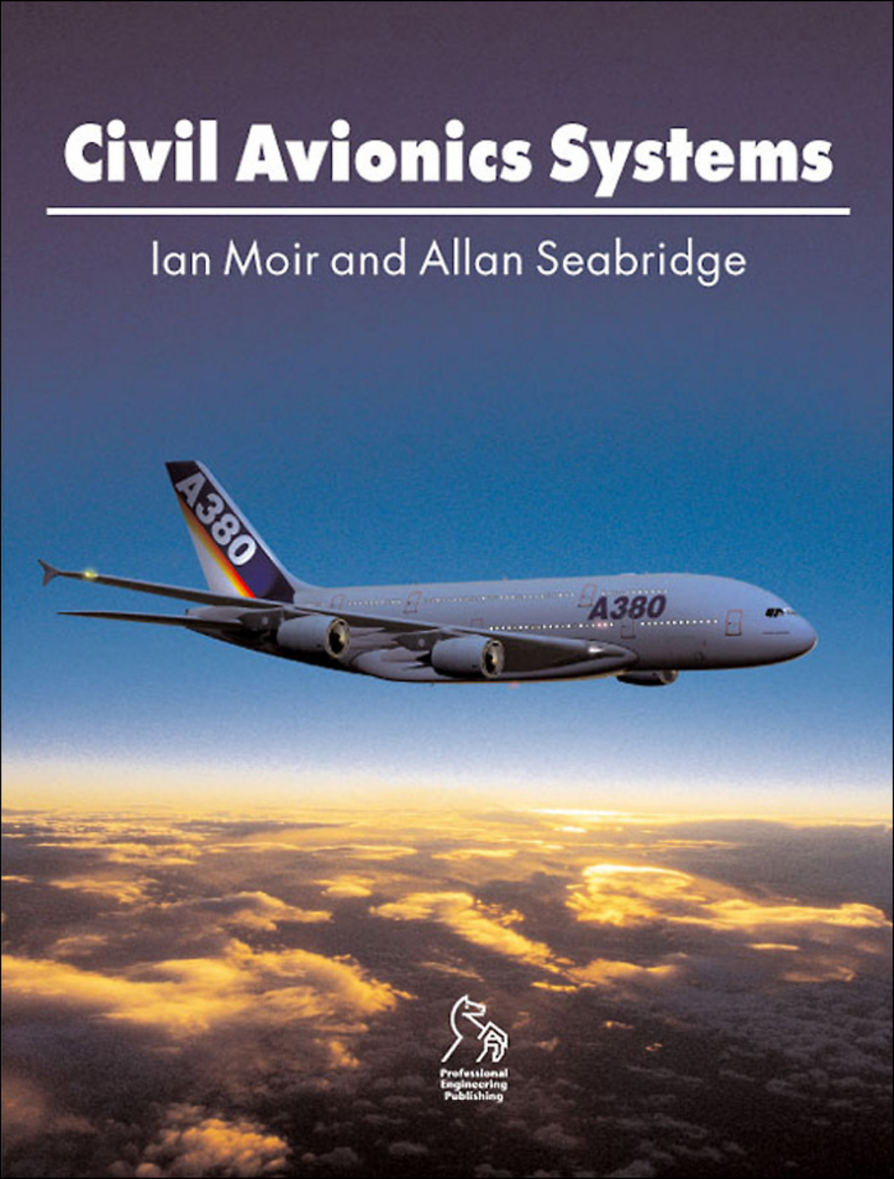
Civil Avionics Systems

Ian Moir and Allan Seabridge



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Engineering
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Ian Moir

Allan G Seabridge

Display chapter contributed by Malcolm Jukes

Professional Engineering Publishing

Professional Engineering Publishing Limited,
London and Bury St Edmunds, UK

This edition published 2003 by Professional Engineering Publishing, UK.
Published in USA by American Institute of Aeronautics and Astronautics, Inc.

ISBN 1 86058 342 3

A CIP catalogue record for this book is available from the British Library.

Printed and bound in Great Britain by St Edmundsbury Press Limited, Suffolk, UK

Cover image © Airbus

About the Authors

Ian Moir *BSc, CEng, FRAeS, FIEE* served twenty years in the Royal Air Force as an Engineering Cadet/Officer, retiring with the rank of Squadron Leader. He then went on to work for eighteen years at Smiths Industries, Cheltenham, UK. Here he had responsibilities for the introduction of avionics technology into aircraft utilities systems on both military and civil aircraft. He was Programme Manager for the integrated Utilities Management System on the UK Experimental Aircraft Programme (EAP); and technology demonstrator for the European Fighter Aircraft. Ian's principal successes at Smiths Industries included the selection and development of new integrated systems for the McDonnell Douglas/Boeing AH-64C/D Longbow Apache attack helicopter and Boeing 777 (Queens Award for Technology – 1998), both of which are major production programmes.

Ian has over 40 years' experience in the aerospace industry. He is currently an International Aerospace consultant, operating in the areas of aircraft electrical and utilities systems and avionics.

Allan Seabridge *BA, MPhil* is currently the Chief Flight Systems Engineer at BAE SYSTEMS, a position held since 1998. Before that he was the Avionics Integrated Product Team Leader on the Nimrod MRA4 programme for five years. He has worked in the aerospace industry for over 30 years in flight systems and avionic systems engineering, business development, and project management. He has been involved in a wide range of military fast jet, trainer, and ground or maritime surveillance aircraft projects.

Allan has worked in many international collaborative programmes in Europe and the United States, and he has led a number of national and international engineering teams. This has led to an interest in all aspects of system engineering capability – the practice of engineering, the processes and tools employed, and the people and skills required.

Malcolm Jukes *BSc, FRAeS, FIEE* has over 35 years' experience in the aerospace industry, mostly working for the Smiths Group at Cheltenham, UK. Among his many responsibilities as Chief Engineer for Defence Systems Cheltenham, Malcolm managed the design and experimental flight trials of the first UK Electronic Flight Instrument System (EFIS) and subsequently the development and application of shadow-mask CRT technology for multi-function, head-down displays on the F/A-18, AV8B, Eurofighter Typhoon and EH101 aircraft. In this role, and subsequently as Technology Director, he was responsible for product technical strategy and the acquisition of new technology for Smiths UK aerospace products. One of his most significant activities was the application of AMLCD technology to civil and military aerospace applications.

Malcolm is now an aerospace consultant operating in the areas of displays, display systems, and mission computing.

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Acknowledgements

The authors have been practising aerospace engineers with a collective experience approaching 80 years, much of that time spent in the specification, design, and development of avionics and utilities systems for high-performance military aircraft and civil aircraft. It is a truism that the more experience and knowledge gained with modern avionics systems, the more it is realized how extensive the subject is and how very much more there is to learn. The writing of this book would not have been possible without the generous support, advice, and guidance of many colleagues, specialists, and companies on both sides of the Atlantic.

Special mention must be made of Malcolm Jukes, formerly of Smiths Aerospace, who added his considerable and specialized experience to the preparation of the book; especially in the Displays chapter – Chapter 7.

The comprehensive nature of the book would not have been possible without the dedication and persistence of those who reviewed the book and offered advice as to how it could be improved:

Michael I Braasch of Ohio University, US
Derek Bracknell of QinetiQ, UK
Mike Hirst of Loughborough University, UK
Dr Amy Prichard of Georgia Institute of Technology, US

During the writing and review of the book generous help has also been given by a wide range of individuals and specialists who have provided specialized support in their fields of expertise to ensure that the details of all the systems described are accurate.

Invaluable support and assistance have been given by the following companies and organizations:

Airbus
BAE SYSTEMS
Boeing
Bombardier Aerospace
CMC Electronics
Duxford Imperial War Museum
Honeywell
Korry Electronics
Micro Circuit Electronics
Northrop Grumman Corporation
Parker Aerospace
Raytheon
Rockwell Collins
Society of Automobile Engineers (SAE)
Smiths Aerospace
QinetiQ
VDO Luftfahrtgerate Werk

Foreword by B Tucker

Over the past forty years, the development of aircraft for civil applications has been enhanced by the increasing introduction of electronic systems to enable greater operational freedom, safety, and efficiency to be achieved.

These electronic systems, called avionics, have been at the heart of the creation of the sophisticated commercial airliners of today and they will play an important part in the succeeding generations of aircraft in the 21st century. Avionics systems have played a key role in the great advances in air safety, which have been a feature of the past few decades. They have played a major part in providing safe and affordable long-haul travel for business and leisure users.

In addition, parallel developments for military aircraft have offered technologies for both civil and military applications, and techniques developed in one area have transferred to the other with rapidity and ease. Both civil and military avionics systems have utilized, to a very large extent, the revolution in computing that has so much changed the growth of technology since the 1950s.

To keep up to date with these avionics systems can be a formidable task, made more complex by the architectural concepts within which avionics systems lie. Technological advances in processing, memory, displays, and data communication systems have made, and will continue to make, significant changes to individual systems and systems architectures in a constant striving for even better safety, maintainability, and operating costs.

This book offers the reader an opportunity to understand the evolution of the systems being employed today, and it gives detailed explanations of the architecture within which they operate and the concept of operations of the avionics systems.

As electronic systems began to emerge into the commercial aircraft, they essentially operated independently of each other and could be considered and certificated, from that viewpoint. This situation has changed very rapidly over the last two decades and today the interaction between systems is both complex and essential for safe aircraft operation.

The reader will be able to understand these interactions from the text and this will enable a good basis for realizing the implications of even greater integration in new aircraft. The next generation of avionics systems will also achieve much greater integration with the whole Air Traffic Management infrastructure and this book will help in preparing the reader for these developing concepts.

The field of avionics is extremely wide and it is commendable that this book manages to cover such a wide area, and yet give detailed information on many of today's key systems.

I hope that you will enjoy reading the book and that it will aid the reader's understanding of the enormous progress that has been made over the past decades in the systems on aircraft and the opportunity that new avionics systems will bring to safety, operational efficiency, and the comfort and convenience of passengers.

Brian G S Tucker OBE, BSc(Hons), FRAeS, CEng,. Senior Vice President,
Business Acquisition & Customer Relations, BAE SYSTEMS

Foreword by B Cosgrove

There can be few of us who have not been touched in the last four decades by the modern miracle of air travel; the ability to step onto an aeroplane and alight in some exotic and distant location within a matter of hours. As well as serving the business traveller, the availability of cheap and safe air transport has brought worldwide mobility within the scope of most people who wish to travel. As the world becomes smaller, families can re-unite and young folk can travel the world and gain experiences that their parents – and certainly grand-parents – could hardly have dreamt of.

The advances of air travel were caused by new developments in aerodynamics, airplane structures, propulsion systems, and the advances in aircraft avionics systems. Avionics systems embraced the application of electronics to control all the vital aircraft functions. These systems can involve the display of data to the flight crew, navigation of the aircraft, communications, flight control and fly-by-wire, and automatic control of the aircraft flight path. They also control many of the housekeeping functions on the aircraft: fuel, hydraulics, environmental, and other systems vital to the safe flight of the aircraft and the comfort of those on-board. To achieve a comprehensive understanding of this multitude of systems is a daunting task. In particular, aircraft could not operate in the crowded skies of today without the comprehensive and accurate knowledge of aircraft position, height, track, etc. that the avionics systems bring. The accurate navigation capability that modern systems bestow enable fuel economy to be maximized and adverse environmental effects and noise footprints to be minimized. Finally, the development of in-flight entertainment systems and rapidly improving air–ground broad-band connectivity enable the entertainment and business needs of the passengers to be satisfied.

These systems also evolve swiftly as the enabling micro-electronic technology rapidly advances, and it is difficult to capture this development in a single reference. In *Civil Avionics Systems* Ian Moir and Allan Seabridge have succeeded in encapsulating all these topics in a single book, written in a clear, concise, and easily understood way. As well as documenting the newer developments, the authors have also presented a historical perspective to enable the reader to understand why systems have evolved in the way they have. This work will be of benefit to many throughout the Industry, whether they are students just embarking upon their career, or senior managers and engineers who wish to keep abreast of the latest technology improvements.

Benjamin A. Cosgrove, NAE, Fellow of the AIAA and RAeS, 1991 recipient of the Wright Brothers Trophy, Retired Senior Vice President of Engineering, Boeing Commercial Airplane Group

Authors' Preface

Civil Avionics Systems is a companion to our book *Aircraft Systems.* Together the books describe the complete set of systems that form an essential part of modern military and commercial aircraft. There is much common ground – many basic aircraft systems such as fuel, air, flight control, and hydraulics are common to both types, and modern military aircraft are incorporating commercially available avionic systems such as liquid crystal cockpit displays and flight management systems.

Avionics is an acronym that broadly applies to AVIation (and space) electrONICS. Civil avionic systems are a key component of the modern airliner and business jet. They provide the essential aspects of navigation, human–machine interface, and external communications for operation in the busy commercial airways. The civil avionic industry, like the commercial aircraft industry it serves, is driven by regulatory, business, commercial, and technology pressures and it is a dynamic environment in which risk must be managed carefully and balanced against performance improvement. The result of many years of improvement by systems engineers is better performance, improved safety, and improved passenger facilities.

Civil Avionics Systems provides an explanation of avionic systems used in modern aircraft, together with an understanding of the technology and the design process involved. The explanation is aimed at workers in the aerospace environment – researchers, engineers, designers, maintainers, and operators. It is, however, aimed at a wider audience than the engineering population; it will be of interest to people working in marketing, procurement, manufacturing, commercial, financial and legal departments. Furthermore, it is intended to complement undergraduate and post graduate courses in aerospace systems to provide a path to an exciting career in aerospace engineering.

The book is intended to operate at a number of levels:

- providing a top-level overview of avionic systems with some historical background;
- providing a more in-depth description of individual systems and integrated systems for practitioners;
- providing references and suggestions for further reading for those who wish to develop their knowledge further.

We have tried to deal with a complex subject in a straightforward, descriptive manner. We have included aspects of technology and development to put the systems into a rapidly changing context. To fully understand the individual systems and integrated architectures of systems to meet specific customer requirements is a long and complicated business. We hope that this book makes a contribution to that understanding.

Ian Moir and Allan Seabridge 2003

Acronyms and Abbreviations

A429 ARINC 429
A600 ARINC 600
A629 ARINC 629

A Amperes
AC Advisory Circular (FAA)
AC Alternating Current
ACARS Aircraft Communications And Reporting System
ACE Actuator Control Electronics (Boeing 777 flight control system)
ACFD Advanced Civil Flight Deck
ACM Air Cycle Machine
ACMP AC Motor Pump
A/D Analogue to Digital
ADC Air Data Computer
ADD Airstream Direction Detector
ADF Automatic Direction Finding
ADI Attitude Direction Indicator
ADIRS Air Data and Inertial Reference System
ADIRU Air Data and Inertial Reference Unit (Boeing 777)
ADM Air Data Module
ADP Air Driven Pump
ADS-A Automatic Dependent Surveillance – Address Mode
ADS-B Automatic Dependent Surveillance – Broadcast Mode
Aero-C SATCOM operating configuration – PC capability
Aero-H/H+ SATCOM operating configuration – high gain global capability
Aero-I SATCOM operating configuration – medium gain over-land capability
Aero-M SATCOM operating configuration – single channel
AEW Airborne Early Warning
AFCS Automatic Flight Control System
AFDC Autopilot Flight Director Computer
AFDS Autopilot Flight Director System
AFDX Avionics Fast Switched Ethernet
AGCU APU GCU
AHARS Attitude and Heading Reference System
AIMS Airplane Information Management System (Boeing 777)
ALT Barometric Altitude
AM Amplitude Modulation
AMJ Advisory Material Joint
AMLCD Active Matrix Liquid Crystal Display
AMSL Above Mean Sea Level
ANP Actual Navigation Performance
AoA Angle of Attack
AOC Airline Operation Centre
AOR–E Azores Ocean Region - East
AOR–W Azores Ocean Region - West
APB Auxiliary Power Breaker
APEX Application Executive
API Application Implementation
APU Auxiliary Power Unit
ARINC Air Radio INC
ARP Aerospace Recommended Practice
AS Aerospace Standard (SAE)
ASCB Avionics Standard Communication Bus
ASI AirSpeed Indicator
ASIC Application Specific Integrated Circuit
ASTOR Airborne STand Off Radar
ATA Air Transport Association
ATC Air Traffic Control
ATI Air Transport Instrument
ATM Air Transport Management
ATN Aeronautical Telecommunications Network
ATP Advanced Turbo-Prop
ATR Air Transport Radio
ATSU Air Traffic Services Unit
AVM Airframe Vibration Monitor
AWACS Airborne Warning And Control System
BAC British Aircraft Corporation (forefather of BAe)
BAe British Aerospace (UK)
BAES BAE SYSTEMS
BC Bus Controller
BCAG Boeing Commercial Airplane Group
BCD Binary Coded Decimal
BCU Bus Control Unit (Boeing 747-400)
BIT Built-In Test
BITE Built-In Test Equipment
BNR Binary
BR Bus Request (1553B)
BPCU Bus Power Control Unit
BRNAV Basic RNAV
BSCU Brake System Control Unit (Boeing 777)
BTB Bus Tie Breaker
BTC Bus Tie Contactor
BTMU Brake Temperature Monitoring Unit
C Centre
CA Course/Acquisition (GPS code)
CAA Civil Aviation Authority
CADC Central Air Data Computer
CAS Calibrated Airspeed
Cat I Category I approach
Cat II Category II approach
Cat III Category III approach
CBIT Continuous Built-In Test
CBL™ Control-By-Light™ (Raytheon fibre-optic data bus)
CCA Common Cause Analysis
CCB Converter Control Breaker (Boeing 777)
CCD Cursor Control Device
CDI Course Deviation Indicator
CDR Critical Design Review
CD ROM Compact Disc Read Only Memory
CDU Control and Display Unit
CF Course to a fix
CF Constant Frequency
CFIT Controlled Flight Into Terrain
CH Channel
CIV Centre Interconnect Valve
CMA Common Mode Analysis
CMCS Central Maintenance Computing System (Boeing 777)
CMM Capability Maturity Model
CMS Central Maintenance System (Airbus)
CNS Communications, Navigation, Surveillance
C of G**,** CG Centre of Gravity
COTS Commercial Off-The Shelf
CPDLC Controller to Pilot Data Link Communications
CPM Central Processing Module

CPU Central Processing Unit
CRT Cathode Ray Tube
CSD Constant Speed Drive
CSDB Commercial Standard Data Bus
CTC Cabin Temperature Controller
Cu in Cubic Inches
CW Continuous Wave
DA Decision Altitude – referenced to sea level
D/A Digital to Analogue
DAPS Data Access Protocol System (ATC mode S)
DATAC Digital Autonomous Terminal Access Communication
DC Direct Current
DCDU Datalink Display and Control Unit (FANS A)
Def Stan Defence Standard
DEOS Digital Engine Operating System
DF Direct to a Fix
DFDR Digital Flight Data Recorder
DFDRS Digital Flight Data Recording System
DFDAU Digital Flight Data Acquisition Unit
DFGC Digital Flight Guidance Computers
DG Directional Gyro
DGPS Differential GPS
DH Decision Height – referenced to terrain
DLP Digital Light Projector
DMC Display Management Computer
DMD Digital Micromirror Device
DME Distance Measuring Equipment
DoD Department of Defense
DTI Department of Trade and Industry (UK)
DU Display Unit
DVOR Doppler VOR
EADI Electronic Attitude Direction Indicator
EAP Experimental Aircraft Programme
EAS Equivalent Airspeed
EBHA Electrical Back-up Hydraulic Actuator (A380 flight control system)
EC European Community
ECAM Electronic Centralised Aircraft Monitor (Airbus)
ECCM Electronic Counter Counter Measures
ECM Electronic Counter Measures
ECS Environmental Control System
EDP Engine Driven Pump
EEC Electronic Engine Controller
E^{2}PROM Electrically Erasable ProgrammableRead-Only Memory
EFA European Fighter Aircraft
EFIS Electronic Flight Instrument System
EGT Exhaust Gas Temperature
EGNOS European Geo-stationary Navigation Overlay System
EGPWS Enhanced Ground Proximity Warning System
EHA ElectroHydrostatic Actuator
EHF Extremely High Frequency
EHSI Electronic Horizontal Situation Indicator
EICAS Engine Indication and Crew Alerting System (Boeing and others)
ELAC Elevator/Aileron Computer (A320 flight control system)
ELCU Electronic Load Control Unit
elec electrical
ELM Extended Length Messages (ATC mode S)
ELMS Electrical Load Management System (Boeing 777)
EMA Electro-Mechanical Actuator
EMC ElectroMagnetic Compatibility
EMI Electro-Magnetic Interference
EMP Electrical Motor Pump
EPC External Power Contactor
EPIC Honeywell integrated avionics system
EPR Engine Pressure Ratio
EPROM Electrically Programmable Read Only Memory
ESA European Space Agency
ESM Electronic Support Measures
ESS Environmental Stress Screening
ETA Estimated Time of Arrival
ETOPS Extended Range Twin Operations
EU Electronic Unit
EUROCAE European Organisation for Civil Aviation Equipment
EW Electronics Warfare
ext external
FA Fix to an Altitude
FAA Federal Aviation Administration
FAC Flight Augmentation Computer (A320 flight control system)
FADEC Full-Authority Digital Engine Control
FANS Future Air Navigation System
FANS 1 Boeing implementation of FANS functions
FANS A Airbus implementation of FANS functions (A330/340)
FANS B Airbus implementation of FANS functions (A320)
FAR Federal Aviation Regulation
FBW Fly-by-Wire
FCC Flight Control Computer
FCDC Flight Control Data Concentrator (A330/340 flight control system)
FCMC Fuel Control and Monitoring Computer (Airbus A340-500/600)
FCMS Fuel Control and Monitoring System (Airbus A340-500/600)
FCPC Flight Control Primary Computer (A330/340 flight control system)
FCSC Flight Control Secondary Computer (A330/340 flight control system)
FD Flight Director
FDAU Flight Data Acquisition Unit
FDC Fuel Data Concentrator (Airbus A340-500/600)
FDDI Fibre Distributed Data Interface
FDR Flight Data Recorder
FDX Fast Switched Ethernet
FHA Functional Hazard Analysis
FIS Flight Information Services
FL Flight Level – altitudes defined above transition level
FLIR Forward Looking Infra-Red
FMEA Failure Modes and Effects Analysis
FMES Failure Modes and Effects Summary
FMGEC Flight Management Guidance and Envelope Computer (Airbus A330/340 flight control system)
FMQGS Fuel Measurement and Quantity Gauging System (Bombardier Global Express)
FMS Flight Management System
FMSP Flight Mode Selector Panel
FMU Fuel Metering Unit
FOG Fibre-Optic Gyroscope
FQIS Fuel Quantity Indication System
FQPU Fuel Quantity Processing Unit (Boeing 777)
FSEU Flap/Slats Electronics Unit (Boeing 777)
FSK Frequency Shift Keying
FSU File Server Unit
FTA Fault Tree Analysis
FTE Flight Technical Error
Fwd Forward
g Acceleration due to gravity
GA General Aviation
Gallileo Proposed European satellite navigation constellation
GCB Generator Control Breaker
GCR Generator Control Relay
GCU Generator Control Unit
GEC General Electric Company (UK)
Gen Generator
GEO Geo-stationary Earth Orbit
GHz 1 x 10^9 cycles per second
GLC Generator Line Contactor
GLONASS GLObal NAvigation Satellite System – Russian equivalent to GPS
GNSS Global Navigation Satellite System
GNSS-1 EGNOS Concept
GNSS-2 Gallileo System
GPCU Ground Power Control Unit
GPS Global Positioning System
GPWS Ground Proximity Warning System
GS Glide slope
H Earth's Magnet Field
HDD Head-Down Display

Hex Heat Exchanger
HF High Frequency
HF Height to a fix
HFDL High Frequency Data Link
HIRF High Intensity Radio Frequency
HMSU Hydraulic Systems Monitoring Unit (Airbus)
HOL High Order Language
HP High Pressure
HSI Horizontal Situation Indicators
HUD Head-Up Display
HX Holding to Fix
HYDIM Hydraulic system control card (Boeing)
Hz Frequency – cycles per second
I^2t I Squared versus Time – electrical trip characteristic
IAP Integrated Actuator Package
IAS Indicated Airspeed
IBIT Initiated Built In Test
IC Integrated Circuit
ICO Instinctive Cut-Out
ICAO International Civil Aviation Organisation
IDG Integrated Drive Generator
IF Initial Fix
IFE In-Flight Entertainment
IFF Identification Friend or Foe
IFR Instrument Flight Rules
IFSD In-Flight Shut-Down
IGV Inlet Guide Vane
I^2S Integrated Information System (Rockwell Collins)
ILS Instrument Landing System
IMA Integrated Modular Avionics
IN Inertial Navigator
in Inch(es)
INMARSAT International Maritime Satellite Organisation
INS Inertial Navigation System
Inv Inverter
I/O Input/Output
IOC Initial Operational Capability
IOM Input/Output Module
IOR Indian Ocean Region
IP Intermediate Pressure (Rolls-Royce triple-shaft engines)
IPT Integrated Product Team
IR Infra-Red
IRP Integrated Refuelling Panel
IRS Inertial Reference System
ISA Instruction Set architecture
ISIS Integrated Standby Instrument System
ISS Integrated Sensor Suite
IT Information Technology
ITO Indium Tin Oxide
JAA Joint Aviation Authorities
JAR Joint Aviation Regulation
JSF Joint Strike Fighter
JTIDS Joint Tactical Information Distribution System
kHz 1 x 10^3 cycles per second
kVA Kilowatt Volts-Amperes
kW Kilowatt
L Left
L Level (fluid)
LAAS Local Area Augmentation System (GPS enhancement)
LAN Local Area Network
lb Pound(s) - mass
LC Liquid Crystal
LCoS Liquid Crystal on Silicon
LED Light Emitting Diode
LF Low Frequency
LISA Limited Instruction Set Architecture
LIV Left Interconnect Valve
LNA Low Noise Amplifier (SATCOM)
LNAV Lateral Navigation
LORAN LOng RAnge Navigation (LORAN C is latest variant)
LOX Liquid Oxygen
LP Low Pressure
LRM Line Replaceable Module
LROPS Long Range Operations
LRU Line Replaceable Unit
LSB Lower Sideband
LSI Large Scale Integration
L Slat Left Slat (MD-80 AFDS)
MA Markov Analysis
Mach, M Mach Number
MAD Magnetic Anomaly Detector
MAT Maintenance Access Terminal (Boeing 777)
MAU Modular Avionics Unit (Honeywell EPIC)
mb milli-bar(s)
Mb Mega Bit
Mb/sec Mega Bits per second
MCDU Multi-function Control and Display Unit
MCM Multi-Chip Module
MCU Modular Concept Unit
MDA Minimum Decision Altitude
MEA More-Electric Aircraft
MF Medium Frequency
MHRS Magnetic Heading Reference System
MHz 1 x 10^6 cycles per second
MIL-STD Military Standard
MLS Microwave Landing System
M_{mo} Maximum Operating Mach Number
MMR Multi-Mode Receiver
MNPS Minimum Navigation Performance Specification
Mode A ATC Mode A (range and bearing)
Mode C ATC Mode C (range, bearing and altitude)
Mode S ATC Mode S (range, bearing, altitude and unique identifier)
MOPS Minimum Operational Performance Standard
MPA Maritime Patrol Aircraft
MSAS Multifunction Satellite Augmentation System (Japan)
MSL Mean Sea Level
MSI Medium Scale Integration
MTBF Mean Time Between Failure
MTI Moving Target Indication
NAS National Airspace System
NAT North Atlantic
NATO North Atlantic Treaty Organisation
NAV Navigation (Mode)
ND Navigation Display
NDB Non-Directional Beacon
NOTAM Notice to Airmen
nm Nautical Miles
NH or N2 Engine speed – high pressure shaft
Ni-Cd Nickel-Cadmium
NL or N1 Engine speed – low pressure shaft
OAT Outside Air Temperature
OBOGs On-Board Oxygen Generating System
OEM Original Equipment Manufacturer
O/H Overheat
OMS On-board Maintenance System
OP Overhead Panel
P Pressure
P_c Capsule Pressure
PC Personal Computer
PCU Power Control Unit (Boeing 777 flight control system)
P_d Dynamic Air Pressure
PDA Power Distribution Assembly
PDR Preliminary Design Review
PFC Primary Flight Computer
PFCS Primary Flight Control System (Boeing 777)
PFD Primary Flight Display
PMA Permanent Magnet Alternator
PMAT Portable Maintenance Access Terminal
PMG Permanent Magnet Generator
POR Pacific Ocean Region
PPS Precise Positioning System (GPS)
PRA Particular Risks Analysis
Pri Primary
PRNAV Precision Area Navigation
PROM Programmable Read-Only Memory
PRSOV Pressure Reducing Shut-off Valve
P_s Static Air Pressure
PSEU Proximity Switch Electronics Unit (Boeing 777)
PSR Primary Surveillance Radar
PSSA Preliminary System Safety Analysis
PSU Power Supply Unit
P_t Total Air Pressure
PTU Power Transfer Unit
QFE Altimeter Setting relating to a specific feature eg airport

QNH Altimeter Setting related to sea level
R Right
RA Resolution Advisory (TCAS II only)
Rad Alt Radar Altimeter
RAE Royal Aircraft Establishment (UK)
RAF Royal Air Force
RAIM Receiver Autonomous Integrity Monitor
RAM Random Access Memory
RAT Ram Air Turbine
RCT Rear Cargo Tank (Airbus A340-500)
Rcv Receive
R&D Research and Development
RDCP Refuel/Defuel Control Panel (Bombardier Global Express)
Recirc Recirculation
RF Radio Frequency
RF Route to a fix
RFI Request For Information
RFP Request For Proposal
RFU Radio Frequency Unit (SATCOM)
RGB Red; Green; Blue
RISA Reduced Instruction Set Architecture
RLG Ring Laser Gyro
RMI Radio Magnetic Indicator
RNAV Area Navigation
RNP Required Navigation Performance
ROM Read-Only Memory
RS Electronic Industries Association Recommended Standard
R Slat Right Slat (MD-80 AFDS)
RSS Root Sum Squared
RT Remote Terminal
RTA Required Time of Arrival
RTCA Radio Technical Committee Association
RTZ Return to Zero
RVR Runway Visual Range
RVSM Reduced Vertical Separation Minima
RW Runway
SAARU Secondary Attitude Air Data Reference Unit (Boeing 777)
SAE Society of Automobile Engineers
SAFEbus® Proprietary Backplane Bus (Honeywell AIMS)
SAR Synthetic Aperture Radar
SAS Standard Altimeter Setting (29.92inHg/1013.2mb)
SAT Static Air Temperature
SATCOM SATellite COMmunications
SB Sideband
SCR Silicon Controlled Rectifier
S/D Synchro to Digital
SDR System Design Review
SDU Satellite Data Unit (SATCOM)
Sec Secondary
SEC Spoiler/Elevator Computers (A320 flight control system)
SELCAL SELective CALling
SFCC Slat/Flap Control Computer (A330/340 flight control system)
SG Symbol Generator
SG Synchronisation Gap (ARINC 629 data bus)
SHF Super High Frequency
SI Smiths Industries (UK), now Smiths Aerospace
SIB System Isolation Breaker
SID Standard Instrument Departure
SIAP Standard Instrument Approach Procedure
SIGINT SIGnals INTelligence
SIM Serial Interface Module
SIOM Standard Input/Output Module
SIU Secure Interface Unit
SLAR Sideways Looking Aperture Radar
SLR Sideways Looking Radar
SMP Systems Management Processor
SOV Shut-Off Valve
SPS Standard Positioning System (GPS)
SRR System Requirements Review
SS Sub-System (1553B)
SSA System Safety Analysis
SSB Single Sideband
SSB Split System Breaker (Boeing 747-400)
SSI Small Scale Integration
SSPC Solid-State Power Controller
SSR Secondary Surveillance Radar
SSR Software Specification Review
STAR Standard Terminal Approach Routes
STC Supplementary Type Certificate
STCM Stabiliser Trim Control Module (Boeing 777 flight control system)
SV Solenoid Valve
SW Switch
T Temperature
TA Traffic Advisory (TCAS I and II)
TAB Tape Automated Bonding
TACAN TACtical Air Navigation system
TACCO TACtical COmmander
TADS Triple Air Data System
TAS True Airspeed
TAT Total Air Temperature
TAWS Terrain Avoidance Warning System
TBD To Be Determined
TCAS Traffic Collision and Avoidance System
TDMA Time Division Multiple Access
TF Track to a fix
TFT Thin Film Transistor
TG Terminal Gap
THS Tailplane Horizontal Stabiliser
TLS Transponder Landing System
TPMU Tyre Pressure Monitoring Unit
T/R Transmit/Receive
TRU Transformer Rectifier Unit
TSO Technical Service Order (FAA)
TURB Turbulence Mode – Weather Radar
TV Television
UHF Ultra High Frequency
UK United Kingdom
UPS United Parcels Service
ULD Underwater Locating Device
US United States
usa Useable Screen Area
USB Upper Sideband
UV Ultra-Violet
VAC Volts AC
VCS Voice Command System
VDC Volts DC
VDL VHF Data Link
‘V’ Diagram Validation and Verification Procedure
VDR VHF Digital Radio
VF Variable Frequency
VFR Visual Flight Rules
VG Vertical Gyro
VHF Very High Frequency
VHFDL Very High Frequency Data Link
VHPIC Very High Performance Integrated Circuit
VLF Very Low Frequency
VLSI Very Large Scale Integration
V_{mo} Maximum Operating Speed
VMS Vehicle Management System
VNAV Vertical Navigation
VOD Video On-Demand
VOR Very High Frequency Omni-Range
VOR/TAC VOR/TACAN
VS Vertical Speed
VSCF Variable Speed Constant Frequency
VSI Vertical Speed Indicator
VSV Variable Stator Vane
W Watt
WAAS Wide Area Augmentation System (GPS enhancement)
WAP Wireless Access Protocol
wef With effect from
WGS World Geodetic System
WOW Weight-On-Wheels
Xmt Transmit
Xfr Transfer
XPC External Power Contactor (Boeing 747-400)
XVGA X Video Graphics Adaptor
ZSA Zonal Safety Analysis

CHAPTER 1

Introduction

This book, *Civil Avionics Systems,* is intended to introduce the reader to an industry that has to deal with issues that are complex and sophisticated, market and technology driven, safety conscious, high integrity, and environmentally influenced. The industry is driven by market factors and trends in public mobility, global business travel needs, and domestic leisure needs. These factors can be seriously perturbed by changes in the world financial situation, terrorist activity, political tension, or public loss of confidence resulting from a perception of poor safety. Nevertheless, the industry has recently weathered some serious downturns and is spurred on to obtain ever-greater standards of performance, passenger convenience, and safety.

The use of electronics on aircraft began in the 1930s, gaining increasing impetus during World War II, and a further increase in momentum from 1980 onwards, when digital technology became available for the first time in the civil transport arena.

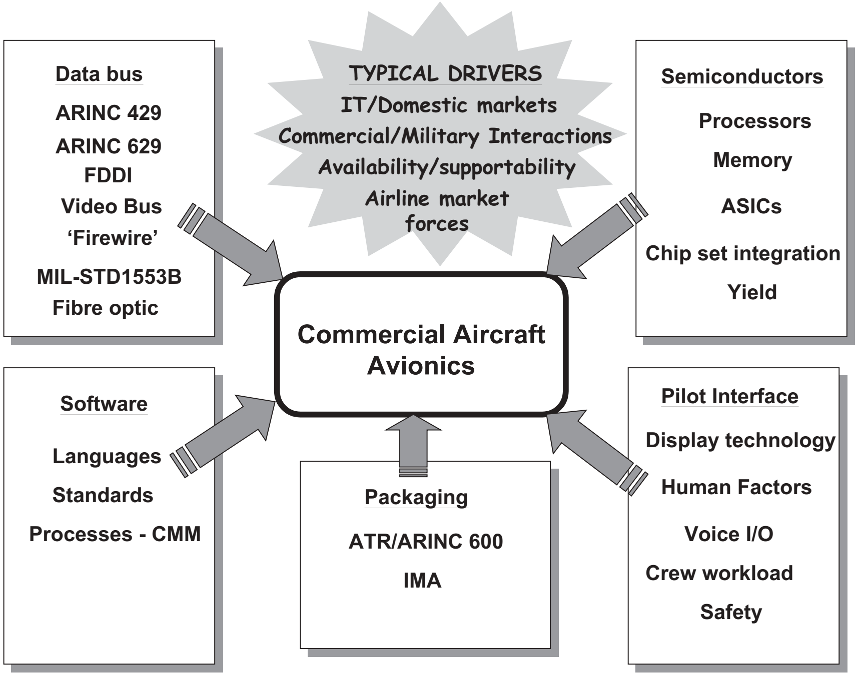
With the advent of jet-powered civil air transport in the late 1950s and early 1960s, aircraft had to rely upon analogue means of navigation, display, and flight control. Synchro-resolvers were the common method of signalling rotary position from one piece of equipment to another. Long-range navigation relied upon methods that would now be considered crude; for example, the VC10 that entered revenue service in the 1960s used a sextant and later LORAN as long-range navigation aids. The aircraft was flown by a four-man crew – captain, first officer, navigator, and flight engineer – compared with the two-man crew used today.

The impetus for change and dramatic increase in performance and capability was the availability of digital electronics, initially developed for military aircraft, starting in the 1960s. The advent of relatively cheap and high-speed computation devices together with the introduction of digital data buses to link equipment enabled the introduction of digital data interchanges between system units. This technology was sufficiently mature and cost effective to be introduced into the Airbus A300 and A310 and the Boeing

757/767 in the early 1980s. As well as the improvements in computation and accuracy that digital technology offered, the introduction of digital data buses using data multiplexing techniques enabled huge quantities of wiring to be removed from aircraft, saving weight, reducing cost, and improving reliability.

It is worth examining the influences that the modern world in general exerts upon commercial aircraft avionics. Figure 1.1 illustrates some of the influences and prime drivers that affect and constrain the development of modern avionics systems in commercial aircraft. Many features of a civil avionics system have their origins in military avionics systems: inertial navigation systems and global satellite communication/navigation systems are good examples of early adoption of military technology in the civil field. Other features such as Head-Up Displays (HUDs) have taken much longer to gain acceptance, and their use is by no means widespread.

Fig. 1.1 *Typical drivers for commercial avionics*



The civil aviation community has been driven primarily by cost constraints whereas the military have historically been more performance driven. However, the advent of highly accurate satellite navigation systems and the need to increase air traffic density while maintaining and improving safety margins have changed the emphasis. The Future Air Navigation System (FANS) concepts enabled by modern avionics technology and being progressively implemented worldwide mean that the airlines are also being performance as well as cost driven. VHF and HF data-link communications with greater bandwidth and information-bearing capabilities are beginning to augment and supplant conventional voice communications.

Figure 1.1 also illustrates the technical factors that influence the civil avionics system. Digital technology in the form of semiconductors and digital data buses is increasingly relying upon the adoption of commercially available technology from the information technology and telecommunications industries rather than the development of bespoke applications, as in the past. This has drawbacks as well as advantages: while such technology, termed Commercial Off-The-Shelf (COTS), may be cheap and allow rapid prototyping, it also needs to be ruggedized to survive in demanding airborne applications. Piece parts such as Integrated Circuits (ICs) may not have been developed to the rigorous requirements specified by the aerospace industry and may need redesigning. Perhaps the most serious concern is the relatively short life cycle of commercial electronics compared with those of aircraft avionics systems: component obsolescence is a huge problem that the industry continually has to address.

Many lessons have been learned in terms of the software languages, standards, and processes that need to be adopted and followed when designing, coding, and testing software for aerospace applications; in many cases the software is flight critical. Packaging means have generally been proven, but there are areas such as Integrated Modular Avionics (IMA) where standards are not generally in place at the present time.

The flight crew interface has improved with the introduction of the ‘glass cockpit’ in lieu of the conventional round dial instruments. Colour displays using active matrix liquid crystal display technology enable complex data to be displayed in an unambiguous manner to the flight crew for flight control and navigation purposes. The expansion of avionics technology into the management of aircraft systems such as fuel, environmental, and hydraulic systems means that information-rich system synoptic displays and status and maintenance data pages relating to these systems may be readily made available to the flight crew. More recently, HUD and voice command systems have been introduced into some of the top-level business jet avionics suites to assist in reducing crew workload.

Finally, the introduction of systems such as the Traffic Collision and Avoidance System (TCAS) and Terrain Avoidance Warning System (TAWS) has improved and will continue to improve flight crew situational awareness, especially in and around airport approaches. The need for the flight crew to be aware of their aircraft location and flight path in relation to other aircraft and adjacent terrain is vital in preserving and improving flight safety.

This book is intended to provide the basis of an appreciation of the magnitude and wide-ranging effects of the application of modern microelectronics to the safe and timely flight around the globe that we all take for granted, be it for business or pleasure. The growth of air transport in the past two decades, and that projected for the future, means that the airspace is becoming increasingly crowded. To continue to assure safe air travel, significant improvements will be required in systems that are already highly capable.

These improvements will arise from even more effective and integrated avionics resulting from the application of systems engineering principles. They will be based upon the application of best practice and world standards to improve functional performance, physical design, and packaging, together with highly effective data communications.

CHAPTER 2

Avionics Technology

The first major impetus for the use of electronics in aviation occurred during World War II. Communications were maturing and the development of airborne radar using the magnetron and associated technology occurred at a furious pace throughout the conflict (**1**).

Transistors followed in the late 1950s and 1960s and supplanted thermionic valves for many applications. The improved cost effectiveness of transistors led to the development of digital aircraft systems throughout the 1960s and 1970s, initially in military combat aircraft where it was used for Nav/Attack systems (see Fig. 2.1).

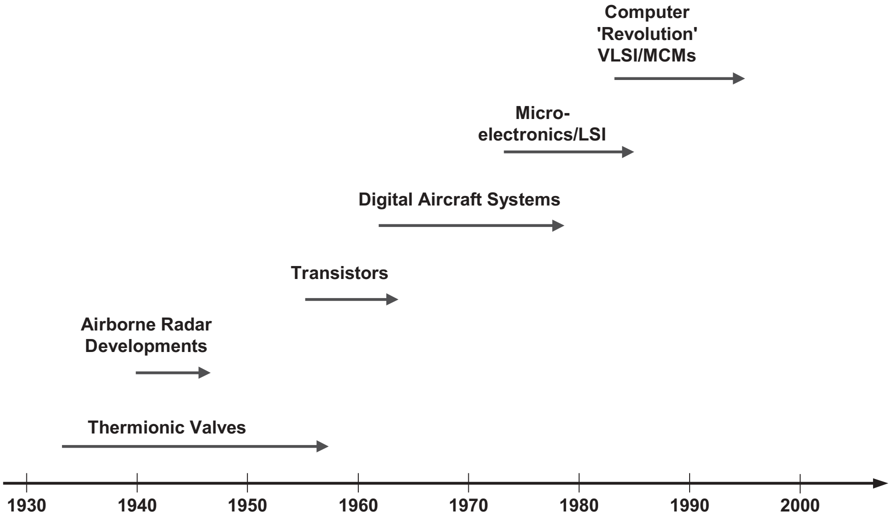


Fig. 2.1 *Major electronics developments in aviation since 1930*

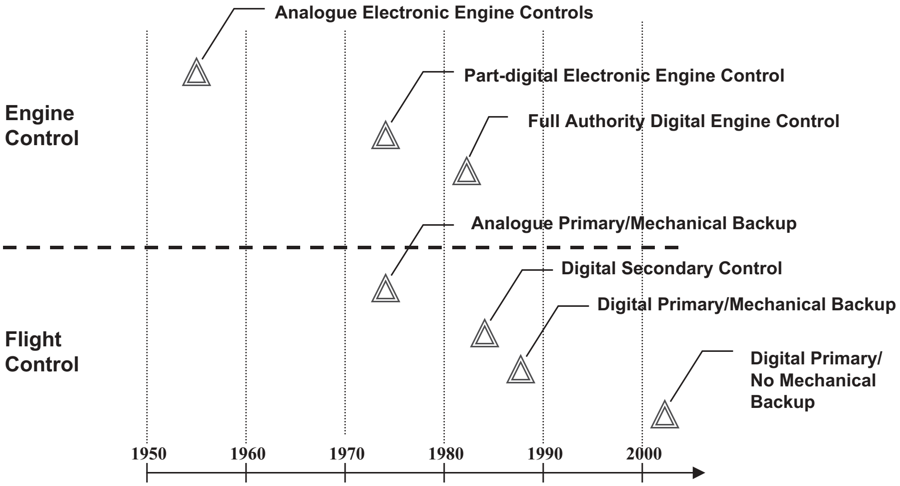
For many years the application of electronics to airborne systems was limited to analogue devices and systems, with signal levels and voltages generally being related in some linear or predictive way. This type of system was generally prone to heat soak, drift, and other non-linearities. The principles of digital computing had been understood for a number of years before the techniques were applied to aircraft. The development of thermionic valves (or vacuum tubes) enabled digital computing to be accomplished, but at the expense of vast amounts of hardware. During World War II a code-breaking machine called Colossus employed thermionic valves on a large scale. The machine was physically enormous and quite impracticable for use in any airborne application.

The first aircraft to be developed in the United States using digital techniques was the North American A-5 Vigilante, a US Navy carrier-borne bomber that became operational in the 1960s. The first aircraft to be developed in the United Kingdom that was intended to use digital techniques on any meaningful scale was the ill-fated TSR 2, which was cancelled by the UK Government in 1965. The technology employed by the TSR 2 was largely based upon solid-state transistors, then in comparative infancy. In the United Kingdom it was not until the development of the Anglo-French Jaguar and the Hawker Siddeley Nimrod in the 1960s that weapon systems began seriously to embody digital computing.

Since the late 1970s/early 1980s, digital technology has become increasingly used in the control of aircraft systems as well as for mission-related systems. A key driver in this application has been the availability of capable and cost-effective digital data buses such as ARINC 429, MIL-STD-1553B, and ARINC 629. This technology, coupled with the availability of cheap microprocessors and more advanced software development tools, has led to the widespread application of avionics technology throughout the aircraft. This has advanced to the point where virtually no aircraft system – including the toilet system – has been left untouched.

The evolution and increasing use of avionics technology for civil applications of engine controls and flight controls since the 1950s is shown in Fig. 2.2. Engine

Fig. 2.2 *Evolution of electronics in flight and engine control*



analogue controls were introduced by Ultra in the 1950s, which comprised electrical throttle signalling used on aircraft such as the Bristol Britannia. Full-Authority Digital Engine Control (FADEC) became commonly used in the 1980s. Digital primary flight control with a mechanical back-up has been used on the Airbus A320 family, A330/A340 using side-stick controllers, and on the B777 using a conventional control yoke. Aircraft such as the A380 are adopting flight control without any mechanical back-up but with electrically signalled back-up. Research in the military field is looking at integration of propulsion and flight control to achieve more effective ways of demanding changes in attitude and speed that may lead to more fuel-efficient operations.

The application of digital techniques to other aircraft systems – utilities systems – began later, as will be described in this chapter. Today, avionics technology is firmly embedded in the control of virtually all aircraft systems. Therefore, an understanding of the nature of avionics technology is crucial in understanding how the control of aircraft systems is achieved.

Avionics technology is influenced strongly by external factors within the aerospace industry – commercial, military, and space – which drive towards ever more exacting standards of performance. This is measured in aircraft performance, crew workload reduction, and safety improvements, as well as supportability. This is perceived by passengers in terms of more reliable transport, fewer delays and cancellations, and more comfort.

External factors also play an important role, with both the commercial and domestic electronic equipment markets dominated by the advances in telecommunications, miniaturization, computing, and display technologies, many of which become incorporated into aerospace systems design**.**

The nature of microelectronic devices

The development of a wholly digital control system has to accommodate interfaces with the ‘real world’, which is analogue in nature. Figure 2.3 shows how the range of microelectronic devices is used in different applications within a digital system.

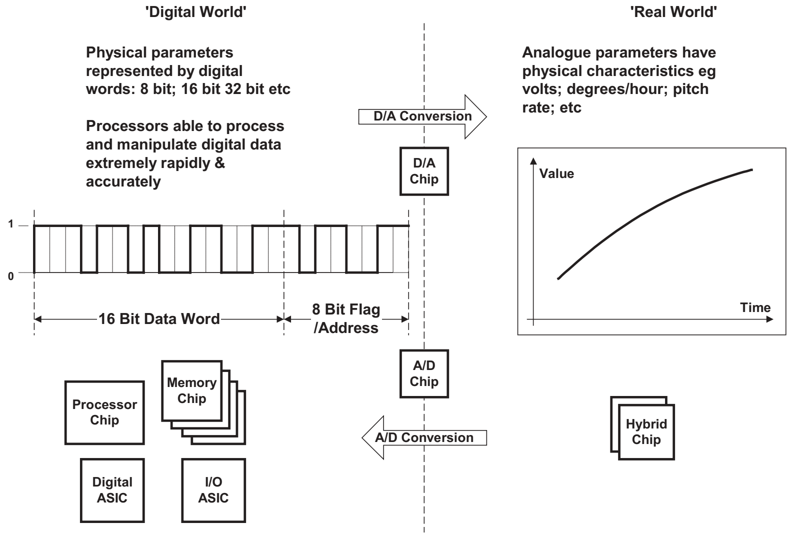


Fig. 2.3 *The nature of microelectronic devices*

Hybrid chips and Input/Output (I/O) Application-Specific Integrated Circuits (ASICs) are key technologies associated with interfacing to the analogue world. A/D and D/A devices undertake the conversion from analogue to digital and digital to analogue signals respectively. Processor and memory devices, together with digital ASICs, perform the digital processing tasks associated with the application (see Fig. 2.3).

A typical interface Integrated Circuit (IC) used for FADEC application is shown in Fig. 2.4.

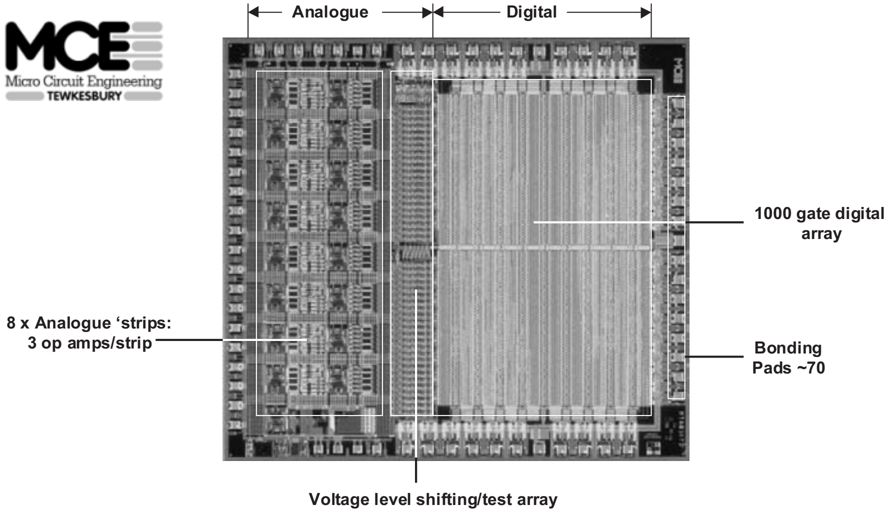


Fig. 2.4 *Typical engine interface hybrid IC (Smiths Aerospace – Micro Circuit Engineering)*

This IC chip comprises two distinct portions: the left side is analogue in nature while the right is digital. Eight analogue channels ('strips'), each comprising three operational amplifiers (op-amps), condition the incoming signal. A voltage shifting/test array in the centre of the device matches the signal to the 1000-gate digital array on the right. Around the extremities of the IC are the bonding pads, of which there are around 70, whereby the device is connected to the host electronic module.

Microelectronic devices are produced from a series of masks that shield various parts of the semiconductor during the processing stages. The resolution of most technology is of the order of 1–3 microns (1 micron is 10^{-6} metres, i.e. one-millionth of a metre or one-thousandth of a millimetre), so the physical attributes are minute. Thus, a device or die of about 0.4 inches square could have hundreds of thousands of transistors/gates to produce the functionality required of the chip. Devices are produced many at a time on a large circular semiconductor wafer. Some devices at the periphery of the wafer will be incomplete, and some of the remaining devices may be flawed and defective (Fig. 2.5). However, the remainder of the good dice may be trimmed to size, tested, and mounted within the device package. The size of the dice, the complexity and maturity of the overall semiconductor process, and the quality of the material will determine the number of good dice yielded by the wafer, and this yield will eventually be reflected in the cost and availability of the particular device.

