avionics - The Future International Scene, RAeS Proc., 17 March, 1988.

120. [Thomas 1983]
Thomas R. E. 'A Standardized Digital Bus for Business Aviation', Proc. of IEEE, 1983.

121. [Thompson 1985]
Thompson K. 'Lockheed-Georgia and Electric Primary Flight Control Systems', NAECON 1985.

122. [Thomson 1983]
Thomson K. 'Demonstration of Electromechanical Actuation Technology for Military Air Cargo Transport', NAECON 1983.

123. [Thomson 1987]
Thomson K. 'The Electrical Control of Large Aeroplanes', NAECON 1987.

124. [Todd 1988]
Todd J.R. 'Toward Fly-by-Light Aircraft', SPIE, Vol. 989, Fibre Optic Systems for Mobile Platforms II, 1988.

125. [Todd 1989]
Todd J. R. 'Development of Fly-by-Light System', SPIE, Vol. 1173, Fibre Optic Systems for Mobile Platforms III, 1989.

126. [Tucker 1988]
Tucker B. 'Advances in Flying Control Systems', Civil Avionics - The Future International Scene, RAeS Proc. 17 March, 1988.

127. [Turk 1984]
Turk P. 'Pave Pillar', International Defence Review, Special Electronics, No. 2, 1984.

128. [Tzafestas 1989]

Tzafestas S. G. 'System Fault Diagnosis using the Knowledge based Methodology', in Fault Diagnosis in Dynamic Systems: Theory and Application, Edited By Patton R, Frank P and Clark R, Prentice Hall, 1989.

129. [Uhlhorn 1991]

Uhlhorn R. W. 'The Fibre-Optic High-Speed Data Bus for a New Generation of Military Aircraft', IEEE LCS, February 1991.

130. [Uhlhorn et. al. 1990]

Uhlhorn R. W, McDermott T. A and Goldman P. C. 'An Overview of the Fibre-Optic Active Star Coupler Program', NAECON '90.

131. [Uhlhorn 1987]

Uhlhorn R. W. 'Design of a Passive Star-Coupled Fibre Optic High Speed Data Bus for Military Aircraft', NAECON '87.

132. [Uhlhorn AGARD]

Uhlhorn R. W. 'Fibre Optic Buses and Networks for Advanced Avionics Architectures', AGARD-LS-158.

133. [Ward 1983]

Ward D. 'The All-Electric Helicopter', NAECON 1983.

134. [White 1987]

White J. A. P. 'The Development of Electromechanical Actuation for Aircraft Systems' Aerospace, November 1987.

135. [Xu 1991]

Xu Y. M. 'Knowledge based Fault Monitoring for Large Complex Systems', PhD Thesis, Department of Aeronautics and Astronautics, Southampton University, 1991.

136. [Yount 1985]

Yount L. J. 'Generic Fault - Tolerance Techniques for Critical Avionics System, AIAA Paper 85 -1980.

137. [Yount et. al. 1985]

Yount L. J, Liebel K. A and Hill B. H. 'Fault Effect Partition and Partitioning for Fly-by-Wire/Fly-by-Light Avionics Systems', AIAA Paper No. 85-6002.

138. [Yount 1984]

Yount L. J. 'Architectural Solutions to Safety Problems of Digital Flight - Critical Systems for Commercial Transports', AIAA Paper No. 84-2603.

139. [Zenter 1983]

Zenter J.C. 'Aircraft EMC Problems and their Relationship to Subsystem EMI Requirements', NAECON '83, Dayton Convention Centre, 17-19 May, 1983.