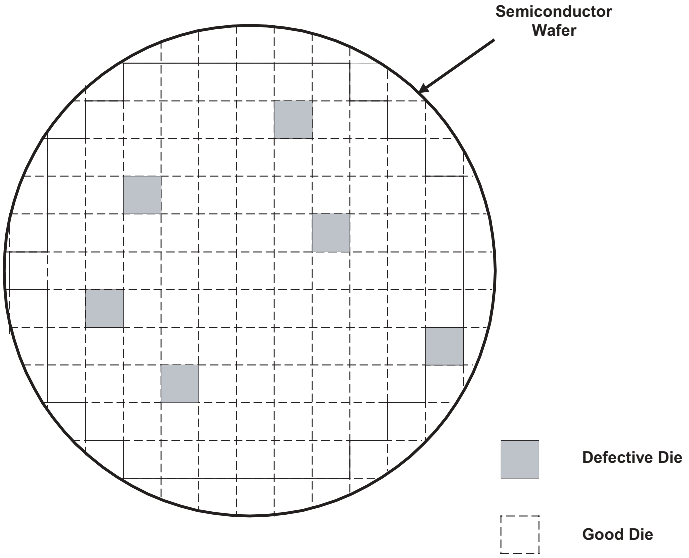


Fig. 2.5 *Semiconductor wafer yield*

Microelectronics devices are environmentally screened according to the severity of the intended application; usually, three levels of screening are applied, in increasing levels of test severity:

- Commercial grade.
- Industrial grade.
- Aerospace military grade – also used in many cases for civil aerospace applications.

There is little doubt that in the past this screening technique has helped to improve the maturity of the manufacturing process and quality of the devices. However, as an increasingly small proportion of devices overall are used for aerospace applications, full military screening is difficult to assure for all devices. There is a body of opinion that believes that screening is not beneficial, and adds only to the cost of the device. It is likely that avionics vendors will have to take more responsibility in future for the quality of the devices used in their product. There is an increasing and accelerating trend for aerospace microelectronics to be driven by the computer and telecommunications industries.

The extent of the explosion in IC developments can be judged by reference to Fig. 2.6. This shows a greater than tenfold increase per decade in the number of transistors per chip. Another factor to consider is the increase in the speed of device switching. The speed of operation is referred to as gate delay; gate delay for a thermionic valve is of the order of 1000 ns (1ns is 10^{-9} or one-thousandth of one-millionth of a second); transistors are about 10 times quicker at 100 ns. Silicon chips are faster again at ~ 1ns. This gives an indication of how powerful these devices are and why they have had such an impact upon our daily life.

Another area of major impact for the IC relates to power consumption, which can vary depending upon the IC function; memory and processor devices tend to consume higher levels of power and may provide cooling challenges in some applications. Consumption is related to the technology type and speed of operation. The higher the

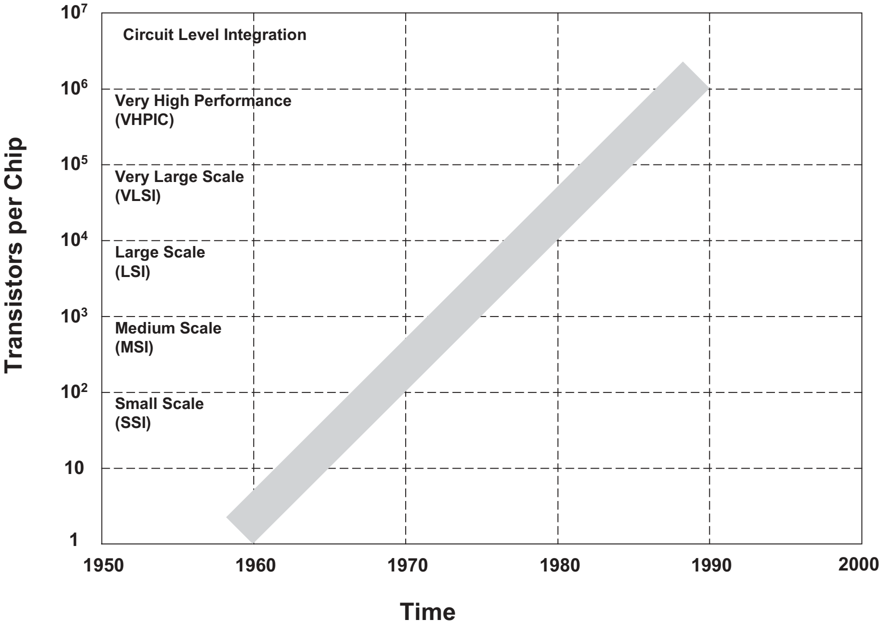


Fig. 2.6 *Trends in integrated circuit development*

speed of operation, the greater is the power required, and vice versa. The main areas where avionics component technology has developed are:

- Processors.
- Memory.
- Data buses.

Processors

Digital processor devices became available in the early 1970s as 4 bit devices. By the late 1970s 8 bit processors had been superceded by 16 bit devices; these led in turn to 32 bit devices such as the Motorola 68000, which have been widely used on the Typhoon and Boeing 777. The pace of evolution of processor devices does present a significant concern owing to the risk of the chips becoming obsolescent, leading to the prospect of an expensive redesign and recertification programme.

Following adverse experiences with its initial ownership of microprocessor-based systems, the US Air Force pressed strong standardization initiatives based upon the MIL-STD-1750A microprocessor with a standardized Instruction Set Architecture (ISA), variously known as limited instruction set or Reduced Instruction Set Architectures (RISAs). The MIL-STD-1750A initiative was successful in that it did introduce standardization into the processing suites of several US Air Force applications but it was expensive to maintain and eventually led to the Department of Defense (DoD) adopting COTS applications. For some types of application, starting with the adoption of the Motorola 68020 on Typhoon, the industry is making extensive use of commercially developed microprocessor or microcontroller products when they are replaced.

Memory devices

Memory devices have experienced a similar explosion in capability. Memory devices comprise two main categories: Read Only Memory (ROM) represents the memory used to host the application software for a particular function; as the term suggests, this type of memory may only be read but not written to. A particular version of ROM used frequently was Electrically Programmable Read Only Memory (EPROM), but this suffered the disadvantage that memory could only be erased by irradiating the device with ultraviolet (UV) light. For the last few years, EPROM has been superceded by the more user-friendly Electrically Erasable Programmable Read Only Memory (E^2PROM). This type of memory may be reprogrammed electrically with the memory module still resident within the Line Replaceable Unit (LRU), and using this capability it is now possible to reprogram many units *in situ* on the aircraft via the aircraft digital data buses.

Random Access Memory (RAM) is read–write memory that is used as program working memory storing variable data. Early versions required a power back-up in case the aircraft power supply was lost. More recent devices are less demanding in this regard.

Portable memory means of data are also used using a Portable Maintenance Access Terminal (PMAT) (see Chapter 10). An example of this is the navigation database which may be updated each month to introduce new routes, obstructions, airfield limitations, and Notices To AirMen (NOTAM).

Digital data buses

The advent of standard digital data buses began in 1974 with the specification by the US Air Force of MIL-STD-1553. The ARINC 429 data bus became the first standard data bus to be specified and used for civil aircraft, being applied throughout the Boeing 757 and 767 and Airbus A300/A310 in the late 1970s and early 1980s. ARINC 429 (A429) is widely used on a range of civil aircraft today, as will become apparent during this chapter. In the early 1980s, Boeing developed a more capable digital data bus termed Digital Autonomous Terminal Access Communication (DATAC) which later became an ARINC standard as A629; the Boeing 777 is the first and at present the only aircraft to use this more capable data bus. At the same time, these advances in digital data bus technology were matched by advancements in processor, memory, and other microelectronic devices such as A/D and D/A devices, logic devices, etc., which made the application of digital technology to aircraft systems possible.

The largest single impact of microelectronics on avionic systems has been the introduction of standardized digital data buses, which greatly improve the intercommunication between aircraft systems. Previously, large amounts of aircraft wiring were required to connect each signal with the other equipment. As systems became more complex and more integrated, so this problem was aggravated. Digital data transmission techniques use links that send streams of digital data between equipment. These data links comprise only two or four twisted wires, and therefore the interconnecting wiring is greatly reduced.

Common types of serial digital data transmission are:

- Single source–single sink. This is the earliest application and comprises a dedicated link from one piece of equipment to another. This was developed in the 1970s for use on Tornado and Sea Harrier avionics systems.
- Single source–multiple sink. This describes a technique where one piece of transmitting equipment can send data to a number of recipient pieces of equipment (sinks). ARINC 429 is an example of this data bus which is widely used by civil transport and business jets.
- Multiple source–multiple sink. In this system, multiple transmitting sources may transmit data to multiple receivers. This is known as a full-duplex system and is widely employed by military users (MIL-STD-1553B) and by the B777 (ARINC 629).

The major digital data buses in widespread use today in avionics are:

- ARINC 429 (A429).
- MIL-STD-1553B, also covered by UK Def Stan 00-18/Parts 1 and 2 and NATO STANAG 3838.
- ARINC 629 (A629).

Other buses include:

- Avionics Standard Communications Bus (ASCB), available in several forms and based upon Ethernet protocols. ASCB was developed by Honeywell and is used in General Aviation (GA) and business jet applications.
- Commercial Standard Data Bus (CSDB) developed by Rockwell Collins for use in GA applications.
- Avionics Full Duplex Ethernet (AFDX) based upon commercial Fast Switched Ethernet (FDX) technology and adopted for the Airbus A380.
- In some applications, commercial RS 232 and RS 422 buses are also used.

Of these, A429 and A629 are commonly in use on civil aircraft. MIL-STD-1553B is a military standard somewhat similar in bus topology, encoding, and data encoding to A629, though the command protocol is different. For reasons of brevity, only A429, A629, and MIL-STD-1553B will be described in detail.

A429 data bus

The characteristics of ARINC 429 were agreed among the airlines in 1977–78, and it was first used throughout the B757/B767 and Airbus A300 and A310 aircraft. ARINC, short for Aeronautical Radio INC., is a corporation in the United States whose stockholders comprise US and foreign airlines and aircraft manufacturers. As such it is a powerful organization central to the specification of equipment standards for known and perceived technical requirements.

The A429 bus operates in a single-source, multiple-sink mode, so that a source may transmit to a number of different terminals or sinks, each of which may receive the data message. However, if any pieces of the sink equipment need to reply, then they will each require their own transmitter and a separate physical bus to do so, and they cannot reply down the same wire pair. This half-duplex mode of operation has certain disadvantages. If it is desired to add additional equipment as shown in Fig. 2.7, a new set of buses may be required – up to a maximum of eight new buses in this example if each new link needs to operate in bidirectional mode.

The A429 bus is by far the most common data bus in use on civil transport aircraft, regional jets, and executive business jets today. Since its introduction on the Boeing 757/767 and Airbus aircraft in the early 1980s, hardly an aircraft has been produced that does not utilize this data bus.

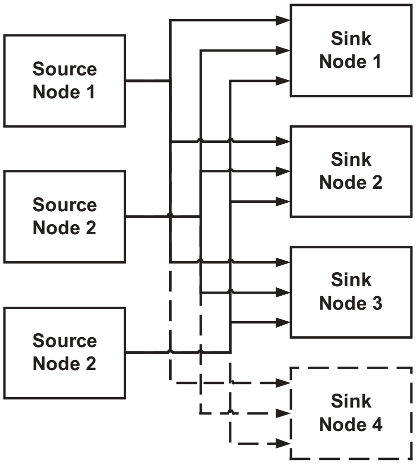


Fig. 2.7 *A429 topology and the effect of adding units*

The physical implementation of the A429 data bus is a screened, twisted wire pair, with the screen earthed at both ends and at all intermediate breaks. The transmitting element shown on the left in Fig. 2.8 is embedded in the source equipment and may interface with up to 20 receiving terminals in the sink equipment. Information may be transmitted at a low rate of 12–14 kilobyte/s or a higher rate of 100 kilobit/s; the higher rate is by far the most commonly used. The modulation technique is bipolar Return To Zero (RTZ), as shown in the box in Fig. 2.8. The RTZ modulation technique has three signal levels: high, null, and low. A logic state 1 is represented by a high state returning

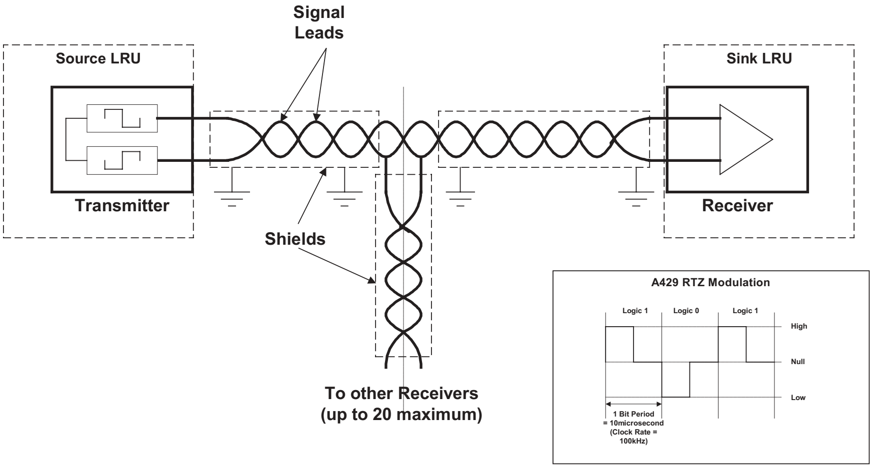


Fig. 2.8 *A429 data bus and encoding format*

to zero; a logic state 0 is represented by a low state returning to null. Information is transmitted down the bus as 32 bit words, as shown in Fig. 2.9.

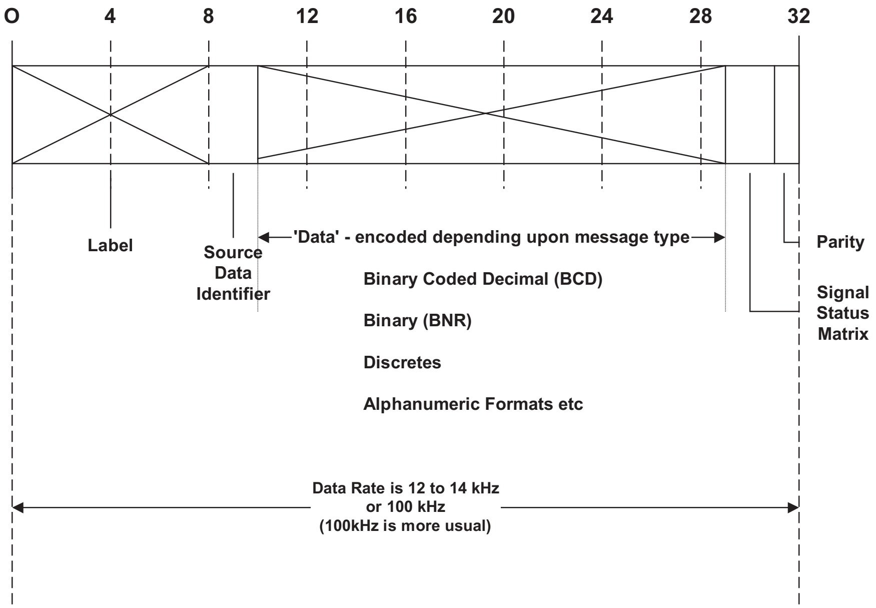


Fig. 2.9 *A429 data word format*

The standard embraces many fixed labels and formats, so that a particular type of equipment always transmits data in a particular way. This standardization has the advantage that all manufacturers of particular equipment know what data to expect. Where necessary, additions to the standard may also be implemented. Further reading for A429 may be found in references (**2**)–(**4**).

MIL-STD-1553B

MIL-STD-1553B has evolved since the original publication of MIL-STD-1553 in 1973. The standard has developed through the 1553A standard issued in 1975 to the present 1553B standard issued in September 1978. The basic layout of an MIL-STD-1553B data bus is shown in Fig. 2.10. The data bus comprises a screened, twisted wire pair along which data combined with clock information are passed. The standard generally supports multiple redundant operation, with dual-redundant operation being by far the most common configuration actually used.

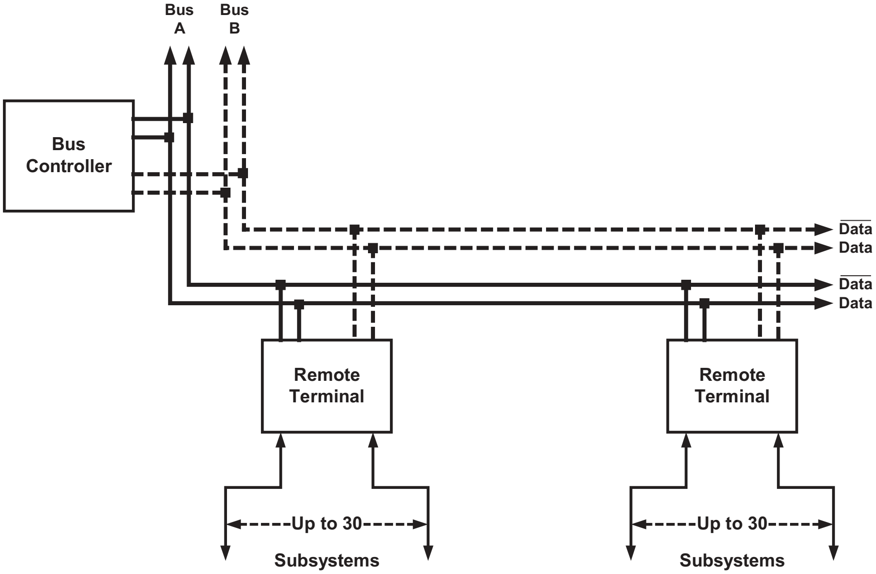


Fig. 2.10 MIL-STD-1553B data bus

Control of the bus is performed by a Bus Controller (BC) which communicates with a number of Remote Terminals (RTs) (up to a maximum of 31) via the data bus. The RTs only perform the data bus related functions and interface with the host (user) equipment they support. In early systems the RT comprised one or more circuit cards, whereas nowadays it is usually an embedded chip or hybrid module within the host equipment. Data are transmitted at 1 MHz using a self-clocked Manchester biphase digital format. The transmission of data in true and complement form down a twisted screened pair offers an error detection capability. Words may be formatted as data words, command words, or status words, as shown in Fig. 2.11. Data words encompass a 16 bit digital word, while the command and status words are associated with the data bus transmission protocol. Command and status words are compartmented to include various address, subaddress, and control functions as shown in the figure.

MIL-STD-1553B is a command–response system in which transmissions are conducted under the control of a single bus controller at any one time; although only one bus controller is shown in these examples, a practical system will employ two bus controllers to provide control redundancy.

Two typical transactions are shown in Fig. 2.12. In a simple transfer of data from RT A to the BC, the BC sends a transmit command to RT A, which replies after a short interval, known as the response time, with a status word, followed immediately by one or more data words up to a maximum of 32 data words. In the example shown in the upper part of Fig. 2.12, transfer of one data word from RT A to the BC will take approximately 70 μs (depending on the exact value of the response time plus propagation time down the bus cable). For the direct transfer of data between two RTs, as shown from RT A to RT B, the BC sends a receive command to RT B followed by a transmit command to RT A. RT A will send its status word plus the data (up to a maximum of 32 words) to RT B, which will then respond by sending its status word to the BC, thereby concluding the transaction. In the simple RT to RT transaction shown in Fig. 12.11, the total elapsed time is around 120 μs for the transmission of a single data word, which appears to be rather expensive owing to the overhead of having to transmit two command words and two status words as well. However, if the maximum number of data words had been transmitted (32), the same overhead of two command and two status words would represent a much lower percentage of the overall message time. For

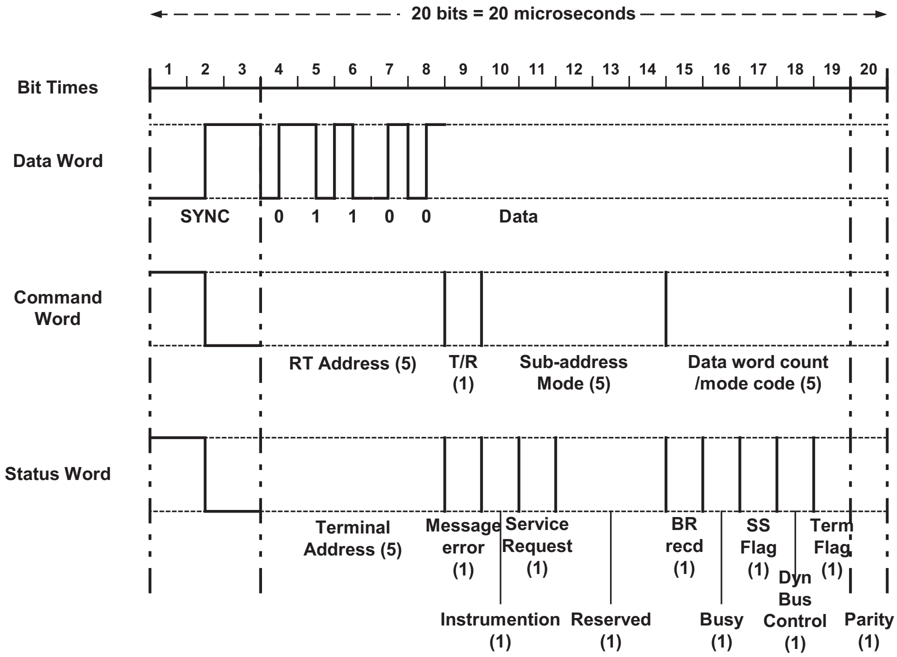


***Fig. 2.11** MIL-STD-1553B data bus word formats*

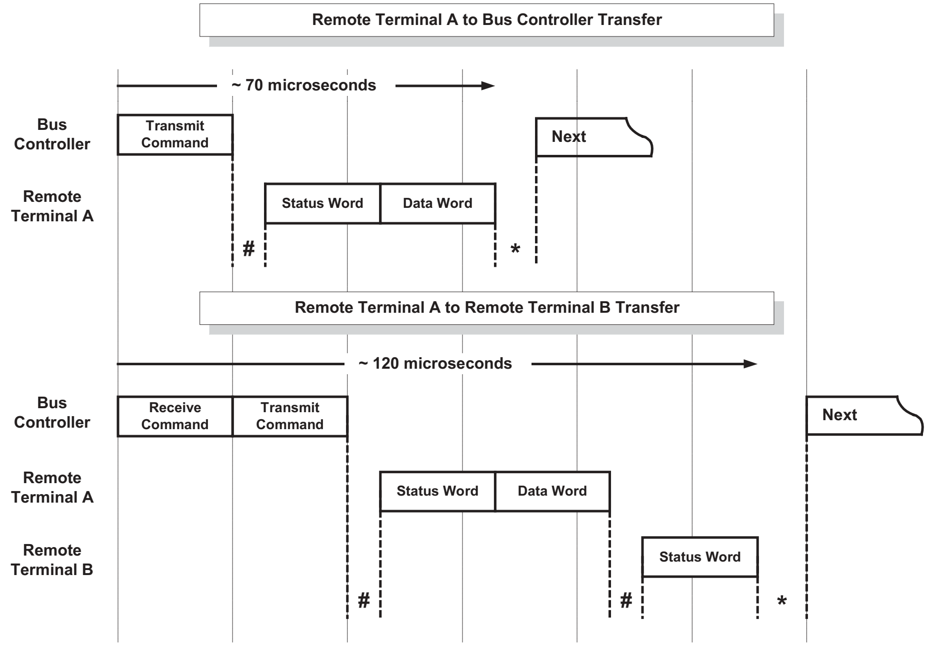
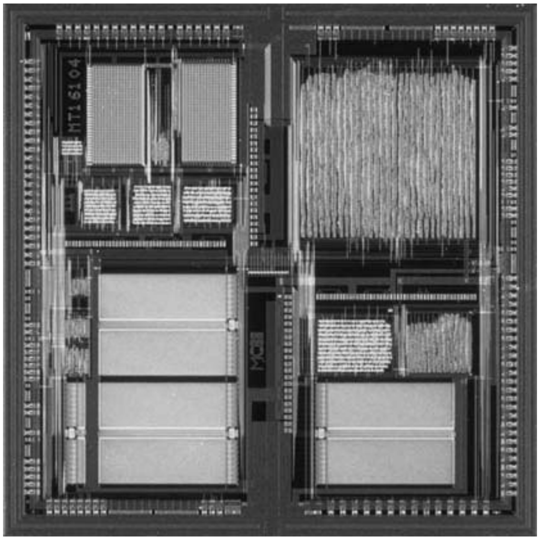


***Fig. 2.12** MIL-STD-1553B typical data transactions*

further reading on MIL-STD-1553B, see reference (**5**).

Most bus controllers and RTs nowadays use embedded chip sets or hybrid terminals available from a number of manufacturers. At the heart of these terminals is an IC that may perform either the RT or the BC MIL-STD-1553B function. A typical digital IC is illustrated in Fig. 2.13.

MCE
Micro Circuit Engineering
TEWKESBURY



1553B full Remote Terminal (RT) and Bus Controller Function

Two transceiver die may be Mounted on top to drive the 1553 bus transformers

Contains a lot of RAM for the storage of 1553 bus messages

Fig. 2.13 *MIL-STD-1553B combined RT/BC IC (Smiths Aerospace – Micro Circuit Engineering)*

ARINC 629 data bus

Like MIL-STD-1553B, the A629 bus is a true data bus in that the bus operates as a multiple-source, multiple-sink system (see Fig. 2.14). That is, each terminal can transmit data to, and receive data from, every other terminal on the data bus. This allows much more freedom in the exchange of data between units in the avionics system than the single-source, multiple-sink A429 topology.

Furthermore, the data rates are much higher than for the A429 bus, where the highest data rate is 100 kbit/s. The A629 data bus operates at 2 Mbit/s or 20 times that of A429 and twice the rate of 1553B. The true data bus topology is much more flexible in that additional units can be fairly readily accepted physically on the data bus. A further attractive feature of A629 is the ability to accommodate up to a total of 128 terminals on a data bus, though in a realistic implementation the high amount of data bus traffic would probably preclude the use of this large number of terminals. The first use of A629 was on the Boeing 777 and is described below.

Figure 2.15 portrays a quadruple data bus implementation as opposed to the dual-redundant MIL-STD-1553B format shown in Fig. 2.10. In fact, on the Boeing 777, A629 data buses are used in the following formats:

- Quadruple-redundant: engine interfaces on to aircraft systems data buses (left, centre 1, centre 2, right).
- Triple-redundant: flight control data buses and a number of other systems on the aircraft systems buses (left, centre, right).

Fig. 2.14 *MIL-STD-1553 and A629 data bus topology*

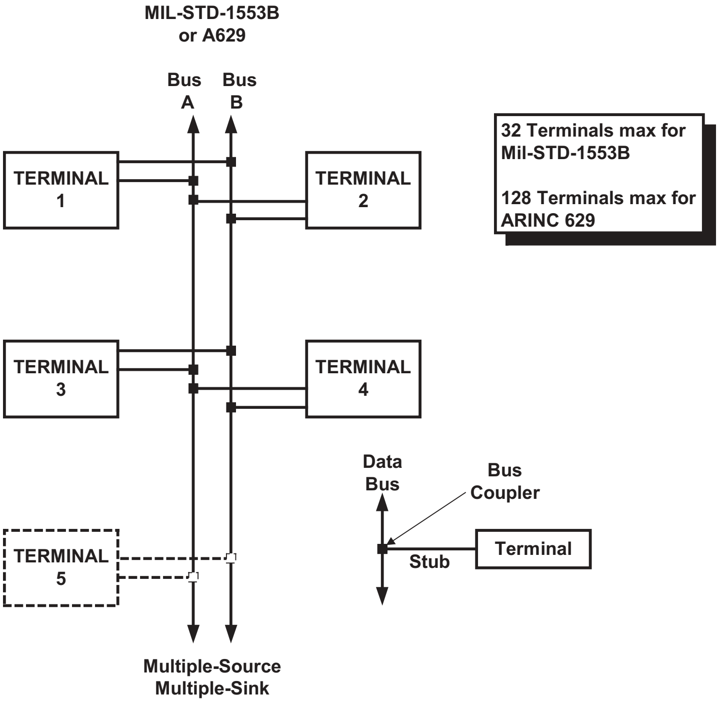
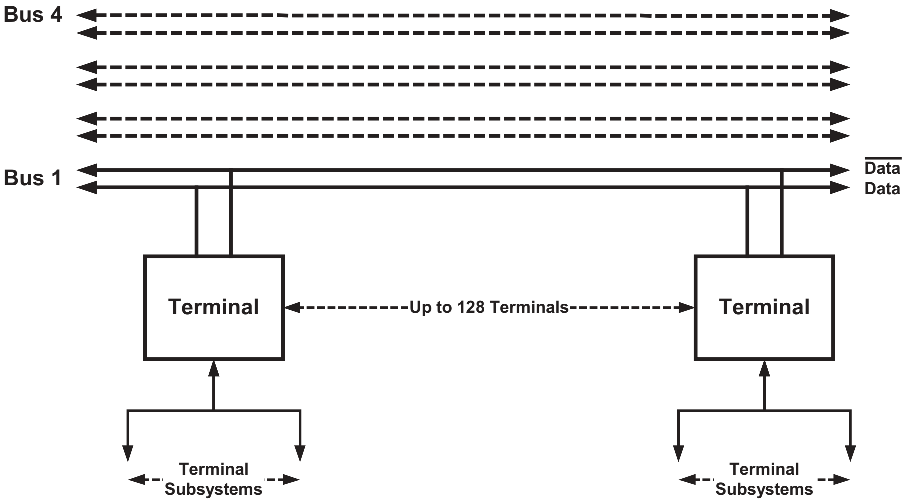


Fig. 2.15 *A629 data bus*



- Dual-redundant: some systems on the aircraft systems data buses (left and right).

The protocol utilized by A629 is a time-based, collision-avoidance concept in which each terminal is allocated a particular time slot to access the bus and transmit data on to the bus. Each terminal will autonomously decide when the appropriate time slot is available through the use of several control timers embedded in the bus interfaces (described below) and transmit the necessary data (see Fig. 2.16). This protocol was the civil aircraft industry's response to the military MIL-STD-1553B data bus that utilizes a dedicated controller to decide what traffic passes down the data bus. It was felt for civil aircraft that the centralized control philosophy of MIL-STD-1553B was inappropriate and difficult to certificate, whereas the A629 concept removes the centralized controller and distributes bus access control around all the terminals on the bus. After each terminal has transmitted its data package, a Terminal Gap (TG) message is transmitted by the next terminal in the sequence, followed by its data bus message. This procedure continues until all terminals have had a time slot allocated to them. At this point a Synchronization Gap (SG) message is transmitted, followed by a TG message; after a short interval the total sequence is repeated. The overall sequence (where every terminal has had access to the bus) equates to one major cycle on the data bus. The order in which terminals access the data bus is determined by the personality embedded in memory located in each terminal chip set. In a way, this technique represents a distributed message control protocol, as opposed to MIL-STD-1553B where message control protocol is centralized and effected only by the bus controller.

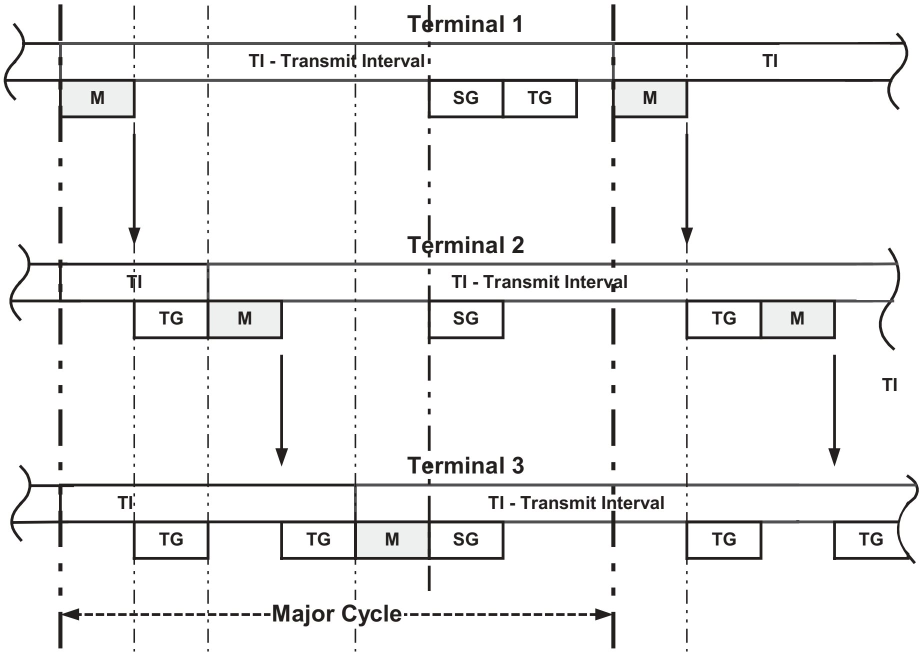
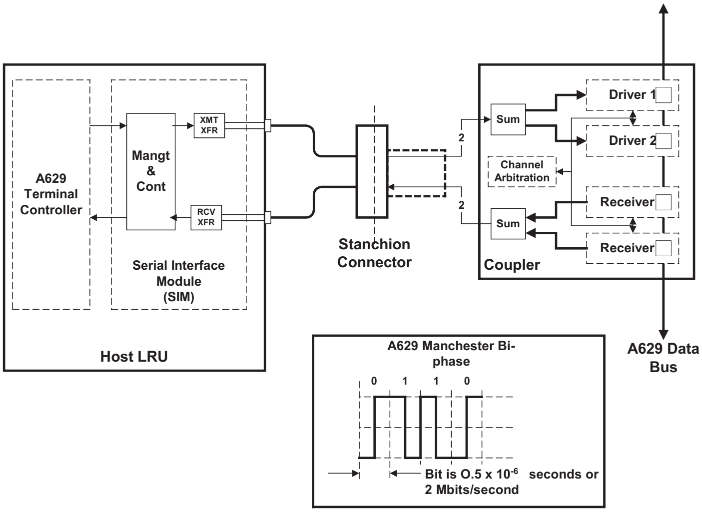


Fig. 2.16 *A629 message sequence*

Because of the higher data rates and higher technology baseline, the A629 bus coupler arrangement is slightly more involved than for A429. Figure 2.17 shows how the host LRU connects to the A629 data bus via the Serial Interface Module (SIM), embedded in the LRU, and via a stanchion connector to the coupler itself. Owing to the transmit/receive nature of the A629 protocol, there are separate channels for transmit and receive. Transformer coupling is used owing to concern that a single bus failure could bring down all the terminals connected to the same data bus. Somewhat oddly, the bus couplers are all grouped in a fairly low number of locations to ease the installation issues.

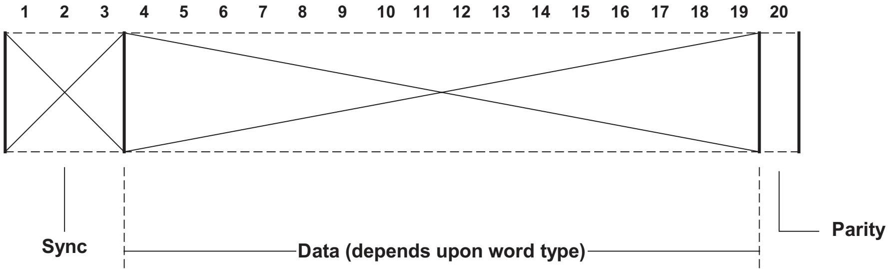
Fig. 2.17 A629 bus coupler interface and encoding format



Also shown within the box in Fig. 2.17 is a simplified portrayal of the Manchester biphase encoding which the A629 data bus (and MIL-STD-1553B) uses. In this protocol a logic 0 is signified when there is a negative to positive change in signal; this change of state occurs mid-way during the particular bit duration. Similarly, logic 1 is denoted when there is a positive to negative change in signal during the bit period. This timing is aided by the fact that the first three bits in a particular data word act as a means of synchronization for the whole of the word. The data are said to be 'self-clocked' on a word by word basis, and therefore these rapid changes in signal state may be accurately and consistently recognized with minimal risk of misreads.

Figure 2.18 shows the typical A629 20 bit data word format which is very similar to

Fig. 2.18 A629 digital word format



A629 Word Formats:

General Format

System Status Word

Function Status Word

Parameter Validity Word

Binary (BNR) Word

Discrete Word

MIL-STD-1553B (see Fig. 2.11). The first three bits are related to word time synchronization, as already described. The next 16 bits are the data contents, and the final bit is a parity bit. The data words may have a variety of formats depending on the word function; there is provision for general formats, systems status, function status, parameter validity, and binary and discrete data words. Therefore, although the data format is simpler than for A429, the system capabilities are more advanced as the data rate is some 20 times faster than the fastest (100 kbit/s) option for A429.

The only aircraft utilizing A629 data buses so far is the Boeing 777. The widespread application of technology such as A629 is important, as more widespread application drives component prices down and makes the technology more cost effective. Certainly that has been the case with A429 and MIL-STD-1553B, with A429 in almost all modern civil aircraft and MIL-STD-1553B in almost all military aircraft around the world. For more detail on A629, see references (**6**)–(**8**).

Data bus examples – integration of aircraft systems

The increasing cost effectiveness offered by system integration using digital data buses and microelectronic processing technologies has led to a rapid migration of the technology into the control of aircraft systems. Three examples have been chosen below to discuss further and highlight how all-embracing this process has become:

- MIL-STD-1553B – Experimental Aircraft Programme (EAP) utilities management system
- A429 – Airbus A330/340 aircraft systems
- A629 – Boeing 777 aircraft systems

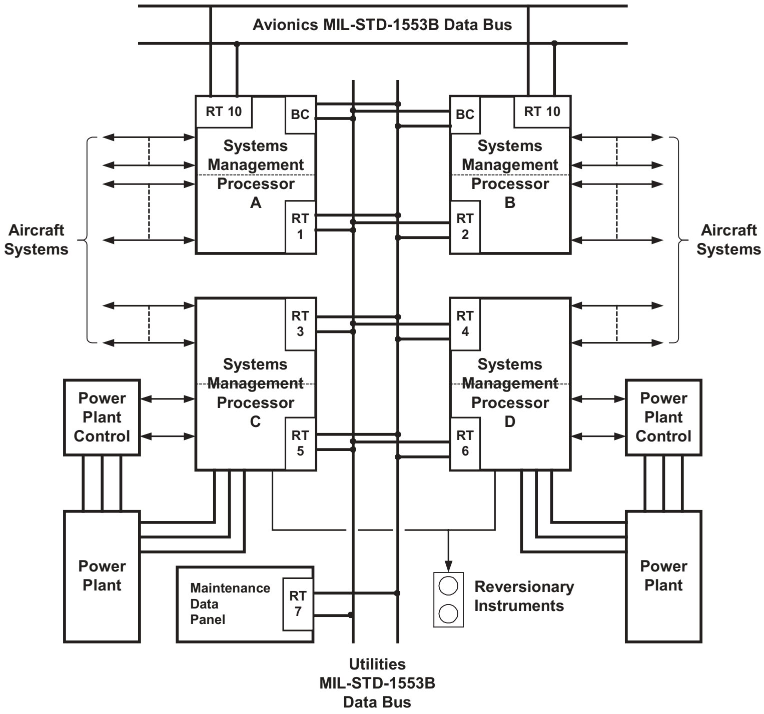
Experimental Aircraft Programme

The first aircraft to utilize MIL-STD-1553B for the integration of aircraft utility systems as opposed to avionics systems was the UK Experimental Aircraft Programme which was a technology-demonstrator forerunner to the Typhoon. This aircraft first flew in August 1986 and was demonstrated at the Farnborough Air Show the same year, also being flown at the Paris Air Show the following year. This system is believed to be the first integrated system of its type, dedicated purely to the integration of aircraft utility systems. The system encompassed the following functions:

- Engine control and indication.
- Fuel management and fuel gauging.
- Hydraulic system control and indication, undercarriage indication and monitoring, wheel brakes.
- Environmental control systems, cabin temperature control, and later an On-Board Oxygen Generating System (OBOGS).
- Secondary power system.
- Liquid Oxygen (LOX) contents, electrical generation and battery monitoring, probe heating, emergency power unit.
- Interfaces with cockpit and avionic systems.

The system comprised four LRUs – Systems Management Processors (SMPs) – which also housed the power switching devices associated with operating motorized valves, solenoid valves, etc. These four units comprised a set of common modules or building blocks, replacing a total of 20–25 dedicated controllers and six power switching relay units that a conventional system would use. The system contained several novel features, offering a level of integration of the utilities system that has not been equalled since (Fig. 2.19).

Fig. 2.19 EAP utilities system architecture



The technology and techniques applied to aircraft utilities systems demonstrated on EAP have since been used successfully on Typhoon and Nimrod MRA4. The lessons learned on each aircraft have been passed on through the generations of utilities management systems, and are now being used on a number of aircraft projects under the heading of Vehicle Management Systems.

Airbus A330/340

The two-engined A330 and four-engined A340 make extensive use of A429 data buses to integrate aircraft utility systems with each other and with the avionics and displays. Table 2.1 lists some of the major subsystems and control units.

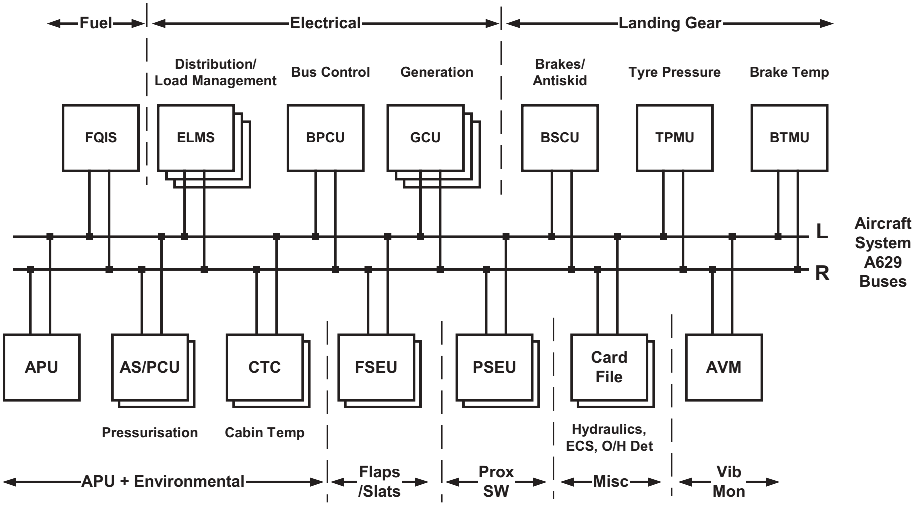
Table 2.1 A330/A340 Typical Aircraft System Controllers

Control unit	A330	A340	Remarks
Bleed air control	2	4	One per engine
Fuel control	2	2	
Landing gear control	2	2	
Flight control:			
Flight control primary computer	2	2	
Flight control secondary computer	2	2	
Flight control data concentrator	2	2	
Slat/flap control computer	2	2	
Probe heat3	3		
Zone controller	1	1	
Window heat control	2	2	
Cabin pressure control	2	2	
Pack controller	2	2	
Avionics ventilation computer	1	1	
Generator control unit	2	4	One per engine
Full-Authority Digital Engine Control (FADEC)	2	4	One per engine
Flight warning computer	2	2	
Central maintenance computer	2	2	
Hydraulic control	1	1	

Boeing 777

The B777 makes extensive use of the A629 digital data bus to integrate the avionics, flight controls, and aircraft systems. Figure 2.20 depicts a simplified version of the B777 aircraft systems that are integrated using A629 buses. Most equipment is connected to the left and right aircraft system buses, but some are also connected to a centre bus. Exceptionally, the engine electronic engine controllers (EECs) are connected to left, right, centre 1, and centre 2 buses to give true dual–dual interface to the engines. The systems so connected embrace the following:

Fig. 2.20 *B777 aircraft systems integration using A629 data buses*



- Fuel Quantity Indication System (FQIS).
- Electrical:
  - Electrical Load Management System (ELMS**)**,
  - Bus Power Control Unit (BPCU),
  - Generator Control Unit (GCU).
- Landing gear:
  - Brakes and Antiskid Control Unit (BSCU),
  - Tyre Pressure Monitoring Unit (TPMU),
  - Brake Temperature Monitoring Unit (BTMU).
- Auxiliary Power Unit (APU) and environmental control:
  - APU controller,
  - Air System and Pressurization Control Unit (AS/PCU),
  - Cabin Temperature Control (CTC).
- Flap/slats Electronics Unit (FSEU).
- Proximity Switch Electronics Unit (PSEU).
- Card files: Boeing produced modules used for the management of hydraulics, overheat detection, and environmental and other functions (see Chapter 11).
- Airframe Vibration Monitor (AVM).

Regional aircraft/business jets

The previous examples relate to fighter and civil transport aircraft. The development of regional aircraft and business jet integrated avionics systems is rapidly expanding to include the aircraft utilities functions. The Honeywell EPIC system being developed for the Hawker Horizon and Embraer ERJ-170/190 is an example of how higher levels of system integration are being achieved (see Fig. 2.21).

This is a much more closed architecture than the ones already described, which utilize open, internationally agreed standards. This architecture uses a proprietary Avionics Standard Communication Bus (ASCB) – a variant D data bus developed exclusively by Honeywell, originally for GA applications. Previous users of ASCB

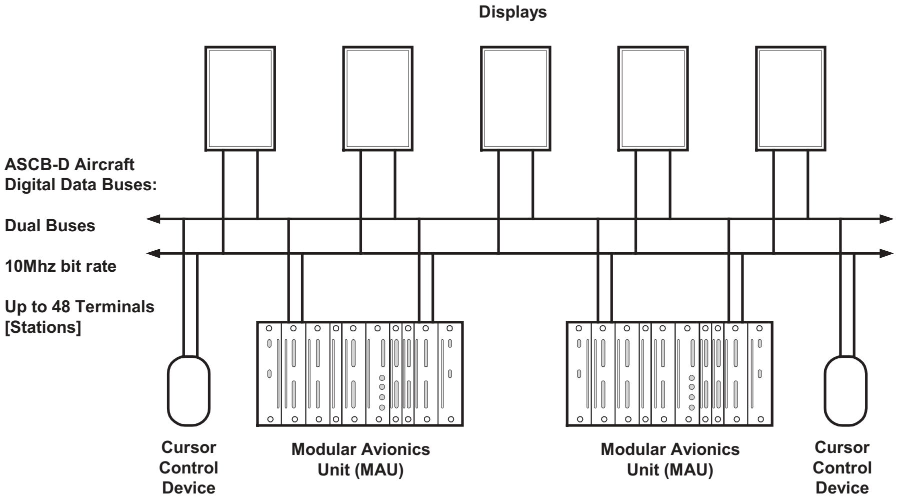


Fig. 2.21 *Honeywell EPIC system – typical*

have been Cessna Citation, Dassault Falcon 900, DeHavilland Dash 8, and Gulfstream GIV. The Honeywell EPIC system is described in Chapter 11.

The key characteristics of ASCB-D are:

- Dual data bus architecture.
- 10 MHz bit rate – effectively 100 times faster than the fastest A429 rate (100 kbit/s).
- Up to 48 terminals may be supported.
- The architecture has been certified for flight-critical applications.

This example shows two Modular Avionics Units (MAUs), but it is more typical to use four such units to host all the avionics and utilities functions. It can be seen from this example that the ambitions of Honeywell in wishing to maximize the return on their EPIC investment is driving the levels of system integration in the regional aircraft and business jets to higher levels than the major Original Equipment Manufacturers (OEMs) such as Boeing and Airbus. A publication that addresses ASCB and compares it with other data buses is reference (**9**). This reference contains much useful data regarding the certification of data bus systems. For an example of a fuel system integrated into the EPIC system, see reference (**10**), which describes the integration of the Hawker Horizon fuel system into the Honeywell EPIC system.

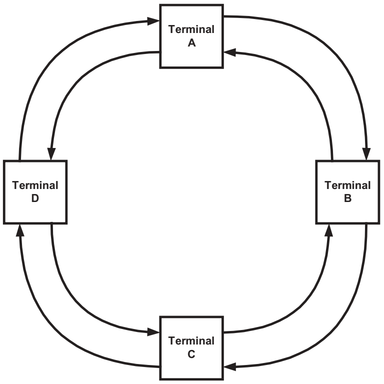
Fibre-optic buses

The examples described so far relate to electrically signalled (conducting) data buses. Fibre-optic interconnections offer an alternative to the electrically signalled bus that is much faster and more robust in terms of electromagnetic interference (EMI). Fibre-optic techniques are widely used in the telecommunications industry, and those used in cable networks serving domestic applications may typically operate at around 50–100 MHz.

A major problem with fibre-optic communication is that it is unidirectional (half-duplex). That is, the signal may only pass in one direction, and if bidirectional (full duplex) communication is required then two fibres are needed. There is also no ‘T-junction’ in fibre optics, and communication networks have to be formed by

'Y-junctions' or ring topologies. An example of the ring topology is shown in Fig. 2.22, in which the bidirectional interconnection between four terminals requires a total of eight unidirectional fibres. This network does have the property that inter-unit communication is maintained should any terminal or fibre fail. Fibre optics also allow the use of wavelength division multiplexing, which allows many channels of data transfer including full duplex (bidirectional) transfers if required.

Fig. 2.22 Fibre-optic ring topology



Fibre- Optic Ring Topology:

- **Each Link transmits Uni-directional Data**
- **Ring Topology allows Data to be passed from Terminal to Terminal**
- **Dual Ring Topology allows Data top be passed following a Terminal Failure**
- **Data transfer @ 1.25 MBits/sec**

This particular topology is similar to that adopted by the Raytheon Control-By-Light™ (CBL™) system which has been demonstrated in flight, controlling the engine and thrust reversers of a Raytheon business jet. In this application the data rate is a modest 1.25 Mbit/s, which is no real improvement over conventional buses such as MIL-STD-1553B and indeed is slower than A629. A fibre-optic bus does have the capability of operating at much higher data rates. It appears that the data rate in this case may have been limited by the protocol (control philosophy) which is an adaptation of a US PC/industrial Local Area Network (LAN) protocol widely used in the United States.

Fibre-optic standards have been agreed and utilized on a small scale within the avionics community, usually for On-board Maintenance System (OMS) applications. The Boeing 777 uses a 10 Mbit/s FDDI, as described in Chapter 10.

Avionics packaging – Line Replaceable Units

Line Replaceable Units (LRUs) were developed as a way of removing functional elements from an avionics system with minimum disruption. LRUs have logical functional boundaries associated with the task they perform in the aircraft. LRU formats were standardized to the following standards:

1. Air Transport Radio (ATR). The origins of ATR standardization may be traced back to the 1930s when United Airlines and ARINC established a standard racking system called the air transport radio (ATR) unit case. ARINC 11 identified three sizes defined by box width: ½ ATR, 1 ATR, 1½ ATR, with the same height and length. In a similar time-scale, standard connector and pin sizes were specified for the wiring connections at the rear of the unit. The use of standardized form boxes

led to the use of standard card sizes, connector types, mounting tray assemblies, and internal layout design. The US military and the military authorities in the United Kingdom adopted these standards although to differing degrees, and they are still in use in military parlance today. Over the period of usage, ATR 'short' ~ 12.5" length and ATR 'long' ~ 19.5" length have also been derived. ARINC 404A developed the standard to the point where connector and cooling duct positioning were specified to give true interchangeability between units from different suppliers. The relatively dense packaging of modern electronics means the ATR 'long' boxes are seldom used. The range of module sizes increased to include half-height boxes known as dwarf and elfin.

2. Modular Concept Unit (MCU). The civil airline community took the standardization argument further by developing the MCU. An 8 MCU box is virtually equivalent to 1 ATR width, and boxes are sized in MCU units. A typical small aircraft systems control unit today might be 2 MCU, while a larger avionics unit such as an Air Data and Inertial Reference System (ADIRS), combining the Inertial Reference System (IRS) with the air data computer function, may be 8 or 10 MCU (1 MCU is roughly equivalent to ~ $1^{1}/_{4}$", but the true method of sizing an MCU unit is given in Fig. 2.23). An 8 MCU box will therefore be 7.64" high × 12.76" deep × 10.37" wide. The adoption of this concept was in conjunction with the most recent ARINC 600 standard, which specifies connectors, cooling air inlets, etc., in the same way as ARINC 404A did earlier.

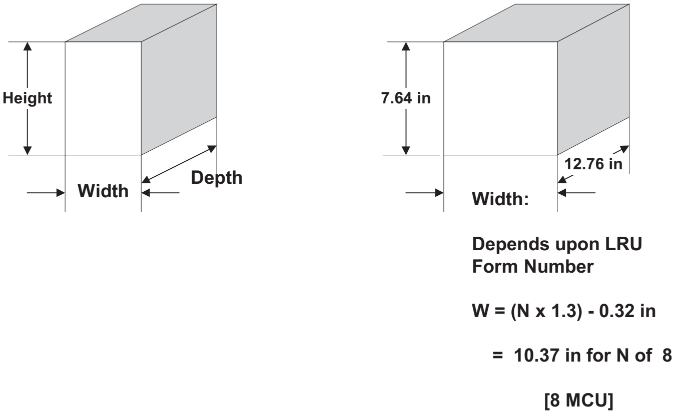
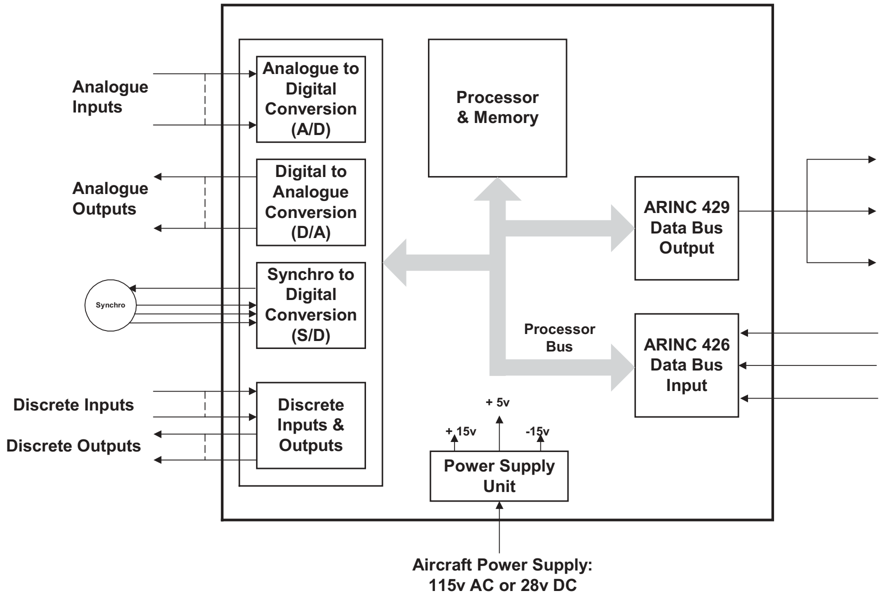


Fig. 2.23 *MCU sizing*

Typical LRU architecture

The architecture of a typical avionics LRU is shown in Fig. 2.24. This shows the usual interfaces and component elements. The unit is powered by a Power Supply Unit (PSU) which converts either 115 VAC or 28 VDC aircraft electrical power to low-voltage DC levels (+5 V and ± 15 V are typical) for the predominant microelectronic devices. In some cases where commercially driven technology is used, +3.3 V may also be required. The processor/memory module communicates with the various I/O modules via the processor bus. The 'real world' to the left of the LRU interfaces with the processor bus via a variety of I/O devices which convert true analogue values to/from a digital format. The right portion of the LRU interfaces with other LRUs by means of digital data buses; in this example, A429 is shown, and it is certainly the most common data bus in use in civil avionics systems today.

Fig. 2.24 *Typical LRU architecture*



Environmental conditions

One of the shortcomings exhibited by microelectronics devices is their susceptibility to external voltage surges and static electricity. Extreme care must be taken when handling the devices outside the LRU, as the release of static electricity can irrevocably damage the devices. The environment that the modern avionics LRU has to withstand and be tested to withstand is onerous, as will be seen later.

The environmental and EMI challenges faced by the LRU in the aircraft can be quite severe, typically including the following:

- Electromagnetic interference
  - EMI – produced by sources external to the aircraft; surveillance radars, high-power radio stations, and communications. This is also known as High-Intensity Radio Frequency (HIRF) in the civil aerospace community;
  - internal EMI – interference between on-board avionics equipment or by passenger-carried laptops, games machines, or mobile phones;
  - lightning effects.

  MIL-STD-461 and 462 are useful military references.

- Physical effects due to one or more of the following:
  - Vibration – sinusoidal or random in three orthogonal axes,
  - temperature,
  - altitude,
  - temperature and altitude,
  - temperature, altitude, and humidity,
  - salt fog,
  - dust,

– sand,
– fungi,
– shock,
– decompression,
– contaminants – fuel, oils, etc.

Figure 2.25 illustrates the construction of a typical LRU; most avionics suppliers within the industry adopt this approach or similar techniques to meet the EMI requirements being mandated today. The EMI sensitive electronics is located in an enclosure on the left which effectively forms a screened enclosure 'Faraday cage'. This enclosed EMI 'clean' area is shielded from EMI effects such that the sensitive microelectronics can operate in a protected environment. All signals entering this area are filtered to remove voltage spikes and surges. To the right of the EMI boundary are the EMI filters and other 'dirty' components such as the PSU. These components are more robust than the sensitive electronics and can successfully operate in this environment. Finally, in most cases the external wiring will be shielded and grounded to screen the wiring from external surges or interference induced by lightning and, more recently and perhaps more certainly, from emissions from passengers' laptop computers, mobile telephones, and hand-held computer games.

Standards exist for determining the EMI performance of on-board equipment and include tests for radiated susceptibility and conducted susceptibility, as well as testing the units for their ability to interfere with other on-board equipment.

A typical test plan for modern avionics units will include some of the above tests as part of the LRU/system production test, as opposed to the aircraft certification process. Additionally, production units may be required to undergo an Environmental Stress Screening (ESS), a stress screening process carried out during production testing. This typically includes 50 h of testing involving temperature cycling and/or vibration testing to detect 'infant mortality' prior to units entering full-time service.

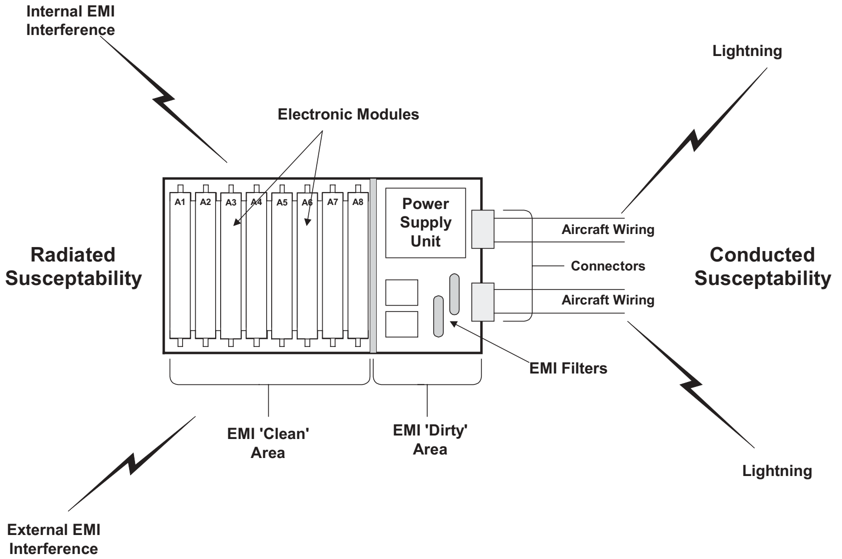


Fig. 2.25 *LRU EMI hazards*

Integrated Modular Avionics

Integrated Modular Avionics (IMA) is a new avionics architecture and packaging technique that could move electronic packaging beyond the ARINC 600 era. ARINC 600, as described earlier, relates to the specification of recent transport aircraft LRUs, and this is the packaging technique used by many aircraft flying today. However, a move towards a more integrated solution is being sought as the avionics technology increasingly becomes smaller and the benefits to be attained by greater integration become very attractive. Therefore, the advent of IMA introduces an integrated cabinet approach where the conventional ARINC 600 LRUs are replaced by fewer units.

The IMA concept is shown in Fig. 2.26. The diagram depicts how the functionality of ten ARINC 600 LRUs (LRUs A–J) may instead be installed in an integrated rack or cabinet as ten Line Replaceable Modules (LRMs) (LRMs A–J). In fact the integration process is likely to be more aggressive than this, specifying common modules and allowing multiple processing tasks within common processor modules. Therefore, the IMA approach is more than merely mapping old LRU functionality onto new smaller modules/LRMs. It is intended to map multiple functions onto one or more LRMs and to allow system reconfiguration to enable spare processing capacity to be used should one processor/LRM fail. An important aspect of this architecture is a well-designed standard back plane bus that is robust and expandable (see the comments on open architectures in Chapter 11).

As Fig. 2.26 suggests, there are a number of obvious potential advantages to be realized by this integration:

- Volume and weight savings.
- Sharing of resources, such as power supplies, across a number of functional modules.
- Standard module designs yielding a more unified approach to equipment design.
- LRMs are more reliable than LRUs.

These advantages must be weighed against the disadvantages:

- Possibly more expensive overall to procure.
- Possibly more difficult to certificate and more risky owing to additional certification requirements (not adopting or modifying an existing unit).
- May pose proprietary problems by having differing vendors working more closely together.
- Segregation considerations (more eggs in one basket).
- Will an ‘open’ or ‘closed’ architecture prevail?
- What standards will apply – given the fact that a lot of effort has been invested in ARINC 600?
- Who takes responsibility for systems integration?

Clearly, there are some difficult issues to be answered. Also, critics might ask: as the technology is becoming more reliable anyway, is the reliability increase due to the concept or the technology? Nevertheless, this approach is gaining credence in both military and civil fields, and the EPIC system described earlier has adopted this approach.

The issues associated with IMA systems are examined more fully in Chapter 11.

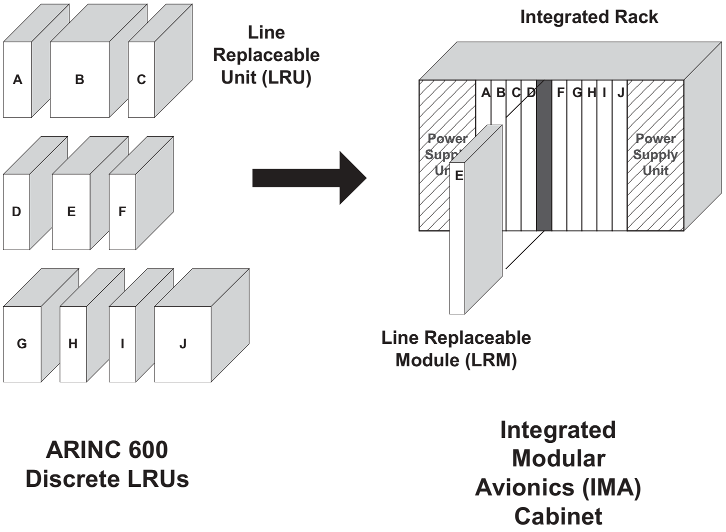


Fig. 2.26 *LRU and integrated modular cabinet comparison*

Software

Many avionic functions are supported by, or entirely provided by, software implementations. The uncertainty of exhaustive testing of software to ensure that there are no latent design errors has largely been overcome by a rigorous and systematic approach to design. The application of standards such as RTCA DO-178B (**11**), together with the application of sound systems engineering practices, has led to the certification of software for safety-critical systems such as propulsion control and flight control systems.

Levels of criticality of software are defined by the development processes described in Chapter 3. Generally, software levels will be defined according to the effect of a software failure on the aircraft (Table 2.2).

High-Order Languages (HOLs) have become widely accepted, and autocode generation techniques coupled with validated compilers are used to produce high-quality software with higher productivity. The language becoming widely accepted for the generation of commercial avionic software is C or C++.

The Capability Maturity Model (CMM) process is encouraging systems designers to set in place an organizational structure and a culture that support good software design. The process has been set up to enable organizations to advance through levels of demonstrated process and capability improvement (CMM levels 1–5) to ensure credibility in the overall software development process from prime contractor down to module suppliers.

Table 2.2 Software levels according to the effect of software failure on the aircraft

Level	Contribution to resultant failure condition for the aircraft
A	Catastrophic
B	Hazardous
C	Major
D	Minor
E	No effect

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CHAPTER 3

Systems Development

As the reader will judge from the contents of this book, aircraft systems are becoming more complex and more sophisticated for a number of technology and performance reasons. In addition, avionics technology, while bringing the benefits of improved control by using digital computing and greatly increased integration by the adoption of digital data buses, is also bringing greater levels of complexity to the development process. The disciplines of avionics system development, including hardware and software integration, are now being applied to virtually every aircraft system.

The increasing level of system sophistication and the increased interrelation of systems are also making the development process more difficult. The ability to capture all of the system requirements and interdependences between systems has to be established at an early stage in the programme. Safety and integrity analyses have to be undertaken to ensure that the system meets the necessary safety goals, and a variety of other trades studies and analytical activities have to be carried out.

These increasing strictures need to be met by following a set of rules, and this chapter gives a brief overview of the regulations, development processes, and analyses that are employed in the development of modern aircraft systems, particularly where avionics technology is also extensively employed.

The design of an aircraft system is subject to many rigours and has to satisfy a multitude of requirements derived from specifications and regulations. There are also many development processes to be embraced. The purpose of this chapter is not to document these *ad nauseam* but to give the reader an appreciation of the depth and breadth of the issues that need to be addressed:

1. Systems design. There are not only references to some of the better known specifications and requirements, but also attempts to act as a tutorial in terms of giving examples of how the various design techniques and methods are applied. As

the complexity of and increasing interrelationship and reliance between aircraft systems has progressed, it has become necessary to provide a framework of documents for the designer of complex aircraft systems.

2. Development processes. An overview of a typical life cycle for an aircraft or equipment is given, and the various activities are described. Furthermore, some of the programme management disciplines are briefly visited.

System design

Key documentation is applied under the auspices of a number of agencies. A list of the major documents that apply are included in the reference section of this chapter, and it is not intended to dwell in great detail on those documents in this brief overview. There are several agencies that provide material in the form of regulations, advisory information, and design guidelines whereby aircraft and system designers may satisfy mandatory requirements.

Key agencies and documentation

These agencies include:

- Society of Automobile Engineers (SAE):
  ARP 4754 (**1**),
  ARP 4761 (**2**).
- Federal Aviation Authority (FAA):
  AC 25.1309-1A (**3**).
- Joint Airworthiness Authority (JAA):
  AMJ 25.1309 (**4**).
- Air Transport Association (ATA):
  ATA-100 (**5**).
- Radio Technical Committee Association (RTCA):
  RTCA DO-160C,
  RTCA DO-178B (**6**),
  RTCA DO-254 (**7**).

This list should not be regarded as exhaustive but merely indicative of the range of documentation that exists. In addition, it is worth mentioning that for military aircraft both US MIL standards and UK Def standards apply.

Design guidelines and certification techniques

References (**1**) and (**2**) offer a useful starting point in understanding the interrelationships of the design and development process:

- Reference (**1**) – ARP 4754: System Development Processes.
- Reference (**2**) – ARP 4761: Safety Assessment Process Guidelines and Methods.
- Def Stan 00-970 for military aircraft.

Figure 3.1 shows the interplay between the major techniques and processes associated with the design and development process. This figure, which is presented as part of the SAE ARP 4754 document, gives an overview of the interplay between some of the major references/working documents that apply to the design and development process.

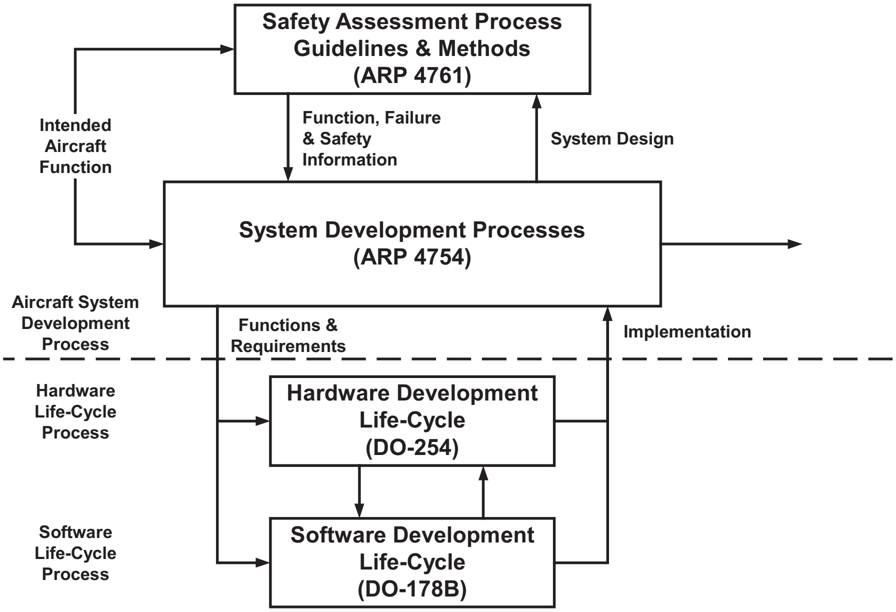


Fig. 3.1 *SAE ARP 4754 system development process*

In summary:

- SAE ARP 4754 is a set of development processes.
- SAE ARP 4761 represents a set of tools and techniques.
- RTCA DO-178B offers advice for the design and certification of software.
- RTCA DO-254 offers guidance for hardware design and development.

Serious students or potential users of this process are advised to procure an updated set of these documents from the appropriate authorities. The main subject headings are summarized below:

Certification considerations for complex aircraft systems – SAE ARP 4754

Key areas are:

- System development.
- Certification process and coordination.
- Safety assessment process.
- Validation of requirements.
- Implementation verification.
- Configuration management.
- Process assurance.
- Modified aircraft.
- Requirements determination and assignment of development level assurance.

Guidelines and methods for conducting the safety assessment – SAE ARP 4761

Major elements include:

- Functional Hazard Analysis (FHA).
- Preliminary System Safety Analysis (PSSA).
- System Safety Analysis (SSA).
- Fault Tree Analysis (FTA).
- Dependency Diagrams.
- Markov Analysis (MA).
- Failure Modes and Effects Analysis (FMEA).
- Failure Modes and Effects Summary (FMES).
- Zonal Safety Analysis (ZSA).
- Particular Risks Analysis (PRA).
- Common Mode Analysis (CMA).
- Contiguous safety assessment process example.

Software considerations – RTCA DO-178B

Major development areas include:

- Introduction.
- System software development.
- Software life cycle.
- Software planning process.
- Software development process.
- Software verification process.
- Software configuration management process.
- Software quality assurance process.
- Certification liaison process.
- Overview of aircraft and engine certification.
- Software life cycle data.
- Additional considerations.

Hardware development – RTCA DO-254

Key points in this recently released documentation map almost directly onto the RTCA DO-178B software equivalent. They are:

- Introduction.
- System aspects of hardware design assurance.
- Hardware design life cycle.
- Hardware planning process.
- Hardware design process.
- Validation and verification.
- Configuration management.
- Process assurance.
- Certification liaison.
- Design life cycle data.
- Additional considerations.

Equivalence of US and European specifications

To the uninitiated, the abundance of US and European development specifications can be most confusing. To alleviate this problem, Table 3.1 establishes equivalence between these specifications.

Table 3.1. Equivalence of US and European specifications

Specification topic	US Specification	European EUROCAE* specification
Systems development process	SAE ARP 4754	ED-79
Safety assessment process guidelines and methods	SAE ARP 4761	
Software design	RTCA DO-178B	ED-12
Hardware design	RTCA DO-254	ED-80
Environmental test	RTCA DO-160	ED-14

Footnote * European Organization for Civil Aviation Equipment.

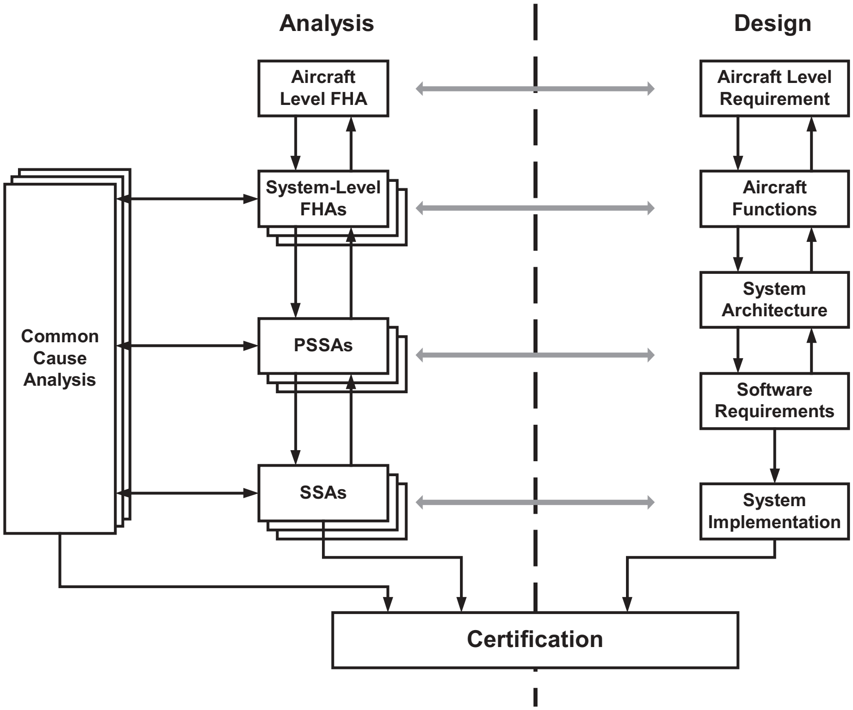
Interrelation of processes

There are a number of interrelated processes that are applied during the safety assessment of an aircraft system. These are:

- Functional Hazard Analysis (FHA).
- Preliminary System Safety Analysis (PSSA).
- System Safety Analysis (SSA).
- Common Cause Analysis (CCA).

Figure 3.2 shows a simplified version of the interplay between these processes as the system design evolves and eventually the system achieves certification. The diagram effectively splits into two sections: design activities on the right and analysis on the left. As the system evolves from aircraft level requirements, aircraft functions are evolved. These lead in turn to system architectures which in turn define software requirements and the eventual system implementation. At corresponding stages of the design, various analyses are conducted to examine the design in the light of the mandated and recommended practices. At every stage, the analyses and the design interact in an evolutionary manner as the design converges upon a solution that is both cost effective and able to be certificated, meeting all the safety requirements.

Fig. 3.2 Simplified portrayal of safety processes



Functional Hazard Analysis (FHA)

An FHA is carried out at both aircraft and system levels; one flows down from the other. The FHA identifies potential system failures and the effects of these failures. Failures are tabulated and classified according to their possible effects, and the safety objectives are assigned according to the criteria briefly listed in Table 3.2.

Table 3.2. Overview of failure classification and safety objectives

Failure condition classification	Development assurance level	Safety objectives	Safety objectives quantitative requirement (probability per flight hour)
Catastrophic	A	Required	$< 1 \times 10^{-9}$
Hazardous/ severe/major	B	May be required	$< 1 \times 10^{-7}$
Major	C	May be required	$< 1 \times 10^{-5}$
Minor	D	Not required	$< 1 \times 10^{-3}$
No safety effect	E	Not required	None

The FHA identifies the data in the first two columns of the table: the failure condition classification and the development assurance level. These allow the safety objectives to be assigned for that particular condition and a quantitative probability requirement to be assigned.

For a failure that is identified as having a catastrophic effect, the highest assurance level A will be assigned. The system designer will be required to implement fail-safe features in his or her design and will have to demonstrate by appropriate analysis that the design is capable of meeting or exceeding a probability of failure of less than 1×10^{-9} per flight hour. In other words, the particular failure should occur less than once per 1 000 000 000 flight hours or once per 1000 million flight hours. This category of failure is assigned to systems such as flight controls, structure, etc., where a failure could lead to the loss of the aircraft and to death or serious injury to crew, passengers, or overflown population. The vast majority of aircraft systems are categorized at much lower levels where little or no safety concerns apply.

A more user-friendly definition quoted in words as used by the Civil Aviation Authority (CAA) may be:

- Catastrophic: less than 1×10^{-9}, extremely improbable.
- Hazardous: between 1×10^{-9} and 1×10^{-7}, extremely remote.
- Major: between 1×10^{-7} and 1×10^{-5}, remote.
- Minor: between 1×10^{-5} and 1×10^{-3}, reasonably probable; greater than 1×10^{-3}, frequent.

Preliminary System Safety Analysis (PSSA)

The PSSA examines the failure conditions established by the FHA(s) and demonstrates how the system design will meet the specified requirements. Various techniques such as FTA, Markov diagrams, etc., may be used to identify how the design counters the effects of various failures and may point towards design strategies that need to be incorporated in the system design to meet the safety requirements. Typical analyses may include the identification of system redundancy requirements – the number of channels, the control strategies that could be employed, and the need for dissimilarity of control, e.g. dissimilar hardware and/or dissimilar software implementation. The PSSA is therefore part of an iterative process that scrutinizes the system design and assists the system designers in ascribing and meeting risk budgets across one or a number of systems. Increasingly, given the high degree of integration and the interrelationship between major aircraft systems, this is likely to be a multisystem, multidisciplinary exercise coordinating the input of many systems specialists.

System Safety Analysis (SSA)

The SSA is a systematic and comprehensive evaluation of the system design using similar techniques to those employed during the PSSA activities. However, whereas the PSSA identifies the requirements, the SSA is intended to verify that the proposed design does in fact meet the specified requirements as identified during the FHA and PSSA analyses conducted previously. As may be seen in Fig. 3.2, the SSA occurs at the point in the design cycle where the system implementation is concluded or finalized and prior to system certification.

Common Cause Analysis (CCA)

The CCA begins concurrently with the system FHA and is interactive with this activity and subsequent PSSA and SSA analyses. The purpose of the CCA is, as the name suggests, to identify common cause or common mode failures in the proposed design and assist in directing the designers towards strategies that will obviate the possibility of such failures. Such common cause failures may include:

- Failure correctly to identify the requirement.
- Failure correctly to specify the system.
- Hardware design errors.
- Component failures.
- Software design and implementation errors.
- Software tool deficiencies.
- Maintenance errors.
- Operational errors.

The CCA is therefore intended to scrutinize a far wider range of issues than the system hardware or software process. Indeed, it is meant to embrace the whole process of developing, certifying, operating, and maintaining the system throughout the life cycle.

Requirements capture

It can be seen from the foregoing that requirements capture is a key activity in identifying and quantifying all the necessary strands of information that contribute to a complete and coherent system design. It is vital that the system requirements are established in a complete, consistent, and clear manner, or significant problems will be experienced during development and into the service life. There are a number of ways in which the requirements capture may be addressed. Two main methods are commonly used:

- Top-down approach.
- Bottom-up approach.

Top-down approach

The top-down approach is shown in Fig. 3.3. This represents a classical way of tackling the requirements capture by decomposing the system requirements into smaller functional modules. These functional modules may be further decomposed into functional submodules. This approach tends to be suited to the decomposition of large software tasks where overall requirements may be flowed down into smaller functional software tasks or modules. This would apply to a task where the hardware boundaries are fairly well understood or inferred from the overall system requirement. An example might be the definition of the requirements for an avionics system such as a Flight Management System (FMS). In such a system, the basic requirement – the need to improve the navigation function – is well understood, and the means by which the various navigation modes are implemented [Inertial Navigation System (INS), Global Positioning System (GPS), Very High Frequency OmniRange (VOR), etc.] are well defined.

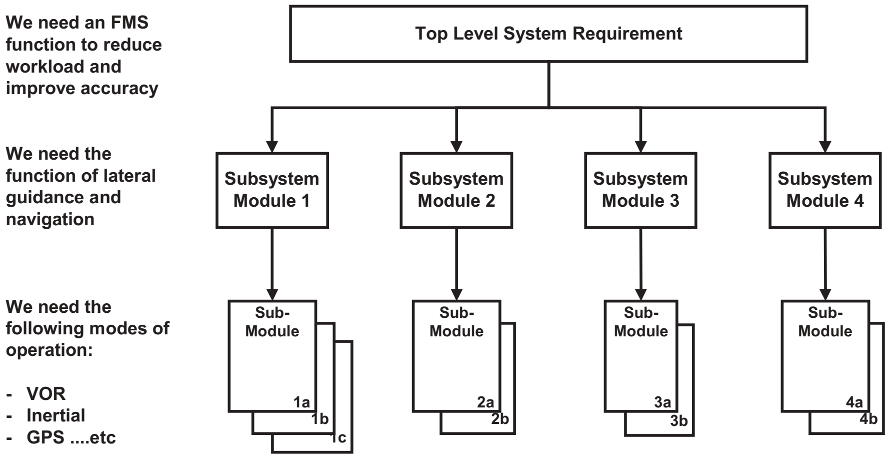


Fig. 3.3 *Top-down approach*

Bottom-up approach

The bottom-up method is shown in Fig. 3.4. The bottom-up approach is best applied to systems where some of the lower level functions may be well understood and documented and represented by a number of submodules. An example of this is adding a new functional element to an established system design. However, the process of integrating these modules into a higher subset presents difficulties as the interaction between the individual subsystems is not fully understood. In this case, building up the top level requirements from the bottom may well enable the requirements to be fully captured. An example of this type might be the integration of aircraft systems into an integrated utilities management system. In this case the individual requirements of the fuel system, hydraulic system, environmental control system, etc., may be well

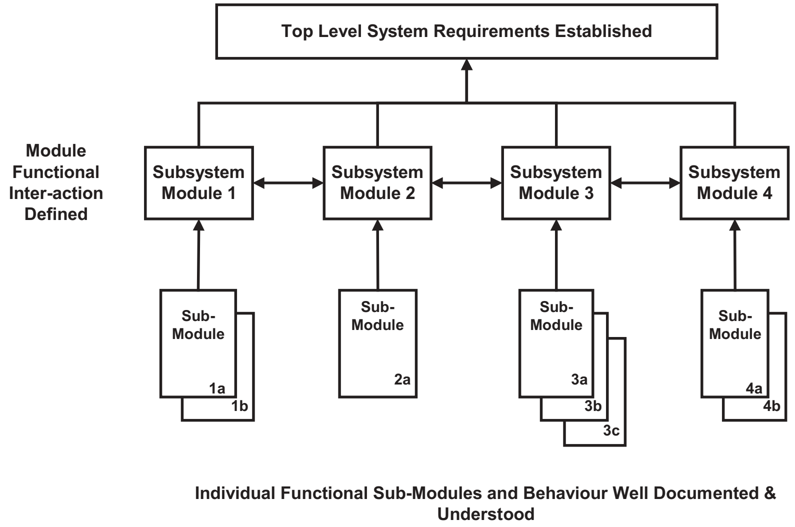


Fig. 3.4 *Bottom-up approach*

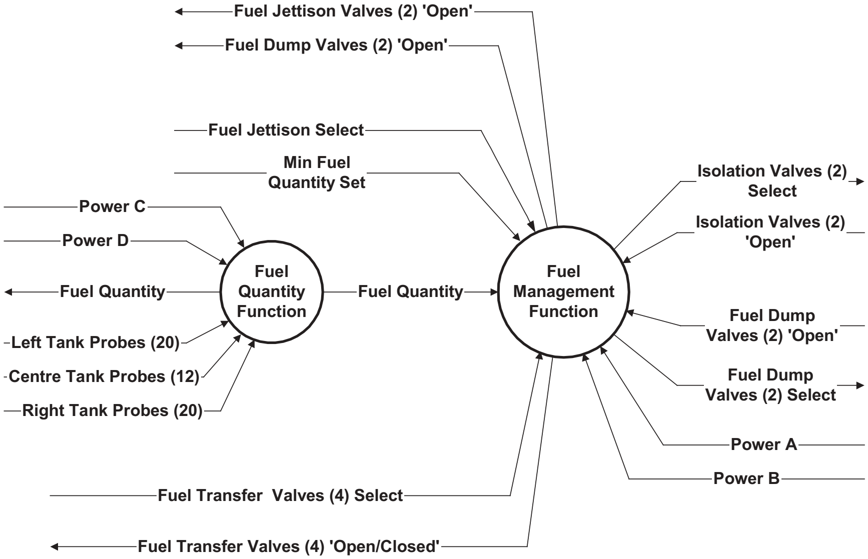
understood. However, the interrelationships between the candidate systems and the implications of adopting integration may be better understood and documented by working bottom up.

In fact, most development projects may use a combination of these two approaches to best capture the requirements.

Requirements capture example

The example given in Fig. 3.5 shows a functional mapping process that identifies the elements or threads necessary to implement a fuel jettison function. Two main functional subsystems are involved: the fuel quantity measurement function and the fuel management function. Note that this technique merely identifies the data threads that are necessary to perform the system function. No attempt is made at this stage to ascribe particular functions to particular hardware or software entities. Neither is any attempt made to determine whether signals are hardwired or whether they may be transmitted as multiplexed data as part of an aircraft system data bus network.

Fig. 3.5 *System requirements capture example*



The system requirements from the flight crew perspective are:

- The flight crew need to jettison excess fuel in an emergency situation in order that the aircraft may land under the maximum landing weight.
- The flight crew wish to be able to jettison down to a preselected fuel quantity.
- The crew wish to be given indications that fuel jettison is under way.

The information threads associated with the flight crew requirements are shown in the upper-centre portion of the diagram. It may be seen that, although the system requirements are relatively simple when stated from the flight crew viewpoint, many other subsystem information strands have to be considered to achieve a cogent system design:

1. Fuel quantity function. The fuel quantity function measures the aircraft fuel quantity by sensing fuel in the aircraft fuel tanks; in the example given in Fig. 3.5, a total of 52 probes are required to sense the fuel held in three tanks. The fuel quantity calculations measure the amount of fuel that the aircraft has on-board, taking account of fuel density and temperature. It is usual in this system, as in many others, to have dual power supply inputs to the fuel quantity function to assure availability in the event of an aircraft electrical system busbar failure. Finally, when the calculations have been completed, they are passed to the flight deck where the aircraft fuel quantity is available for display to the flight crew. Fuel quantity is also relayed to the fuel management function so that, in the event of fuel jettison, the amount of fuel on-board may be compared with the preset jettison value. The fuel quantity function interfaces to:
   - The fuel quantity system measurement probes and sensors.
   - The flight deck multifunction displays.
   - The fuel management system.
   - The aircraft electrical system.
2. Fuel management function: The fuel management function accepts information regarding the aircraft fuel state from the fuel quantity function. The flight crew inputs a ‘fuel jettison select’ command and the minimum fuel quantity that the crew wishes to have available at the end of fuel jettison. The fuel management function accepts flight crew commands for the fuel transfer valves (4), fuel dump (jettison) valves (2), and fuel isolation valves (2). It also provides ‘open’/‘closed’ status information on the fuel system valves to the flight crew. As before, two separate power inputs are received from the aircraft electrical system. The fuel management function interfaces with:
   - The fuel system valves.
   - The flight deck multifunction displays and overhead panel.
   - The fuel quantity function.
   - The aircraft electrical system.

This example shows how a relatively simple function interfaces to various aircraft systems and underlines some of the difficulties that exist in correctly capturing system requirements in a modern integrated aircraft system. This simple example illustrates how this system, like most systems, can operate in different configurations or ‘modes’ of operation, and that the desired behaviour of a system needs to be specified for all eventualities.

Fault Tree Analysis

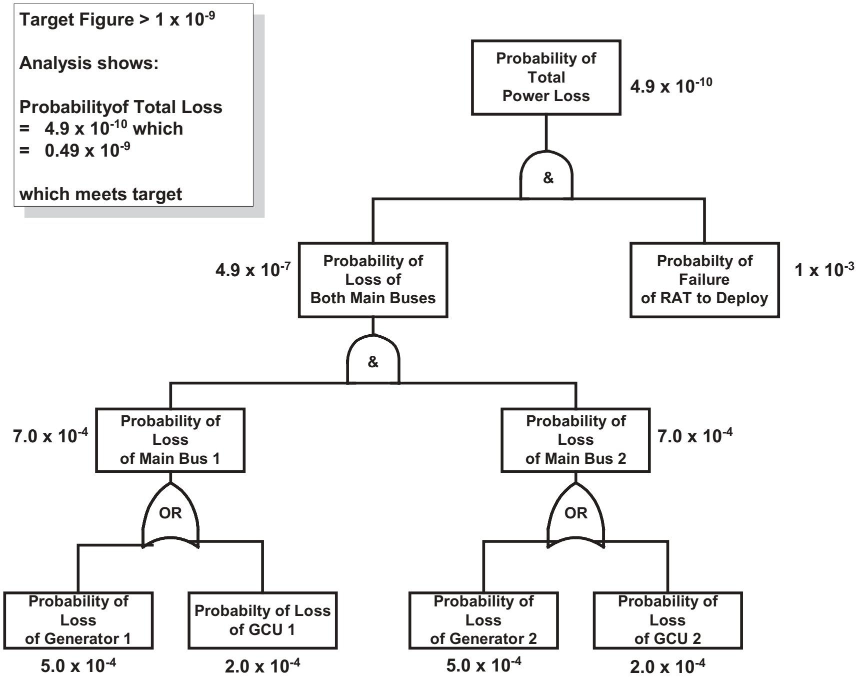
Fault Tree Analysis (FTA) is one of the tools described in SAE document ARP 4761 (**2**). This analysis technique uses probability to assess whether a particular system configuration or architecture will meet the mandated requirements. For example, assume that the total loss of aircraft electrical power on-board an aircraft has catastrophic failure consequences as identified by the functional hazard analysis (Fig. 3.2 and Table 3.2). Then the safety objective quantitative requirement established by FAR/JAR 25.1309 and as amplified in ARP 4754 will be such that this event cannot occur with a probability greater than 1×10^{-9} per flight hour (or once per 1000 million flight hours). The ability of a system design to meet these requirements is established by an FTA using the following probability techniques.

In the example it is assumed:

- That the aircraft has two independent electrical power generation systems, the main components of which are the generator and the Generator Control Unit (GCU) which governs voltage regulation and system protection.
- The aircraft has an independent emergency system such as a Ram Air Turbine (RAT).
- That the failure rates of these components may be established and agreed on account of the availability of in-service component reliability data or sound engineering rationale which will provide a figure acceptable to the certification authorities.

The FTA analysis (very much simplified) for this example is shown in Fig. 3.6.

Fig. 3.6 *Simplified FTA for an aircraft electrical system*



Starting in the bottom left-hand portion of the diagram: the Mean Time Between Failure (MTBF) of a generator is 2000 h – this means that the failure rate of generator 1 is 1/2000 or 5.0×10^{-4} per flight hour. Similarly, if the MTBF of the generator controller GCU 1 is 5000 h, then the failure rate of GCU 1 is 1/5000 or 2.0×10^{-4} per flight hour. The combined failure rate gives the probability of loss of electrical power to main bus 1. This is calculated by summing the failure rates of the generator and controller, as either failing will cause the loss of main bus 1

$5.0 \times 10^{-4} \quad + \quad 2.0 \times 10^{-4} \quad = \quad 7 \times 10^{-4}$ per flight hour
(generator 1) (GCU 1) (main bus 1)

Similarly, assuming that generator channels 1 and 2 are identical, the failure rate of main bus 2 is given by

5.0×10^{-4} + 2.0×10^{-4} = 7×10^{-4} per flight hour
(generator 2) (GCU 2) (main bus 2)

(Note that at this state the experienced aircraft systems designer would be considering the effect of a common cause or common mode failure.)

The probability of the two independent channels failing (assuming no common cause failure) is derived by multiplying the respective failure rates. Thus, the probability of both main buses failing is

$(7 \times 10^{-4}) \times (7 \times 10^{-4}) = 49 \times 10^{-8}$ or 4.9×10^{-7} per flight hour
(main bus 1) (main bus 2)

Therefore, the two independent electrical power channels alone will not meet the requirement. Assuming the addition of the RAT emergency channel as shown in Fig. 3.6, the probability of total loss of electrical power is

$(4.9 \times 10^{-7}) \times (1 \times 10^{-3}) = 4.9 \times 10^{-10}$ per flight hour
(main bus 1 (RAT failure)
and main bus 2)

which meets the requirements.

This very simple example is illustrative of the FTA which is one of the techniques used during the PSSA and SSA processes. However, even this simple example outlines some of the issues and interactions that need to be considered. Real systems are very much more complex, with many more system variables and interlinks between a number of aircraft systems.

Failure Modes and Effects Analysis (FMEA)

The example given is a useful tool for examining total system integrity using a bottom-up approach. Certain parts of a system may be subject to scrutiny as they represent single-point failures and as such warrant more detailed analysis. The analysis used in this situation is the FMEA.

Again, the process used in the FMEA is best illustrated by the use of a simple example – the case of an electrical generator feeding an aircraft main electrical busbar via an electrical power line contactor. The line contactor is operated under the control of the GCU as shown in Fig. 3.7.

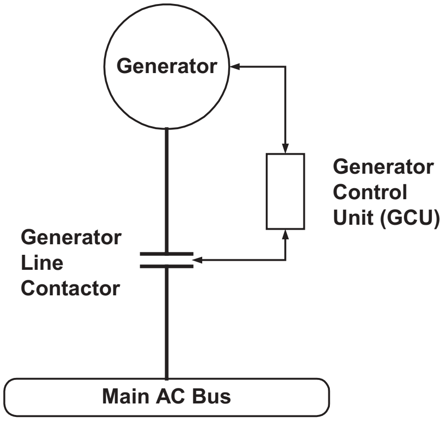


Fig. 3.7 *Main generator, GCU, and power contactor relationship*

An FMEA on this portion of the aircraft electrical system will examine the possible failures of all the elements:

- The generator failures and effects; in other words, examine in detail all the failures that contribute to the generator failure rate of 5×10^{-4} per flight hour, as used in the previous analysis and the effects of those failures.
- The GCU failures and effects: examining all the failures that contributed to the overall failure rate of 2×10^{-4} per flight hour, as used above, and the effects of those failures.
- The line contactor failures and effects. If a line contactor has an MTBF of 100 000 h with a failure rate of 1×10^{-5} per flight hour, the ways in which the contactor may fail are ascribed portions of this failure rate for the different failures and effects:
  - the contactor may fail open,
  - the contactor may fail closed,
  - the contactor may fail with one contact welded shut but the others open
  - and so on until all the failures have been allocated a budget.

This process is conducted in a tabular form such that:

- Failure modes are identified.
- Mode failure rates are ascribed.
- Failure effects are identified.
- The means by which the failure is detected is identified.

An FMEA should therefore respond to the questions asked of the system or element under examination in a quantitative manner.

Component reliability

In the analyses described, a great deal of emphasis is placed upon the failure rate of a component or element within the system under review. This clearly calls into question how reliability values for different types of component are established. There are two main methods for determining component reliability:

- Analytical by component count.
- Historical by means of accumulated in-service experience.

Analytical methods

MIL-STD-781 was a standard – now superceded by MIL-HDBK-781 – developed by the US military over a period of years to use an analytical bottom-up approach to predicting reliability. This method uses a component count to build up an analysis of the reliability of a unit. This approach has probably best been applied to electronic equipment over the years, as the use of electronic components within a design tends to be replicated within a design and across a family of designs. This method uses type of component, environment, and quality factor as major discriminators in predicting the failure of a particular component, module, and ultimately subsystem. Component failure rates, λ, are extracted from the US military standard and then applied with the appropriate factors to establish the predicted value as shown in the simplified example:

$$\lambda = \pi_Q \times (K_1 \pi_T + K_2 \times \pi_E) \times \pi_L$$

where π_Q is a device quality factor,
π_T is a temperature factor, π_E is an environmental factor, π_L is a maturity factor, and K_1 and K_2 are constants.

There are a number of issues associated with this method:

1. It is only as good as the database of components and the factors used.
2. Experience has generally shown that, if anything, predicted values are generally pessimistic, thereby generating predicted failure rates worse than might be expected in real life.
3. The technique has merit in comparing competing design options in a quantitative manner when using a common baseline for each design – the actual numerical values are less important than the comparison.
4. It is difficult to continue to update the database, particularly with the growing levels of integration with ICs, which makes device failure rates difficult to establish.
5. The increasing number of COTS components also confuses the comparison.
6. The technique is particularly valuable when it can be compared with in-service experience and appropriate correction factors can be applied.

Reference (**8**) is a paper presented at a recent international aerospace conference that gives a very good overview of this technique when applied to power electronics.

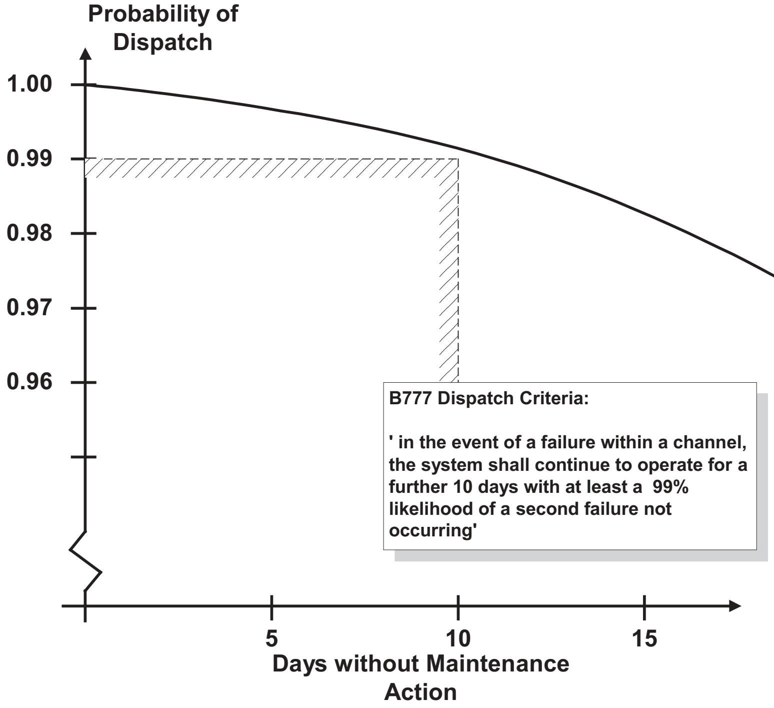
In-service data

The use of in-service data is the best way of approaching the assessment of mechanical components used in the same environment. It does depend upon correspondence between the components that the design is contemplating with the in-service database being used. Any significant variation in component usage, technology baseline, or location in the aircraft/environment may nullify the comparison. Nevertheless, when used in conjunction with other methods, this is a valid method. The manufacturers of civil and fighter aircraft and helicopters and their associated suppliers will generally be able to make 'industry standard' estimates using this technique.

Dispatch reliability

Dispatch availability is key to an aircraft fulfilling its mission, whether a military or civil aircraft, and is also known as dispatch reliability. The ability to be able to continue to dispatch an aircraft with given faults has been given impetus by the commercial pressures of the air transport environment, where multiple redundancy for integrity reasons has also been used to aid aircraft dispatch. On the Boeing 777 the need for high rates of dispatch availability was specified in many systems, and in some systems this led to the adoption of dual redundancy for dispatch availability reasons rather than for reasons of integrity. A simplified version of the dispatch requirements is shown in Fig. 3.8.

This means of specifying the dispatch requirement of part of an aircraft system leads to an operational philosophy far beyond a 'get-you-home' mode of operation. In fact it is the first step towards a philosophy of no unscheduled maintenance. For an aircraft flying up to 14 h per day – a typical utilization for a wide-bodied civil transport – this definition dictates a high level of availability for up to a 120 h flying period. The ability to stretch this period in the future – perhaps to a 500 h operating period – as more reliable systems become available could lead to a true system of unscheduled maintenance. A 500 h operating period roughly equates to 7 weeks of flying, at which



B777 Dispatch Criteria:

' in the event of a failure within a channel, the system shall continue to operate for a further 10 days with at least a 99% likelihood of a second failure not occurring'

Fig. 3.8 *Simplified dispatch criteria*

time the aircraft will probably be entering the hangar for other specified maintenance checks.

This leads to a more subtle requirement to examine the ability of the system to meet integrity requirements when several failures have already occurred, and this requires different techniques.

Markov Analysis

Another technique used to assist in system analysis is Markov Analysis (MA). This approach is useful when investigating systems where a number of states may be valid and are also interrelated. This could be the case in a multichannel system where certain failures may be tolerated but not in conjunction with some failure conditions. The question of whether a system is airworthy is not a simple mathematical calculation as in previous analyses but depends upon the relative states of parts of the system. The simple methods used are insufficient in this case, and another approach is required. Markov analysis is the technique to be applied in these circumstances when it is necessary to address the operation of a system in different modes and configurations.

As before, a simple example will be used to illustrate the MA technique, in this case the dual-channel Full Authority Digital Engine Control (FADEC) example outlined in Fig. 3.9

This simplified architecture is typical of many dual-channel FADECs. There are two independent lanes: lane A and lane B. Each lane comprises a command and a monitor portion, which are interconnected for cross-monitoring purposes, and undertakes the task of metering the fuel flow to the engine in accordance with the necessary control laws to satisfy the flight crew thrust command. The analysis required

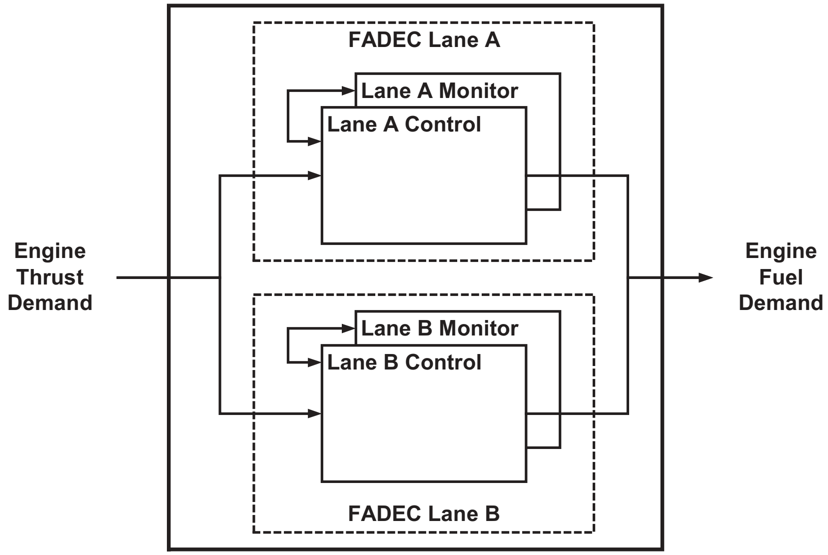


Fig. 3.9 *Simplified FADEC architecture*

to decide upon the impact of certain failures in conjunction with others requires a Markov model in order to be able to understand the dependences.

Figure 3.10 depicts a simple Markov model which equates to this architecture. By using this model, the effects of interrelated failures can be examined. The model has a total of 16 states, as shown by the number in the bottom right-hand corner of the appropriate box. Each box relates to the serviceability state of the lane A command (Ca) and monitor (Ma) channels and lane B command (Cb) and monitor (Mb) channels. These range from the fully serviceable state in box 1 through a series of failure conditions to the totally failed state in box 16. Clearly, most normal operating conditions are going to be in the left-hand region of the model.

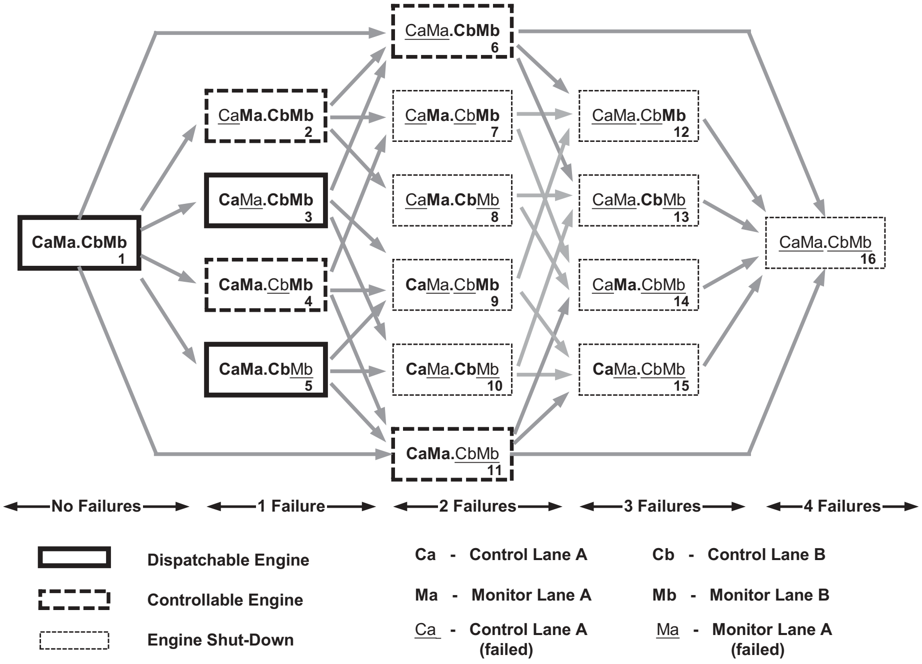
Concentrating on the left-hand side of the model, it can be seen that the fully serviceable state in box 1 can migrate to any one of six states:

- Failure of command channel A results in state 2 being reached.
- Failure of monitor channel A results in state 3 being reached.
- Failure of command channel B results in state 4 being reached.
- Failure of monitor channel B results in state 5 being reached.
- Failure of the cross-monitor between command A and monitor A results in both being lost simultaneously and reaching state 6.
- Failure of the cross-monitor between command B and monitor B results in both being lost simultaneously and reaching state 11.

All of these failure states result in an engine that may still be controlled by the FADEC. However, further failures beyond this point may result in an engine that may not be controllable either because both control channels are inoperative or because the 'good' control and monitor lanes are in opposing channels. The model shown above is constructed according to the following rules: an engine may be dispatched as a 'get-you-home' measure provided that only one monitor channel has failed. This means that states 3 and 5 are dispatchable, but not states 2, 4, 6, or 11, as subsequent failures could result in engine shutdown.

By knowing the failure rates of the command channels, monitor channels, and cross-monitors, quantitative values may be inserted into the model and probabilities assigned to the various states. Summing the probabilities so calculated enables numerical values to be derived.

Fig. 3.10 *Use of Markov analysis to examine engine dispatch reliability*



Development processes

The product life cycle

Figure 3.11 shows a typical aircraft product life cycle from concept through to disposal at the end of the product's useful life. Individual products or equipment may vary from this model, but it is a sufficiently good portrayal to illustrate the role of systems engineering and the equipment life cycle. The major phases of this model are:

- Concept phase.
- Definition phase.
- Design phase.
- Build phase.
- Test phase.
- Operate phase.
- Refurbish or retire.

This model closely approximates to the Downey cycle used by the UK Ministry of Defence for the competitive procurement of defence systems. The model is as equally applicable for systems used in commercial aircraft as it is for military applications. It is used to describe the role of systems engineering in each phase of the product life cycle.

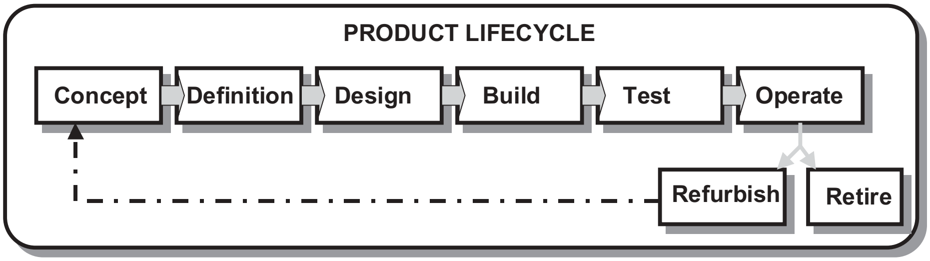


Fig. 3.11 *Typical aircraft product life cycle*

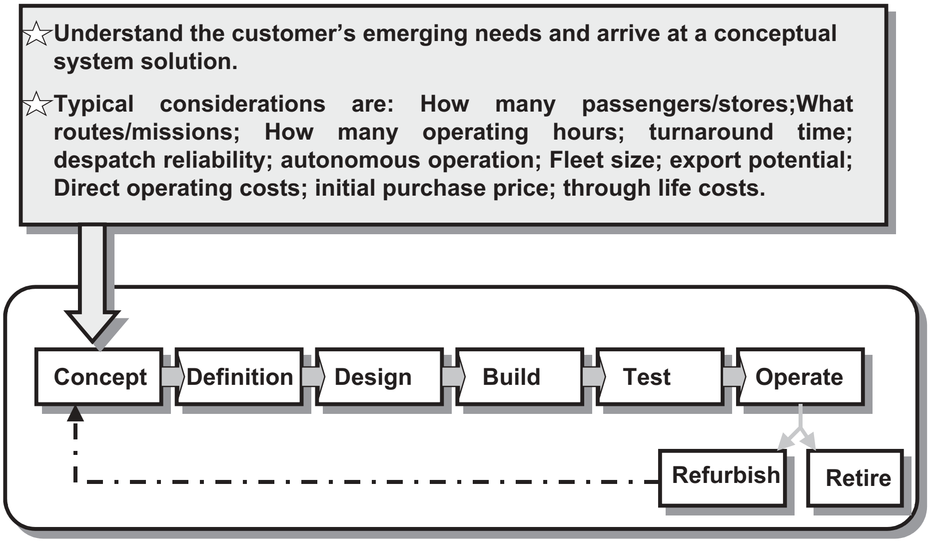
Concept phase

The concept phase is about understanding the customer’s emerging needs and arriving at a conceptual model of a solution to address those needs. The customer continuously assesses his or her current assets and determines their effectiveness to meet future requirements. The need for a new military system can arise from a change in the local or world political scene that requires a change in defence policy. The need for a new commercial system may be driven by changing national and global travel patterns resulting from business or leisure traveller demands.

The customer’s requirement will be made available to industry so that solutions can be developed specifically for that purpose, or adapted from the current Research and Development (R&D) base. This is an ideal opportunity for industry to discuss and understand the requirements to the mutual benefit of customers and their industrial suppliers, and to understand the implications of providing a fully compliant solution or one that is aggressive and sympathetic to marketplace requirements (see Fig. 3.12).

Typical considerations at this phase are:

- Establishing and understanding the primary role and functions of the required system.
- Establishing and understanding desired performance and market drivers such as:
  - range,
  - endurance,



☆ **Understand the customer’s emerging needs and arrive at a conceptual system solution.**

☆ **Typical considerations are: How many passengers/stores;What routes/missions; How many operating hours; turnaround time; despatch reliability; autonomous operation; Fleet size; export potential; Direct operating costs; initial purchase price; through life costs.**

Fig. 3.12 *Concept phase*

  - routes or missions,
  - technology baseline,
  - operational roles,
  - number of passengers,
  - mass, number, and type of weapons,
  - availability and dispatch reliability,
  - fleet size to perform the role or satisfy the routes,
  - purchase budget available,
  - operating or through-life costs,
  - commonality or model range,
  - market size and export potential,
  - customer preference.

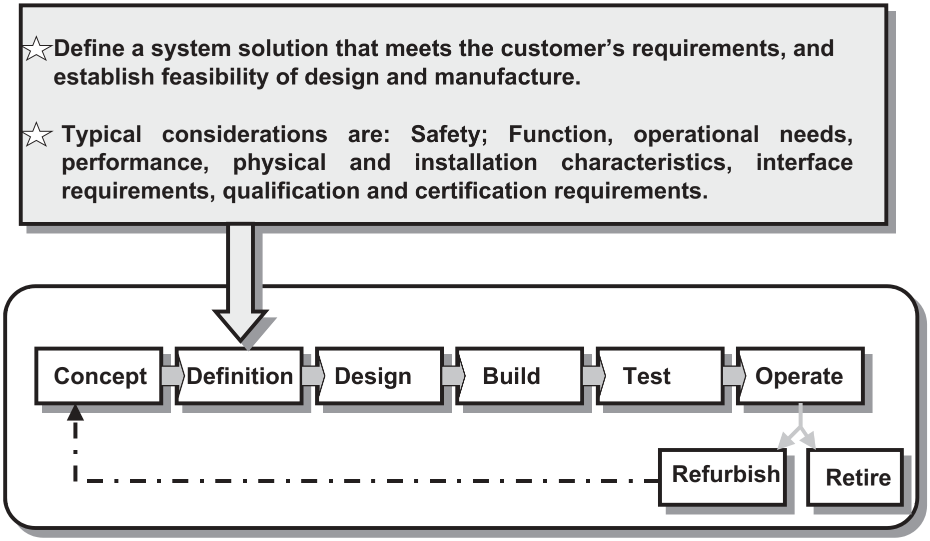
This phase is focused on establishing confidence that the requirement can be met within acceptable commercial or technological risk. The establishment of a baseline of mature technologies may be first solicited by means of a Request For Information (RFI). This process allows possible vendors to establish their technical and other capabilities and represents an opportunity for the platform integrator to assess and quantify the relative strengths of competing vendors and also to capture mature technology of which he or she was previously unaware for the benefit of the programme.

It is in this phase that important decisions are made that determine whether a type is modified or extended (e.g. Boeing 737-400, -500, -600) or whether a new type emerges (e.g. Airbus A380).

Definition phase

See Fig. 3.13. Customers will usually consolidate all the information gathered during the concept phase to firm up their requirements. A common feature used more frequently by platform integrators is to establish engineering joint concept teams to establish the major system requirements. These teams are sometimes called Integrated Product Teams (IPTs). They may develop a cardinal points specification; perhaps even undertake a preliminary system or baseline design against which all vendors might bid. This results in the issue of a specification or a Request For Proposal (RFP). This allows industry to develop their concepts into a firm definition, to evaluate the technical, technological, and commercial risks, and to examine the feasibility of completing the design and moving to a series production solution. Typical considerations at this stage are:

- Developing the concept into a firm definition of a solution.
- Developing system architectures and system configurations.
- Re-evaluating the supplier base to establish what equipment, components, and materials are available or may be needed to support the emerging design.
- Defining physical and installation characteristics and interface requirements.
- Developing operational and initial safety models of the individual systems.
- Quantifying key systems performance such as:
  - Mass,
  - Volume,
  - Growth capability,
  - Range/endurance.



☆Define a system solution that meets the customer's requirements, and establish feasibility of design and manufacture.

☆ Typical considerations are: Safety; Function, operational needs, performance, physical and installation characteristics, interface requirements, qualification and certification requirements.

Fig. 3.13 *Definition phase*

The output from this phase is usually in the form of feasibility study reports, performance estimates, sets of mathematical models of the behaviour of individual systems, and an operational performance model. This may be complemented by breadboard or experimental models, or laboratory or technology demonstrators. Preliminary design is also likely to examine installation issues with mock-ups in three-dimensional computer model form (using tools such as CATIA) which replaces in the main the former wooden and metal models.

Design phase

If the outcome of the definition phase is successful and a decision is made to proceed further, then industry embarks on the design phase within the programme constraints as described later in the chapter. Design takes the definition phase architectures and schemes and refines them to a standard that can be manufactured (see Fig. 3.14).

Detailed design of the airframe ensures that the structure is aerodynamically sound, is of appropriate strength, and is able to carry the crew, passengers, fuel, and systems that are required to turn it into a useful product. As part of the detailed design, cognizance needs to be taken of mandated rules and regulations that apply to the design of an aircraft or airborne equipment. Three-dimensional solid modelling tools are used to produce the design drawings in a format that can be used to drive machine tools to manufacture parts for assembly.

Systems are developed beyond the block diagram architectural drawings into detailed wiring diagrams. Suppliers of bought-in equipment are selected, and they become an inherent part of the process start to design equipment that can be used in the aircraft or systems. Indeed, in order to achieve a fully certifiable design of many of the complex and integrated systems found on aircraft today, an integrated design team comprising platform integrators and supplier(s) is essential. In fact, many of these processes are iterative, extending into and even beyond the build and test phases.

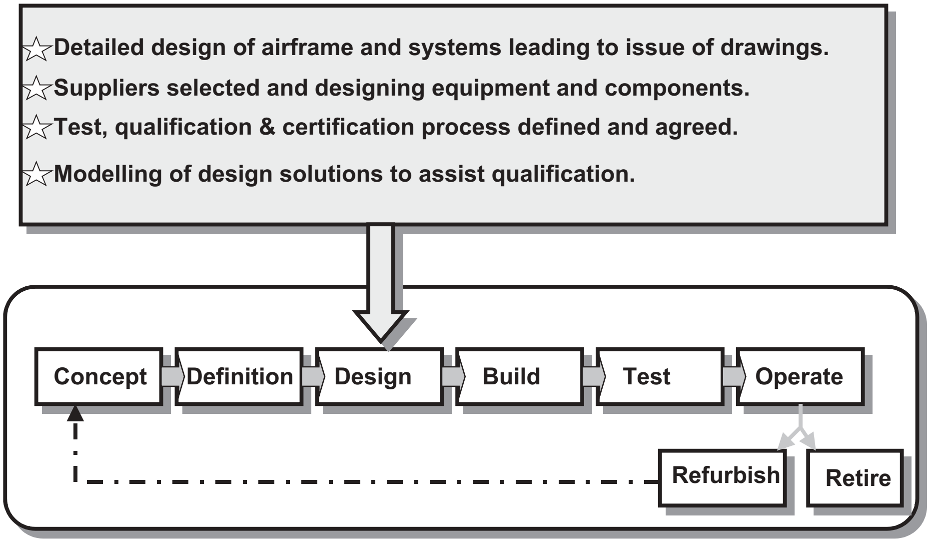


Fig. 3.14 *Design phase*

- ☆ Detailed design of airframe and systems leading to issue of drawings.
- ☆ Suppliers selected and designing equipment and components.
- ☆ Test, qualification & certification process defined and agreed.
- ☆ Modelling of design solutions to assist qualification.

Build phase

The aircraft is manufactured to the drawings and data issued by the design as shown in Fig. 3.15. During the early stages of the programme, a delivery schedule would have been established. Some long lead time items – those which take a long time to build – may need to be ordered well ahead of aircraft build commencing. In the case of some of the more complex, software-driven equipment, design will be overlapping well into the test phase. This is usually accommodated by a phased equipment delivery embracing the following:

- Electrical models – equipment electrically equivalent to the final product but not physically representative.

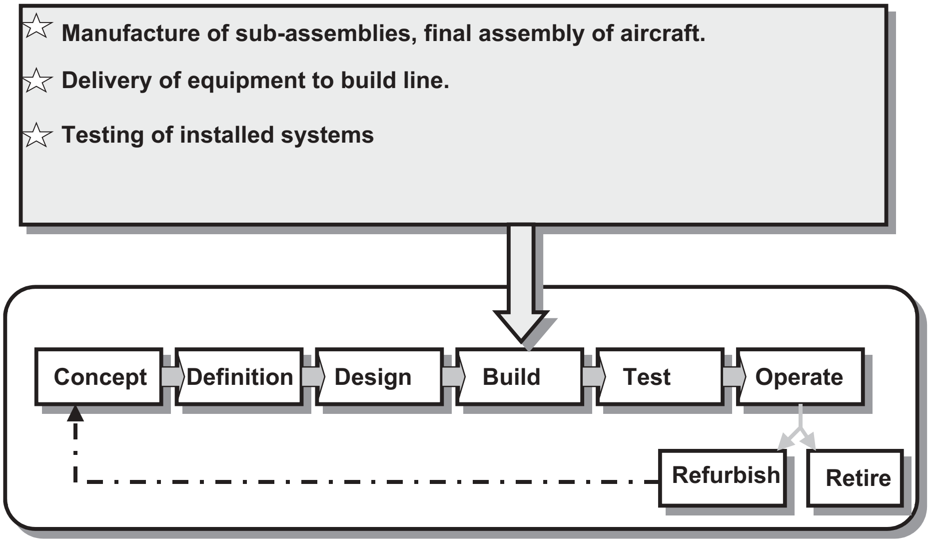


Fig. 3.15 *Build phase*

- ☆ Manufacture of sub-assemblies, final assembly of aircraft.
- ☆ Delivery of equipment to build line.
- ☆ Testing of installed systems

- Red label hardware – equipment that is physically representative but not cleared for flight.
- Black label hardware – equipment that is physically representative and is cleared for flight by virtue of the flightworthy testing carried out and/or the software load incorporated.

These standards are usually accompanied with a staged software release that enables the software load progressively to become more representative of the final functionality.

Test phase

The aircraft and its components are subject to a rigorous test programme to verify their fitness for purpose, as shown in Fig. 3.16. This phase includes testing and integration of equipment, components, subassemblies, and, eventually, the complete aircraft. Functional testing of equipment and systems on the ground and flight trials verify that the performance and operation of the equipment is as specified. Conclusion of the test programme and the associated design, analysis, and documentation process lead to certification of the aircraft or equipment.

In the event of a new aircraft, responsibility for the certification of the aircraft lies with the aircraft manufacturer. However, where equipment is to be improved or modified in the civil arena, equipment suppliers or other agencies can certify the equipment by means of the Supplementary Type Certificate (STC) in a process defined by the certification authorities. This permits discrete equipment, for example, more accurate fuel quantity gauging in a particular aircraft model, to be changed without affecting other equipment.

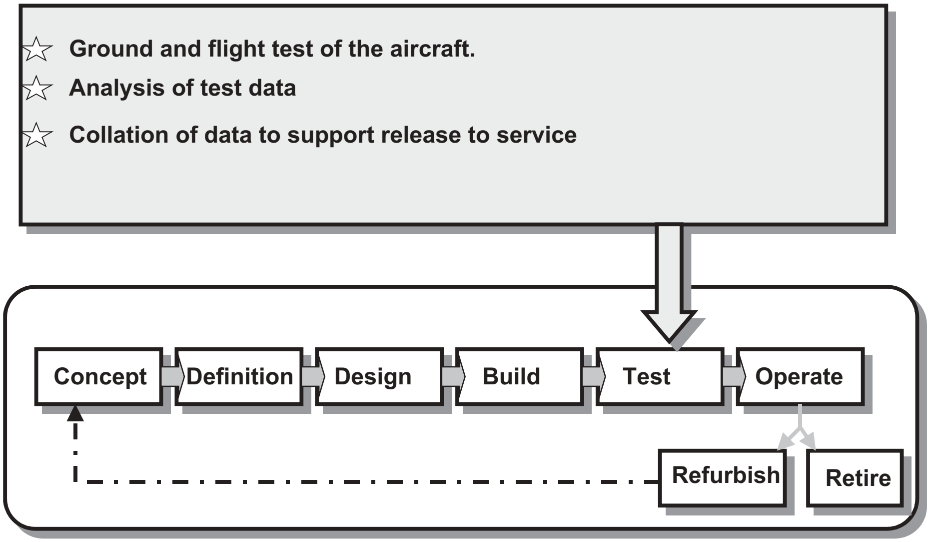


Fig. 3.16 *Test phase*

Operate phase

During the operate phase (Fig. 3.17) the customer is operating the aircraft on a routine basis. Its performance will be monitored by means of a formal defect reporting process, so that any defects or faults that arise are analysed by the manufacturer. It is possible to attribute causes to faults such as random component failures, operator mishandling, or design errors. The aircraft manufacturers and their suppliers are expected to participate in the attribution and rectification of problems arising during aircraft operations, as determined by the contract.

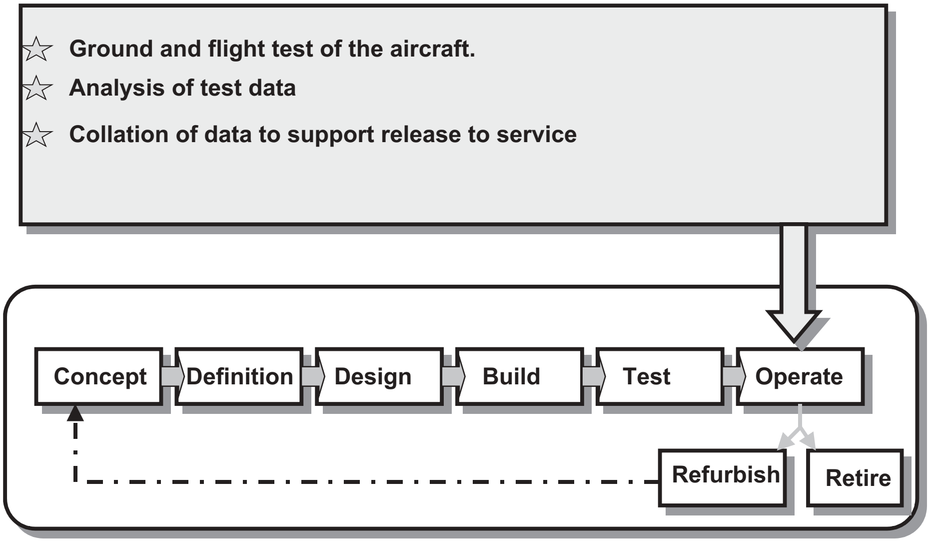


Fig. 3.17 Operate phase

Disposal or refurbish

At the end of the useful or predicted life of the aircraft, decisions have to be made about its future, as depicted in Fig. 3.18. The end of life may be determined by unacceptably high operating costs, unacceptable environmental considerations (noise, pollution, etc.), or predicted failure of mechanical or structural components determined by the supplier's test rigs. If it is not possible to continue to operate the aircraft, then it may be disposed of – sold for scrap or for an alternative use, e.g. to an aircraft enthusiast or for use as a gate guardian at an operational base.

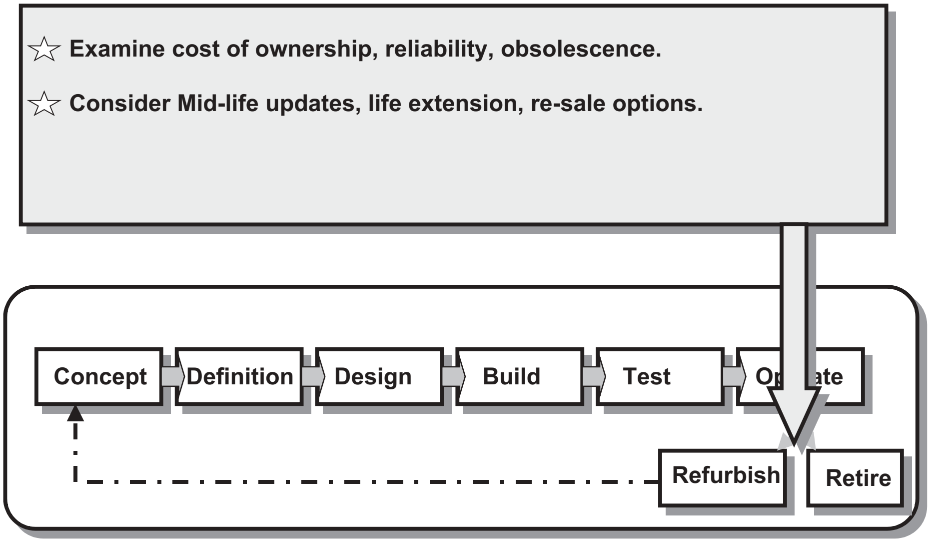


Fig. 3.18 Disposal or refurbishment

If the aircraft still has some residual and commercially viable life, then it may be refurbished. This latter activity is often known as a mid-life update, or even a conversion to a different role, e.g. a VC10 passenger aircraft converted to in-flight refuelling use, as has happened with the Royal Air Force. Similarly, in the civil arena, many former passenger aircraft are being converted to a cargo role.

Development programme

So far, the processes, methods, and techniques used during aircraft system design have been described. However, these need to be applied and controlled within an overall programme management framework. Figure 3.19 shows the major milestones associated with the aircraft systems development process. It is assumed, as is the case for the majority of aircraft systems developed today, that the system has electronics associated with the control function and that the electronics has a software development content.

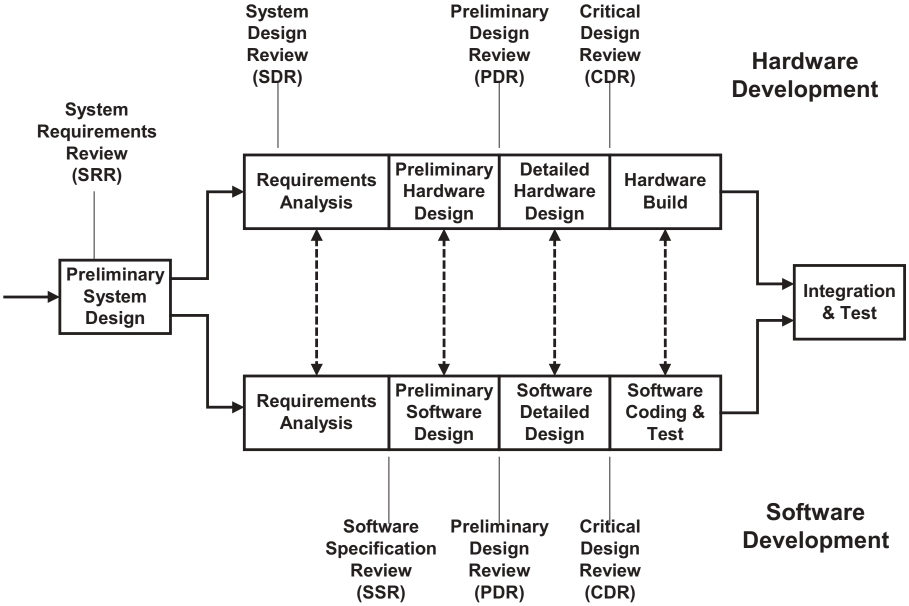


Fig. 3.19 *Typical development programme*

The main characteristic of the development is the bifurcation of hardware and software development processes into two separate paths, though it can be seen that there is considerable interaction between the two. The key steps in the avionics development programme that are primarily designed to contain and mitigate against risk are:

1. System Requirements Review (SRR). The SRR is the first top-level, multidisciplinary review of the perceived system requirements. It is effectively a sanity check upon what the system is required to achieve; a top-level overview of requirements and review against the original objectives. Successful attainment of this milestone leads to a preliminary system design leading in turn to the parallel development of hardware and software requirements analysis, albeit with significant coordination between the two.
2. System Design Review (SDR)**.** The hardware SDR immediately follows the

preliminary design phase and will encompass a top-level review of the system hardware characteristics such that preliminary design may proceed with confidence. Key hardware characteristics will be reviewed at this stage to ensure that there are no major mismatches between the system requirements and what the hardware is capable of supporting.

3. Software Specification Review (SSR). The SSR is essentially a similar process to the hardware SDR but applying to the software when a better appreciation of the software requirements has become apparent, and possibly embracing any limitations such as throughput, timing, or memory that the adopted hardware solution may impose. Both the SDR and SSR allow the preliminary design to be developed up to the preliminary design review (PDR).
4. Preliminary Design Review (PDR). The preliminary design review process is the first detailed review of the initial design (both hardware and software) versus the derived requirements. This is usually the last review before committing major design resource to the detailed design process. This stage in the design process is the last before major commitment to providing the necessary programme resources and investment.
5. Critical Design Review (CDR). By the time of the CDR, major effort will have been committed to the programme design effort. The CDR offers the possibility of identifying final design flaws or, more likely, trading the risks of one implementation path versus another. The CDR represents the last opportunity to review and alter the direction of the design before very large commitments and final design decisions are taken. Major changes in system design – both hardware and software – after the CDR will be very costly in terms of financial and schedule loss, to the total detriment of the programme.

The final stages following CDR will realize the hardware build and software coding and test processes which bring together the hardware and software into the eventual product realization. Even following system validation and equipment certification, it is unusual for there to be a period free of modification either at this stage or later in service when airlines may demand equipment changes for performance, reliability, or maintainability reasons.

‘V’ diagram

The rigours of software development are particularly strict and are dictated by reference (**6**). For obvious reasons, the level of criticality of software used in avionics systems determines the rigour applied to the development process. The levels of software are generally defined according to the operational impact of a software failure. These are defined in Table 3.3.

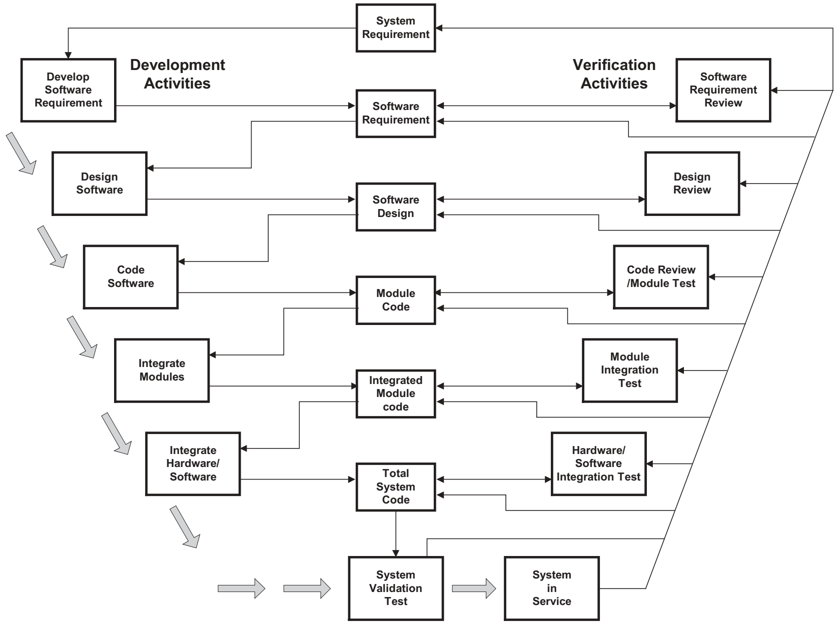
Table 3.3. Software levels

Level	Contribution to resultant failure condition for the aircraft
A	Catastrophic
B	Hazardous
C	Major
D	Minor
E	No effect

The software development process is generally in the form presented in Fig. 3.20, which shows the development activities evolving down the left of the diagram and the verification activities down the right. This shows how the activities eventually converge in the software validation test at the foot of the diagram, that is, the confluence of hardware and software design and development activities. Down the centre of the diagram the various development software stages are shown. It can be seen that there is considerable interaction between all the processes that represent the validation of the requirements and of the hardware and software design at each level. Any problems or issues discovered during the software validation tests are fed back up the chain, if necessary back into the top level. Therefore, any minor deviations are reflected back into all the requirement stages to maintain a consistent documentation set and a consistent hardware and software design.

Whereas the earlier stages of software development and testing might be hosted in a synthetic software environment, it is increasingly important as testing proceeds to undertake testing in a representative hardware environment. This testing represents the culmination of functional testing for the LRU or equipment short of flight testing.

Fig. 3.20 'V' diagram



ETOPS and LROPS requirements

Extended Range Twin Operations (ETOPS) allowing twin-engine aircraft to operate for up to 180 min from a suitable diversion airfield have been in force for a number of years. ETOPS operations are addressed by reference (**9**). The increasing long-range capability of some new transport aircraft and the ability to navigate accurately over the entire surface of the world is leading to the opening up of longe-range transpolar, oceanic, and wilderness routes that require the drafting of new regulations.

At present, both the FAA and the JAA are preparing draft documents to discuss and develop the emerging regulations. The new regulations will address the factors associated with Long Range Operations (LROPS).

The new regulations will categorize operations according to the severity of conditions that may be expected to prevail on the route in question. These categories are, in ascending order of severity:

- Benign.
- Demanding.
- Severe.

Figure 3.21 illustrates the categories that are perceived to exist at various locations around world, and Table 3.4. gives a precise geographical definition of the severe areas of operation.

Fig. 3.21 *Transport aircraft operating conditions around the world*

Table 3.4. Severe climatic areas – geographical boundaries

Area	Limits
North Atlantic	North of 70º N worldwide
North America	North of 55º N
Siberia	Landmass north of 40º N and east of 60º E
Southern Polar	South of 60º S
Himalayas	Area bounded by: 40º N 66º E; 40º N 102º E; 26º N 105º E; 26º N 78º E; 40º N 66º E

The importance of the Himalayan region relates primarily to the height of the terrain. In the event of an engine loss or pressurization failure, which would cause the aircraft to descend to less than 10 000 ft, a catastrophic situation could occur as the valley floors can be higher than 10 000 ft in the region.

Key issues and definitions that appear to be emerging include the following:

- ETOPS begin at a 60 min diversion time.
- ETOPS of up to 180 min will be controlled using the existing ETOPS advisory circular (**9**).
- LROPS will apply to all two- , three- , and four-engined aircraft beyond 180 min (realistically only 180 min will apply to two-engine aircraft, with the possible exception of 207 min (180 min plus 15 per cent).

The major application is for four-engine aircraft with diversion times of up to 480 min (8 h). A simplified classification of the diversion time versus climatic severity is shown in Fig. 3.22. It can be seen that twin-engine aircraft using ETOPS rules can only fly for up to 180 min over benign or demanding climatic regions. Only four-engine aircraft will be allowed to operate for up to 480 min diversion over demanding and severe regions, and only then when an engine and aircraft systems assessment has been conducted to assure the integrity of operation of vital systems for such long exposure times.

It is upon these systems that the LROPS regulations will have an impact. Although still in the early stages of formulation, it is anticipated that the regulations will address the following aircraft systems issues:

- Engine reliability – In-Flight ShutDown (IFSD) of one or more engines.
- Conditioned air supply/bleed air system integrity and reliability.
- Cargo hold fire suppression capacity and integrity of smoke/fire alerting systems.
- Brake accumulator and emergency braking system capacity/integrity.
- Adequate capacity of time dependent functions.
- Pressurization/oxygen system integrity/reliability/capacity.

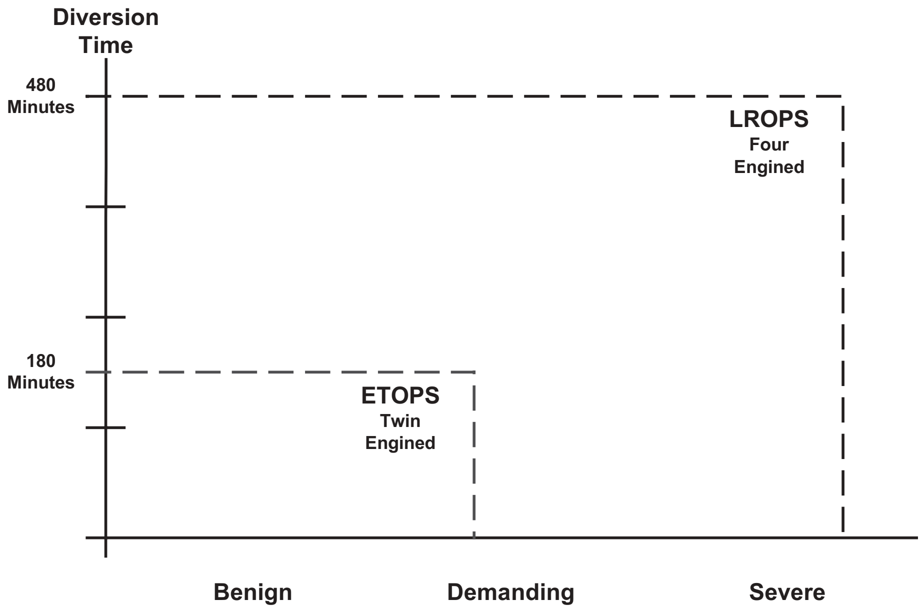


Fig. 3.22 *Relationship of diversion time and climatic severity*

- Integrity/reliability/capacity of back-up systems (electrical, hydraulic).
- Fuel system integrity and fuel accessibility. Fuel consumption with engine failure and other system failures.
- Fuel quantity and fuel used indications and alerts.

References

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(2) SAE ARP 4761, *Guidelines and Methods for Conducting the Safety Assessment Process on Civil Airborne Systems,* Society of Automobile Engineers Inc.

(3) Advisory Circular, AC 25.1309–1A (1988) System Design and Analysis.

(4) Advisory Material Joint, AMJ 25.1309, System Design and Analysis.

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(6) RTCA DO-178B, Software Considerations in Airborne Systems and Equipment Certification.

(7) RTCA DO-254, Design Assurance Guidance for Airborne Electronic Hardware.

(8) **Bonneau, V.** (1998) Dual-Use of variable speed constant frequency (VSCF) cyclo-converter technology, *FITEC' 98,* London.

(9) Advisory Circular, AC120-42A, (1988) Extended Range Operations with Two-Engine Airplanes.

CHAPTER 4

Electrical Systems

Electrical systems have made significant advances over the years as aircraft have become more dependent upon electrically powered services. A typical electrical power system of the 1940s and 1950s was the twin 28 V DC system. This system was used a great deal on twin-engined aircraft; each engine powered a 28 V DC generator which could employ load sharing with its contemporary if required. One or two DC batteries were also fitted, and an inverter was provided to supply 115 V AC and then 26 V AC to the flight instruments.

The advent of the V-bombers in the United Kingdom, and similar developments in the United States changed this situation radically owing to the much greater power requirements imposed by the wide range of systems in these aircraft. In the United Kingdom, one aircraft, the Vickers Valiant, incorporated electrically actuated landing gear. These aircraft were fitted with four 115 V AC generators, one being driven by each engine. To provide the advantages of no-break power, these generators were paralleled which increased the amount of control and protection circuitry. The V-bombers had to power high-power military mission loads such as radar and electronic warfare jamming equipment.

Most aircraft utilization equipment is accustomed to a constant frequency supply of 115 V AC In order to generate constant-frequency 115 V AC at 400 Hz, a Constant Speed Drive or CSD is required to negate the aircraft engine speed variation typically over approximately a 2:1 speed range (full power speed/flight idle speed). These are complex hydromechanical devices that by their very nature are not highly reliable without significant maintenance penalties. Therefore, the introduction of constant-frequency AC generation systems was not without accompanying reliability problems, particularly on fighter aircraft where engine throttle settings are changed very frequently throughout the mission. In modern aircraft the generator and CSD make up a combined unit called an Integrated Drive Generator (IDG).

The advances in high-power, solid-state switching technology together with enhancements in the necessary control electronics have made Variable-Speed/Constant-Frequency (VSCF) systems a viable proposition in the last decade. The VSCF system removed the unreliable CSD portion, the variable-frequency or frequency-wild power from the AC generator being converted to 400 Hz constant-frequency 115 V AC power by means of a solid-state VSCF converter. VSCF systems are now becoming more commonplace: the F-18 fighter uses such a system, and some versions of the Boeing 737-500 did use such a system. In addition, the Boeing 777 airliner utilizes a VSCF system for back-up AC power generation.

In US military circles, great emphasis is being placed by the US Air Force and the US Army on the development of 270 V DC systems. In these systems, high-power generators derive 270 V DC power, some of which is then converted into 115 V AC 400 Hz or 28 V DC required to power specific equipment and loads. This is claimed to be more efficient than conventional methods of power generation, and the amount of power conversion required is reduced with accompanying weight savings. These developments are allied to the 'more-electric aircraft' concept, where it is intended to ascribe more aircraft power system activities to electrical means rather than use hydraulic or high-pressure bleed air, as is presently the case. The fighter aircraft of tomorrow will therefore need to generate much higher levels of electrical power than at present. The use of higher voltages than 115 V AC for civil aircraft is not generally favoured by the certification authorities owing to wiring problems experienced on existing aircraft, but the need to move towards more-electric solutions is recognized. As a general point of interest, many of the electrical power system techniques that have been adopted by the civil community have evolved from military technology development.

At the component level, advances in the development of high-power contactors and solid-state power switching devices are improving the way in which aircraft primary and secondary power loads are switched and protected. These advances are being married to microelectronic developments to enable the implementation of new concepts for electrical power management system distribution, protection, and load switching. The use of electrical power has progressed to the point where the generation, distribution, and protection of electrical power to the aircraft electrical services or loads now comprise one of the most complex aircraft systems. This situation was not always so.

The move towards the higher AC voltage is really driven by the amount of power the electrical channel is required to produce. The sensible limit for DC systems has been found to be around 400 A owing to the limitations of feeder size and high-power protection switchgear, known as contactors. Therefore, for a 28 VDC system delivering 400 A or just over, the maximum power the channel may deliver is about 12 kW, well below the requirements of most aircraft today. This level of power is sufficient for General Aviation (GA) aircraft and some of the smaller business jets. However, the requirements for aircraft power in business jets, regional aircraft, and larger transport aircraft is usually in the range 20–90 kVA per channel and higher. This requirement for more power has been matched in the military aircraft arena.

Aircraft electrical system characteristics

The generic parts of a typical AC aircraft electrical system are shown in Fig. 4.1 and comprise the following:

- Power generation.
- Primary power distribution and protection.
- Power conversion and energy storage using a Transformer Rectifier Unit (TRU) and battery.
- Secondary power distribution and protection.

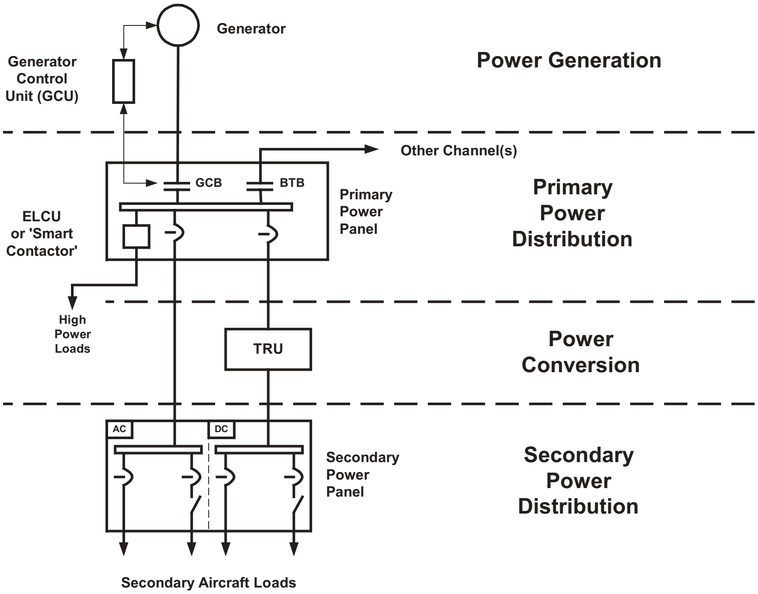


Fig. 4.1 *Generic aircraft AC electrical system*

At this stage it is worth outlining the major differences between AC and DC power generation. Later in the chapter, more emphasis is placed upon more recent AC power generation systems.

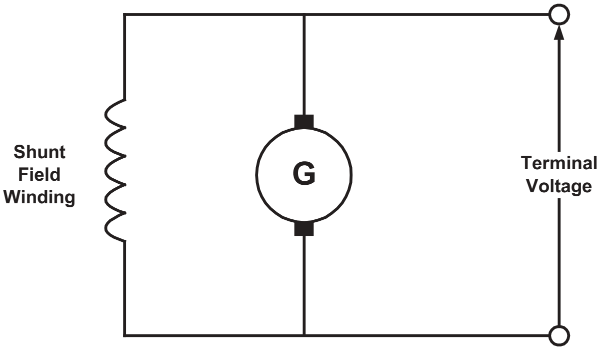
Power generation

DC power generation

DC systems use generators to develop a DC voltage to supply aircraft system loads; usually the voltage is 28 V DC but there are 270 V DC systems in being which will be described later in the chapter. The generator is controlled – the technical term is regulated – to supply 28 V DC at all times to the aircraft loads such that any tendencies for the voltage to vary or fluctuate are overcome. DC generators are self-exciting, in that they contain rotating electromagnets that generate the electrical power. The conversion to DC power is achieved by using a device called a commutator that enables the output voltage, which would appear as a simple sine wave output, to be effectively half-wave rectified and smoothed to present a steady DC voltage with a ripple imposed.

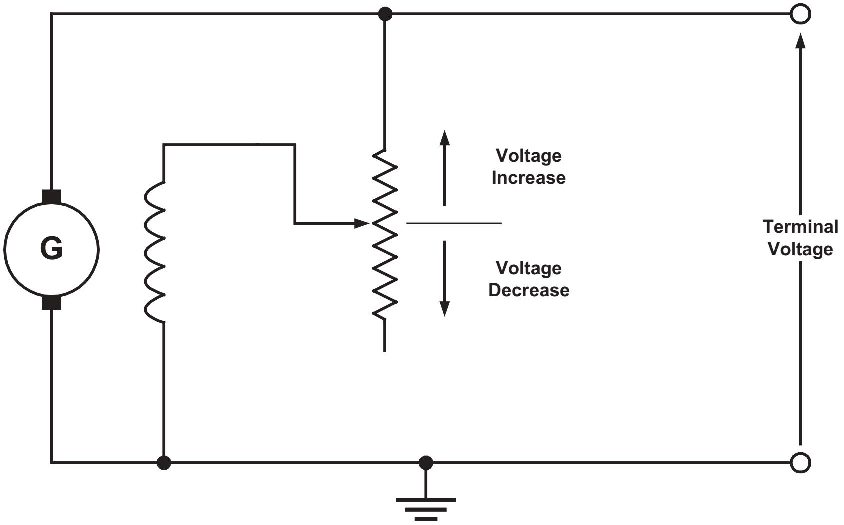
In aircraft applications, the generators are typically shunt wound, the high-resistance field coils being connected in parallel with the armature, as shown in Fig. 4.2.

Fig. 4.2 Shunt-wound DC generator



The natural load characteristic of the shunt-wound generator is for the voltage to 'droop' with the increasing load current, whereas the desired characteristic is to control the output at a constant voltage – nominally 28 V DC For this purpose, a voltage regulator is used which modifies the field current to ensure that terminal voltage is maintained while the aircraft engine speed and generator loads vary. The principle of operation of the DC voltage regulator is shown in Fig. 4.3 and described later in the chapter.

Fig 4.3 DC voltage regulator



AC power generation

An AC system uses a generator to generate a sine wave of given voltage and, in most cases, of constant frequency. The construction of the alternator is simpler than that of the DC generator in that no commutator is required. Early AC generators used slip rings to pass current to/from the rotor windings, but these suffered from abrasion and pitting, especially when passing high currents at altitude. High-altitude operations are more prone to arcing effects owing to the fact that the thinner air acts as a weaker dielectric; also, a phenomenon known as 'corona' can affect higher-voltage operation.

Modern AC generators, commonly called compound machines, work on the principle shown in Fig. 4.4. This AC generator may be regarded as several machines sharing the same shaft. From left to right as viewed on the diagram they comprise:

- A Permanent Magnet Generator (PMG).
- An excitation stator surrounding an excitation rotor containing rotating diodes.
- A power rotor encompassed by a power stator.

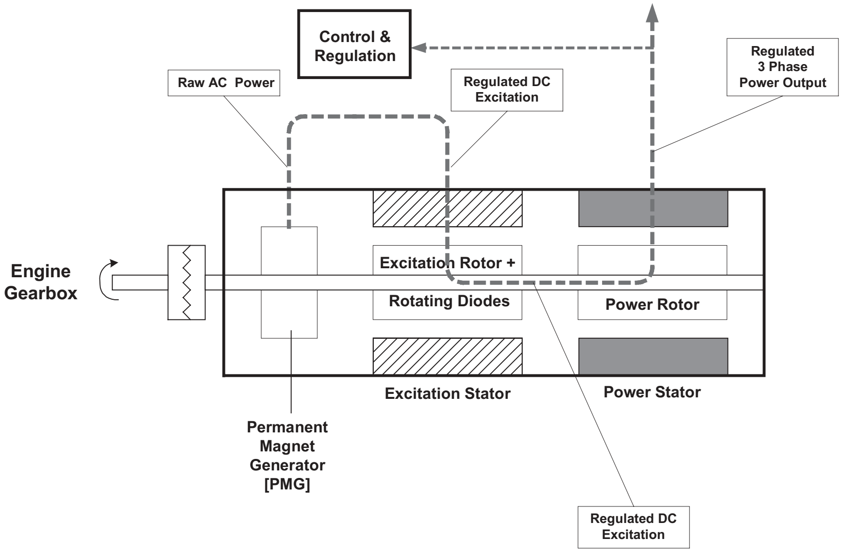
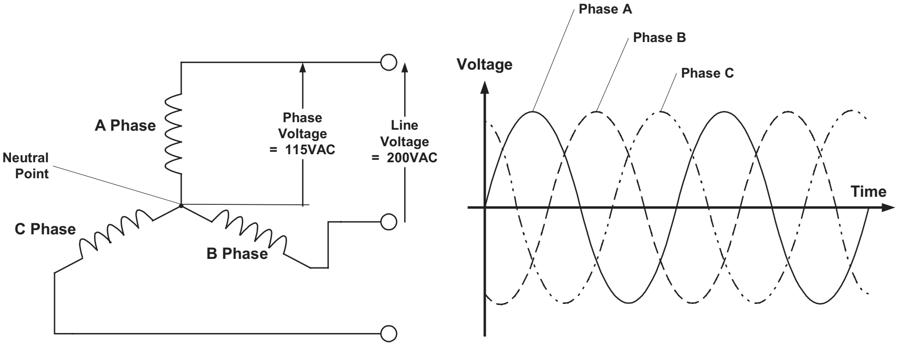


Fig. 4.4 *Principle of operation of modern AC generator*

The flow of power through this generator is highlighted by the dashed line. The PMG generates ‘raw’ (variable-frequency, variable-voltage) power sensed by the control and regulation section that is part of the generator controller. This modulates the flow of DC current into the excitation stator windings and therefore controls the voltage generated by the excitation rotor. The rotation of the excitation rotor within the field produced by the excitation stator windings is rectified by means of diodes contained within the rotor and supplies a regulated and controlled DC voltage to excite the power rotor windings. The rotating field generated by the power rotor induces an AC voltage in the power stator that may be protected by the GCU, as explained later, and supplied to the aircraft systems when operating within acceptable criterian.

Most AC systems used on aircraft use a three-phase system, that is, the generator supplies three sine waves, each phase positioned 120 degrees out of phase with the others. These phases are most often connected in a star configuration, with one end of each of the phases connected to a neutral point, as shown in Fig. 4.5. In this layout the phase voltage of a standard aircraft system is 115 V AC, whereas the line voltage measured between lines is 200 V AC The standard for aircraft frequency-controlled systems is 400 cycle/s or 400 Hz.

Fig. 4.5 Star connected 3 phase AC generator



The descriptions given previously outline the two primary methods of power generation used on aircraft for many years. The main advantage of AC power is that it operates at a higher voltage – 115 V AC rather than 28 V DC for the DC system. The use of a higher voltage is not an advantage in itself, in fact higher voltages require better standards of insulation. It is in the transmission of power that the advantage of higher voltage is most apparent. For a given amount of power transmission, a higher voltage relates to an equivalent lower current. The lower the current, the lower are losses such as voltage drops (proportional to current) and power losses (proportional to current squared). Also, as current conductors are generally heavy, it can be seen that the reduction in current also saves weight, a very important consideration for aircraft systems.

Power generation control

The primary elements of power system control are:

- DC systems:
  - Voltage regulation,
  - parallel operation,
  - protection functions.

- AC systems:
  - Voltage regulation,
  - parallel operation,
  - supervisory functions.

DC system generation control

Voltage regulation

DC generation is by means of shunt-wound self-exciting machines, as already briefly outlined. The principle of voltage regulation has already been outlined in Fig. 4.3. This shows a variable resistor in series with the field winding such that variation in the resistor alters the combined total resistance and therefore the current flowing in the field winding; hence, the field current and output voltage may be varied. In actual fact the regulation is required to be an automatic function that takes account of load and engine speed. The voltage regulation needs to be in accordance with the standard used to specify aircraft power generation systems. The standards specify the voltage at the point of regulation and the nature of the acceptable voltage drops throughout the aircraft distribution, protection, and wiring system. Typical standards for commercial or military use are presented in the electrical power quality section below. DC systems are limited to a maximum of around 400 A or 12 kW per channel for two reasons:

- The size of conductors and switchgear to carry the necessary current becomes prohibitive.
- The brush wear on brushed DC generators becomes excessive, with resulting maintenance costs if these levels are exceeded.

Parallel operation

In multiengined aircraft, each engine will be driving its own generator, and in this situation it is desirable that 'no-break' or uninterrupted power is provided in cases of engine or generator failure. A number of sensitive aircraft instruments and navigation devices that comprise some of the electrical loads may be disturbed and may need to be restarted or re-initialized following a power interruption. In order to satisfy this requirement, generators are paralleled to carry an equal proportion of the electrical load between them. Individual generators are controlled by means of voltage regulators which automatically compensate for variations. In the case of parallel generator operation there is a need to interlink the voltage regulators such that any unequal loading of the generators can be adjusted by means of corresponding alterations in field current. This paralleling feature is more often known as an equalizing circuit and therefore provides 'no-break' power in the event of a major system failure. A simplified diagram showing the main elements of DC parallel operation being supplied by generator 1 (G1) and generator 2 (G2) is given in Fig. 4.6.

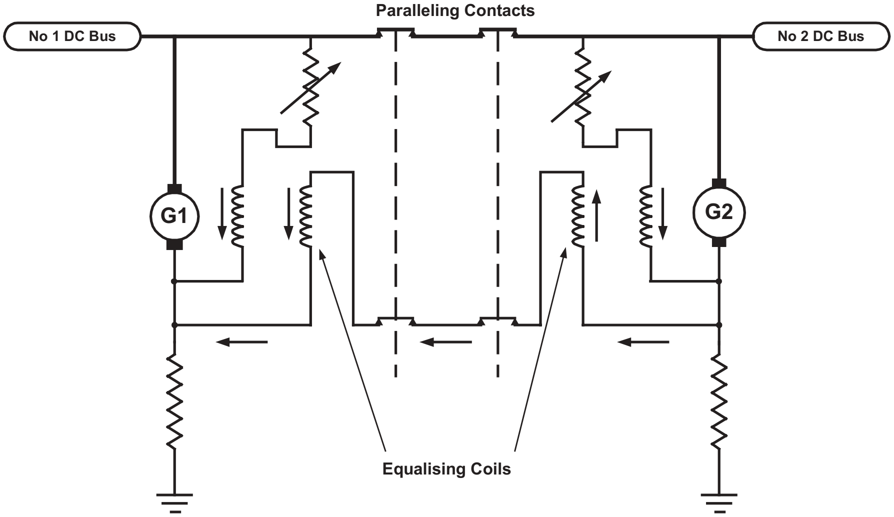


Fig. 4.6 *DC generator parallel operation*

Protection functions

The primary conditions for which protection needs to be considered in a DC system are as follows:

1. Reverse current. In a DC system it is evident that the current should flow from the generator to the busbars and distribution systems. In a fault situation it is possible for current to flow in the reverse direction, and the primary system components need

to be protected from this eventuality. This is usually achieved by means of reverse current circuit breakers or relays. These devices effectively sense reverse current and switch the generator out of circuit, thus preventing any ensuing damage.

2. Overvoltage protection. Faults in the field excitation circuit can cause the generator to overexcite and thereby regulate the supply voltage to an erroneous overvoltage condition. This could then result in the electrical loads being subject to conditions that could cause permanent damage. Overvoltage protection senses these failure conditions and opens the line contactor, taking the generator off-line.
3. Undervoltage protection. In a single-generator system, undervoltage is a similar fault condition to the reverse current situation already described. However, in a multigenerator configuration with paralleling by means of an equalizing circuit, the situation is different. Here, an undervoltage protection capability is essential as the equalizing circuit is always trying to raise the output of a lagging generator; in this situation the undervoltage protection is an integral part of the parallel load sharing function.

AC power generation control

Voltage regulation

As has already been described, AC generators differ from DC machines in that they require a separate source of DC excitation for the field windings, although the system described earlier does allow the generator to bootstrap the generation circuits. The subject of AC generator excitation is a complex topic for which the technical solutions vary according to whether the generator is frequency wild or constant frequency. Some of these solutions comprise sophisticated control loops with error detectors, preamplifiers, and power amplifiers.

Parallel operation

In the same way that DC generators are operated in parallel to provide ‘no-break’ power, AC generators may also be controlled in a similar fashion. This technique only applies to constant-frequency AC generation as it is impossible to parallel frequency-wild or Variable Frequency (VF) AC generators. In fact, many of the aircraft loads such as anti/de-icing heating elements driven by VF generators are relatively frequency insensitive, and the need for ‘no-break’ power is not nearly so important. To parallel AC machines, the control task is more complex as both real and reactive (imaginary) load vectors have to be synchronized for effective load sharing.

The sharing of real load depends upon the relative rotational speeds and hence the relative phasing of the generator voltages. Constant-speed or constant-frequency AC generation depends upon the tracking accuracy of the constant-speed drives of the generators involved. In practice, real load sharing is achieved by control laws that measure the degree of load imbalance by using current transformers and error detection circuitry, thereby trimming the constant-speed drives such that the torques applied by all generators are equal.

The sharing of reactive load between the generators is a function of the voltage generated by each generator, as for the DC parallel operation case. The generator output voltages depend upon the relevant performance of the voltage regulators and field excitation circuitry. To accomplish reactive load sharing requires the use of special transformers called mutual reactors, error detection circuitry, and preamplifiers/power amplifiers to adjust the field excitation current. Therefore, by using a combination of trimming the speed of the Constant-Speed Drives (CSDs) and balancing the field excitation to the generators, real and reactive load components may be shared equally between the generators (see Fig. 4.7). This has the effect of providing a powerful single vector AC power supply to the aircraft AC system, providing a very 'stiff' supply in periods of high power demand. Perhaps the biggest single advantage of paralleled operation is that all the generators are operating in phase synchronism, and therefore, in the event of a failure, there are no changeover transients.

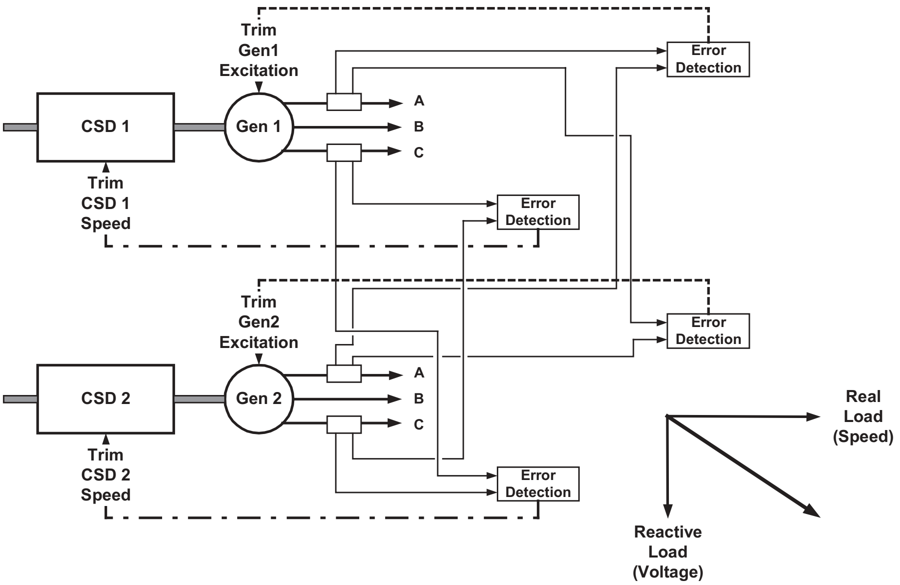


Fig. 4.7 *AC generator parallel operation*

Supervisory and protection functions

Typical supervisory or protection functions undertaken by a typical AC Generator Control Unit (GCU) are listed below:

- Over voltage.
- Under voltage.
- Under/overexcitation.
- Under/overfrequency.
- Differential current protection.

The overvoltage, undervoltage, and under/overexcitation functions are similar to the corresponding functions described for DC generation control. Under- or overfrequency protection is effectively executed by the real load-sharing function already described above for AC parallel operation. Differential current protection is designed to detect a short-circuit busbar or feeder line fault which could impose a very high current demand on the short-circuited phase. Differential current transformers sense the individual phase currents at different parts of the system. These are connected so that detection circuitry will sense any gross difference in phase current (say in excess of 20 A per phase) resulting from a phase imbalance and disconnect the generator from the busbar by tripping the Generator Control Breaker (GCB).

Modern electrical power generation types

So far, basic DC and AC power generating systems have been described. The DC system is limited by currents greater than 400 A, and the constant-frequency AC method using an Integrated Drive Generator (IDG) has been mentioned. In fact, there are many more power generation types in use today. A number of recent papers have identified the issues and projected the growth in aircraft electric power requirements in a civil aircraft setting, even without the advent of more-electric systems. However, not only are aircraft electrical system power levels increasing but the diversity of primary power generation types is increasing.

The different types of electrical power generation currently being considered are shown in Fig. 4.8. The constant-frequency (CF) 115 V AC , three-phase, 400 Hz options are typified by the IDG, VSCF cycloconverter, and DC link options. The VF 115 V AC, three-phase power generation – sometimes termed ‘frequency wild’ - is also a more recent contender, but, although a relatively inexpensive form of power generation, it has the disadvantage that some motor loads may require motor controllers. Military aircraft in the United States are inclining towards 270 V DC systems. Permanent Magnet Generators (PMGs) are used to generate 28 V DC emergency electrical power for high-integrity systems.

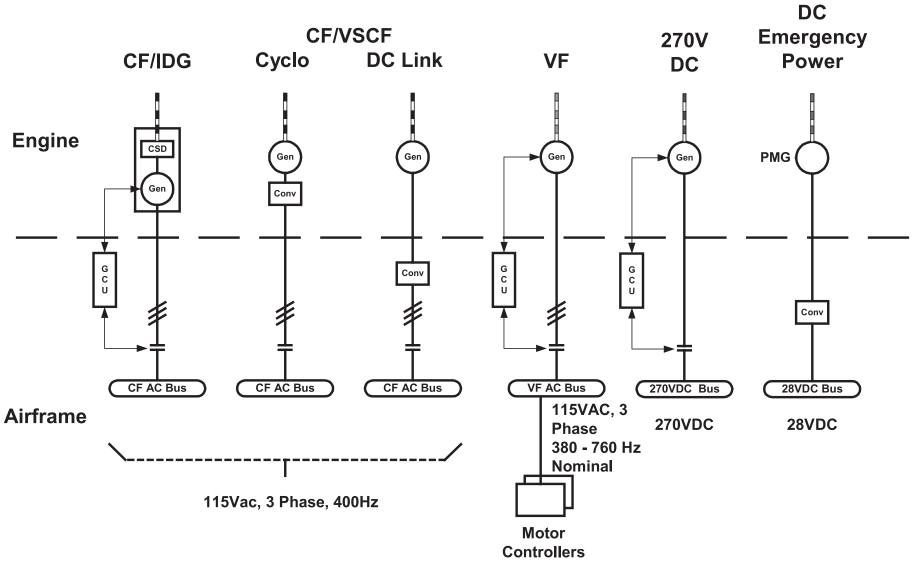


Fig. 4.8 *Electrical power generation types*

Figure 4.8 is also interesting in that it shows the disposition between generation system components located on the engine and those within the airframe. Without being drawn into the partisan arguments regarding the pros and cons of the major types of power generation in use or being introduced today, it is worth examining the main contenders:

- Constant-Frequency power generation using an IDG.
- Variable-Frequency power generation.
- Variable-Speed/Constant-Frequency power generation.

Constant frequency/IDG generation

The main features of CF/IDG power are shown in Fig. 4.9. In common with all the other power generation types, this has to cater for a 2:1 ratio in engine speed between maximum power and ground idle. The CSD in effect acts as an automatic gearbox, maintaining the generator at a constant shaft speed which results in a constant-frequency output of 400 Hz, usually within ~10 Hz or less. The drawback of the hydromechanical CSD is that it needs to be correctly maintained in terms of oil charge level and oil cleanliness. Also, to maintain high reliability, frequent overhauls may be necessary.

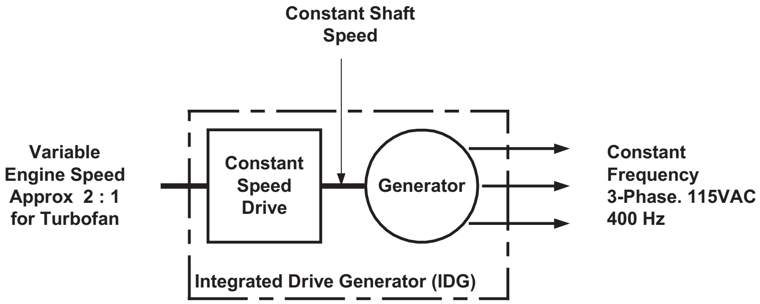


Fig. 4.9 *Constant frequency/IDG generation*

Features:

Constant frequency AC power is most commonly used on turbofan aircraft today

System is expensive to purchase & maintain; primarily due to complexity of Constant Speed Drive (CSD)

Single company monopoly on supply of CSD/IDG

Alternate methods of power generation are under consideration

That said, the IDG is used to power the majority of civil transport aircraft today, as shown in Table 4.1 (page 75).

Variable-frequency generation

Variable-Frequency power generation, as shown in Fig. 4.10, is the simplest and most reliable form of power generation. In this technique no attempt is made to nullify the effects of the 2:1 engine speed ratio, and the power output, though regulated to 115 V AC, suffers a frequency variation typically from 380 to 720 Hz. This wide-band VF power has an effect on frequency-sensitive aircraft loads, the most obvious being the effect on AC electric motors that are used in many aircraft systems. There can therefore be a penalty to be paid in the performance of other aircraft systems such as fuel, Environmental Control System (ECS), and hydraulics. In many cases, variations in motor/pump performance may be accommodated, but in the worst cases a motor controller may be needed to restore an easier control situation.

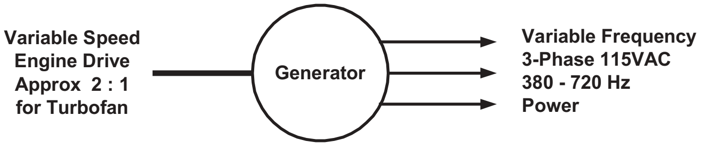


Fig. 4.10 *Variable frequency power generation*

Features:

Simplest form of generating power, cheapest and most reliable

Variable frequency has impact upon other aircraft subsystems

Motor controllers may be needed for certain aircraft loads

Beginning to be adopted for new programmes: gains outweigh disadvantages

Variable Frequency is being widely adopted in the business jet community as their power requirements take them above the 28 V DC/12 kW limit of twin 28 V DC systems. Aircraft such as Global Express had VF designed in from the beginning, and VF power has been established as the baseline for the Airbus A380 project.

VSCF generation

Figure 4.11 shows the concept of the VSCF converter. In this technique the variable -frequency power produced by the generator is electronically converted by solid-state power switching devices to constant-frequency 400 Hz, 115 V AC power. Two options exist:

1. DC link. In the DC link the raw power is converted to an intermediate DC power stage – the DC link – before being electronically converted to three-phase AC power. DC link technology has been used on the B737, MD-90, and B777 but has yet to rival the reliability of CF or VF power generation.
2. Cycloconverter. The cycloconverter uses a different principle. Six phases are generated at relatively high frequencies in excess of 2000 Hz and the solid-state devices switch between these multiple phases in a predetermined and carefully controlled manner. The effect is electronically to commutate the input and provide three phases of constant-frequency 400 Hz power. Though this appears to be a complex technique, it is in fact quite elegant, and cycloconverter systems have been successfully used on military aircraft in the United States: F-18, U-2, and the F-117 Stealth fighter. There are no civil applications.

As suggested earlier in Fig. 4.8, each of these techniques may locate the power conversion section on the engine or in the airframe. Reference (**1**) examines the implications of moving the VSCF converter from the engine to the airframe in a civil aircraft context.

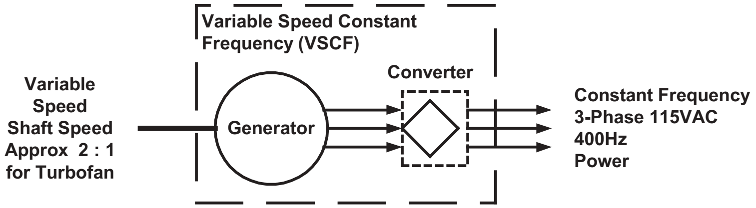


Fig. 4.11 *VSCF power generation*

Features:

Conversion of VF electrical power to CF is accomplished by electronic controlled power switching

DC Link & Cycloconverter options available

Not all implementations have proved to be robust/reliable - Cycloconverter shows most promise

Still unproven in transport market

Table 4.1. Recent civil and military aircraft power system developments

Generation type	Civil application		Military application	
IDG/CF [115 V AC /400 Hz]	B777	2 × 120 kVA		
	A340	4 × 90 kVA		
	B737NG	2 × 90 kVA		
	MD-12	4 × 120 kVA		
	B747-X	4 × 120 kVA		
	B717	2 × 40 kVA		
	B767-400	2 × 120 kVA		
	Do728	2 × 40 kVA		
VSCF (Cycloconverter) (115 V AC /400 Hz)			F-18E/F	2 × 60/65 kVA
VSCF (DC link) (115 V AC /400 Hz)	B777 (Backup)	2 × 20 kVA		
	MD-90	2 × 75 kVA		
VF (115 V AC /380 – 760Hz typical)	Global Ex	4 × 40 kVA	Boeing JSF (X-32A/B/C)	2 × 50 kVA
	Horizon	2 × 20/25 kVA		
	A380	4 × 150 kVA		
270 V DC			F-22 Raptor	2 × 70 kVA
			Lockheed-Martin JSF (X-35A/B/C)	2 × 50 kVA

Table 4.1 lists the power generation types developed and proposed for civil and military (fighter) aircraft platforms throughout the 1990s. Not only are the electrical power levels increasing in this generation of aircraft, but the diversity of electrical power generation methods introduces new aircraft system issues that need to be addressed. For example, the B777 standby VSCF and the MD-90 VCSF converters, being located in the airframe, increase the ECS requirements since waste heat is dissipated in the airframe, whereas the previous IDG solution rejected heat into the engine oil system. Similarly, the adoption of VF can complicate motor load and power conversion requirements. The adoption of 270 V DC systems by the US military has necessitated the development of a family of 270 V DC protection devices since conventional circuit breakers cannot be used at such high voltages.

Electrical power quality

The quality of the electrical power on-board an aircraft is carefully controlled in a number of regards. Power quality will be defined by the nature of the electrical power generation system, the distribution system, and the nature of loads – particularly the high-power loads that are connected. Power quality is defined by a number of specifications that are similar in many regards:

- RTCA DO-160C. The generally recognized US civil specification described in reference (**2**).
- ADB-0100. The Airbus specification defined in reference (**3**).
- MIL-STD-704E. The US military specification defined in reference (**4**). Although apparently disassociated from either of the above, it becomes highly relevant when fitting military equipment on-board an aircraft that is predominantly civilian in origin (civil aircraft such as Boeing 767 or A330s) and that has been designed using a civil electrical power standard. A topical example is the UK or US military programmes in which these aircraft platforms are being seen as ideal candidates for adoption as military tanker aircraft. For an exposé on some of these issues, see Chapter 13.

These electrical power system references typically refer to specification of the following defined, though not exclusive, parameters:

- Voltage transients – both normal and abnormal – for AC and DC networks.
- Normal voltage excursions – 115 V AC and 28 V DC networks.
- Normal frequency excursions – CF systems.
- Voltage spikes – 115 V AC and 28 V DC networks.
- Power quality – harmonic distortion – AC systems.
- Power factor limits – AC systems.
- Emergency power requirements – both AC and DC. These requirements demand a rigorous assessment of the aircraft power electrical generation system, distribution system, and loading. Load assessments have to be conducted to assure that the system will meet or exceed the supply of electrical power at the necessary voltage and power levels while also meeting the integrity of certain flight-critical recipient equipment such as Fly-By-Wire (FBW) and others.

Many of the requirements are similar in concept though not necessarily identical in approach. For a detailed review of the differing requirements, readers are recommended to refer to the appropriate electrical power specifications for the aircraft type in mind.

Primary power distribution

The primary power distribution system consolidates the aircraft electrical power inputs. In the case of a typical civil airliner, the aircraft may accept power from the following sources:

- Main aircraft generator, by means of a Generator Control Breaker (GCB) under the control of the GCU.
- Alternate aircraft generator – in the event of generator failure – by means of a bus tie breaker under the control of a Bus Power Control Unit (BPCU).
- APU generator, by means of an APU breaker under the control of the BPCU.
- Ground power, by means of an external power contactor (EPC) under the control of the ground power monitor.
- Back-up converter, by means of a converter control breaker (CCB) under the control of the VSCF converter (B777 only).
- RAT generator when deployed by the emergency electrical system.

The power switching used in these cases is a power contactor or breaker. These are special high-power switches that usually switch power in excess of 20 A per phase. As well as the power switching contacts, auxiliary contacts are included to provide contactor status – 'open' or 'closed' – to other aircraft systems.

Higher-power aircraft loads are increasingly switched from the primary aircraft bus bars by using Electronic Load Control Units (ELCUs) or 'smart contactors' for load protection. Like contactors, these are used where normal rated currents are greater than 20 A per phase, i.e., for loads of around 7 kVA or greater. Figure 4.12a shows a line contactor such as a GCB, and an ELCU or 'smart contactor' is shown in Fig. 4.12b**.** The latter has in-built current-sensing coils which enable the current of all three phases to be measured. Associated electronics allow the device trip characteristics to be more closely matched to those of the load. Typical protection characteristics embodied within the electronics are I^2t, modified I^2t, and differential current protection. For a paper explaining more about 'smart contactors' refer to reference (**5**).

Fig 4.12a *Power contactor*

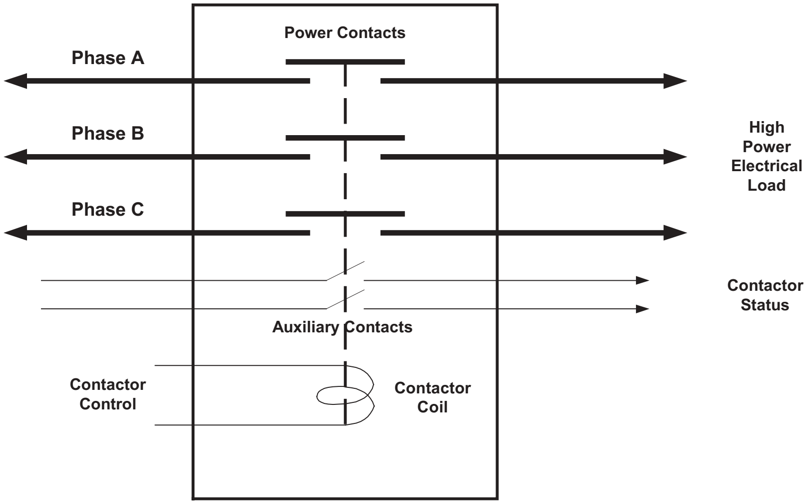
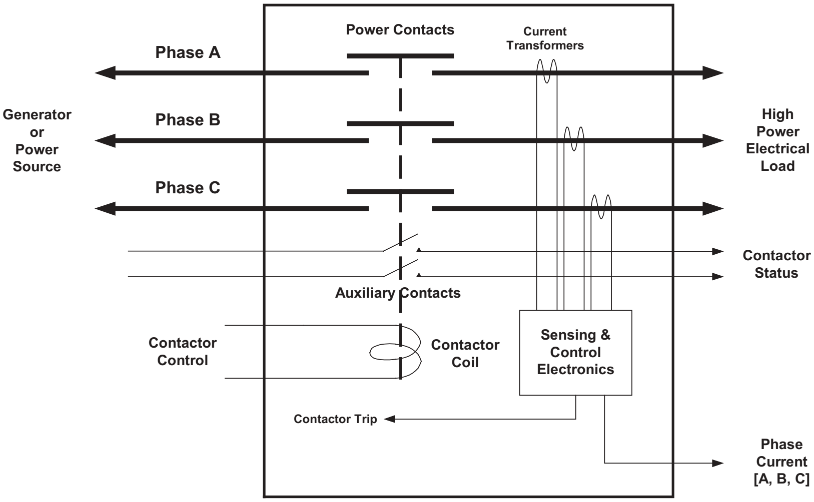


Fig 4.12b *ELCU or 'smart contactor'*



Power conversion and energy storage

This chapter so far has addressed the primary generation of electrical power and primary power distribution and protection. There are, however, many occasions within an aircraft electrical system where it is required to convert power from one form to another. Typical examples of power conversion are:

1. Conversion from DC to AC power. This conversion uses units called inverters to convert 28 VDC to 115 VAC single-phase or three-phase power.
2. Conversion from 115 VAC to 28 VDC power. This is a much-used conversion using units called Transformer Rectifier Units (TRUs).
3. Conversion from one AC voltage level to another; a typical conversion would be from 115 VAC to 26 VAC

4. Battery charging. As previously outlined, it is necessary to maintain the state of charge of the aircraft battery by converting 115 V AC to a 28 V DC battery charge voltage.

Inverters

Inverters convert 28 V DC power into 115 V AC single-phase electrical power. This is usually required in a civil application to supply the captain or first officer's instruments following an AC failure. Alternatively, under certain specific flight conditions, such as autoland, the inverter may be required to provide an alternative source of power to the flight instruments in the event of a power failure occurring during the critical autoland phase. Some years ago the inverter would have been a rotary machine with a DC motor harnessed in tandem with an AC generator. More recently the power conversion is likely to be accomplished by means of a static inverter where the use of high-power, rapid switching, Silicon-Controlled Rectifiers (SCRs) will synthesize the AC waveform from the DC input. Inverters are therefore a minor though essential part of many aircraft electrical systems.

Transformer Rectifier Units

TRUs are probably the most frequently used method of power conversion on modern aircraft electrical systems. Most aircraft have a significant 115 V AC three-phase AC power generation capability inherent within the electrical system, and it is usual to convert a significant portion of this to 28 V DC by the use of TRUs. TRUs comprise start primary and dual star/delta secondary transformer windings together with three-phase full wave rectification and smoothing to provide the desired 115 V AC/28 V DC conversion. A typical TRU will convert a large amount of power, for example, the Boeing 767 uses two TRUs, each of which supplies a rated load of 120 A (continuous) with a 5 min rating of 180 A. TRUs dissipate a lot of heat and are therefore forced air cooled. The Boeing 767 unit is packaged in a 6 MCU ARINC 600 case and weighs around 24 lb. Figure 4.13 shows a typical TRU.

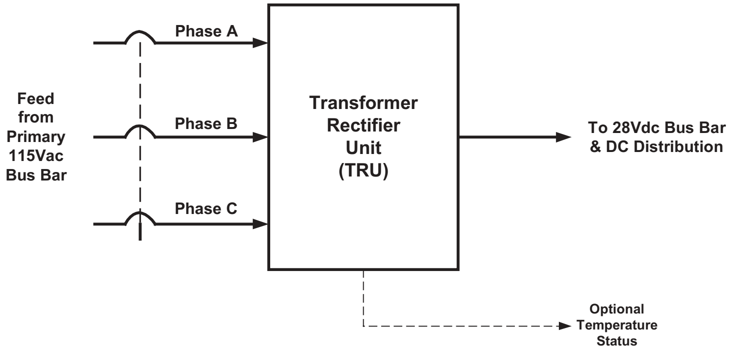


Fig. 4.13 *Transformer rectifier unit (TRU)*

TRUs are usually simple, unregulated units, that is, the voltage is not controlled to 28 V DC as load is increased, and accordingly the load characteristic tends to 'droop'. In some specialist military applications this feature is not desirable and regulated TRUs are used. TRUs are usually operated in isolation, but when regulated they may also be configured to operate in parallel in a similar way to the parallel operation of DC generators. Reference (**6**) is a paper relating to the development of a regulated TRU.

Autotransformers

In certain parts of an electrical system, simple autotransformers may be used to provide a simple voltage step-up or step-down conversion. For example, the 115 V/26 V AC transformation used to provide 26 V AC aircraft lighting supplies direct from main 115 V AC busbars in the easiest way.

Battery chargers

Battery chargers share many of the attributes of TRUs and are in fact dedicated units whose function is purely that of charging the aircraft battery. In some systems the charger may also act as a standby TRU providing a boosted source of DC power to the battery in certain system modes of operation. Usually, the task of the battery charger is to provide a controlled charge to the battery without overheating, and for this reason battery temperature is usually closely monitored.

Batteries

The majority of this section has described power generation systems, both DC and AC However, it neglects an omnipresent element – the battery. This effectively provides an electrical storage medium independent of the primary generation sources. Its main purposes are:

- To assist in damping transient loads in the DC system.
- To provide power in system start-up modes when no other power source is available.
- To provide a short-term, high-integrity source during emergency conditions while alternative/back-up sources of power are being brought on line.

The capacity of the aircraft battery is limited and is measured in terms of ampere-hours. This parameter effectively describes a current/time capability or storage capacity. Thus a 40 A h battery when fully charged would have the theoretical capacity of feeding a 1 A load for 40 h or a 40 A load for 1 h. In fact the capacity of the battery depends upon the charge sustained at the beginning of the discharge, and this is a notoriously difficult parameter to quantify. Most modern aircraft systems utilize battery chargers to maintain the battery charge at moderately high levels during normal system operation, thereby assuring a reasonable state of charge should solo battery usage be required.

The battery most commonly used is the Nickel–Cadmium (Ni–Cd) type, which depends on the reaction between nickel oxides for the anode and cadmium for the cathode, and operates in a potassium hydroxide electrolyte. Lead–acid batteries are not favoured in modern applications owing to corrosive effects. To preserve battery health, it is usual to monitor its temperature which gives a useful indication of overcharging and if thermal runaway is likely to occur.

Batteries are susceptible to cold temperatures and lose capacity rapidly. This can be a problem on autonomous operations in cold climates, after a cold soak or a prolonged period of operation at low temperature where starting APUs or engine starter motors can be troublesome.

Secondary power distribution

Power switching

In order to reconfigure or to change the state of a system, it is necessary to switch power at various levels within the system. At the high power levels that prevail at the primary power part of the system, power switching is achieved by using high-power electromagnetic devices called contactors. These devices can switch hundreds of amperes and are used to switch generator power on to the primary busbars in both DC and AC systems. The devices may be arranged so that they magnetically latch, that is, they are magnetically held in a preferred state or position until a signal is applied to change the state. In other situations a signal may be continuously applied to the contactor to hold the contacts closed, and removal of the signal causes the contacts to open. Primary power contactors and ELCUs have been described earlier in the chapter.

For switching currents below 20 A or so, relays are generally used. These operate in a similar fashion to contactors but are lighter, simpler, and less expensive. Relays may be used widely throughout the primary electrical system, usually for switching of medium- and high-power secondary aircraft loads or services.

For lower currents still, where the indication of device status is required, simple switches can be employed. These switches may be manually operated by the crew or they may be operated by other physical means as part of the aircraft operation. Such switches are travel limit switches, pressure switches, temperature switches, and so on.

Load protection

Circuit breakers

Circuit breakers perform the function of protecting a circuit in the event of an electrical overload. Circuit breakers serve the same purpose as fuses or current limiters. A circuit breaker comprises a set of contacts that are closed during normal circuit operation. The device has a mechanical trip mechanism that is activated by means of a bimetallic element. When an overload current flows, the bimetallic element causes the trip mechanism to activate, thereby opening the contacts and removing power from the circuit. A push button on the front of the unit protrudes, showing that the device has tripped. Pushing in the push button resets the breaker, but, if the fault condition still exists, the breaker will trip again. Physically pulling the button outwards can also allow the circuit breaker to break the circuit, perhaps for equipment isolation or aircraft maintenance reasons. Circuit breakers are rated at different current values for use in differing current-carrying circuits. This enables the trip characteristic to be matched to each circuit. The trip characteristic also has to be selected to coordinate with the feeder trip device upstream. Circuit breakers are literally used by the hundred in aircraft distribution systems; it is not unusual to find 500–600 devices throughout a typical aircraft system.

Circuit breakers have a finite trip time, being a thermal device. This time must be taken into account in systems design, especially with regard to switch contacts. Under fault conditions the contacts must carry the fault load until the breaker trips. A circuit breaker compatibility test is normally specified for switch and relay contacts.

Solid state power controllers

The availability of high-power solid-state switching devices has been steadily increasing for a number of years, both in terms of variety and rating. More recent developments have led to the availability of solid-state power-switching devices that provide a protection capability as well as switching power. These devices, known as Solid-State Power Controllers, or SSPCs, effectively combine the function of a relay or switch and a circuit breaker. There are disadvantages with the devices available at present; they are readily available up to a rating of 22.5 A for use with DC loads, but the switching of AC loads may only be carried out at lower ratings and with a generally unacceptable power dissipation. Another disadvantage of SSPCs is that they are expensive and costwise may not be comparable with the relay/circuit breaker combination they replace. They are, however, predicted to be more reliable than conventional means of switching and protecting small and medium-sized electrical loads and are likely to become far more prevalent in use in some of the aircraft electrical systems presently under development. SSPCs are also advantageous when used in high duty cycle applications where a relay may wear out.

Present devices are rated at 5, 7.5, 12.5 and 22.5 A and are available to switch 28 V DC and 270 V DC. A recent paper (**7**) summarizes the development and capabilities of SSPCs and power management units embodying SSPCs.

Electrical loads

Once the aircraft electrical power has been generated and distributed, then it is available to the aircraft services. These electrical services cover a range of functions spread geographically throughout the aircraft depending upon their task. While the number of electrical services is legion, they may be broadly subdivided into the following categories:

1. Motors and actuation.
2. Lighting services.
3. Heating services.
4. Subsystem controllers and avionics systems.

Motors and actuation

Motors are obviously used where motive force is needed to drive a valve or an actuator from one position to another depending upon the requirements of the appropriate aircraft system. Typical uses for motors are:

1. Linear actuation: electrical position actuators for engine control; trim actuators for flight control systems.
2. Rotary actuation: electrical position actuators for flap/slat operation; fuel cocks.
3. Control valve operation: electrical operation of fuel control valves, hydraulic control valves, air control valves, and control valves for ancillary systems.
4. Starter motors: provision of starting for engines, APUs, and other systems that require assistance to reach self-sustaining operation.
5. Pumps: provision of motive force for fuel pumps and hydraulic pumps; pumping for auxiliary systems.
6. Gyroscope motors: provision of power to run gyroscopes for flight instruments and autopilots.

7. Fan motors: provision of power to run cooling fans for the provision of air to passengers or equipment.

Many of the applications for which electric motors are used are not continuously rated; that is, the motor can only be expected to run for a small proportion of the time. Others, such as the gyroscope and cooling fan motors may be run continuously throughout the period of operation of the aircraft, and the sizing/rating of the motor has to be chosen accordingly. The following categorizes the characteristics of the DC and AC motor types commonly used for aircraft applications.

DC motors

A DC motor is the inverse of the DC generator described earlier in this chapter. It comprises armature field windings and commutator/brushgear and is similarly self-excited. The main elements of importance in relation to motors are the speed and torque characteristics, i.e. the variations in speed and torque with load respectively. Motors are categorized by their field winding configuration (as for generators), and typical examples are series wound, shunt wound, and compound wound (a combination of series and shunt wound). Each of these types of motor offers differing performance characteristics that may be matched to the application for which they are intended.

A specialized form of series motor is the split-field motor where two sets of series windings of opposite polarity are each used in series with the armature but parallel with each other. Either one set of field windings may receive power at any one time, and therefore the motor may run bidirectionally depending upon which winding is energized. When used in conjunction with suitable switches or relays, this type of motor is particularly useful for powering loads such as fuel system valves where there may be a requirement to change the position of various valves several times during flight. Limit switches at the end of the actuator travel prevent the motor/actuator from overrunning once the desired position has been reached. Split-field motors are commonly used for linear and rotary position actuators when used in conjunction with the necessary position feedback control.

DC motors are most likely to be used for linear and rotary actuation, fuel valve actuation, and starter functions.

AC motors

AC motors used for aircraft applications are most commonly of the ‘induction motor’ type. An induction motor operates upon the principle that a rotating magnetic field is set up by the AC field current supplied to two or more stator windings (usually three-phase). A simple rotor, sometimes called a ‘squirrel cage’, will rotate under the effects of this rotating magnetic field without the need for brushgear or slip rings; the motor is therefore simple in construction and reliable. The speed of rotation of an induction motor depends on the frequency of the applied voltage and the number of pairs of poles used. The advantage of the induction motor for airborne uses is that there is always a source of constant-frequency AC power available, and for constant rated applications it offers a very cost-effective solution. Single-phase induction motors also exist, but these require a second set of phase windings to be switched in during the start phase, as single-phase windings can merely sustain and not start synchronous running.

AC motors are most likely to be used for continuous operation, i.e. those applications where motors are continuously operating during flight, such as fuel booster pumps, flight instrument gyroscopes, and air conditioning cooling fans.

Lighting

Lighting systems represent an important element of the aircraft electrical services. A large proportion of modern aircraft operating time occurs during night or low-visibility conditions. The availability of adequate lighting is essential to the safe operation of the aircraft. Lighting systems may be categorized as follows:

1. External lighting systems:
    (a) Navigation lights.
    (b) Strobe lights/high-intensity strobes/anti-collision beacons.
    (c) Landing/taxi lights.
    (d) Formation lights.
    (e) Inspection lights (wing/empennage/engine anti-ice).
    (f) Emergency evacuation lights.
    (g) Logo lights.
    (h) Searchlights (for search and rescue or police aircraft).
2. Internal lighting systems:
    (a) Cockpit/flight deck lighting (general, spot, flood, and equipment panel).
    (b) Passenger information lighting.
    (c) Passenger cabin general and personal lighting.
    (d) Emergency/evacuation lighting.
    (e) Bay lighting (cargo or equipment bays for servicing).
    (f) Floodlighting/emergency floods (usually red).
    (g) Wander lights.

Lighting is generally powered by 28 V DC or by 26 V AC provided by autotransformer from the main AC buses and is mainly achieved by means of conventional filament bulbs. Specialized lights may use 115 V AC supplies. These filaments vary from around 600 W for landing lights to a few watts for minor internal illumination uses. Some aircraft instrument panels or signs may use electroluminescent lighting which is a phosphor layer sandwiched between two electrodes, the phosphor glowing when supplied with AC power.

Internal lighting systems must be designed to provide a balanced illumination across the flight deck, especially when dimming is used at night, so that all panels and display surfaces have an even distribution of illumination. This is to avoid eyestrain and to avoid the crew concentrating on any one display at the expense of any other.

Heating

The use of electrical power for heating purposes on aircraft can be extensive. The highest power usage relates to electrically powered anti-icing or de-icing systems which can consume many tens of kVAs. This power does not have to be frequency stable and can be frequency wild and therefore much easier and cheaper to generate. Anti/de-icing elements are frequently used on the tailplane and fin leading edges, intake cowls, propellers, and spinners. The precise mix of electrical and hot air (using bleed air from the engines) anti/de-icing methods varies from aircraft to aircraft. Electrical anti/de-icing systems are high current consumers and require controllers to time, cycle, and switch the heating current between heater elements to ensure optimum use of the heating capability and to avoid local overheating.

Windscreen heating is another important electrical heating service. In this system the heating element and the controlling thermostat are embedded in the windscreen itself. A dedicated controller maintains the temperature of the element at a predetermined value which ensures that the windscreen is kept free of ice at all times.

Subsystem controllers and avionics systems

As aircraft have become increasingly complex, so has the sophistication of the aircraft subsystems also increased. Many have dedicated controllers for specific system control functions. For many years the aircraft avionics systems, embracing display, communication, and navigation functions, have been packaged into Line Replaceable Units (LRUs) which permit rapid removal should a fault occur. Many of the aircraft subsystem controllers are now packaged into similar LRUs owing to increased complexity and functionality and for the same reasons of rapid replacement following a failure. These LRUs may require DC or AC power depending upon their function and modes of operation. Many may utilize dedicated internal power supply units to convert the aircraft power to levels better suited to the electronics that require ±15 V DC and +5 V DC and also, more recently, +3.3 V DC. These LRUs represent fairly straightforward and, for the most part, fairly low-power, non-reactive loads. However, there are many of them and a significant proportion may be critical to the safe operation of the aircraft. Therefore, two important factors arise: first, the need to provide independence of function by distribution of critical LRUs across several aircraft busbars, powered by both DC and AC supplies to prevent a single power supply failure or fault leading to a loss of critical functions. Second, the need to provide adequate sources of emergency power such that, should a dire emergency occur, the aircraft has sufficient power to supply critical services to support a safe return and landing, including the safe evacuation of the crew and passengers.

Ground power

For much of the period of aircraft operation on the ground, a supply of power is needed. Ground power may be generated by means of a motor–generator set where a prime motor drives a dedicated generator supplying electrical power to the aircraft power receptacle.

The usual standard for ground power is 115 V AC three-phase 400 Hz, that is, the same as the aircraft AC generators. In some cases, and this is more the case at major airports, an electrical conversion set adjacent to the aircraft gate supplies 115 V AC three-phase power that has been derived from the national electricity grid. The description given later in this chapter of the Boeing 767 system explains how ground power can be applied to the aircraft by closing the EPC.

The aircraft system is protected from substandard ground power supplies by means of a ground power monitor. This ensures that certain essential parameters are met before enabling the EPC to close. In this way the ground power monitor performs a similar function to a main generator GCU. Typical parameters that are checked are undervoltage, overvoltage, frequency, and correct phase rotation.

Emergency power generation

In certain emergency conditions the typical aircraft power generation system already described may not meet all the airworthiness authority requirements, and additional sources of power generation may need to be used to power the aircraft systems. The aircraft battery offers a short-term power storage capability, typically up to 30 min. However, for longer periods of operation the battery is insufficient. The operation of twin-engined passenger aircraft on Extended-Range Twin OperationS (ETOPS) flights means that the aircraft must be able to operate on one engine while up to 180 min from an alternative or diversion airfield. This has led to modification of some of the primary aircraft systems, including the electrical system, to ensure that sufficient integrity remains to accomplish the 180 min diversion while still operating with acceptable safety margins. The three standard methods for providing back-up power on civil transport aircraft are:

- Ram Air Turbine (RAT).
- Back-up converters.
- Permanent Magnet Generators (PMGs).

Ram Air Turbine

The Ram Air Turbine (RAT) is deployed when most of the conventional power generation system has failed or is unavailable for some reason, such as total engine flame-out or the loss of all generators. The RAT is an air-driven turbine, normally stowed in the aircraft ventral or nose section, that is extended either automatically or manually when the emergency commences. The passage of air over the turbine is used to power a small emergency generator of limited capacity, usually enough to power the crew's essential flight instruments and a few other critical services (see Fig. 4.14). Typical RAT generator sizing may vary from 5 to 15 kV A depending upon the aircraft. The RAT also powers a small hydraulic power generator for similar hydraulic system emergency power provision. Once deployed, the RAT remains extended for the duration of the flight and cannot be restowed without maintenance action on the ground. The RAT is intended to furnish the crew with sufficient power to fly the aircraft while attempting to restore the primary generators, reaching the engine relight envelope, or

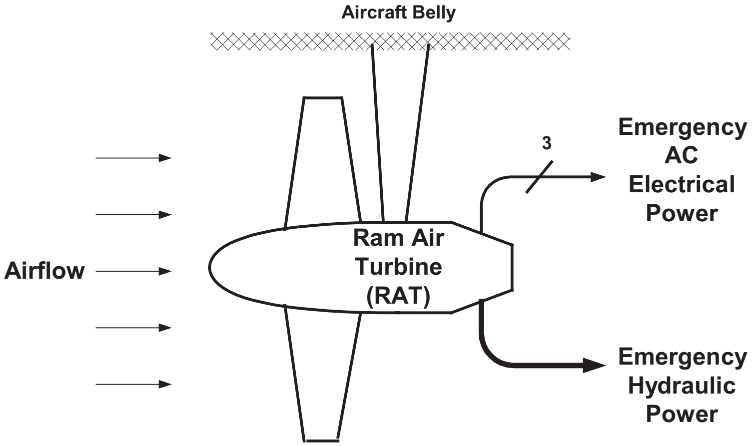


Fig. 4.14 *Ram Air Turbine (RAT)*

carry out a diversion to the nearest airfield. It is not intended to provide significant amounts of power for a lengthy period of operation. Owing to the importance of the supply of emergency power, the RAT will normally be expected to deploy and come on line in around 5 s.

Back-up converters

The requirements for ETOPS have led to the need for an additional method of back-up power supply, short of deploying the RAT which should occur in only the direst emergency. The use of a back-up converter satisfies this requirement and is used on the B777. Back-up generators are driven by the same engine accessory gearbox but are quite independent of the main IDGs (see Fig. 4.15).

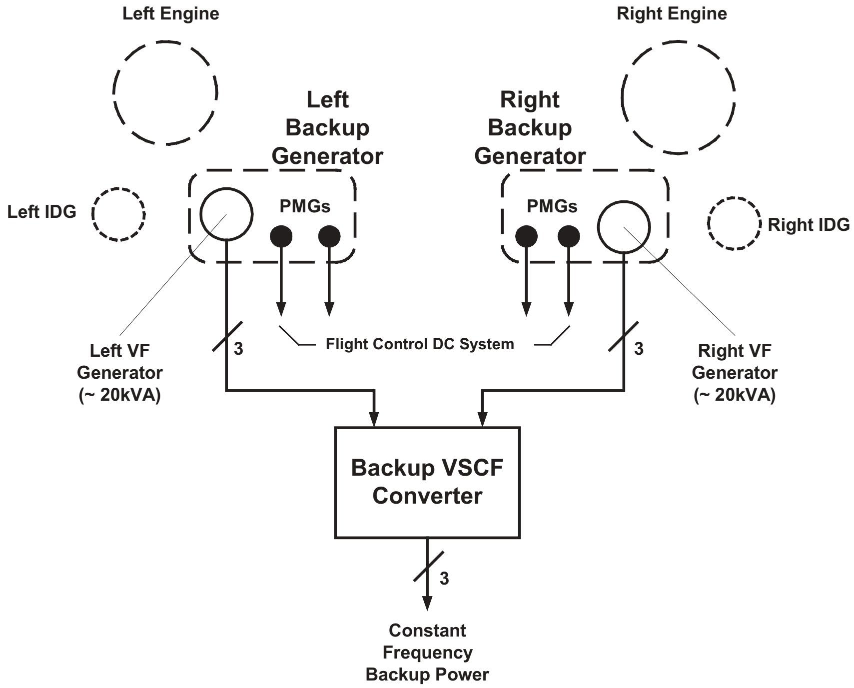


Fig. 4.15 Simplified back-up VSCF converter system

The back-up generators are VF and therefore experience significant frequency variation as engine speed varies. The VF supply is fed into a back-up converter which, using the DC link technique, first converts the AC power to DC by means of rectification. The converter then synthesizes three-phase 115 V AC 400 Hz power by means of sophisticated solid-state power switching techniques. The outcome is an alternative means of AC power generation that may power some of the aircraft AC busbars; typically, the 115 V AC transfer buses in the case of the Boeing 777. In this way, substantial portions of the aircraft electrical system may remain powered even though some of the more sizeable loads such as the galleys and other non-essential loads may need to be shed by the Electrical Load Management System (ELMS).

Permanent magnet generators

The use of PMGs to provide emergency power has become prominent over the last decade or so. As can be seen from the description of the back-up converter above, the back-up generator hosts PMGs which may supply several hundred watts of independent generated power to the flight control DC system where the necessary conversion to 28 V DC is undertaken. It was already explained earlier in the chapter that AC generators include a PMG to bootstrap the excitation system. Additionally, PMGs – also called Permanent Magnet Alternators (PMAs) – are used to provide dual independent on-engine supplies to each lane of the FADEC. As an indication of future trends, on an aircraft such as the B777 there are a total of 13 PMGs/PMAs across the aircraft critical control systems – flight control, engine control, and electrical systems (see Fig. 4.16.)

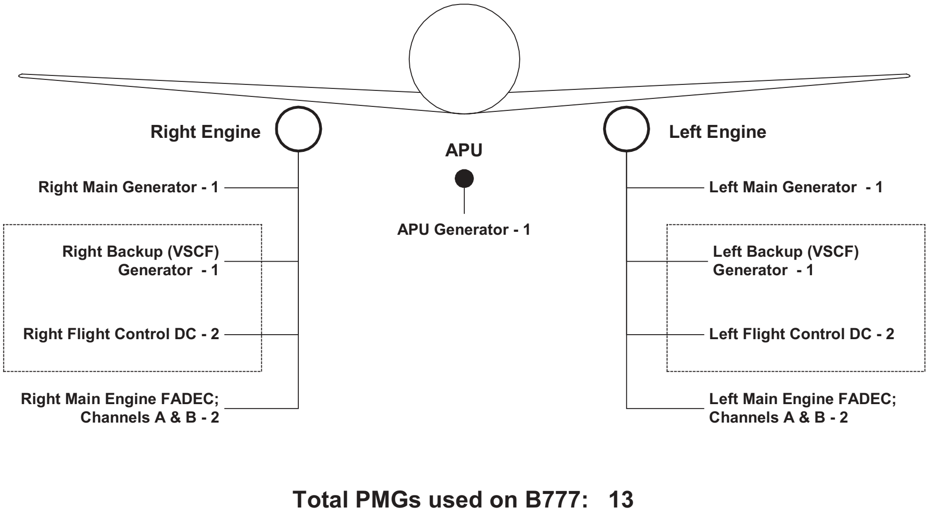


Fig. 4.16 *Boeing 777 PMG/PMA complement*

Reference (**8**) is an early paper describing the use of a PMG, and reference (**9**) describes some of the work undertaken in looking at higher levels of PMG power generation.

Typical aircraft DC system

A generic distribution system is shown in Fig. 4.17. In this case a twin 28 V DC system is shown which might be typical for a twin-engine commuter aircraft requiring less than ~12 kW per channel. This type of system could typically also be used on smaller business jets.

The main elements of this electrical system are:

- Two 28 V DC generators operating in parallel to supply No. 1 and No. 2 main DC busbars. These busbars feed the non-essential DC services.
- Two inverters operate, one off each of the DC busbars, to provide 115 V AC 400 Hz to non-essential AC services.
- Both No. 1 and No. 2 busbars feed power to a centre or essential busbar which provides DC power for the aircraft essential DC services. An inverter powered off this busbar feeds essential 115 V AC loads. A 28 V DC external power source may

also feed this busbar when the aircraft is on the ground without the engines running.

- The aircraft battery feeds the battery busbar from which vital services are fed. The battery may also be connected to the DC essential busbar if required.

To enable a system such as this is to be provided with suitable protection, several levels of power switching and protection are required:

- Primary power generation protection of the type described earlier, which includes reverse current and under/overvoltage protection under the control of the voltage regulator. This regulator controls the generator feed contactors which switch the generator output onto the No. 1/No. 2 DC busbars.
- The protection of feeds from the main buses, i.e. the protection of the feeds to the essential busbar. This may be provided by a circuit breaker, or a 'smart' contactor may be used to provide the protection.
- The use of circuit breakers to protect individual loads or groups of loads fed from the supply or feeder busbars.

The cardinal principle is that fault conditions should be contained with the minimum of disruption to the electrical system. Furthermore, faults that cause a load circuit breaker to trip should not cause the next level of protection to trip, causing a cascade failure. Thus, the trip characteristics of all protection devices should be coordinated to ensure that this does not occur.

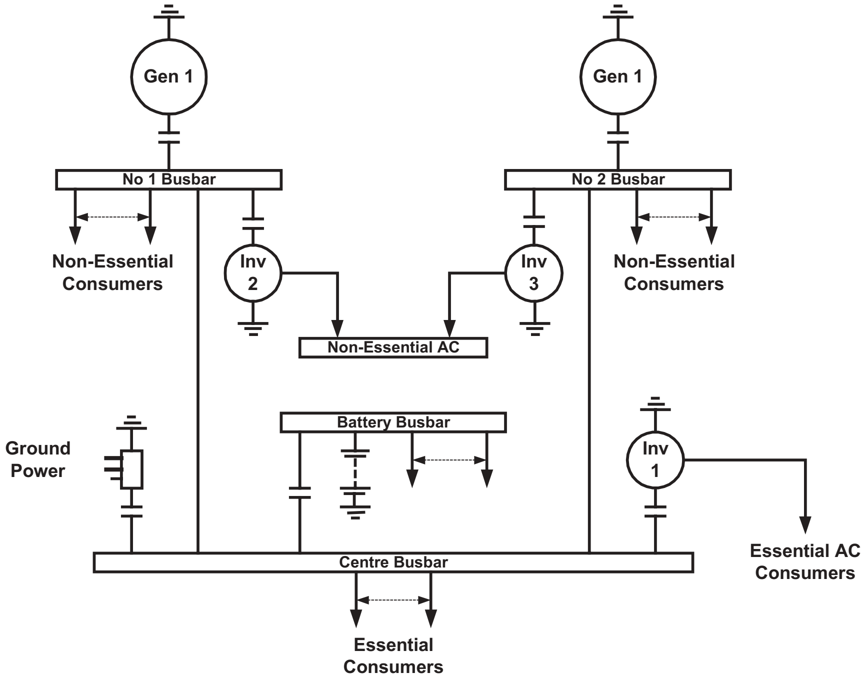
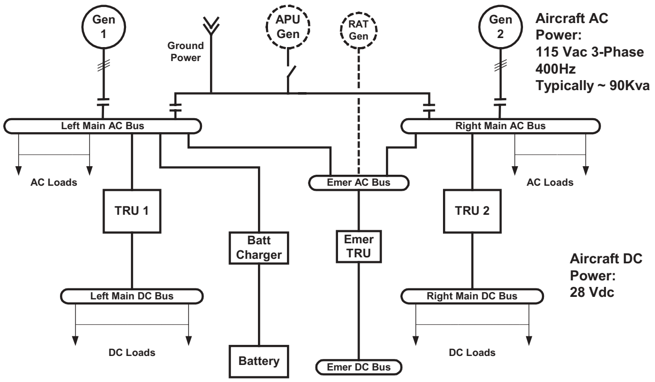


Fig. 4.17 *Typical twin 28 VDC system*

Typical civil transport aircraft systems

A typical civil transport electrical power system is shown in Fig. 4.18. This is a simplified example of a twin-engine transport aircraft and is useful to describe some generic aspects of the system.



Notes:

Most High power loads are 115 Vac

Most Electronic/Avionic Loads are 28 Vdc

Fig. 4.18 *Typical AC electrical power system*

The key features of this system are common to many electrical power systems:

- Two 115 V AC, three-phase, 400 Hz generators feed left and right main AC buses.
- There is an ability to feed both AC buses on the ground from an external power source or from an APU generator. In some systems, the APU can be started in flight to provide an additional source of power.
- A RAT is provided to generate emergency power.
- Two TRUs convert 115 V AC to 28 V DC to supply the left and right DC buses. In most systems the DC supply is unregulated.
- A battery charger maintains battery charge. A second battery (not shown) is usually provided to start the APU.
- In some systems an emergency TRU may be provided to derive an emergency DC supply.
- Some systems may use a static inverter to supply back-up single- or three-phase AC, e.g. for essential flight instruments.

This generic architecture is also expanded to include four generators and AC buses on a four-engine aircraft.

Civil aircraft electrical system examples

To examine the different architectures used on board modern transport aircraft, the following examples will be described:

- Boeing 767.
- Boeing 747-400.
- Airbus A330/340.
- Boeing 777.
- Airbus A380.

This list is chronological in terms of when the aircraft was certified, with the exception of the A380 which is in early development. Therefore, the illustrations represent a historical perspective as well as outlining differing implementations.

Boeing 767

A simplified representation of the Boeing 767 aircraft electrical power system which is described in detail in reference (**10**) is given in Fig. 4.19. The primary AC system comprises identical left and right channels. Each channel has an Integrated Drive Generator (IDG) driven from the accessory gearbox of the respective engine. Each AC generator is a three-phase 115 VAC 400 Hz machine producing 90 kVA and is controlled by its own Generator Control Unit (GCU). The GCU controls the operation of the GCB, closing the GCB when all operating parameters are satisfactory and opening the GCB when fault conditions prevail. Two Bus Tie Breakers (BTBs) may be closed to tie both buses together in the event that either generating source is lost. The BTBs can also operate in conjunction with the External Power Contactor (EPC) or the Auxiliary Power Breaker (APB) to supply both main AC buses with power, or the 90 kVA APU generator may also feed the ground handling and ground servicing buses by means of changeover contactors. The control of the BTBs, EPB, and the ground handling and ground servicing contactors is carried out by a unit called a Bus Power Control Unit (BPCU).

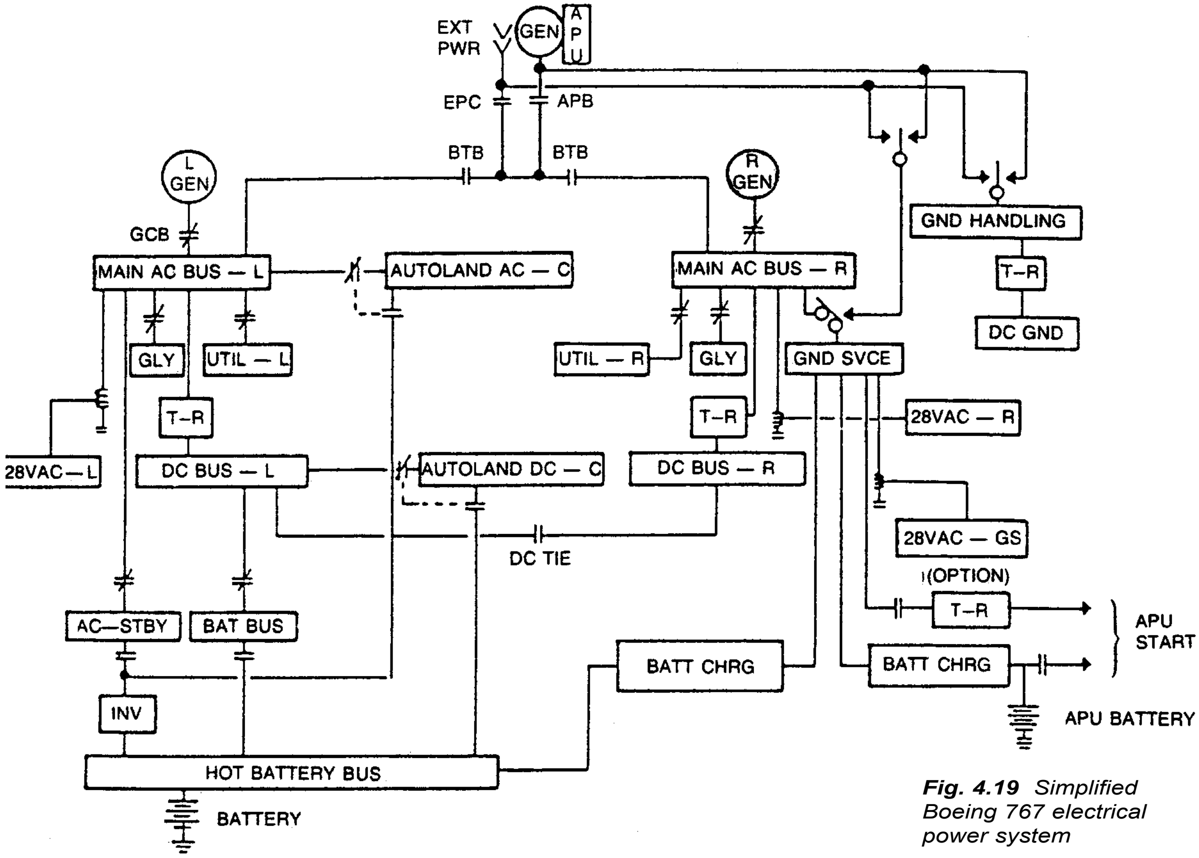


Fig. 4.19 Simplified Boeing 767 electrical power system

The APU may also be used as a primary power source in flight on certain aircraft in the event that either left or right IDG is lost.

Each of the main AC buses feeds a number of sub-buses or power conversion equipment. TRUs convert 115 V AC to 28 V DC to feed the left and right DC buses respectively. In the event that either main AC bus or TRU should fail, a DC Bus Tie Contactor (BTC) closes to tie the left and right DC buses together. The main AC buses also feed the aircraft galleys (a major electrical load) by means of 'smart' contactors. The utility buses are also fed via contactors from each of the main AC buses. In the event of a major electrical system failure, the galley loads and non-essential utility bus loads may be shed under the supervision of the BPCU. Both main AC buses feed 26 V AC buses via autotransformers and 28 V DC buses via TRUs. Other specific feeds from the left main AC bus are a switched feed to the autoland AC bus (interlocked with a switched feed from the standby inverter) and a switched feed to the AC standby bus. Dedicated feeds from the right main AC bus are via the air/ground changeover contactor to the ground services bus feeding the APU TRU and battery charger, and via the main battery charger to the hot battery bus. The left DC bus also supplies a switched feed to the autoland DC bus (interlocked with a switched feed from the hot battery bus). The hot battery bus also has the capability of feeding the autoland AC bus via the standby inverter.

To the uninitiated this may appear to be overly complex, but the reason for this architecture is to provide three independent lanes of AC and DC conversion for use during autoland conditions. These are:

1. Left main AC bus (disconnected from the autoland AC bus) via the left TRU to the left DC bus (which in this situation will be disconnected from the autoland DC bus).
2. Right main AC bus via the right TRU to the right DC bus.
3. Right main AC bus via the ground services bus and main battery charger to the hot battery bus and thence to the autoland DC bus (now disconnected from the left DC bus). Also, from the hot battery bus via the standby inverter to the autoland AC bus (now disconnected from the left main AC bus).

This provides the three independent lanes of electrical power required. It might be argued that two lanes are initially derived from the right main AC bus and therefore the segregation requirements are not fully satisfied. In fact, as the hot battery is fed from the main aircraft battery, this represents an independent source of stored electricity, provided that an acceptable level of charge is maintained. This latter condition is satisfied as the battery charger is fed at all times when the aircraft is electrically powered from the ground services bus from either an air or ground source. The battery capacity is such that all standby loads may be powered for 30 min following primary power loss.

Boeing 747-400

The Boeing 747-400 system comprises four IDGs and two APU generators and has the capability of accepting two external power sources, as shown in Fig. 4.20. All four 115 V AC buses may be tied together via the BTBs and the Split System Breaker (SSB) to provide parallel operation of all four 90 kV A generators. The two APU generators and external ground power sources may be connected via APU power breakers APB 1 and APB 2 and external power contactors XPC 1 and XPC 2 controlled via Bus Control Units (BCUs) I and 2 respectively to apply APU or external ground power to all four

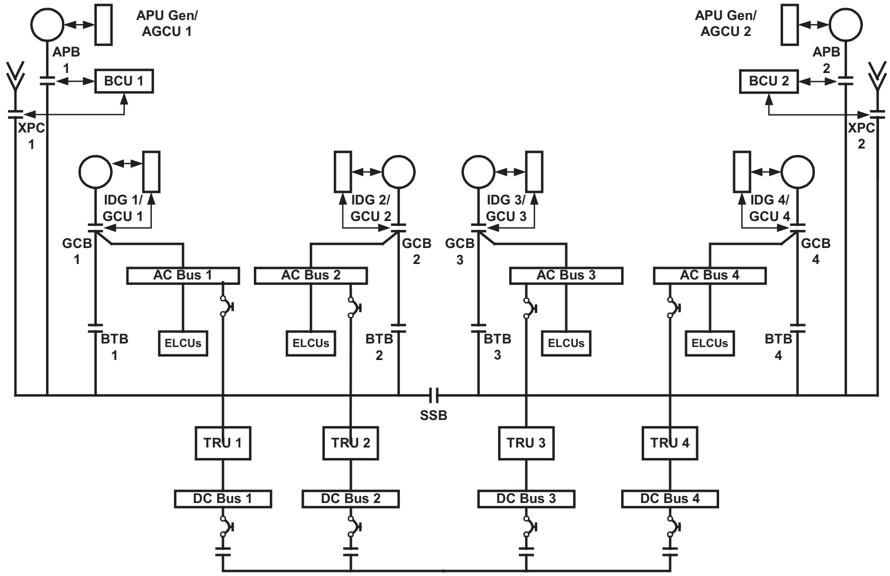


Fig. 4.20 *Boeing 747-400 electrical power system (Boeing)*

AC buses. Each 115 V AC bus feeds a 28 V DC bus via a dedicated TRU.

Each GCU provides the following control functions:

- Generator Control Relay (GCR) control and indication.
- Generator Control Breaker (GCB) control.
- BTB control and indication.
- Autoparalleling/dead bus control.
- Auto bus isolation (autoland) control.
- DC isolation relay control.
- Cooling valve control.
- Utility load management.
- Bus off indication for the Engine Indication and Crew Alerting System (EICAS).
- System no-break power transfers.

Each BCU is capable of performing the following:

- APU generator field control and indication.
- APB control and power status.
- XPC power status and indication.
- Ground service and ground handling bus control (not shown in Fig. 4.20 for clarity).
- Utility and galley load management control.
- DC load management.
- Auto bus isolation (autoland) control.
- SSB control.
- APU and AGCU interface.
- Autoparalleling.
- Frequency reference.
- DC and standby system control.

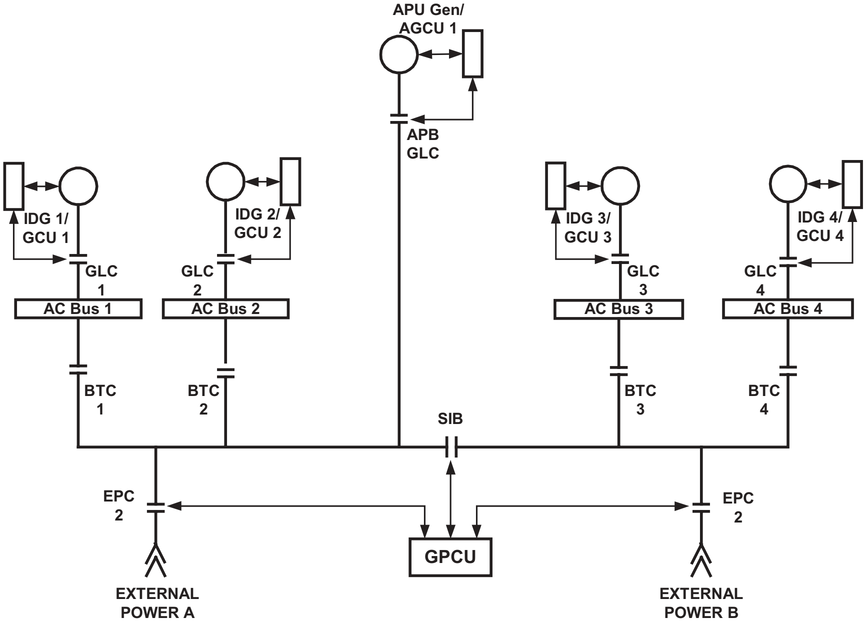
Power transfers with the exception of APU to APU and ground power sources are no-break transfers where generator phases are synchronized to meet the paralleling requirements. The four GCUs and the two BCUs are interconnected via ARINC 429 data buses such that system control and fault may be readily interchanged and electrical system data gathered for display on the aircraft EICAS displays.

For a more detailed description, see references (**11**) and (**12**).

Airbus A330/340

The A340 electrical power system (Fig. 4.21) is somewhat similar to that for the Boeing 747-400, with the exception that only one APU power source is provided. The names of the primary power breakers are different though their function is the same. The Generator Line Contactors (GLCs) connect the IDG and APU generators to their respective bus when all the necessary power quality and protection functions are satisfied. The Bus Tie Contactor (BTC) connects power to/from the AC bus to other power sources. The External Power Contractors (EPCs) connect the external power sources, and the System Interconnect Breaker (SIB) allows the two halves of the system to be connected as required. The A330 power architecture is similar, the difference being that only two generators are provided as the A330 is a twin-engine aircraft.

Fig. 4.21 *A330/340 electrical power system (Airbus)*



The A330/340 systems also have the ability to achieve no-break power transfers between the main generators and the APU generator, though not between the external ground power source and the main generators/APU. An IDG to IDG transfer is effected once either of the IDG GCUs recognizes that a transfer is to take place. A common 400 Hz reference is sent from the Ground Power Control Unit (GPCU) to both GCUs which both report when synchronization has occurred. At this point the appropriate breakers are opened or closed to effect the power transfer which occurs within 100 ms. As the APU generator does not have the ability to trim generator frequency, an IDG to

APU transfer can be completed, but an APU to IDG power transfer cannot. Similarly, an IDG to ground power transfer may be carried out, but not vice versa.

It is possible, by trimming the APU speed governer, for the APU generator to be synchronized with the external power source to complete a no-break power transfer between the external source and the APU. During normal operation the IDGs are not paralleled as is the case on the Boeing 747-400.

The digital GCUs are common between the A330/A340 and A321. However, because of the need to maintain certification commonality between the A321 and A320, with the latter not having a no-break power capability, the no-break power capability on the A321 is not used. For a more detailed explanation, see reference (**13**)

Boeing 777

The Boeing 777 electrical system is shown in Fig. 4.22, albeit with a different perspective. In this diagram an emphasis is placed upon the top-level LRUs which comprise the electrical system and which are located in the electrical equipment bay. By this means, some idea of the size of the system may be gained. For another perspective on the Boeing 777 Electrical Load Management System (ELMS), see Figs 11.6–11.8. in Chapter 11 and the accompanying text.

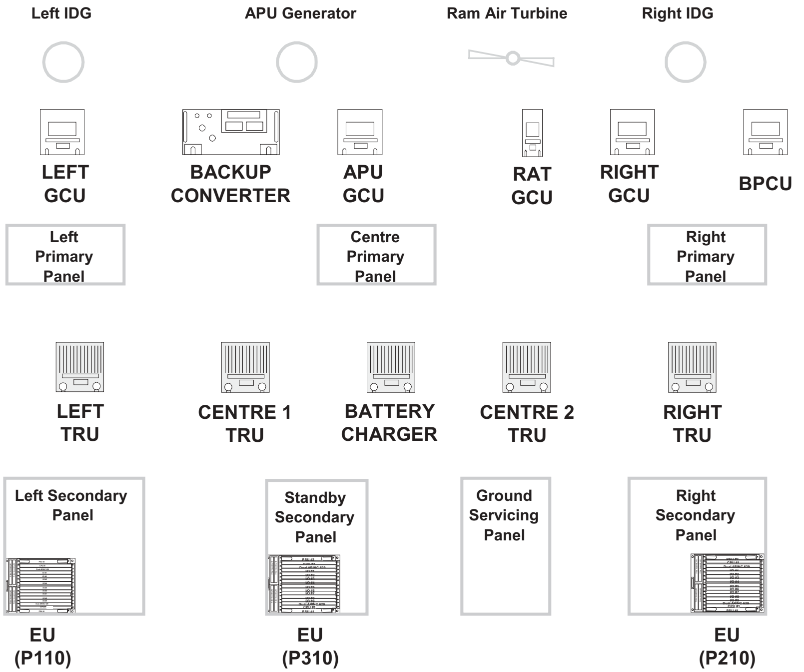


Fig. 4.22 *Boeing 777 electrical power system (Boeing)*

The Boeing 777 units comprise:

- Left, right, and APU GCUs.
- A Bus Power Control Unit (BPCU).
- A RAT GCU.
- Back-up converter which provides two channels of 20 kV A provided by a DC-link VSCF converter.

- Four TRUs.
- A battery charger
- Three Electronic Units (EUs) located in three of the secondary power panels as part of the ELMS described below.

For a comprehensive description of the B777 electrical power system, see reference (**14**).

Electrical Load Management System (ELMS)

The Boeing 777 ELMS developed and manufactured by Smiths Aerospace sets new standards for the Industry in terms of electrical load management. The general layout of the ELMS is shown in Fig. 4.22. The system represents the first integrated electrical power distribution and load management system for a civil aircraft.

The system comprises seven power panels, three of which are associated with primary power distribution:

- P100 – left primary power panel distributes and protects the left primary loads.
- P200 – right primary power panel distributes and protects the right primary loads.
- P300 – auxiliary power panel distributes and protects the auxiliary primary loads.

The secondary power distribution function is undertaken by four secondary power panels:

- P110 – left power management panel distributes and protects power and controls loads associated with the left channel.
- P210 – right power management panel distributes and protects power and controls loads associated with the right channel.
- P310 – standby power management panel distributes and protects power and controls loads associated with the standby channel.
- P320 – ground servicing/handling panel distributes and protects power associated with ground handling.

Load management and utilities systems control is exercised by mean of EUs mounted within the P110, P210, and P310 power management panels. Each of these EUs interfaces with the left and right aircraft systems ARINC 629 digital data buses and contains a dual redundant architecture for reasons of dispatch availability. The EUs contain a modular suite of LRMs that can readily be replaced when the door is open. A total of six module types are utilized to build a system comprising an overall complement of 44 modules across the three EUs. This highly modular construction with multiple use of common modules reduced development risk and resulted in highly accelerated module maturity at a very early stage of airline service. LRMs typically have a mature in-service MTBF ~ 200 000 h, as reported in reference (**15**).

The load management and utilities control features provided by ELMS are far in advance of any equivalent system in airline service. Approximately 17–19 electrical load control units ELCUs – depending upon aircraft configuration – supply and control loads directly from the aircraft main AC buses. These loads can be controlled by the intelligence embedded within the ELMS EUs. A major advance is the sophisticated load shed/load optimization function which closely controls the availability of functions should a major electrical power source fail or become unavailable. The system is able to reconfigure the loads to give the optimum distribution of available power. In the event that electrical power is restored, the system is able to reinstate loads according to a number of different schedules. The system is therefore able to make the optimum use of power at all times rather than merely shed loads in an emergency.

The benefits conferred by ELMS have proved to be considerable, with significant reduction in volume, wiring and connectors, weight, relays, and circuit breakers. On account of the in-built intelligence, the use of digital data buses, maintainability features, and the extensive system Built-In Test (BIT), the system build and on-aircraft test time turned out to be ~30per cent of that experienced by contemporary systems.

Airbus A380

The Airbus A380, now in the early stages of development, will be the first large aircraft for many years to use VF electrical power. The aircraft will utilize a 150 kVA VF generator located on each of the four engines. Power from these generators will be organized to provide electrical power in two channels, as shown in Fig. 9.37 in Chapter 9. The flight control system will be a novel combination of conventional hydraulic actuators, Electro Hydrostatic Actuators (EHAs), and Electric Back-up Hydraulic Actuators (EBHAs) as also described in Chapter 9.

As described earlier in this chapter, VF electrical power offers a cheap solution for generating large quantities of power, but the variable nature of the power can cause problems for some of the aircraft loads, particularly inductive and motor loads.

More-Electric Aircraft (MEA)

For at least the last ten years a number of studies have been under way in the United States that have examined the all-electric aircraft. As stated earlier, aircraft developed in the United Kingdom in the late 1940s/early 1950s, such as the V-Bombers, utilized electric power to a greater extent than present-day aircraft. In the 1980s, a number of studies promoted by the NASA and US Navy and US Air Force development agencies, and undertaken by Lockheed and Boeing, examined the concept in detail. The concept addresses more energy-efficient ways of converting and utilizing aircraft power in the broadest sense and therefore has a far-reaching effect upon overall aircraft performance.

Electrical system displays

The normal method for displaying electrical power system parameters to the flight crew has been via dedicated control and display panels. On a fighter or twin-engined commuter aircraft the associated panel is likely to be fairly small. On a large transport aircraft the electrical systems control and display would have been achieved by a large systems panel forming a large portion of the flight engineer's panel showing the status of all the major generation and power conversion equipment. With the advent of two crew flight deck operations, of which the Boeing 757, 767, 747-400, and Airbus A320, and indeed most modern aircraft, are typical examples, the electrical system selection panel was moved into the flight crew overhead panel. EICAS or Electronic Centralized Aircraft Monitor (ECAM) systems now permit the display of a significant amount of information by the use of:

- Synoptic displays.
- Status pages.
- Maintenance pages.

These displays show in pictorial form the system operating configuration together with the status of major system components, key system operating parameters, and any degraded or failure conditions that apply. The maximum use of colour, as described in Chapter 7, greatly aids the flight crew in assimilating the information displayed. The overall effect is vastly to improve the flight crew/system interface, giving the pilots a better understanding of the system operation while reducing the crew workload.

Aircraft wiring

Aircraft electrical wiring represents the sinew that ties the aircraft electrical system and all its recipient equipment together and as such requires a lot of attention. Wiring is usually categorized by different types, depending on the type of power or signal being carried. Typical categorizations are:

- Generation – generation power feeders.
- Power supply – loads greater than 15 A.
- Miscellaneous – non-sensitive wires.
- Sensitive – interference-sensitive wires.
- Audio – audio.
- Co-axial – antenna-related wiring.
- Bonding and earthing.

Wiring is also segregated in differing wiring runs or routes so that important functions are not carried in the same routing. This physical segregation prevents all of the key wiring being damaged if localized damage occurs by a fire or overheating. Certain critical wires such as the engine power feeders, fire detection and suppression, and hydraulic shut-off solenoids may be routed throughout the aircraft without a cable break to assure integrity and prevent breakdown or arcing within a connector. Routing will also be organized such that sensitive signal cable runs do not run alongside power-emitting cable types. Sensitive signal wires will also be screened to prevent unwanted signal interactions.

The standards of bonding and earthing are carefully controlled so that unwanted effects do not occur. Bonding and earthing leads must be kept as short as possible, and bonding points must be made to provide the lowest practicable resistance to ground (fuselage). This is to reduce the risk of personnel hazard under fault conditions (electrical shock – potentially lethal with high voltages), and to reduce the possibility of earth loops and elevated common rail voltages which will lead to impaired or unpredictable system performance.

The purpose of the load distribution protection devices – circuit breakers and SSPCs – is to protect the aircraft wiring and not the load, since, once installed in the aircraft, wiring is virtually impossible to replace. Wire sizes are calculated to ensure that the voltage drop between the busbar and the equipment terminal does not exceed 2 V. This calculation must take into account the self-heating effect of current-carrying cables in bundles which leads to higher resistance and hence a higher voltage drop.

A considerable amount of effort is being exercised in surveying and assessing aged aircraft wiring to determine the impact of age on insulation and wire condition, insulation breakdown, and circuit breaker and fuse performance. This is to estimate how much risk there is of a short circuit and arcing leading to fire or the risk of igniting inflammable vapours in enclosed bays. This is a serious potential hazard as increasing reliability is leading to longer in-service lives of aircraft.

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CHAPTER 5

Sensors

The primary autonomous sensing methods available to the modern civil airliner are:

- Air data.
- Magnetic.
- Inertial.
- Radar sensors.

Each of the on-board sensor families have their own attributes, and associated strengths and shortcomings. A modern system will take account of these attributes, matching the benefits of one particular sensor type against the deficiencies of another. On-board autonomous sensors are also used in conjunction with external navigation aids and systems to achieve the optimum performance for the navigation system. As will be seen, the capabilities of modern integrated navigation systems blend the inputs of multisensor types to attain high levels of accuracy that permit new navigation and approach procedures to be used. Indeed, in the crowded skies that prevail today, such highly integrated systems are becoming essential to assure safe and smooth traffic flow.

Air data

Air data, as the name suggests, involve the sensing of the medium through which the aircraft is flying. Typical sensed parameters are dynamic pressure, static pressure, rate of change in pressure, and temperature. Derived data include barometric altitude (ALT), indicated airspeed (IAS), vertical speed (VS), Mach (M), Total Air Temperature (TAT), and True AirSpeed (TAS). Static Air Temperature (SAT) is derived. The simplest system provides ALT and IAS as a minimum, but modern jet aircraft require Mach, VS, maximum operating speed, V_{mo}, maximum operating Mach, M_{mo}, SAT, TAT, and TAS to satisfy the aircraft requirements. The evolution of the high-performance commercial and business jet aircraft of today, together with an increase in traffic on congested routes, has significantly influenced the design of the air data system in the following ways:

- By extending the dynamic range of the sensors involved with higher altitudes, higher airspeeds, and greater temperatures.
- By increasing the use of air data on-board the aircraft, not just for navigation but for engine control, flight control, and a whole range of other aircraft subsystems.
- The adoption of higher cruise altitudes has introduced a more severe environment for equipment located outside the pressurized cabin. Increasing complexity, density, and functional requirements have also led to more complexity within the cabin.
- Demand for reduced vertical separation minima requires higher accuracy of height sensing and methods of maintaining height within strict limits.

Air data pressure parameters are sensed by means of pitot static probes and static sensors as shown in Fig. 5.1. Dynamic pressure is the head of pressure created by the forward movement of the aircraft during flight. The dynamic pressure varies according to the square of the forward velocity of the aircraft. Static pressure is the local pressure surrounding the aircraft at a given altitude and may be used to determine the aircraft altitude. Static pressure may be measured by means of ports in the side of the pitot static probe or by means of static ports in the side of the aircraft skin. The exact relationship between these parameters is shown later.

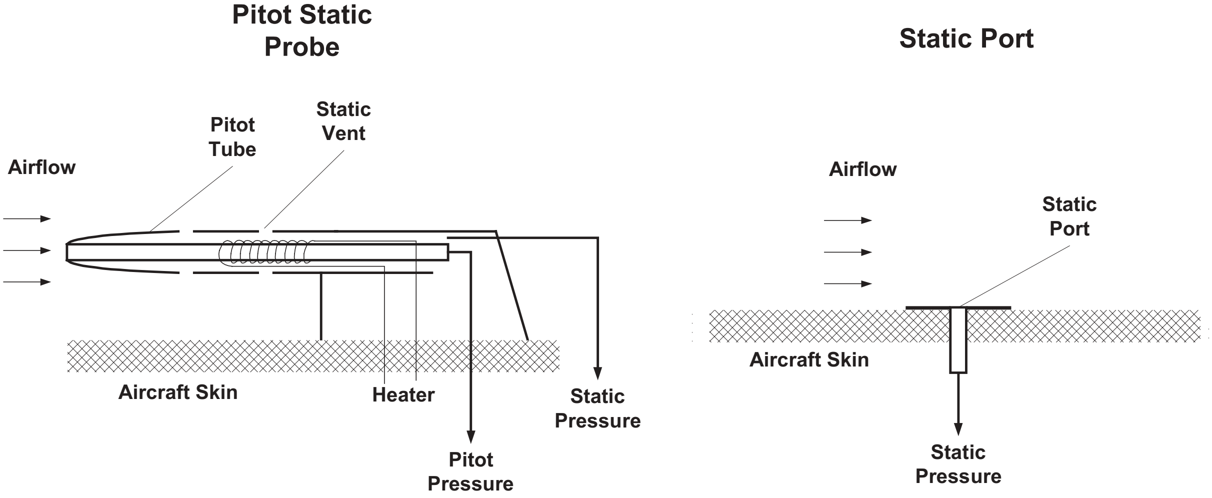


Fig. 5.1 Air data probes

There are several types of error that affect these static pressure sensors. To avoid errors when the aircraft yaws and errors due to changes in the aircraft angle of attack, static ports are located on both sides of the aircraft. There is inevitably some error associated with the less than ideal positioning of the static ports – this is known as 'static source error' and will need correction as described later.

Temperature sensing involves positioning a probe in the airflow and sensing the change in resistance associated with temperature (see Fig. 5.2). Outside Air Temperature (OAT) affects aircraft performance in a number of ways. During take-off it directly affects the thrust available from the engines and the lift due to air density, both of which can significantly affect the aircraft take-off distance and operational

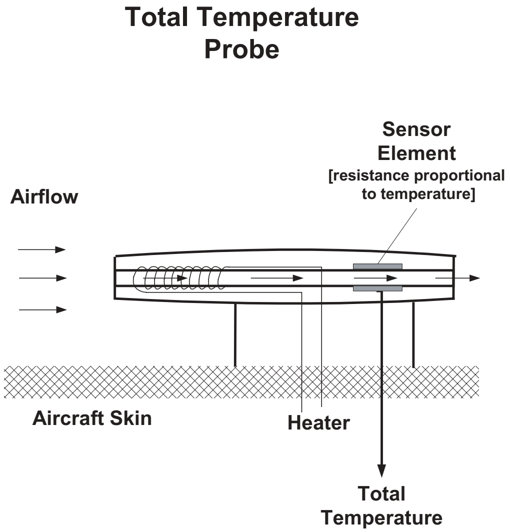


Fig. 5.2 *Total temperature probe*

margins. In the cruise, engine performance and fuel consumption are affected, and TAT needs to be calculated. In adverse weather, SAT indicates the potential for icing, while TAT is closer to the leading edge temperature and is more indicative of icing accretion. The OAT has to be corrected to represent SAT or TAT.

Both pitot and total temperature probes are susceptible to icing and so are equipped with heating elements to prevent this from happening without affecting the accuracy of the sensing elements. Since the blockage of a pneumatic pipe will lead to erroneous air data, the correct operation of the heating element needs to be continuously monitored.

In a basic air data instrumentation system, the combination of sensed pitot and static pressure may be used to derive aircraft data as shown in Fig. 5.3. By using the capsule

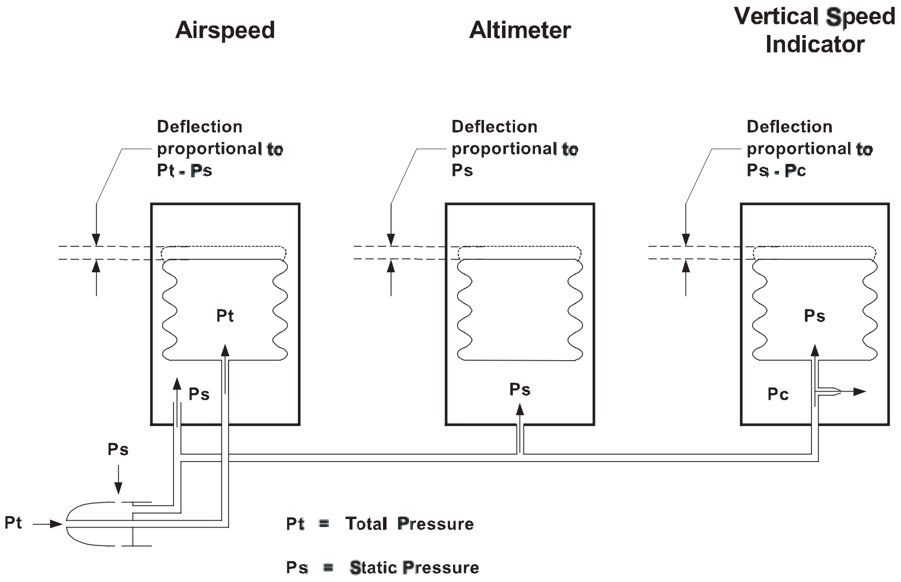


Fig. 5.3 *Use of pitot and static pressure to derive simple indications*

arrangement shown in Fig. 5.3, dynamic pressure is fed into the capsule while static pressure is fed into the case surrounding the capsule. The difference between these two parameters, represented by the deflection of the capsule, represents the aircraft airspeed. This permits airspeed to be measured.

In the centre capsule configuration, static pressure is fed into the case of the instrument while the capsule itself is sealed. Here, capsule deflection is proportional to changes in static pressure and therefore aircraft altitude. This allows aircraft barometric altitude to be measured. A typical aircraft altimeter is shown in Fig. 5.4. In the arrangement shown in the right of Fig. 5.3, static pressure is fed into the capsule. It is also fed via a calibrated orifice into the sealed case surrounding the capsule. In this situation the capsule deflection is proportional to the rate of change in altitude. This permits the aircraft rate of ascent or descent to be measured.

Fig. 5.4 *Typical mechanical altimeter display*

Originally, the portrayal of aircraft airspeed, altitude, and rate of change in altitude was accomplished using discrete instruments: airspeed indicator, altimeter, and Vertical Speed Indicator (VSI). In these simple instruments any scaling required was accomplished by means of the mechanical linkages.

Determination of altitude from pressure measurements is based upon a standard atmosphere in which pressure, density, and temperature are functions of altitude. The altitude resulting from these calculations is called the pressure altitude and represents the altitude above sea level under these standard atmospheric conditions. As a standard atmosphere rarely exists, certain steps are necessary to establish other useful and reliable operating datums for aircraft altitude and height. The particular example of an altimeter shown in Fig. 5.4 has a combination of rolling digits in the centre of the display, together with a pointer display, with increments of 100 ft, but there are many other portrayals and many varied display profiles exist. The barometric set knob at the bottom right allows various static datums to be set; the reason for this is described below.

Figure 5.5. defines the basic criteria for the various altimeter and height settings:

- Altitude or QNH is defined as the barometric height between the aircraft and mean sea level (MSL). Therefore, an altimeter with a QNH setting indicates the height above mean sea level.
- Elevation relates to the height of a particular feature – in this case the height of an airfield above sea level. This is geographically fixed such that the height of the airfield will be fixed in relation to sea level.
- Height or QFE relates to the height of the aircraft above a particular feature

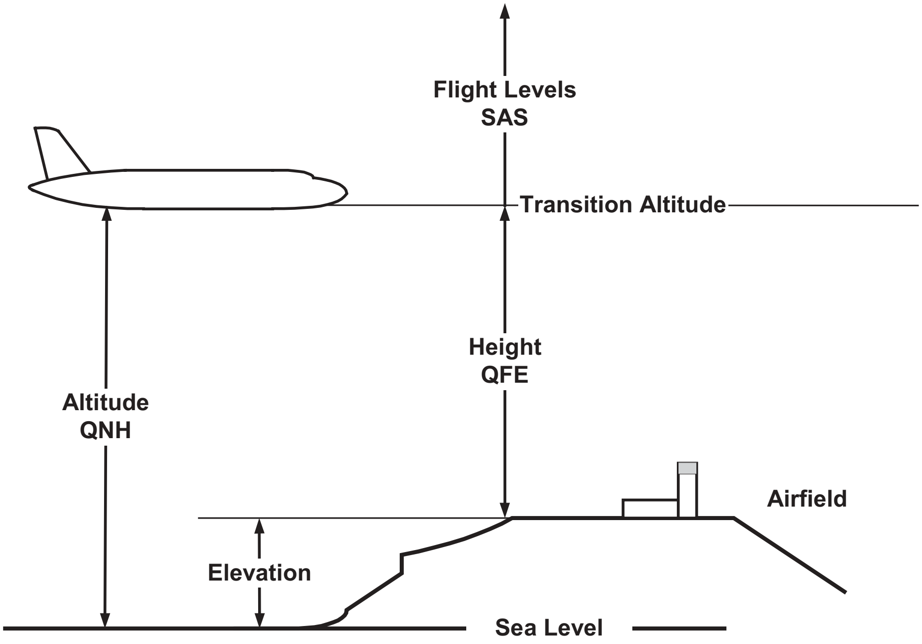


Fig. 5.5 *Altimeter settings*

(airport). An altimeter with the QFE correctly set will read zero at the appropriate airfield.
- Above a transition altitude, which depends upon the country and the height of the local terrain, all aircraft altimeters are set to a nominal Standard Altimeter Setting (SAS) of 29.92 inHg/1013.2 mbar (see Fig. 5.5.), which ensures that all the altimeters of aircraft in a locality are set to a common datum and therefore altitude conflicts due to dissimilar datum settings may be avoided. Above the transitional altitude, all aircraft altitudes are referred to in terms of Flight Levels (FLs) to reduce further ambiguity.

The importance of these definitions will become more apparent in later sections when Reduced Vertical Separation Minima (RVSM) and autoland Decision Height (DH) and Decision Altitude (DA) criteria are described.

As aircraft systems became more complex and more sophisticated propulsion and flight control laws were adopted, the number of systems that required air data increased. Therefore, the provision of air data in various aircraft navigation, flight control, and other subsystems required a more integrated approach. The computation tasks involved the following:

- Conversion of the sensed parameters into a more useful form, e.g. static pressure in terms of millibars or in inches of mercury into altitude (in feet), and dynamic pressure in inches of mercury into airspeed (in knots).
- Combination of two or more parameters to obtain a third parameter, e.g. airspeed and altitude to obtain Mach, or Mach and temperature to obtain true airspeed.
- To correct for known errors as far as possible.

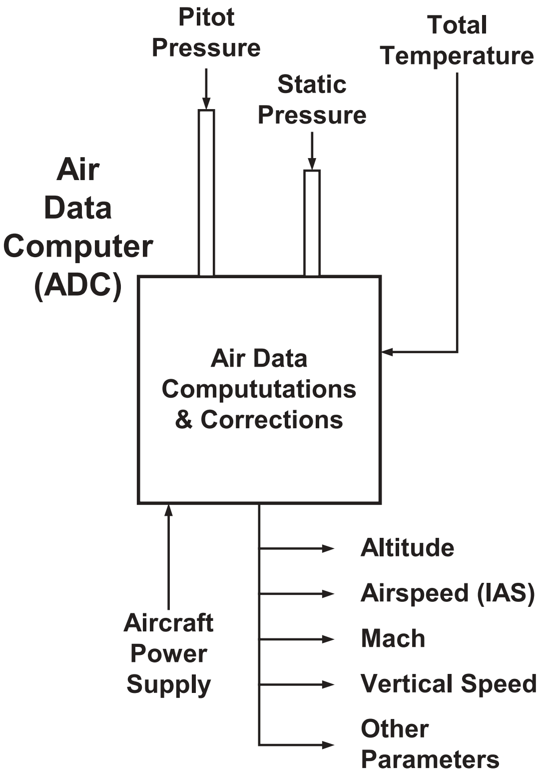
This led to the introduction of one or more Air Data Computers (ADCs), which centrally measured air data and provided corrected data to the recipient subsystems.

This had the advantage that, while the pilot still had the necessary air data presented, more accurate and more relevant forms of the data could be provided to the aircraft systems. Initially, this was achieved by analogue signalling means, but with the evolution of digital data buses in the late 1970s, data were provided to the aircraft subsystems by this means, notably by the use of standard ARINC 429 data buses. The function of an air data computer is to provide the outputs shown in Fig. 5.6. The unit contains the capsules necessary to measure the raw air data parameters and the computing means to calculate the necessary corrections. In earlier implementations, analogue computing techniques were utilized. With the advent of cost-effective digital processing, together with low-cost digital data buses, alternative methods of implementation became possible. The most effective combination of ADCs is the Triple Air Data System (TADS) which allows a majority voting technique to be used to isolate failures and use the best available information.

Airspeed measurements in the ADC are derived according to the computations summarized in Fig. 5.7 and described below:

1. Indicated Airspeed. IAS is the parameter proportional to pitot minus static or dynamic pressure and is directly related to the aerodynamic forces acting on the wings (lift) and control surfaces. IAS is therefore very useful for defining aircraft aerodynamic performance and structural limitations. Mach number is derived from a combination of IAS and altitude and is expressed as a ratio of the aircraft speed to the speed of sound at that particular altitude. Typical reference speeds based upon IAS or Mach are listed below:

Fig. 5.6 Air data computer



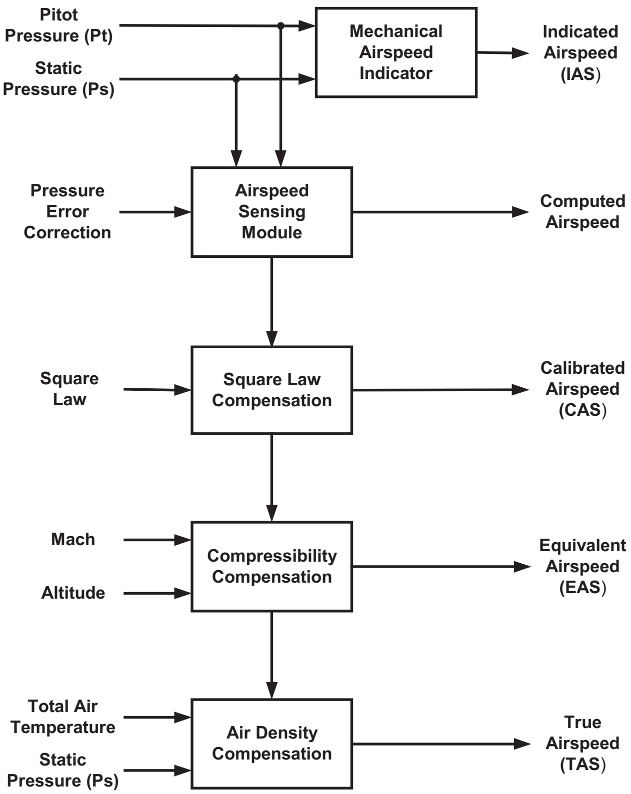


Fig. 5.7 *Derivation of airspeed parameters*

V_{stall} = aircraft stall speed
V_{rotation} = aircraft rotation speed
$V_{\text{gear extend}}$ = aircraft gear extension maximum
$V_{\text{flaps extend}}$ = aircraft flaps extension maximum
$V_{\text{maximum operating}}$ = aircraft maximum operating speed
$M_{\text{maximum operating}}$ = aircraft maximum operating Mach number

These reference speeds are declared in the aircraft operating manual. They are used by the aircrew to operate the aircraft within its structural and performance limitations for continued safe operation. For crucial limiting speeds placards will be placed at prominent positions on the flight deck.

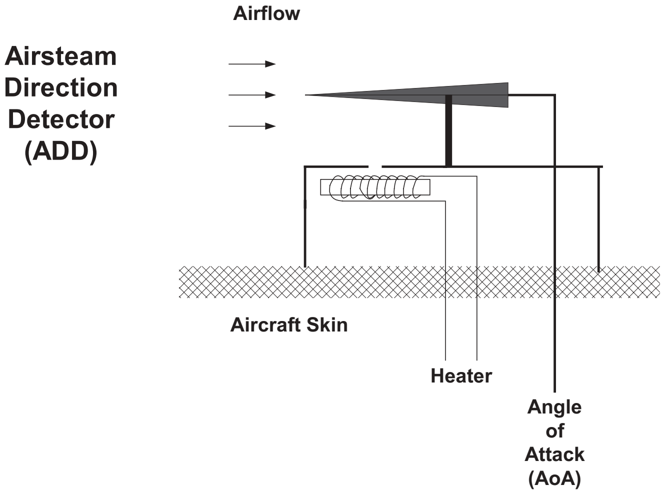
2. Computed airspeed. Computed airspeed is the IAS corrected for static pressure errors. It is not used as a parameter in its own right but is used as a basis for further corrections and calculations.
3. Calibrated airspeed (CAS). CAS is the computed airspeed with further corrections applied for non-linear/square law effects of the airspeed sensing module.
4. Equivalent airspeed (EAS). EAS is achieved by modifying CAS to allow for the effects of compressibility at the pitot probe, thereby obtaining corrected airspeed for varying speeds and altitudes.
5. True airspeed. The most meaningful parameter relating to navigation is TAS. TAS represents the true velocity of the aircraft in relation to the air mass and in still air

would be representative of the aircraft speed over the ground (groundspeed). TAS is calculated from Mach using SAT. In practice, the air mass is seldom stationary and is almost always moving with reference to the ground, and one of the primary functions of the navigation system is to make the necessary allowances.

Typical parameters provided by an ADC are:
- Barometric correction.
- Barometric corrected altitude.
- Altitude rate.
- Pressure altitude.
- Computed airpeed.
- Mach number.
- True airspeed.
- Static air temperature.
- Total air temperature.
- Impact pressure.
- Total pressure.
- Static pressure.
- Indicated angle of attack.
- Overspeed warnings.
- Maximum operating speeds.
- Maintenance information.

Other air data parameters of interest included the Airstream Direction Detector (ADD). This measures the direction of the airflow relative to the aircraft and permits the angle of incidence – also referred to as angle of attack – to be detected. Typically, this information is of use in the flight control system to modify the pitch control laws or in a stall warning system to warn the pilot of an impending stall. A typical airstream detection sensor is shown in Fig. 5.8.

Fig. 5.8 *Airstream Direction Detector*



The small bore pneumatic sensing lines associated with routing the sensed pitot or static pressure throughout the aircraft posed significant engineering and maintenance penalties. As the air data system is critical to the safe operation of the aircraft, it was typical for 3 or 4 or more alternative systems to be provided. The narrow bore of the sensing lines necessitated the positioning of water drain traps at low points in the system where condensation could be drained off, avoiding the blockage of the lines as a result of moisture accumulation. Finally, following the replacement of an instrument or disturbance of any section of tubing, pitot–static leak checks were mandated to ensure that the sensing lines were intact and leak-free and that no corresponding instrumentation sensing errors were likely to occur.

Correct flight instrumentation interpretation relating to air data derived is paramount to safe flight, though a number of accidents have occurred where the flight crew misinterpreted their instruments when erroneous data were presented. Reference (**1**) gives a good overview of how to understand and counter these effects. Angle of attack measurements and the associated display data have been used in the military fighter community for many years; reference (**2**) outlines how to get the best use of this parameter in an air transport setting.

The advent of digital computing and digital data buses such as ARINC 429 meant that computation of the various air data parameters could be accomplished in Air Data Modules (ADMs) closer to the pitot–static sensing points. Widespread use of the ARINC 429 data buses enabled these data to be rapidly disseminated throughout all the necessary aircraft systems. Now, virtually all civil transport aircraft designed within the last 15 years or so have adopted the air data module implementation. Figure 5.9 shows a modern air data system for an Airbus aircraft, while Fig. 5.10 shows a typical ADM.

The Airbus system comprises three pitot and six static sensors. Pitot sensors 1–3 and static sensors 1 and 2 on each side of the aircraft are connected to their own air data module. Each of these seven air data modules provides pitot or static derived air data to the display, flight control, and navigation systems, among others. Static sensors 3 on each side are connected to a common line and a further air data module. Pitot 3 and the combination of the static 3 sensors are also used to provide pitot and static pressure to the aircraft standby airspeed indicator and standby altimeter.

Air data are generally regarded as accurate, but in the longer term rather than the short term. The fact that air data sensing involves the use of relatively narrow bore tubing and pneumatic capsules means that there are inherent delays in the measurement of air data as opposed to some other forms of air data. The use of ADMs situated close to, or integrated into, the probes will reduce such errors, as well as eliminating condensation and icing with consequent maintenance benefits.

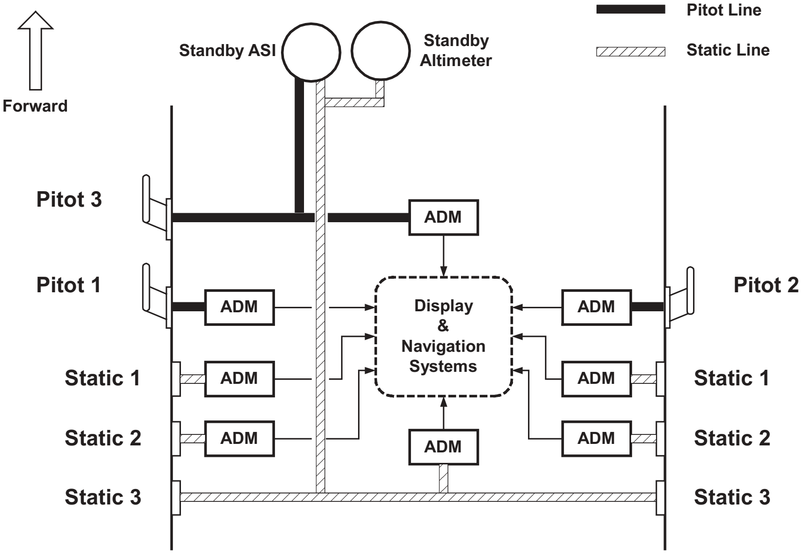


Fig. 5.9 *Airbus data system*

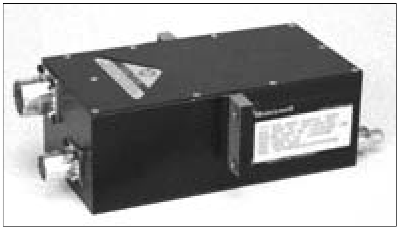


Fig. 5.10 *Typical air data module (Honeywell)*

Magnetic sensing

The use of the Earth's magnetic field to sense direction and the use of north-seeking devices to establish the direction of magnetic north for the purposes of navigation is one of the oldest forms of sensor. The location of a magnetic sensing device – called a flux valve – is usually in the outer section of one of the aircraft wings, well clear of any aircraft-induced sources of spurious magnetism. As may be seen in Fig. 5.11, the axis of the Earth's magnetic field may be considered to be analogous to a simple bar magnet. This magnetic dipole has its field lines originating at a point near the south pole and terminating at a point near the north pole, but the field is skewed from true geodetic north by ~11.5°.

The Earth's field lines enter the earth at a considerable angle to the local horizontal plane, and this angle is called the magnetic angle of inclination (magnetic dip). In the United States and Europe this angle is around 70°. The Earth's magnetic field, $\boldsymbol{H}$, is the vector sum of components H_x, H_y, H_z measured in the orthogonal axis set shown in Fig. 5.12, where the angles of inclination and declination are shown.

The H_x and H_y components, which are in the local horizontal plane, are used to determine the compass heading with reference to the north magnetic pole. However, allowance has to be made for the fact that the magnetic and geodetic poles do not

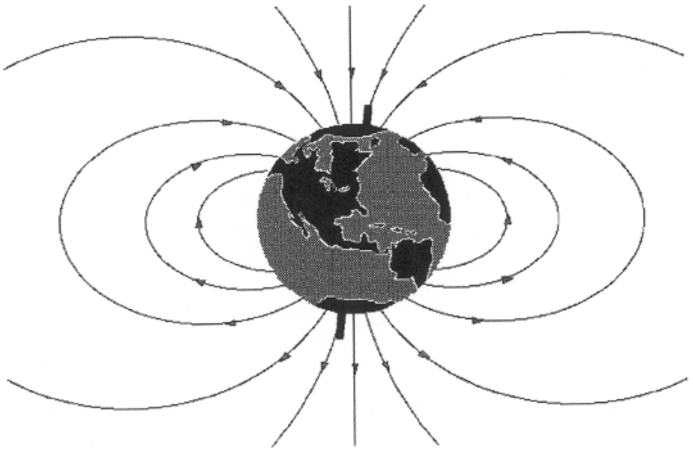


Fig. 5.11 *Earth's magnetic field*

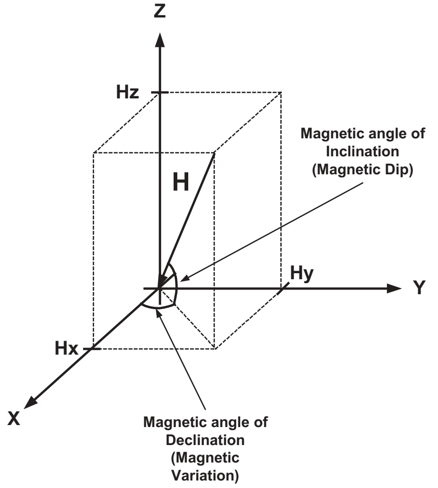


Fig. 5.12 *Earth's magnetic field components*

coincide, and also for the fact that there are considerable variations in the Earth's magnetic characteristics across the globe. These factors are measured and mapped across the globe such that the necessary corrections may be applied. The correction term is called the angle of declination (magnetic variation) and is a corrective angle to be added to/subtracted from the magnetic heading to give a true (geodetic) compass heading. Positive angular declinations represent easterly corrections, while negative angles represent westerly corrections. These corrections are measured, charted, and periodically updated. Figure 5.13 shows the variation in declination from the world magnetic model for 2000, published by the British Geological Survey in conjunction with The United States Geological Survey [see reference (**3**)].

Perhaps easier to understand is a simplified North American map of declination derived a few years previously (Fig. 5.14). In this figure it may be readily noted that declination for the northeastern US seaboard in the vicinity of Boston would be in the region of 14° west (minus 14°), and the corresponding correction for Seattle on the northwestern US seaboard would be 20° east (plus 20°). When the flight time of

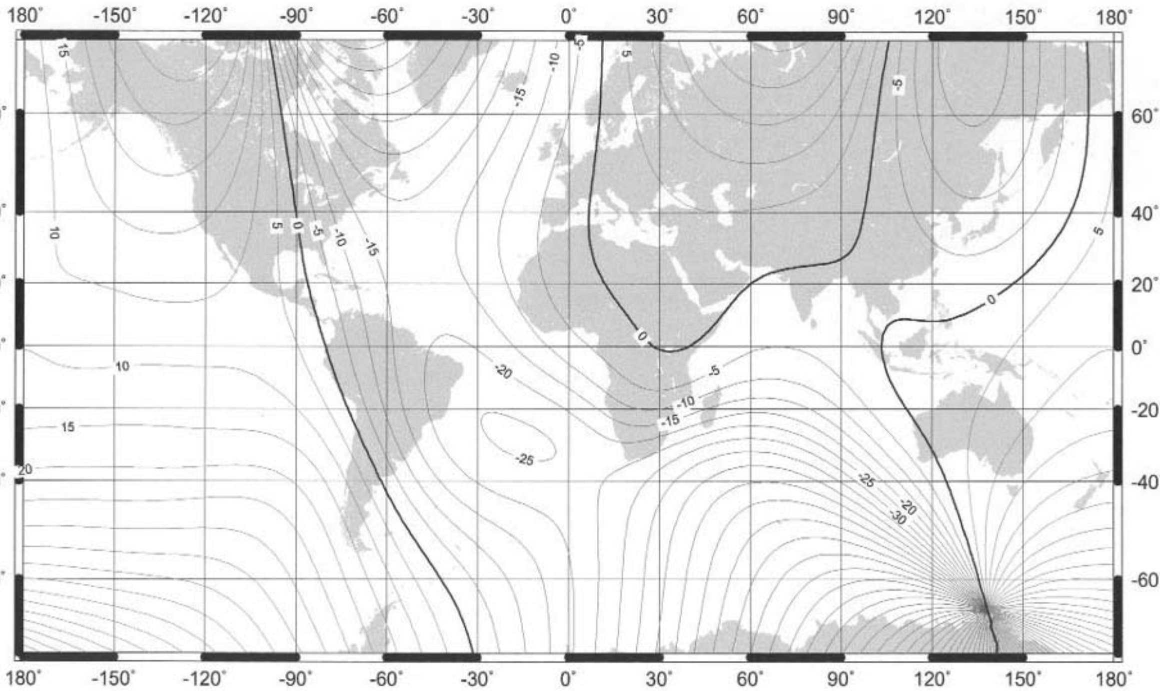


Fig 5.13 *World magnetic map 2000*

~ 4.5–5 h for a typical modern airliner between these points is considered, the magnitude of the change in declination (variation) may be appreciated.

Early-generation flux valves were commonly gimballed in two axes to allow a degree of freedom in pitch and roll but were fixed in azimuth. More recently, three-dimensional solid-state magnetic sensors have become available, which offer a strapdown sensing capability that can fit within the footprint of the existing gimballed design. Figure 5.15 shows the Honeywell HMR2300 strapdown sensor which can resolve the magnetic field into *X*, *Y*, and *Z* components.

Magnetic sensing therefore provides a very simple heading reference system that may be used by virtually all aircraft today, although, in transcontinental and transoceanic aircraft, inertial and Global Navigation Satellite System (GNSS) navigation methods are likely to be preferred. In general aviation aircraft of limited range and performance it is likely to be the only heading reference available. This type of system was commonly used for navigation until the late 1960s/early 1970s, when inertial platforms became common, and it is still in use for many general aviation aircraft.

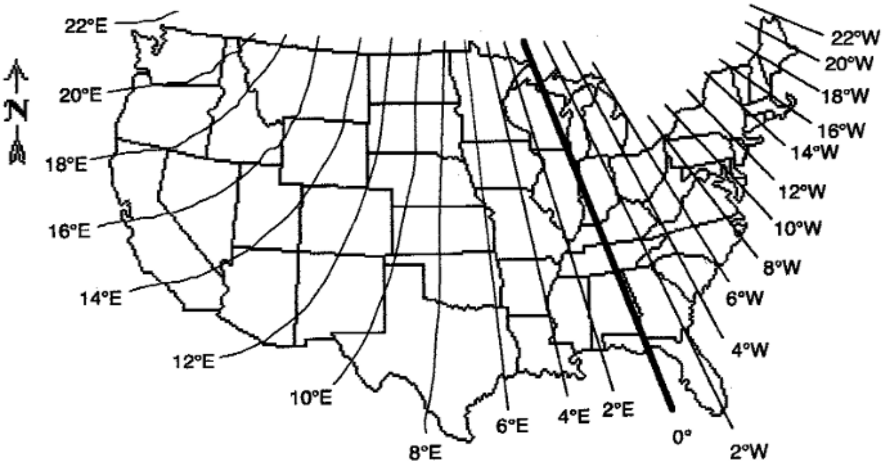


Fig 5.14 *North American declination map*

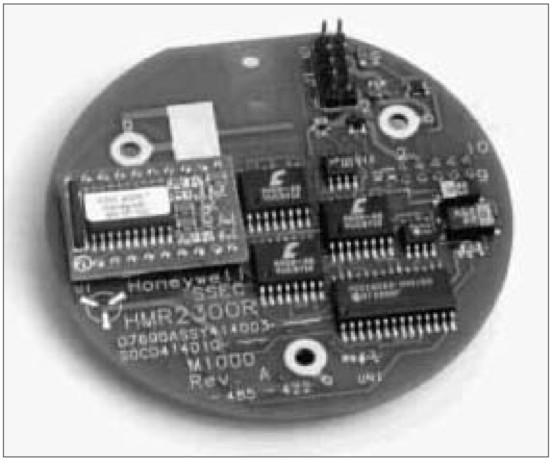


Fig. 5.15 *Honeywell solid state strapdown magnometer (Honeywell)*

Magnetic Heading Reference System (MHRS)

The magnetic sensor or flux valve described earlier will provide magnetic heading and, when combined with a Directional Gyro (DG), which provides an inertial heading reference, can provide a Magnetic Heading and Reference System (MHRS) as shown in Fig. 5.16. The flux valve and DG provide magnetic and inertial heading respectively to the magnetic heading and reference system. According to the heading mode of navigation being used, either magnetic or true (inertial) heading may be selected and displayed on the aircraft heading displays. The system includes a compensation element to provide the necessary corrections as the system encounters differing magnetic field conditions.

The outputs from the MHRS may be fed to a Radio Magnetic Indicator (RMI), to a Horizontal Situation Indicator (HSI) to provide display of heading information, or to the aircraft autopilot/flight director system either by synchro or ARINC 429 data buses.

The Collins AHS-3000A is a smaller, lighter, and more reliable Attitude Heading Reference System (AHRS). The advanced AHRS delivers greater reliability than

Fig. 5.16 *Magentic heading reference system*

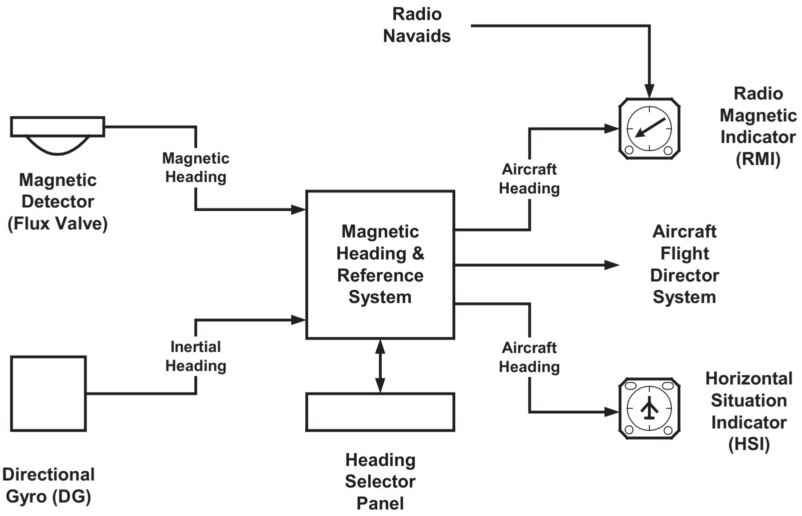
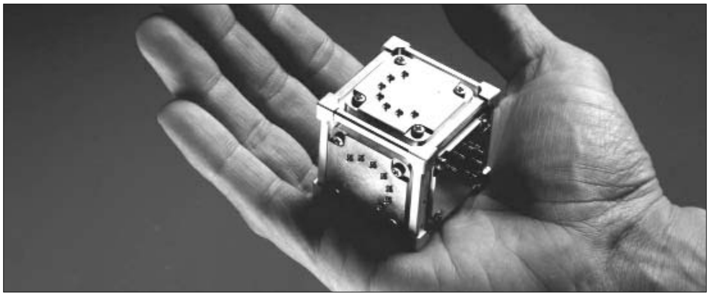


Fig 5.17 *(a) Rockwell Collins AHARS (Rockwell Collins)*

Fig 5.17 *(b) Miniature gyro (Rockwell Collins)*



conventional AHRS, with mean time between failures at greater than 10 000 h. The solid-state system has been selected for inclusion on many new airframes as well as retrofit application on existing aircraft.

Inertial navigation

Inertial sensing

Inertial sensors are associated with the detection of motion in a universal (non-earth) referenced set. Inertial sensors comprise:

- Position gyroscopes.
- Rate gyroscopes.
- Accelerometers.

Position gyroscopes

Gyroscopes are most commonly implemented as spinning masses or wheels tending to hold their position in a space-referenced attitude set. Position gyroscopes or gyros use this property to provide a positional or attitude reference – typically, aircraft pitch position, roll position, or yaw position (heading). Position gyros are used in heading and reference systems to provide the aircraft with vital information regarding the aircraft attitude for a range of aircraft subsystems.

A simple gyroscope may be represented by the simple spinning wheel in Fig. 5.18. The wheel is rotating in the direction shown around the *X* axis and, once it has been spun up to its operating speed, will preserve that orientation in space. The degree of 'stiffness' of the gyro will depend on its angular momentum, which in turn depends on mass and speed of rotation.

In order to preserve the spatial position of a directional gyro, a gimballing system needs to be used as shown in Fig. 5.19. The simple system shown comprises an inner, centre, and outer gimbal mechanism. The vertical pivots between the inner and centre gimbals allow freedom of movement in rotational direction A. The horizontal pivots between the centre and outer gimbal allow freedom of movement in rotational direction B.

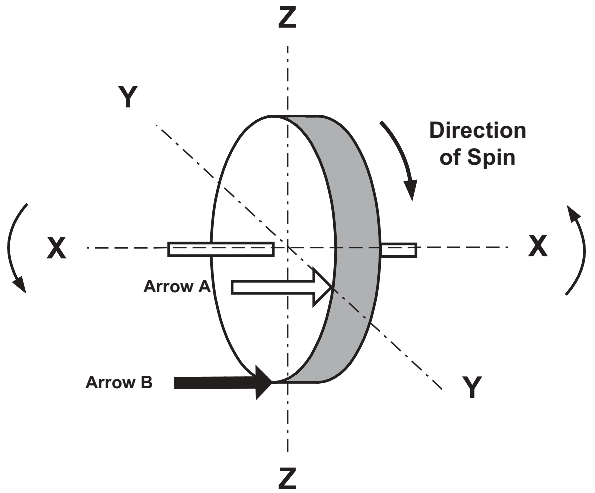
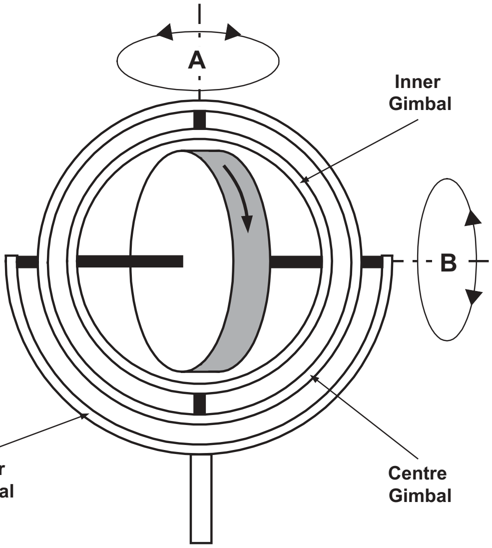


Fig. 5.18 *Simple gyroscope*

Fig. 5.19 Position gyroscope with simple gimballing system



Therefore, if the outer gimbal is fixed to the aircraft frame of axes, the gyroscope will remain fixed in space as the aircraft moves. Hence, pitch, roll, and yaw (heading) attitude may be measured.

Rate gyroscopes

The gyro also has the property of precession when an external force is applied. Referring back to Fig. 5.18, if a force is applied to the forward edge on the *Y* axis (shown by white arrow A), the actual force will not act, as might be imagined, by rotating the *Z* axes. Instead, owing to the properties of the gyro the force will actually act at the bottom of the wheel (shown by black arrow B). This force will cause the gyro *X* axis to tilt counterclockwise in the direction shown. This property of precession allows body rates to be sensed.

Gyroscopes can therefore be used to provide information relating to the aircraft body rates: pitch rate, roll rate, and yaw rate. These rate gyros use the property of gyro precession described. When a gyro is rotating and the frame of the gyro is moved, the gyro moves or precesses owing to the effects of the angular momentum of the gyro. By balancing and measuring this precession force, the applied angular rate of movement applied to the gyroscope frame may be measured. This information, together with the attitude data provided by the position gyros, is crucial in the performance of modern flight control or Fly-By-Wire (FBW) systems, as is described in Chapter 9.

In early systems, gyroscopes were air driven, but later electrically driven gyroscopes became the norm. As these were both rotating devices, bearing friction and wear were major factors in mitigating against high accuracies. In modern systems the gyroscopes used are likely to be fibre-optic devices. The principle of operation of a Ring Laser Gyroscope (RLG) is depicted in Fig. 5.20. The LRG uses laser light in the visible or near-infrared wavelength to sense angular motion. Using a coherent, highly stable laser source and a series of mirrors to create a continuous light path, two travelling-wave laser beams are formed independently, one moving in a clockwise and the other in a counterclockwise direction. When the sensor is stationary in inertial space, both beams

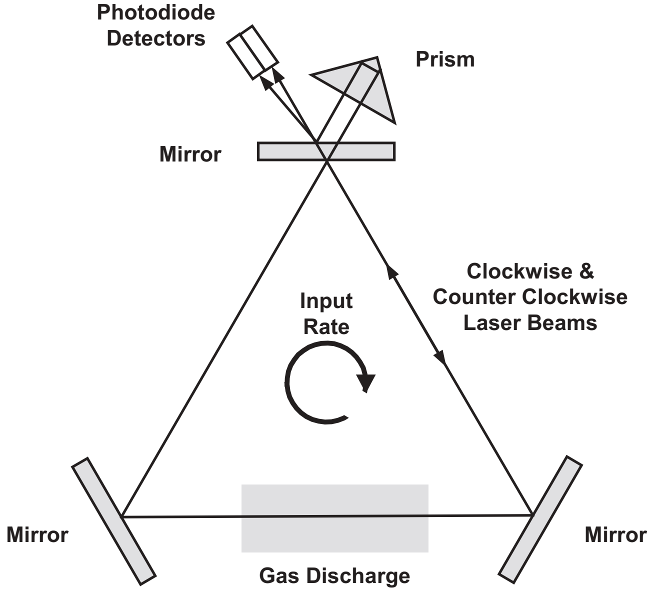


Fig. 5.20 *Ring laser gyroscope – principle of operation*

have the same optical frequency. When the sensor is rotated around the axis perpendicular to the plane containing the beams and mirrors (shown in the figure), differences occur in the frequency of the two beams. The frequency of each beam alters to maintain the resonance necessary for laser action. The path difference between the beams is very small (~1 nm = 1×10^{-9} m), and therefore a source of high spectral purity and stability is required; typically, helium neon gas lasers are used.

The rotational motion is sensed by allowing a small portion of light from each beam to 'leak' through one of the mirrors, and the two beams are combined using a prism to form an interference pattern on a set of photodiodes. The frequency difference between the two beams causes interference fringes to move across the detectors at a frequency proportional to the frequency difference between the beams and hence proportional to the input angular rate. At a very low angular rate the beams can effectively 'lock in' to the same frequency on account of optical backscattering within the device, and this can cause a dead band effect. This may be overcome by mechanically dithering or physically vibrating the entire cavity to obviate the 'lock-in' condition. Alternatively, four beams of differing frequencies may be used within the cavity, which enables the

Fig. 5.21 *Ring laser gyroscope (photograph courtesy of Northrop Grunman Corporation)*

‘lock-in’ effect to be overcome. RLGs can offer a performance in the region of 0.001°/h of drift; less complex and cheaper Fibre-Optic Gyroscopes (FOGs) can produce drift rates of the order of 10°/h or more. An example of an RLG produced by Litton is shown in Fig. 5.21.

Accelerometers

Accelerometers are devices that measure acceleration along a particular axis. The measurement of acceleration can be integrated using computers used to derive aircraft velocity and position. All accelerometers use the principle of sensing the force on a loosely suspended mass, from which the acceleration may be calculated.

A common accelerometer used today is the pendulous force feedback accelerometer shown in Fig. 5.22. The device is fixed to the body structure whose acceleration is to be taken (shown as the structure at the right). As the structure and the pendulous arm move, the pick-off at the end of the pendulous arm moves with respect to two excitation coils. By sensing this movement, a corrective current is applied to the restoring coil that balances the pick-off to the null position. As the restoring coil is balanced between two permanent magnets above and below, the resulting current in the restoring coil is proportional to the applied acceleration – in this example in the vertical direction. Typical accelerometer accuracies may vary from 50 mg (50×10^{-3} g) down to a few μg ($1\ \mu g = 1 \times 10^{-6}$ g).

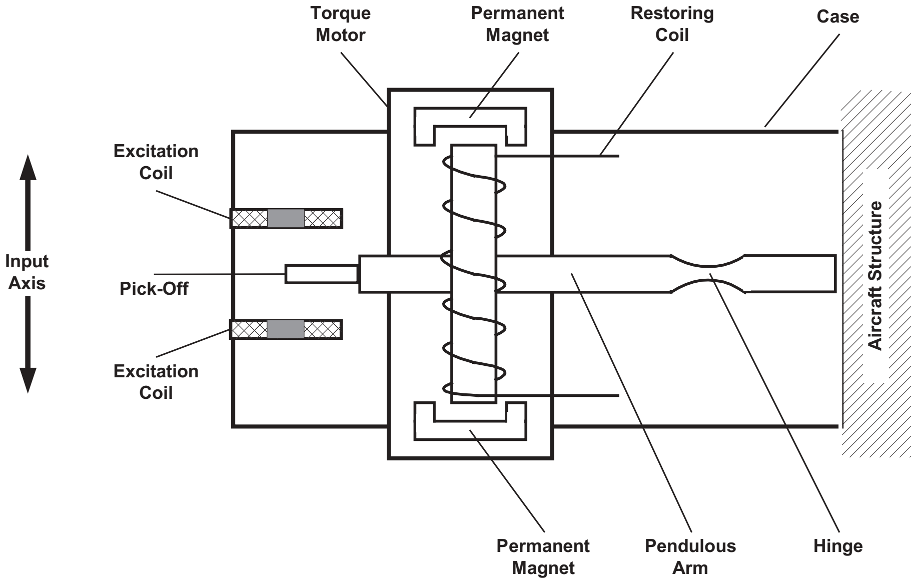


Fig. 5.22 *Pendulous force feedback accelerometer – principle of operation*

Inertial navigation

As for the magnetic sensors, inertial sensors may be arranged such that inertial rates are sensed within an orthogonal axis set as shown in Fig. 5.23.

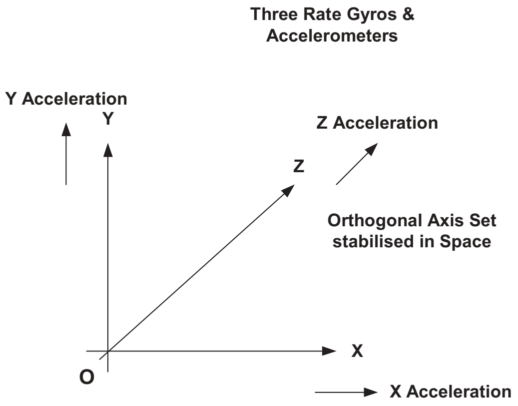


Fig. 5.23 *Inertial reference set*

Features of an Inertial Platform:

Platform is Gyro Stabilised in Space

Sensitive Accelerometers detect acceleration in the direction of the orthogonal axes: Ox, Oy, Oz

Accelerations are integrated to give first velocity and then position in the Ox, Oy, Oz axes

Platform readings can be transformed to relate to earth rather than spatial axes and coordinates

Inertial platforms

Early inertial navigation systems used a gimballed and gyro-stabilized arrangement to provide a stable platform on which the inertial sensors were located. The platform was stabilized using rate information from the gyros to drive torque motors which stabilized the platform in its original frame of axes, independently of movement of the aircraft. Resolvers provided angular information about the aircraft in relation to the platform, and hence aircraft attitude could be determined. A diagram representing a gimballed, stabilized platform is shown in Fig. 5.24.

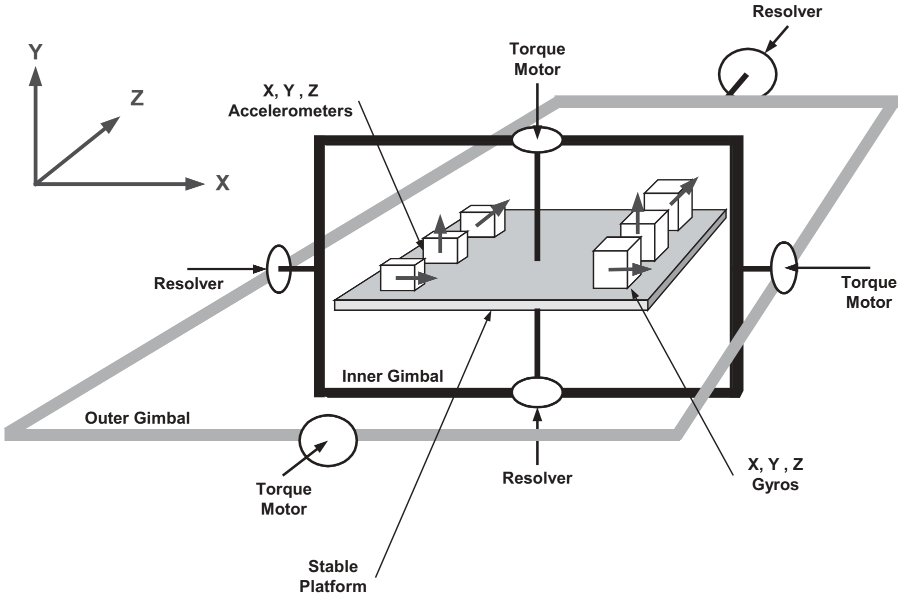
The inertial sensors located on the stable platform preserve their orientation in space and are

X accelerometer	X gyro
Y accelerometer	Y gyro
Z accelerometer	Z gyro

The inertial platform uses a combination of gyros and accelerometers to provide a platform with a fixed reference in space. By using the combined attributes of position and rate gyros and accelerometers, a stabilized platform provides a fixed attitude reference in space and, when fitted in an aircraft, can provide information about aircraft body rates and acceleration in all three axes. Suitable computation can also provide useful information relating to velocity and distance travelled in all three axes.

This is a significant achievement in establishing the movement of the aircraft, but it suffers a major disadvantage. That is, the aircraft body data are derived relative to a reference set in space, whereas to be useful in navigating on the Earth the system needs to be referenced to a global reference set. The development of the Inertial Navigation

Fig. 5.24 Three-gimbal gyro-stabilized platform

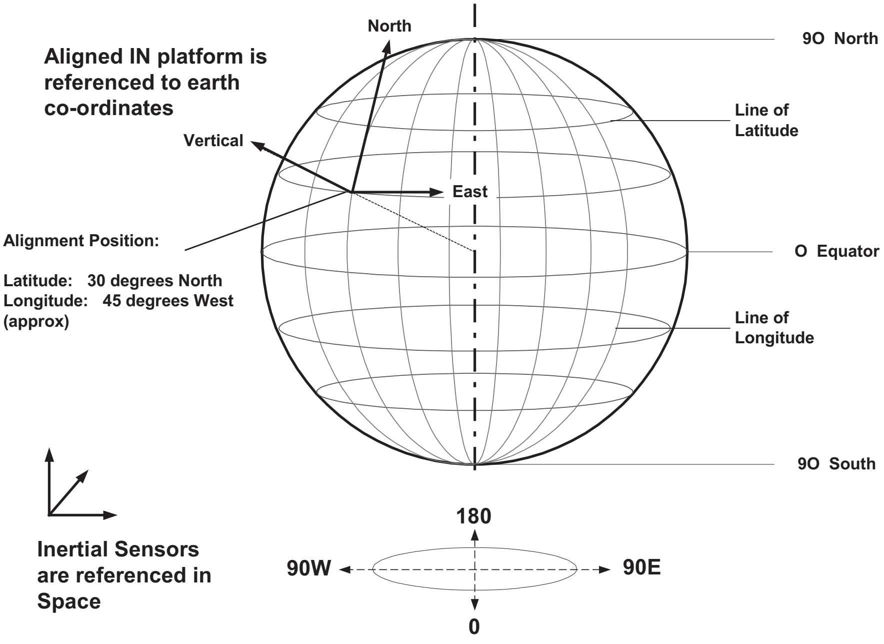


System (INS) provided the additional computation to provide this essential capability (see Fig. 5.25).

The inertial navigation system performs a series of transformations such that the inertial platform is referenced to a meaningful Earth reference set. To do this, it performs the following operations:

- Aligns the vertical axis (*Z* axis) with the local Earth vertical.

Fig. 5.25 Earth-referenced inertial system



- Aligns the horizontal (*Y* axis) with north: the *X* axis will now point east.
- Calculates movement across the surface of the Earth.
- Allows for the Earth's rotation variation with longitude. To do this, it needs an Earth starting point in terms of latitude and longitude.

The alignment of the INS platform follows power-up and usually takes several minutes. During this process, the orthogonal axis set is aligned with north and local vertical and, by implication, east. The platform initialization is undertaken by the flight crew by inserting the latitude and longitude coordinates provided at the aircraft departure gate. This process is completed before aircraft start-up, and the INS is therefore able to perform calculations deriving vital information regarding aircraft position, velocities, and accelerations throughout the flight. Typical information would include:

- Three-axis accelerations.
- Three-axis velocities.
- Present position – latitude and longitude.
- Distance along and across track.
- Angle of drift.
- Time to next waypoint and subsequent waypoints.
- Calculated wind speed and direction, etc.

Inertial systems have the advantage that they are very accurate in the short term – no settling time is required as for air data. Conversely, inertial sensors have a tendency to drift with time and so become progressively less accurate as the flight continues. Often, the flight crew will use other navigation sensors to update the INS during flight and minimize the effects of this characteristic. As will be seen in other chapters, GPS or other satellite-based systems can be used to provide an automatic periodic correction of high accuracy.

The advent of digital computers greatly facilitated the ability to undertake this computation. On an aircraft such as a Boeing 747 Classic, three INSs would be provided and additional calculations performed to establish the best estimate of aircraft position according to information provided by all three systems.

Strapdown systems

Many modern inertial systems are strapdown, in other words, the inertial sensors are mounted directly to the aircraft structure and no gimballed platform is required. The inertial signals are resolved mathematically using a computer prior to performing the necessary navigation calculations. The removal of the stabilized platform with its many high-precision, moving parts and the use of modern RLG sensor technology considerably increase the system reliability. Strapdown systems are claimed to be around five times more reliable than their stabilized platform predecessors.

The Litton LTN-92 strapdown INS was designed to update aircraft from the earlier LTN-72 gyro-stabilized system. The system uses three RLGs, force rebalanced accelerometers, and three high-speed digital microprocessors to provide an RNP-10 (Required Navigation Performance – 95 per cent probability of being within 10 nautical miles of estimated position) navigation capability. The units comprising the LTN-92 system are shown in Fig. 5.26.

With the advent of Air Data Modules (ADMs), a combined unit called the Air Data

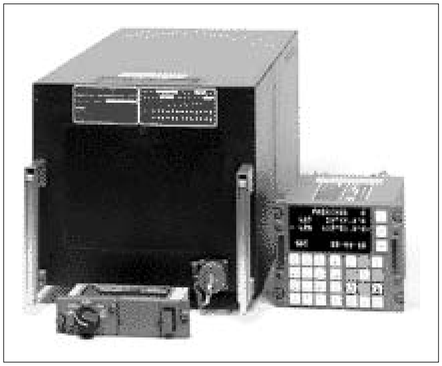


Fig. 5.26 *Litton LTN-92 strapdown INS (phoyograph courtesy of Northrop Grumman Corporation)*

and Inertial Reference System (ADIRS) performs all the necessary calculation on air data and inertial data to provide the navigational information that the aircraft requires. Examples of modern ADIRS systems are addressed in Chapter 8. See reference (**4**) for a more detailed overview of inertial techniques.

Radar sensors

Civil aircraft carry a number of radar sensors that permit the aircraft to derive data concerning the flight of the aircraft. The principle radar sensors in use on civil aircraft are:

- Radar altimeter.
- Doppler radar.
- Weather radar.

Radar altimeter

The radar altimeter (rad alt) uses radar transmissions to reflect off the surface of the sea or the ground immediately below the aircraft. The radar altimeter therefore provides an absolute reading of altitude with regard to the terrain directly beneath the aircraft – absolute distance above terrain. This contrasts with the barometric or air data altimeter where the altitude may be referenced to sea level (altitude) or some other datum such as the local terrain (height). The radar altimeter is therefore of particular value in warning pilots that they are close to the terrain and need to take corrective action. Alternatively, the radar altimeter may provide the flight crew with accurate altitude with respect to terrain during the final stages of a precision approach. Comparison of barometric and radar altitude is shown in Fig. 5.27.

The radar altimeter principle of operation is shown in Fig. 5.28. The oscillator and modulator provide the necessary signals to the transmitter and transmit antennae which direct radar energy towards the terrain beneath the aircraft. Reflected energy is received by the receive antenna and received and passed to a frequency counter. The frequency counter demodulates the received signal and provides a radar altimeter reading to a dedicated display (see Fig. 5.29). Alternatively, the information may be presented on an Electronic Flight Instrument System (EFIS). In modern systems, radar altitude will be provided to a range of systems such as the Flight Management System (FMS), the Enhanced Ground Proximity Warning System (EGPWS), autopilot, etc., as well as being displayed directly to the flight crew. Radar altimeters usually operate over a maximum range of 0–5000 ft; the display shown has a maximum reading of 2000 ft.

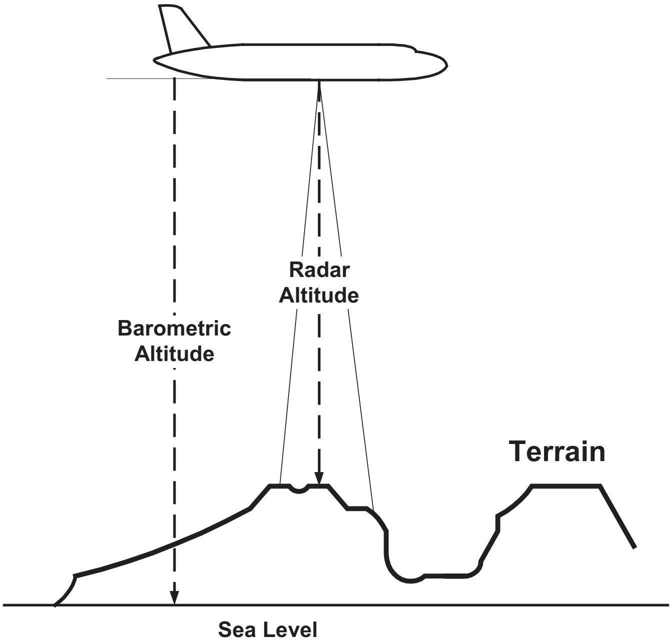


Fig. 5.27 *Radar altimeter compared to barometric altimeter*

Most radar altimeters use a triangular modulated frequency technique on the transmitted energy as shown in Fig. 5.30. The transmitter/receiver generates a Continuous Wave (CW) signal varying from 4250 to 4350 MHz modulated at 100 MHz

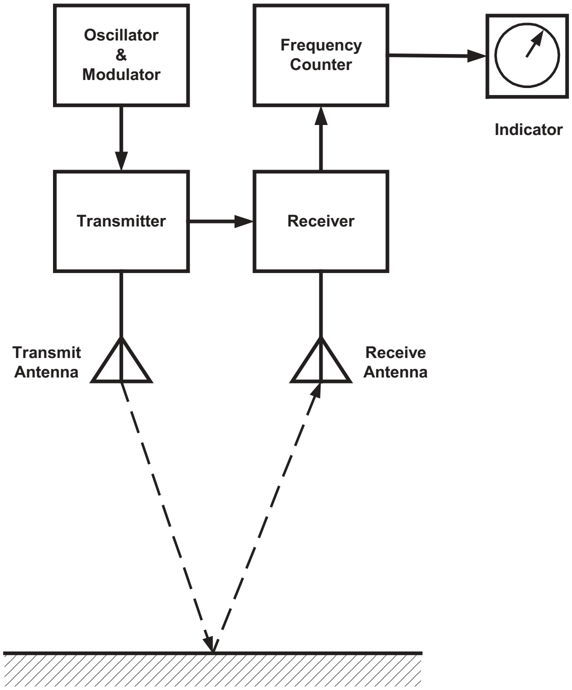


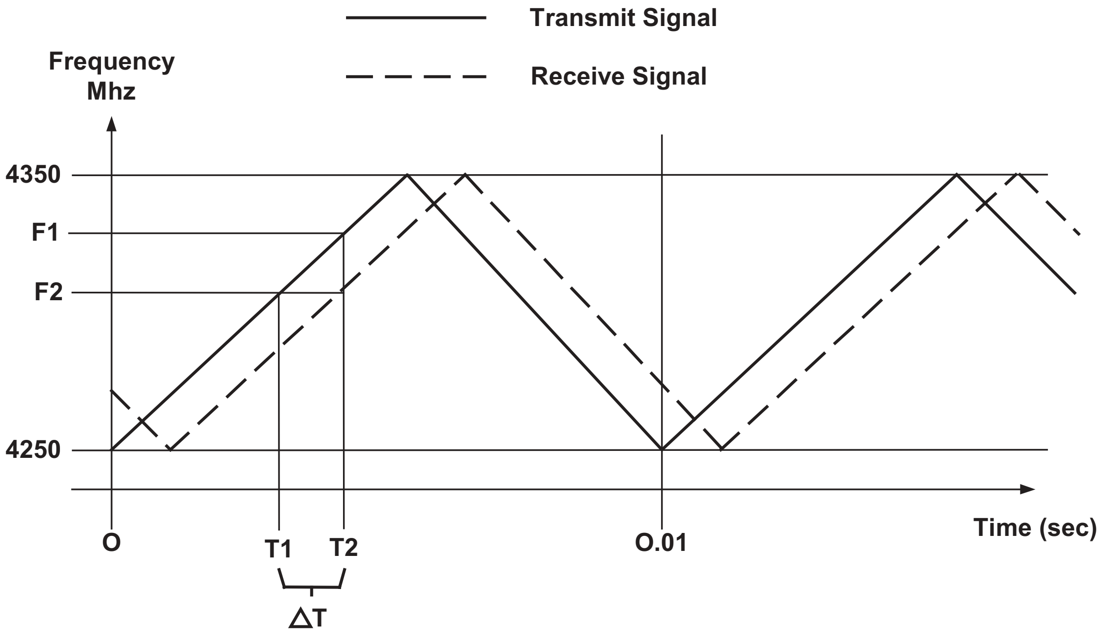
Fig. 5.28 *Radar altimeter – principles of operation*

(period = 0.01 s). Comparison of the frequency of the reflected energy with the transmitted energy – F_1 versus F_2 in the figure – yields a frequency difference that is proportional to the time taken for the radiated energy to return, and hence radar altitude may be calculated.

Radar altimeter installations are calibrated to allow for the aircraft installation delay

Fig. 5.29 *Stand-alone radar altimeter display*

Fig. 5.30 *Radar altimeter frequency modulation technique*



which varies from aircraft to aircraft. This allows compensation for the height of the antenna above the landing gear and any lengthy runs of coaxial cable in the aircraft electrical installation. The zero reading of the radar altimeter is set so that it coincides with the point at which the aircraft landing gear is just making contact with the runway.

Doppler radar

Doppler radar transmits energy in three or four beams skewed to the front and rear of the aircraft, as shown in Fig. 5.31. In this example, a three-beam system is depicted. The beams are also skewed laterally to the sides of the aircraft track. As with the radar altimeter, Doppler radar depends upon the radiated energy being reflected from the terrain within the Doppler beams. As before, a frequency difference between the radiated and reflected energy carries vital information. Owing to the effects of the Doppler principle, energy reflected from beams facing forward will be returned with a higher frequency than the radiated energy. Conversely, energy from a rearward facing beam will have a lower frequency than that radiated.

In Fig. 5.31, beams 2 and 3 will return higher frequencies where the frequency increase is proportional to the aircraft groundspeed, while beam 1 will detect a lower frequency where the frequency decrease is also proportional to groundspeed. This enables the aircraft groundspeed to be derived. If the aircraft is drifting across track on account of a crosswind, then the beams will also detect the lateral frequency difference component, and the cross-track velocity may be measured. Finally, by using computation within the radar, the aircraft V_x, V_y, and V_z velocity components may be determined, as may the overall aircraft velocity vector. Doppler radar velocity outputs may be compared with those from the inertial navigation system, thereby making possible a more accurate estimate of aircraft velocities and position.

Doppler has a number of significant advantages. Velocity is measured directly with respect to the Earth's surface, unlike air data systems which derive velocity relative to the air mass, and inertial systems which are located in an abstract (inertial) reference set. The equipment used is relatively inexpensive and accurate, and does not need an infrastructure of ground stations. It does not undergo an alignment process, as needed

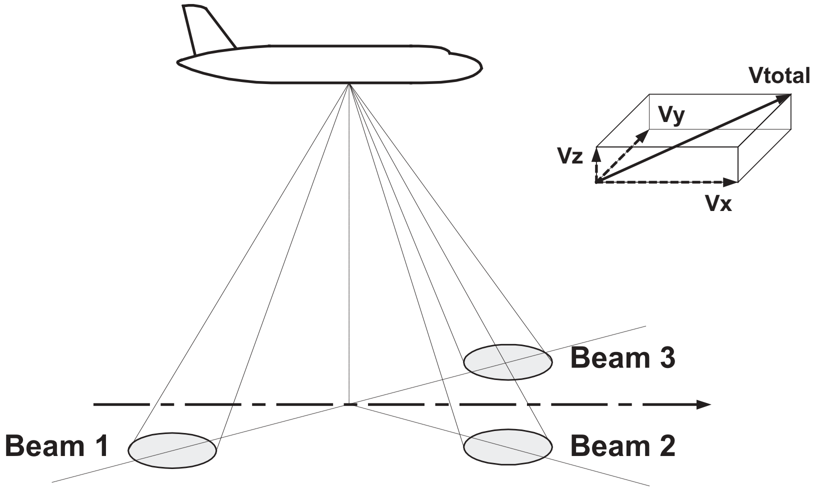


Fig. 5.31 *Principles of Doppler radar*

by an inertial system, and it can provide accurate data at low speeds, which is very useful in helicopter systems.

Doppler radar suffers from one significant drawback. In certain circumstances the terrain over which the aircraft is flying may not reflect enough energy for the aircraft velocities to be determined. Such conditions may be presented while flying over an expanse of water where the surface is very smooth, giving a 'millpond' effect. Similarly, flying over snow-covered or glacial terrain may cause the Doppler radar to 'lose lock', and readings may become unreliable. For autonomous dead reckoning navigation, an external attitude reference source is needed such as an MHRS or the inertial system already described. As for inertial systems, velocity accuracy degrades with time, and there are obvious consequences for long-term navigation.

The choice of the depression angle of the Doppler beams is a compromise between two major considerations. The first is high sensitivity to velocity – in terms of Hz per knot – in which lower values of depression give higher accuracy. This has to be balanced against the fact that, as the depression angle decreases, particularly over water or other terrain with low radar reflectivity, proportionately less energy is returned. Typical values of the depression angle vary from 65 to 80° depending upon the system requirements.

For typical aircraft systems, the sensitivity of Doppler is 30 Hz per knot of speed. For the forward and aft beam geometries of the type shown in Fig. 5.31 – also known as a Janus configuration – the horizontal velocity error is of the order of 0.015 per cent per degree of error in pitch angle. Doppler may be used in either stabilized or strapdown configuration, in which case the overall system error will depend respectively on stabilization accuracy or computational accuracy available.

Doppler radar was commonly used in the 1960s, but the advent of inertial systems and more recently GPS means that this technique is little used in the civil transport and business jet systems produced today. Doppler radar is still commonly used on helicopters.

Weather radar

The weather radar has been in use for over 40 years to alert the flight crew to the presence of adverse weather or terrain in the aircraft's flight path. The weather radar radiates energy in a narrow beam with a beamwidth of ~3° which may be reflected from clouds or terrain ahead of the aircraft. The radar beam is scanned either side of the aircraft centre-line to give a radar picture of objects ahead of the aircraft. The antenna may also be tilted in elevation by around ±15° from the horizontal to scan areas above and below the aircraft.

The principle of operation of a weather radar is shown in Fig. 5.32. This shows a storm cloud directly ahead of the aircraft, with some precipitation below, and also steadily rising terrain. Precipitation can be indicative of severe vertical wind shear which can cause a hazard to the aircraft.

The radar beam (shown in grey) is pointing horizontally ahead of the aircraft with the antenna in its mid- or datum position and will detect the storm cloud through which the aircraft is about to fly. By referring to the weather radar display, the pilot will be able to see if the storm cells can be avoided by altering course left or right. The use of the antenna tilt function is crucial. In the example given, if the antenna is fully raised, the crew will not gain any information relating to storm cloud, precipitation, or rising

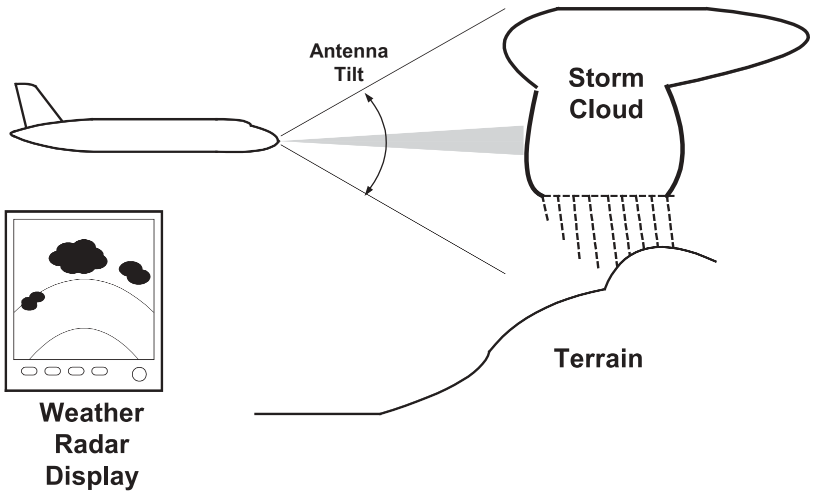


Fig. 5.32 *Operation of weather radar*

terrain. If the antenna is fully depressed, the radar will detect the rising terrain but not the storm cloud or precipitation ahead. For this reason, many weather radars incorporate an automatic tilt feature so that the radar returns are optimized for the flight crew in terms of the returns that are received.

Most modern weather radars can use Doppler processing to detect turbulence ahead of the aircraft. This is a very useful feature as maximum wind shear does not necessarily occur coincidentally with the heaviest precipitation. In fact, some of the most dangerous wind shear can occur in clear air with the aircraft flying nowhere near any clouds or precipitation.

The radar picture may be displayed on a dedicated radar display or overlaid on the pilot or first officer's navigation display. Displays are typically in colour, which helps the flight crew to interpret the radar data. Displays have various selectable range markers and are usually referenced to the aircraft heading. Separate displays may be provided for weather or turbulence modes.

A block diagram of a typical weather radar system is shown in Fig. 5.33. Figure 5.34 shows some typical components for the Honeywell RDR-4B weather radar. The transmitter operates at 9.345 GHz and the system has three basic modes of operation:

- Weather and map with a maximum range of 320 nm.
- Turbulence (TURB) mode out to 40 nm.
- Wind shear detection out to 5 nm.

The radar antenna is stabilized in pitch and roll using aircraft attitude data from an Attitude and Heading Reference System (AHRS) or inertial reference system. The pulse width and pulse repetition frequency vary depending upon mode of operation.

Useful though the weather radar may be, its usefulness does greatly depend upon interpretation by the flight crew. As in other areas, the flight crew are unlikely to depend upon the information provided by the weather radar alone, but are likely to confer with air traffic controllers and take account of status reports from aircraft that have already flown through the area. Reference (**5**) gives a more detailed description of the operation of a modern weather radar.

Fig. 5.33 *Typical weather radar schematic*

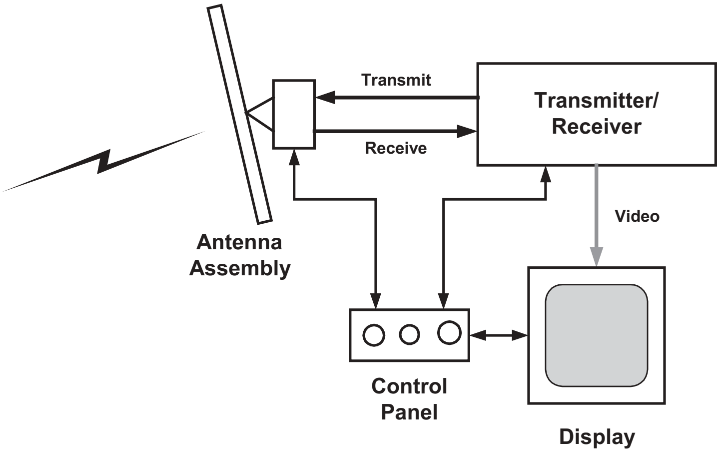
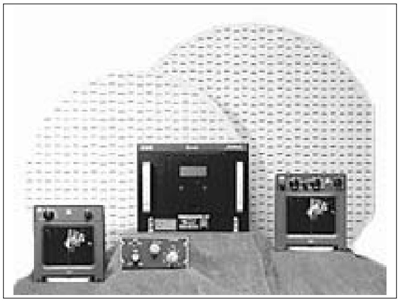


Fig. 5.34 *Weather radar units –Honeywell RDR-4B (Honeywell)*



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(1) Erroneous Flight Instruments, *Boeing Aero* **8**, 1999.

(2) Operational use of Angle of Attack, *Boeing Aero* **12**, 2000.

(3) The derivation of the World Magnetic Model 2000, January 2000, British Geological Survey – Technical Report WM/00/17R.

(4) Modern Inertial Navigation Technology and its Application. *IEE Electronics and Communication Engineering Journal*, 2000.

(5) Pilot’s Handbook – Honeywell Weather Radar RDR-4B.

CHAPTER 6

Communications and Navigation Aids

The sensors described in Chapter 5 are those that are on-board or are autonomous to the aircraft and that do not require the assistance of a third party. However, the aircraft also uses a number of other systems, either for communications or for navigational assistance, that depend upon external agencies in terms of beacons, transmitters, and other support.

Communications systems comprise the following:

- High-Frequency (HF) radio transmit/receive.
- Very High Frequency (VHF) radio transmit/receive and an Aircraft Communications And Reporting System (ACARS).
- Ultra High-Frequency (UHF) radio transmit/receive – mainly used in military communications.
- SATellite COMmunications (SATCOM) including passenger telephone communications.
- Aircraft transponder and Air Traffic Control (ATC) mode A/C and S [also known in the military environment as Identification Friend or Foe/Secondary Surveillance Radar (IFF/SSR)].
- Traffic Collision and Avoidance System (TCAS).
- Communications control.

Common navigation aids are:

- Very High Frequency OmniRange (VOR).
- Distance Measuring Equipment (DME).
- Automatic Direction Finding (ADF).
- TACtical Air Navigation system (TACAN).
- VOR/TACAN (VOR/TAC).
- Hyperbolic navigation systems – typically LORAN C.

- Instrument Landing System (ILS).
- Microwave Landing System (MLS).
- Global Navigation Satellite Systems (GNSSs), of which the Global Positioning System (GPS) is the most notable.

Radio Frequency spectrum

The Radio Frequency (RF) spectrum from 10 kHz (1 × 10^4 Hz) up to 10 GHz (1 × 10^{10} Hz) is shown in a simplified form in Fig. 6.1. This spectrum, stretching over six decades, covers the range in which most of the civil aircraft communications and navigation equipment operate. For military aircraft the spectrum will be wider, as attack radars, Electronics Warfare (EW), and infrared sensors need also to be included. The wide frequency coverage of this spectrum and the nature of radio wave propagation mean that the performance of different equipment varies according to the conditions of operation. Figure 6.1 distinguishes between communications and navigation aids. It can be seen that the part of the spectrum at which aircraft equipment and wiring systems are most susceptible to emissions covers a wide band. Therefore, care has to be taken when designing and operating the aircraft to keep any mutual interference effects to a minimum. The figure also shows that various ground-based domestic equipment can have an adverse effect. This includes: HF communications, VHF TV and radio transmissions, UHF TV transmissions, and ground-based radars.

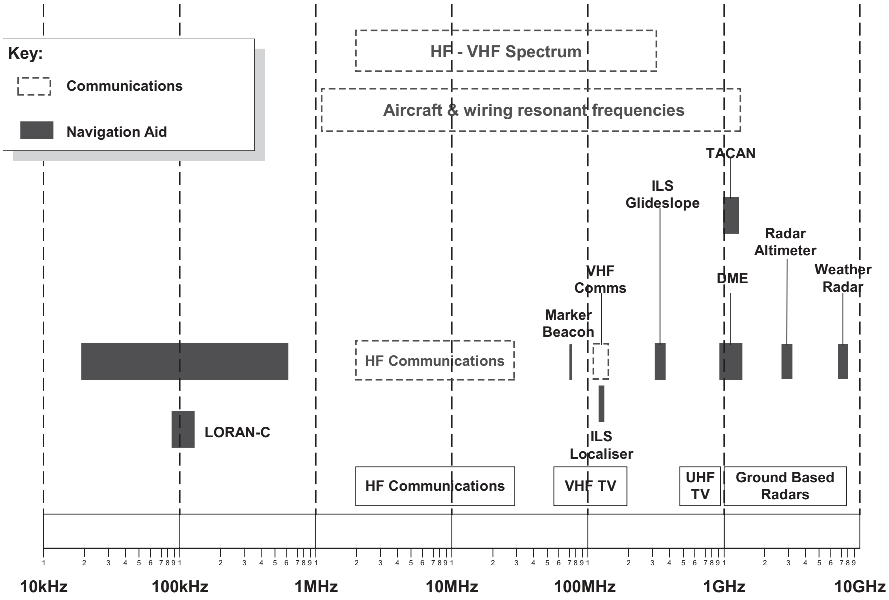


Fig. 6.1 *Simplified radio frequency spectrum – civil use*

The broad categorization of radio frequency bands is as shown in Table 6.1. Some of the higher frequencies are also categorized by a letter designation, as shown in Table 6.2. Note that this designation method is not contiguous as for the notation applied in Table 6.1. Several bands overlap, and this designation system, as well as being historical, tends to categorize bands with similar properties.

Table 6.1. Broad categorization of radio frequency bands

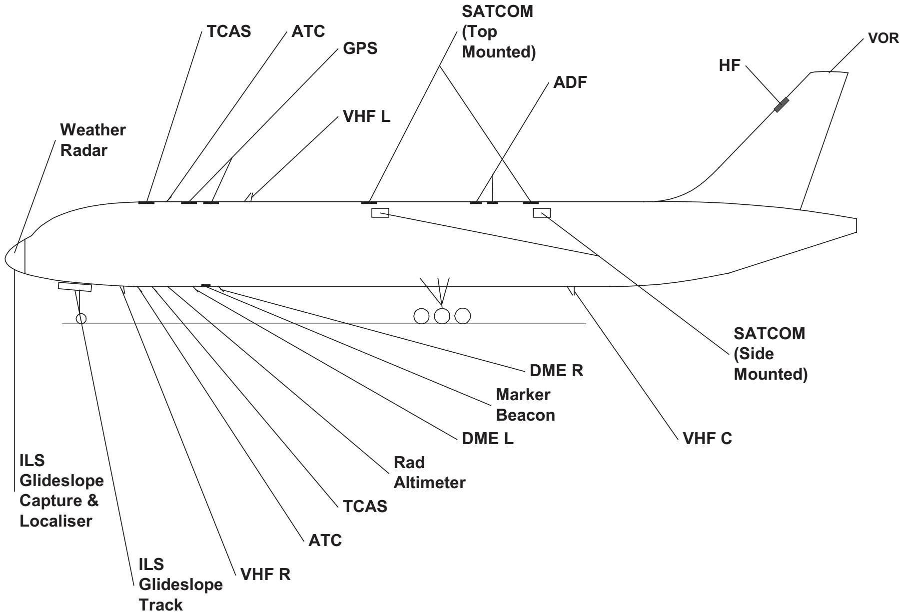
Name	Abbreviation	Frequency
Very low frequency	VLF	3–30 kHz
Low frequency	LF	30–300 kHz
Medium frequency	MF	300–3000 kHz (3 Mhz)
High frequency	HF	3–30 MHz
Very high frequency	VHF	30–300 MHz
Ultrahigh frequency	UHF	300–3000 MHz (3GHz)
Super high frequency	SHF	3–30 GHz
Extremely high frequency	EHF	30–300 GHz

Table 6.2. Letter designation of higher-frequency bands

Letter designation	Frequency range (GHz)
L	0.39–1.55
Ls	0.90–0.95
S	1.55–5.20
C	3.90–6.20
X	5.20–10.90
Xb	6.25–6.90
K[1]	10.90–17.25
Ku	15.35–17.25
Ka	33.00–36.00
Q	36.00–46.00

The number of antennae required on-board an aircraft to handle all the sensors, communications, and navigation aids is considerable. This is compounded by the fact that many of the key pieces of equipment may be replicated in duplicate or triplicate form. This is especially true of VHF, HF, VOR, and DME equipments. Figure 6.2 shows typical antenna locations on a Boeing 777 aircraft; this is indicative of the installation on most civil aircraft operating today, particularly those operating on transoceanic routes. Owing to their operating characteristics and transmission properties, many of these antennae have their own installation criteria. SATCOM antennae, which communicate with satellites, will have the antennae mounted on top of the aircraft so as to have the best coverage of the sky. ILS antennae, associated with the approach and landing phase, will be located on the forward, lower side of the fuselage. Others may require continuous coverage while the aircraft is manoeuvring and may have antennae located on both upper and lower parts of the aircraft; multiple installations are commonplace.

Fig. 6.2 *Boeing 777 antennae locations*



Communications systems

In aviation, communications between the aircraft and the ground (air traffic/local approach/ground handling) have historically been by means of voice communication. More recently, data-link communications have been introduced owing to their higher data rates and in some cases superior operating characteristics. As will be seen, data links are becoming widely used in the HF and VHF bands for basic communications, but also to provide some of the advanced reporting features required by FANS.

After selecting the appropriate communications channel on the channel selector, the pilot transmits a message by pressing the transmit button which connects the microphone to the appropriate radio. The voice message is used to modulate the carrier frequency, and it is this composite signal that is transmitted.

A typical voice signal is shown in the lower part of Fig. 6.3, while the Amplitude Modulated (AM) signal that is transmitted is shown in the upper portion. The receiver demodulates the incoming signal to recover the original voice component. The advantage of this very simple method of transmission is that it is extremely easy to use – all the pilot has to do is speak. A disadvantage is that it occupies a wide bandwidth, typically ~5 kHz, and that speech is not a particularly efficient method of using time and bandwidth compared with data-link applications.

The frequency components associated with amplitude modulation are summarized in Fig. 6.4. The simple AM case is shown on the left, where it can be seen that the carrier is accompanied by upper and lower sidebands (SBs). The example shows the spectrum that would be produced when a carrier of 2100 kHz (2.1 MHz) is being amplitude modulated by a 1 kHz tone. The three constituent elements are:

1. Lower SB (LSB) at (carrier − tone) frequency = 2100 − 1 = 2099 kHz.
2. Carrier component at 2100 kHz.

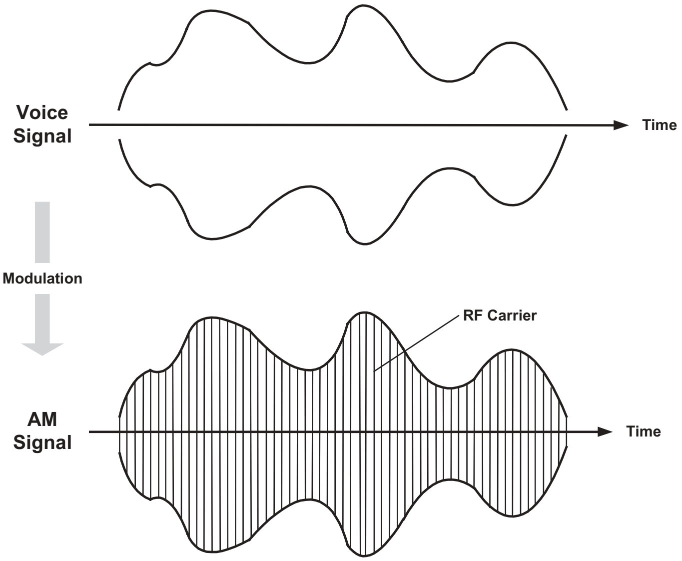


Fig. 6.3 Amplitude modulation (AM)

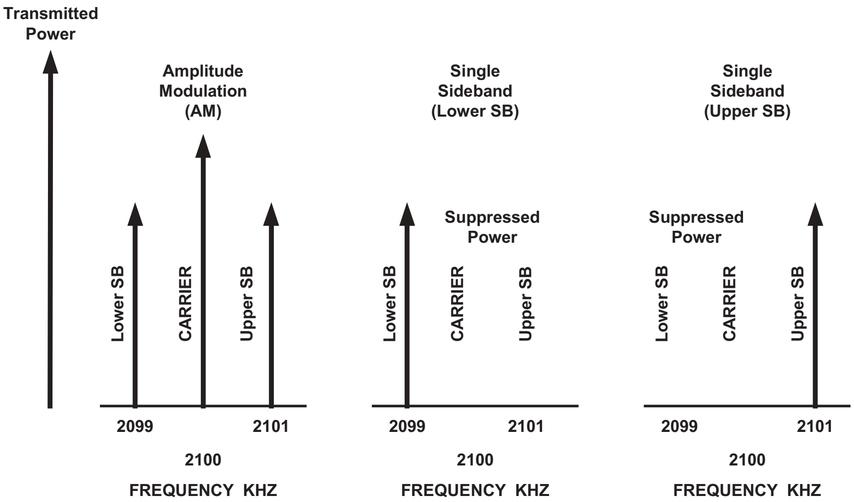


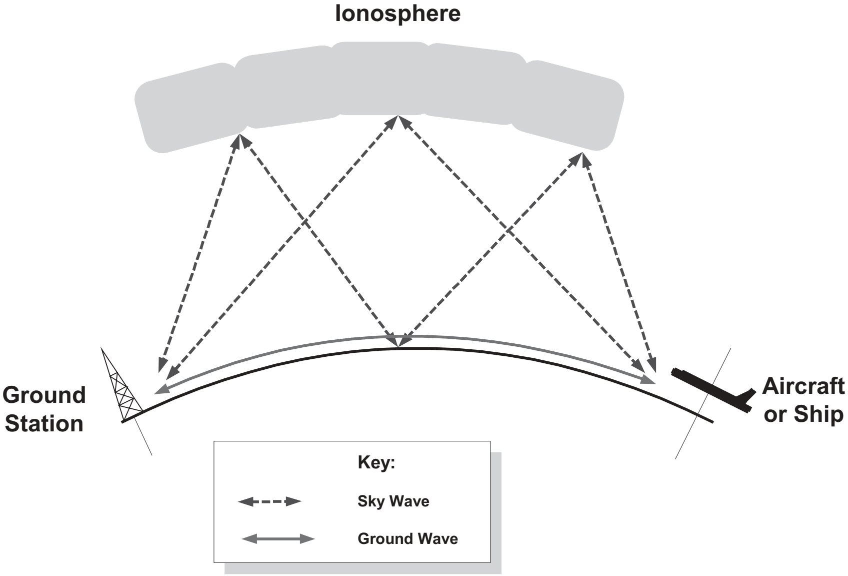
Fig. 6.4 Amplitude modulation and single sideband operation

3. Upper SB (USB) at (carrier + tone) frequency = 2100 + 1 = 2101 kHz.

It can be seen that energy is wasted in that power is being transmitted on both SBs and carrier while the effective signal could be decoded from either LSB or USB. Therefore, the technique of Single SideBand (SSB) has been developed which transmits either the upper or lower SB while suppressing the carrier. This SSB operation can yield effectively 8 times more signal power than AM without any power increase at the transmitter. The SSB techniques are used especially in HF communications: the USB is used extensively for aviation, while the LSB is used for other services such as amateur radio.

The principles of single-sideband LSB and USB operation are shown in the centre and right-hand diagrams respectively of Fig. 6.4.

Fig. 6.5 *HF communication signal propogation*



High Frequency

High Frequency covers the communications band between 3 and 30 MHz and is a very common communications means for land, sea, and air. The utilized band is HF SSB/AM over the frequency range 2.000–29.999 MHz using a 1 kHz (0.001 MHz) channel spacing. The primary advantage of HF communications is that this system offers communication beyond the line of sight. This method does, however, suffer from idiosyncrasies with regard to the means of signal propagation.

Figure 6.5. shows that there are two main means of propagation, known as the sky wave and the ground wave. The sky wave method of propagation relies upon single- or multiple-path bounces between the Earth and the ionosphere until the signal reaches its intended location. The behaviour of the ionosphere is itself greatly affected by radiation falling upon the Earth, notably solar radiation. Times of high sunspot activity are known adversely to affect the ability of the ionosphere as a reflector. It may also be affected by the time of day and other atmospheric conditions. The sky wave as a means of propagation may therefore be severely degraded by a variety of conditions,occasionally to the point of being unusable.

The ground wave method of propagation relies upon the ability of the wave to follow the curvature of the earth until it reaches its intended destination. As for the sky wave, the ground wave may on occasions be adversely affected by atmospheric conditions. Therefore, on occasions HF voice communications may be corrupted and prove unreliable, although HF data links are more resistant to these propagation upsets as described below.

HF communications are one of the main methods of communicating over long ranges between air and ground during oceanic and wilderness crossings when there is no line of sight between the aircraft and ground communications stations. For reasons of availability, most long-range civil aircraft are equipped with two HF sets, with an increasing tendency also to use HF Data Link (HFDL) if polar operations are contemplated.

HF Data Link **(**HFDL**)** offers an improvement over HF voice communications owing to the bit encoding inherent in a data link message format which permits the use of error-correcting codes. Furthermore the use of more advanced modulation and frequency management techniques allows the data link to perform in propagation conditions where HF voice would be unusable or incomphrehensible. An HFDL service is provided by ARINC using a number of ground stations. These ground stations provide coverage out to ~2700 nautical miles and on occasion provide coverage beyond that. Presently, HFDL ground stations are operating at the following locations (see also Fig. 6.6):

1. Santa Cruz, Bolivia.
2. Reykjavik, Iceland.
3. Shannon, Ireland.
4. Auckland, New Zealand.
5. Krasnoyarsk, Russia.
6. Johannesburg, South Africa.
7. Hat Yai, Thailand.
8. Barrow, Alaska.
9. Molokai, Hawaii, USA.
10. Riverhead, New York, USA.
11. San Francisco, California, USA.
12. Bahrain.
13. Gran Canaria, Canary Islands.

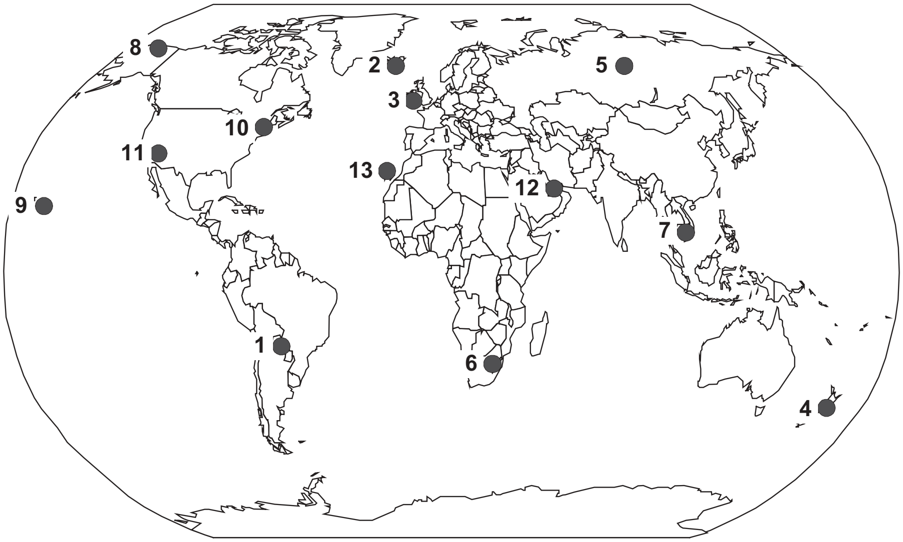


Fig. 6.6 *HF data-link ground stations*

Very High Frequency

Very High Frequency (VHF) voice communication is probably the most heavily used method of communication used by civil aircraft. The VHF band for aeronautical applications operates in the frequency range 118.000–135.975 MHz with a channel spacing in recent years of 25 kHz (0.025 MHz). In recent years, to overcome frequency congestion, and taking advantage of digital radio technology, channel spacing has been reduced to 8.33 kHz (0.00833 MHz), which permits 3 times more radio channels in the available spectrum. Some parts of the world are already operating on the tighter channel spacing – this will be discussed more in Chapter 12.

The VHF band also experiences limitations in the method of propagation. Except in exceptional circumstances, VHF signals will only propagate over line of sight. That is, the signal will only be detected by the receiver when it has line of sight or can ‘see’ the transmitter. VHF transmissions possess neither of the qualities of HF transmission and accordingly neither sky wave nor ground wave properties apply. This line-of-sight property is affected by the relative heights of the radio tower and aircraft (see Fig. 6.7).

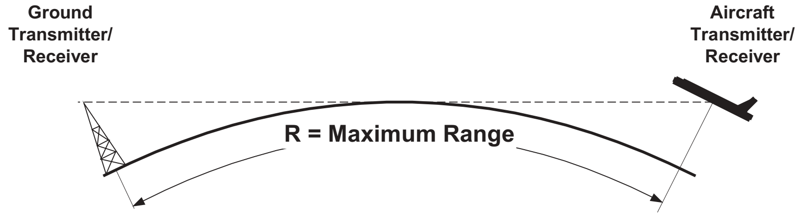


Fig. 6.7 *VHF signal propagation*

The formula that determines the line-of sight range for VHF transmissions is as follows

$$R = 1.2\sqrt{H_t} + 1.2\sqrt{H_a}$$

where R is the range (nautical miles), H_t is the height of the transmission tower (ft), and H_a is the height of the aircraft (ft). Therefore, for an aircraft flying at 35 000 ft, transmissions will generally be received by a 100 ft high radio tower if the aircraft is within a range of around 235 nautical miles.

Additionally, VHF transmissions may be masked by terrain, by a range of mountains for example. These line-of-sight limitations also apply to equipment operating in higher-frequency bands and mean that VHF communications, and other equipment operating in the VHF band or above, such as the navigation aids VOR and DME, may not be used except over large land masses, and then only when there is adequate transmitter coverage. Most long-range aircraft have three pieces of VHF equipment, with one usually being assigned to ARINC ACARS transmissions though not necessarily dedicated to that purpose. The requirements for certifying the function of airborne VHF equipment are given in reference (**1**), while reference (**2**) specifies the necessary Minimum Operational Performance Standards (MOPS).

Both HF and VHF communications incorporate a feature termed SELective CALling (SELCAL). It enables a ground controller to place selective controls on an individual aircraft. If the ground controller wishes to establish communication with an aircraft on a selected frequency, he or she selects a code that relates specifically to the aircraft and initiates the transceiver on a frequency known to be monitored by the crew. When the encoded SELCAL message is received by the aircraft, the message is decoded and, if the correct coding sequence is detected, the crew are alerted by a visual or aural annunciator. The flight crew can then communicate normally with the ground station.

A number of VHF Data Links (VHFDLs) may be used, and these are discussed in more detail in Chapter 12 – FANS. ACARS is a specific variant of VHF communications operating on 131.55 MHz which utilizes a data link rather than voice transmission. As will be seen during the discussion on future air navigation systems, data link rather than voice transmission will increasingly be used for air-to-ground, and air-to-air communications as higher data rates may be used while at the same time reducing flight crew workload. ACARS is dedicated to downlinking operational data to the airline operational control centre. The initial leg is by using VHF communications to an appropriate ground receiver, and thereafter the data may be routed via landlines or microwave links to the airline operations centre. At this point it will be allowed access to the internal airline storage and management systems: operational, flight crew, maintenance, etc.

Originally, only four basic event parameters were transmitted: OUT–OFF–ON–IN, abbreviated to OOOI:

OUT	Aircraft is clear of the gate and ready to taxi
OFF	Aircraft has lifted off the runway
ON	Aircraft has landed
IN	Aircraft has taxied to the ramp area

Now, data such as fuel state, aircraft serviceability, arrival and departure times, weather, crew status, and so on are also included in the data messages. ACARS was introduced to assist the operational effectiveness of an airline; future data-link applications will allow the transfer of more complex data relating to air traffic control routing and flight planning. On-board the aircraft, ACARS introduces a dedicated management unit, control panel, and printer to provide the interface with the flight crew for formatting, dispatching, receiving, and printing messages. This, together with existing VHF equipment and an interface with the Flight Management System (FMS), forms a typical system as shown in Fig. 6.8.

All aircraft and air traffic control centres maintain a listening watch on the international distress frequency (121.5 MHz). In addition, military controllers maintain

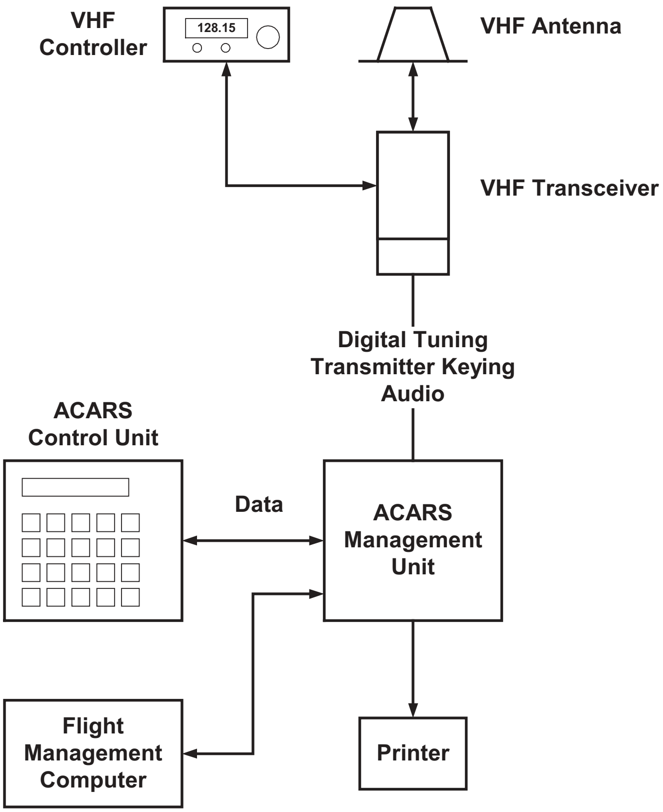


Fig. 6.8 *Typical ACARS configuration*

a listening watch on 243.0 MHz in the UHF band. This is because the UHF receiver could detect harmonics of a civil VHF distress transmission and relay the appropriate details in an emergency (second harmonic of 121.5 MHz (×2) = 243.0 MHz; these are the international distress frequencies for VHF and UHF bands respectively).

Satellite communications

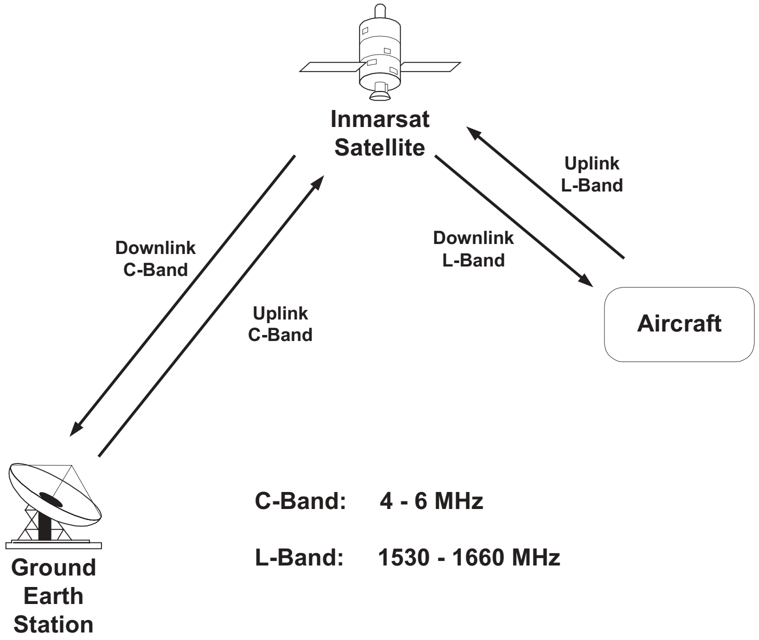
Satellite communications provide a more reliable method of communications using the International Maritime Satellite Organization (INMARSAT) satellite constellation which was originally developed for maritime use. Now, satellite communications, abbreviated to SATCOM, form a useful component of aerospace communications.

The principles of operation of SATCOM are shown in Fig. 6.9. The aircraft communicates via the INMARSAT constellation and remote ground earth station by means of C-band uplinks and downlinks to/from the ground stations and L-band links to/from the aircraft. In this way, communications are routed from the aircraft via the satellite to the ground station and on to the destination. Conversely, communications to the aircraft are routed in the reverse fashion. Therefore, provided the aircraft is within the area of coverage or footprint of a satellite, then communication may be established.

The airborne SATCOM terminal transmits on frequencies in the range 1626.5–1660.5 MHz and receives messages on frequencies in the range 1530.0–1559.0 MHz. Upon power-up, the Radio Frequency Unit (RFU) scans a stored set of frequencies and locates the transmission of the appropriate satellite. The aircraft logs onto the ground earth station network so that any ground stations are able to locate the aircraft. Once logged onto the system-communications between the aircraft and any user may begin. The satellite-to-ground C-band uplink/downlink are invisible to the aircraft, as is the remainder of the Earth support network.

The coverage offered by the INMARSAT constellation was a total of four satellites

Fig. 6.9 *SATCOM principles of operation*



in 2001. Further satellites are planned to be launched. The INMARSAT satellites are placed in Earth geostationary orbit above the equator in the locations shown in Fig. 6.10:

- Two satellites are positioned over the Atlantic: AOR-W at 54º west and AOR-E at 15.5º west.
- One satellite is positioned over the Indian ocean: IOR at 64º east.
- One satellite is positioned over the Pacific ocean: POR at 178º east.

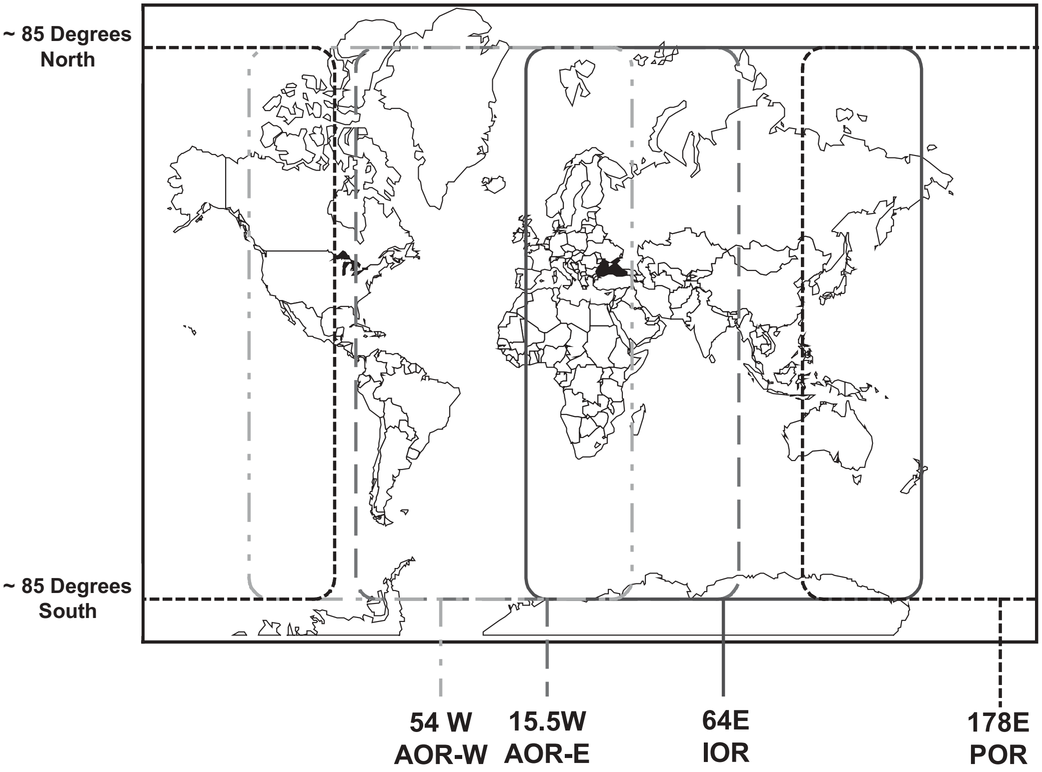


Fig. 6.10 *INMARSAT satellite coverage – 2001*

Blanket coverage is offered over the entire footprint of each of these satellites. In addition there is a spot beam mode which provides cover over most of the land mass residing under each satellite. This spot beam coverage is available to provide cover to lower-capability systems that do not require blanket oceanic coverage.

The geostationary nature of the satellites does impose some limitations. Owing to low grazing angles, coverage begins to degrade beyond 80° north and 80° south and fades completely beyond about 85°. Therefore, no coverage exists in the extreme polar regions, a fact assuming more prominence as airlines seek to expand northern polar routes. A second limitation may be posed by the performance of the on-board aircraft system in terms of antenna installation, and this is discussed shortly. Nevertheless, SATCOM is proving to be a very useful addition to the airborne communications suite and promises to be an important component as Future Air Navigation System (FANS) procedures are developed.

A number of different systems are available as described in Table 6.3. A SATCOM system typically comprises the following units:

- Satellite Data Unit (SDU).
- Radio Frequency Unit.
- Amplifiers, Diplexers/Splitters.
- Low-gain antenna.
- High-gain antenna.

Table 6.3. SATCOM configurations

Configuration	Capabilities
Aero-H/H+	High gain. Aero-H offers a high-gain solution to provide a global capability and is used by long-range aircraft. Aero H+ was an attempt to lower cost by using fewer satellite resources. It provides cockpit data, cockpit voice, and passenger voice services
Aero-I	Intermediate gain. Aero-I offers similar services to Aero-H/H+ for medium- and short-range aircraft. Aero-I uses the spot beam service
Aero-C	Version that allows passengers to send and receive digital messages from a PC
Aero-M	Single-channel SATCOM capability for general aviation users

A typical SATCOM system as installed on the B777 is shown in Fig. 6.11. This example uses extensive ARINC 429 data buses for control and communication between the major system elements. This configuration demonstrates the use of two high-gain conformal antennae which are mounted on the upper fuselage at positions approximately ±20° respectively from the vertical. Conformal antennae lie flush with the aircraft skin, offering negligible additional drag. Alternatively, the system may be configured such that a single top-mounted antenna may be mounted on the aircraft spine. Both systems have their protagonists and opponents. Claims and counterclaims are made for which antenna configuration offers the best coverage. Conformal configurations reportedly suffer from fuselage obscuration dead-ahead and dead-astern, while the top-mounted rival supposedly suffers from poor coverage at low grazing angle near the horizon. Whatever the relative merits, both configurations are widely used by airlines today.

Figure 6.12 shows a top-mounted SATCOM antenna with its associated Low-Noise Amplifier (LNA)/diplexer.

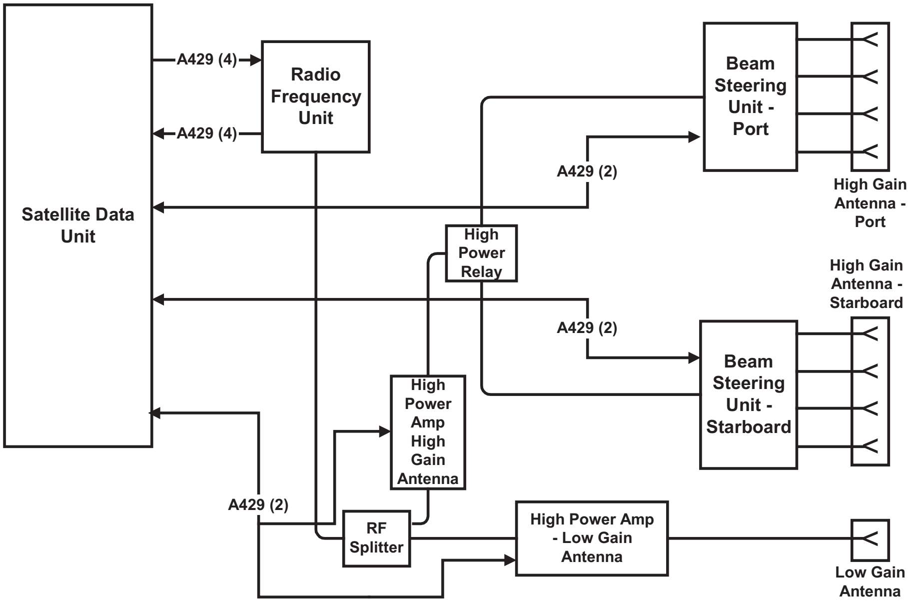


Fig. 6.11 *SATCOM system using conformal antennae*

Fig. 6.12 *Top-mounted SATCOM antenna with LNA/diplexer (CMC electronics)*

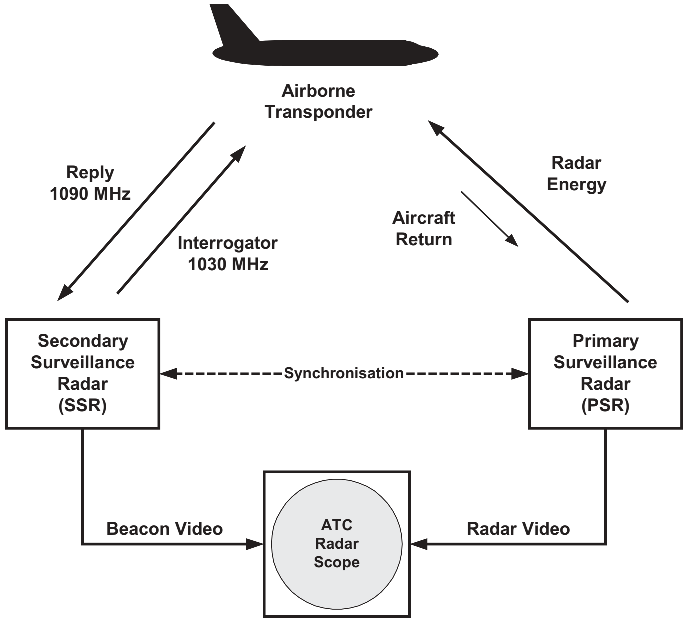
Air Traffic Control (ATC) transponder

As a means to aid the identification of individual aircraft and to facilitate the safe passage of aircraft through controlled airspace, the ATC transponder allows ground surveillance radars to interrogate aircraft and decode data, which enables correlation of a radar track with a specific aircraft. The principle of transponder operation is shown in Fig. 6.13. A ground-based Primary Surveillance Radar (PSR) will transmit radar energy and will be able to detect an aircraft by means of the reflected radar energy – termed the aircraft return. This will enable the aircraft return to be displayed on an ATC console at a range and bearing commensurate with the aircraft position. Coincident with the primary radar operation, a Secondary Surveillance Radar (SSR) will transmit a series of interrogation pulses that are received by the on-board aircraft transponder.

The transponder aircraft replies with a different series of pulses that give information relating to the aircraft, normally aircraft identifier and altitude. If the PSR and SSR are synchronized, usually by being co-boresighted, then both the presented radar returns and the aircraft transponder information may be presented together on the ATC console. Therefore, the controller will have aircraft identification (e.g. BA 123) and altitude presented alongside the aircraft radar return, thereby greatly improving the controller's situational awareness.

The system is also known as Identification Friend or Foe (IFF)/Secondary Surveillance Radar (SSR), and this nomenclature is in common use in the military field.

Fig 6.13 *Principle of transponder operation*



On-board the aircraft, the equipment fit is as shown in Fig. 6.14. The main elements are:

- ATC transponder controller unit for setting modes and response codes.
- Dedicated ATC transponder unit.
- An ATC antenna unit with an optional second antenna. It is usual to utilize both upper and lower mounted antenna to prevent blanking effects as the aircraft manoeuvres.

The SSR interrogates the aircraft by means of a transmission on the dedicated frequency of 1030 MHz which contains the interrogation pulse sequence. The aircraft transponder replies on a dedicated frequency of 1090 MHz with a response that contains the reply pulse sequence with additional information suitably encoded in the pulse stream.

In its present form, the ATC transponder allows aircraft identification – usually the airline call-sign – to be transmitted when using mode A. When mode C is selected, the aircraft will respond with its identifier, together with altitude information.

More recently, an additional mode – mode S or mode Select – has been introduced with the intention of expanding this capability. In ATC mode S, the SSR uses more

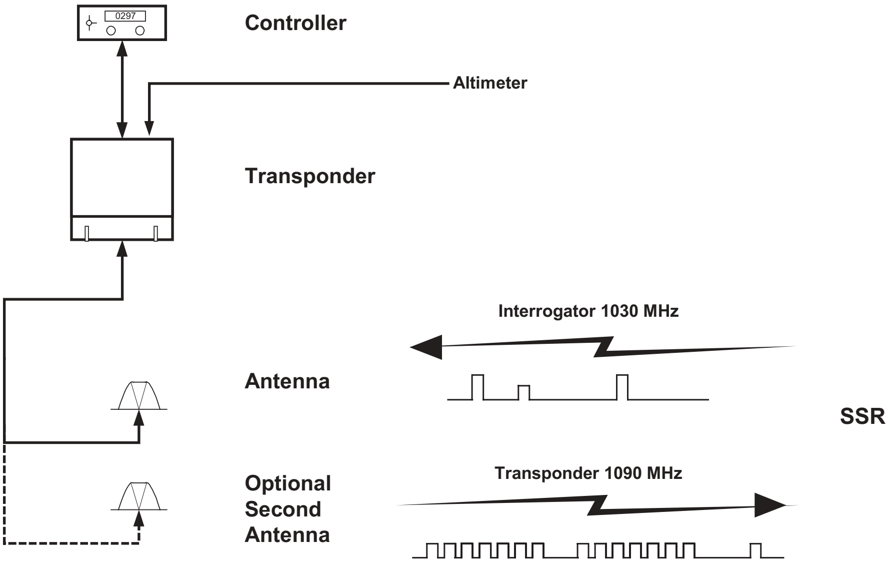


Fig 6.14 *Airborne transponder equipment*

sophisticated monopulse techniques that enable the aircraft azimuth bearing to be determined more quickly. Upon determining the address and location of the aircraft, it is entered into a roll call file. This, together with details of all the other aircraft detected within the interrogator's sphere of operation, forms a complete tally of all the aircraft in the vicinity. Each mode S reply contains a discrete 24 bit address identifier. This unique address, together with the fact that the interrogator knows where to expect the aircraft from its roll call file, enables a large number of aircraft to operate in a busy air traffic control environment – see the next section on the traffic collision avoidance system.

ATC mode S has other features that enable it to provide the following:

- Air-to-air as well as air-to-ground communication.
- The ability of aircraft autonomously to determine the precise whereabouts of other aircraft in their vicinity.

Mode S is an improved conventional secondary radar operating at the same frequencies (1030/1090 MHz). Its 'selectivity' is based on unambiguous identification of each aircraft by the unique 24 bit address. This acts as its technical telecommunications address, but does not replace the mode A code. There are also plans for recovery of the A and C codes via mode S.

Apart from this precise characterization of the aircraft, mode S protects the data it transmits owing to the inclusion of several parity bits that mean that up to 12 erroneous bits may be tolerated by the application of error detection and correction algorithms. For transmission, these parity bits are superimposed on those of the mode S address.

Finally, mode scan can be used to exchange longer, more varied data streams, which can even be completely unplanned. To do this, mode S transmissions between the station and the transponder use highly sophisticated 56 or 112 bit formats called frames. They fall into three main categories: 56 bit surveillance formats, 112 bit communication formats with a 56 bit data field, which are in fact 'extended' surveillance formats (uplink COMM-As and downlink COMM-Bs), and 112 bit communication formats with an 80 bit data field (uplink COMM-Ds and downlink COMM-Ds). This feature will be of use in facilitating the transmission and interchange of flight plans dynamically revised in flight which is one of the longer-term aims of FANS.

When used together with TCAS, ATC mode S provides an important feature for FANS, that of Automatic Dependent Surveillance – A (ADS-A). This capability will assist the safe passage of aircraft when operating in a direct routing mode.

Traffic Collision and Avoidance System

The TCAS was developed in prototype form during the 1960s and 1970s to provide a surveillance and collision avoidance system to help aircraft avoid collisions. It was certified by the FAA in the 1980s and has been in widespread use in the United States in its initial form. TCAS is based on the beacon interrogator and operates in a similar fashion to the ground-based SSR already described. The system comprises two elements: a surveillance system and a collision avoidance system. TCAS detects the range bearing and altitude of aircraft in the near proximity for display to the pilots.

TCAS transmits a mode C interrogation search pattern for mode A and C transponder equipped aircraft and receives replies from all such equipped aircraft. In addition, TCAS transmits one mode S interrogation for each mode S transponder equipped aircraft, receiving individual responses from each one. It will be recalled that mode A relates to range and bearing, while mode C relates to range, bearing, and altitude and mode S relates to range, bearing, and altitude with a unique mode S reply. The aircraft TCAS equipment comprises a radio transmitter and receiver, directional antennae, a computer, and flight deck display. Whenever another aircraft receives an interrogation, it transmits a reply and the TCAS computer is able to determine the range ferom the time taken to receive the reply. The directional antennae enable the bearing of the responding aircraft to be measured. TCAS can track up to 30 aircraft but only display 25, the highest-priority targets being the ones that are displayed.

TCAS is unable to detect aircraft that are not carrying an appropriately operating transponder or that have unserviceable equipment. A transponder is mandated if an aircraft flies above 10 000 ft or within 30 miles of major airports; consequently, all commercial aircraft and the great majority of corporate and general aviation aircraft are fitted with the equipment.

TCAS exists in two forms: TCAS I and TCAS II**.** TCAS I indicates the range and bearing of aircraft within a selected range, usually 15–40 nautical miles forward, 5–15 nautical miles aft, and 10–20 nautical miles on each side. The system also warns of aircraft within ±8700 ft of the aircraft's own altitude.

The collision avoidance system element predicts the time to, and separation at, the intruder's closest point of approach. These calculations are undertaken using range, closure rate, altitude, and vertical speed. Should the TCAS ascertain that certain safety boundaries will be violated, it will issue a traffic advisory (TA) to alert the crew that closing traffic is in the vicinity via the display of certain coloured symbols. Upon receiving a TA, the flight crew must visually identify the intruding aircraft and may alter their altitude by up to 300 ft. A TA will normally be advised between 20 and 48 s before the point of closest approach with a simple audio warning in the flight crew's headsets: 'TRAFFIC, TRAFFIC'. TCAS I does not offer any deconfliction solutions but does provide the crew with vital data in order that they may determine the best course of action.

TCAS II offers a more comprehensive capability, with the provision of Resolution Advisories (RAs). TCAS II determines the relative motion of the two aircraft and determines an appropriate course of action. The system issues an RA via mode S, advising the pilots to execute the necessary manoeuvre to avoid the other aircraft. An RA will usually be issued when the point of closest approach is within 15 and 35 s and the deconfliction symbology is displayed coincident with the appropriate warning.
A total of ten audio warnings may be issued. Examples are:

- ‘CLIMB, CLIMB, CLIMB’.
- ‘DESCEND, DESCEND, DESCEND’.
- ‘REDUCE CLIMB, REDUCE CLIMB’.

Finally, when the situation is resolved, ‘CLEAR OF CONFLICT’.

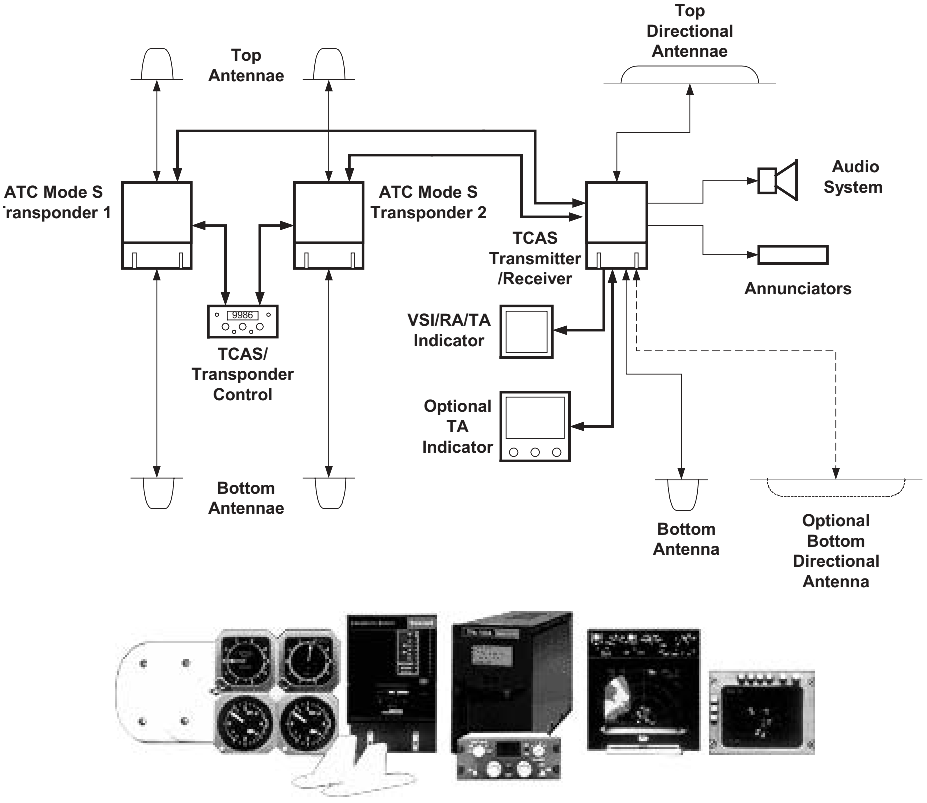


Fig. 6.15 *TCAS architecture showing related equipment and displays (Honeywell)*

TCAS II clearly requires a high level of integration between the active equipment. Figure 6.15 shows the interrelationship between:

- TCAS transmitter/receiver.
- ATC mode S transponders.
- VSI display showing vertical guidance for TAs and RAs.
- Optional horizontal situational indicator for RAs that could be the navigation display.

- Audio system and annunciators.
- Antennae for ATC mode S and TCAS.

This is indicative of the level of integration required between ATC mode S transponders, TCAS, displays, and annunciators. It should be noted that there are a variety of display options, and the system shown does not represent the only TCAS option.

More recently, further changes have been introduced into TCAS II – known as TCAS II Change 7. This introduces software changes and updated algorithms that alter some of the TCAS operating parameters. Specifically, Change 7 includes the following features:

- Elimination of nuisance warnings.
- Improved RA performance in a multi-aircraft environment.
- Modification of vertical thresholds to align with Reduced Vertical Separation Minima (RVSM) (see Chapter 12).
- Modification of RA display symbology and aural annunciations.

The Change 7 modifications became mandatory in Europe for aircraft with 30 seats or more by 31 March 2001. The rest of the world will be following a different but broadly similar time-scale for implementation. Change 7 is not mandated in the United States but it is expected that most aircraft will be equipped to that standard in any case. References (**3**)–(**6**) relate to certification and performance requirements for TCAS II and mode S.

Communications control system

The control of the aircraft suite of communications systems, including internal communications, has become an increasingly complex task. This task has expanded as aircraft speeds and traffic density have increased and the breadth of communications types has expanded. The communications control function is increasingly being absorbed into the flight management function as the management of communications type, frequency selection, and intended aircraft flight path become more interwoven. Now, the flight management system can automatically select and tune the communications and navigation aids required for a particular flight leg, reducing crew workload and allowing the crew to concentrate more on managing the on-board systems.

Navigation aids

As aviation began to expand in the 1930s, the first radio navigation systems were developed. Initially, these were installed at the new growing US airports, and it is interesting to note that the last of these early systems was decommissioned as recently as 1979.

One of the most prominent was the ‘radio range’ system developed in Italy by Bellini and Tosi, which was conceived as early as 1907. The operation of the Bellini–Tosi system relied upon the transmission of Morse characters A (dot-dash) and N (dash-dot) in four evenly spaced orthogonal directions. When flying the correct course, the A and N characters combined to produce a humming noise which the pilot could detect in earphones. Deviation from the desired course would result in either the

A or N characters becoming more dominant, signifying the need for corrective action by turning left or right as appropriate.

Following World War II, the International Civil Aviation Organization (ICAO) produced international standards that led to the definition of the Very High Frequency OmniRange (VOR) system which is in widespread use today and which is described below.

Automatic Direction Finding

Automatic Direction Finding (ADF) involves the use of a loop direction finding technique to establish the bearing to a radiating source. This might be to a VHF beacon or a Non-Directional Beacon (NDB) operating in the 200–1600 kHZ band. NDBs in particular are the most prolific and widely spread beacons in use today. The aircraft ADF system comprises integral sense and loop antennae which establish the bearing of the NDB station to which the ADF receiver is tuned. The bearing is shown on the Radio Magnetic Indicator (RMI) or Electronic Flight Instrument System (EFIS) as appropriate. A typical RMI display is shown in Fig. 6.16.

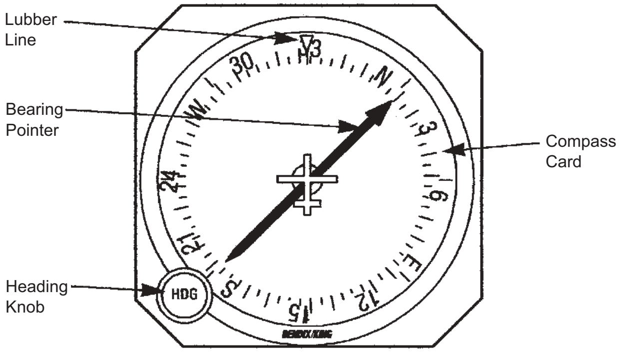


Fig. 6.16 *Typical Radio Magnetic Indicator (RMI)*

Very High Frequency Omnirange

The VOR system was accepted as standard by the United States in 1946 and later adopted by the ICAO as an international standard. The system provides a widely used set of radio beacons operating in the VHF frequency band over the range 108–117.95 MHz with 100 kHz spacing. Each beacon emits a morse code modulated tone which may be provided to the flight crew for the purposes of beacon identification.

The ground station radiates a cardoid pattern that rotates at 30 rev/min, generating a 30 Hz modulation at the aircraft receiver. The ground station also radiates an omnidirectional signal which is frequency modulated with a 30 Hz reference tone. The phase difference between the two tones varies directly with the bearing of the aircraft. At the high frequencies at which VHF operates there are no sky wave effects and the system performance is relatively consistent. VOR has the disadvantage that it can be severely disrupted by adverse weather – particularly by electrical storms – and as such it cannot be used as a primary means of navigation for a civil aircraft.

Overland in the North American continent and Europe, VOR beacons are widely situated to provide an overall coverage of beacons. Usually these are arranged to coincide with major airway waypoints and intersections in conjunction with DME stations (see below) so that the aircraft may navigate for the entire flight using the extensive route/beacon structure. By virtue of the transmissions within the VHF band, these beacons are subject to the line-of-sight and terrain masking limitations of VHF communications. Reference (**7**) lays out a method for complying with the airworthiness rules for VOR/DME/TACAN.

Table 6.4 VOR error budget

Error component	Ascribed value
Radial signal error, E_g (based in practice on measured beacon data)	±1.4°
Airborne component error, E_a (based on the accuracy achieved by typical avionics systems – modern equipment will achieve higher accuracy)	±3.0°
Instrument error. E_i (based on an analogue system – digital systems would achieve better)	±2.0°
Flight technical error, E_f (assumed to be independent of the other variables – probably pessimistic)	±2.3°
Total error: $= \sqrt{(E_g^2 + E_a^2 + E_i^2 + E_f^2)}$	
	$= \sqrt{(1.4^2 + 3.0^2 + 2.0^2 + 2.3^2)}$ = ± 4.5°

The error experienced by the VOR system is ascribed to various sources and the root sum squared figure is taken to establish the overall error (see Table 6.4). Typical values for 95 per cent error probability are described in reference (**7**). This gives a total system error of ±4.5°. Later Doppler VOR (DVOR) installations, which are now widely used, will offer accuracies at least 10 times better than the example quoted above. The implications of VOR accuracy will be further examined in Chapter 12 – FANS.

Distance Measuring Equipment

Distance Measuring Equipment (DME) is a method of pulse ranging used in the 960–1215 MHz band to determine the distance of the aircraft from a designated ground station. The aircraft equipment interrogates a ground-based beacon and, upon the receipt of retransmitted pulses (unique to the on-board equipment), is able to determine the range to the DME beacon (see Fig. 6.17). DME beacons are able to service requests from a large number of aircraft simultaneously but are generally understood to have the capacity to handle ~200 aircraft at once. Specified DME accuracy is ±3 per cent or ±0.5 nm, whichever is the greater (**7**).

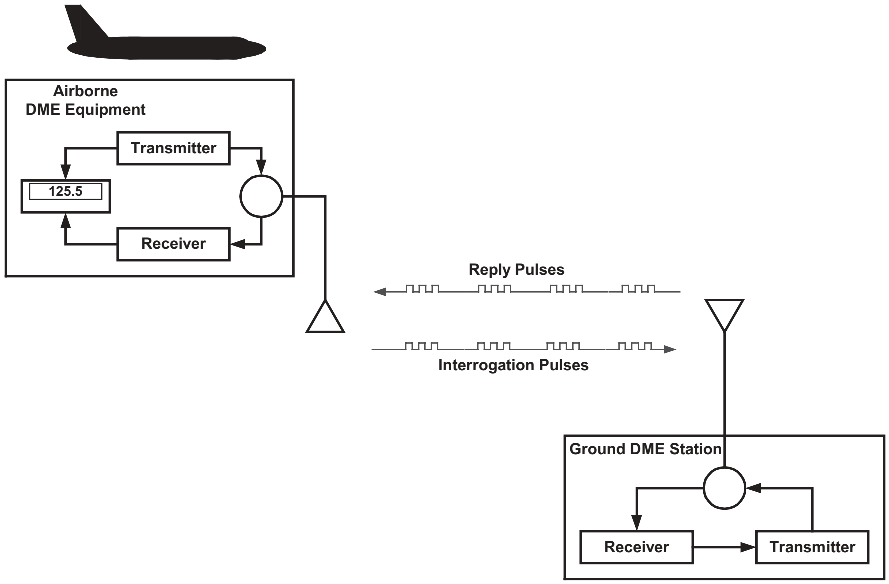


Fig 6.17 *DME principle of operation*

DME and TACAN beacons are paired with ILS/VOR beacons throughout the airway route structure in accordance with the table set out in Appendix 3 of reference (**7**). This is organized such that aircraft can navigate the airways by having a combination of VOR bearing to and DME distance to run to the next beacon in the airway route structure. A more recent development – scanning DME – allows the airborne equipment rapidly to scan a number of DME beacons, thereby achieving greater accuracy by taking the best estimate of a number of distance readings. This combination of VOR/DME navigation aids has served the aviation community well in the United States and Europe for many years, but it does depend upon establishing and maintaining a beacon structure across the land mass or continent being covered. New developments in third-world countries are more likely to skip this approach in favour of the Global Positioning System (GPS) as described in Chapter 8.

TACAN

TACAN (TACtical Air Navigation) is military omnibearing and distance measuring equipment that employs similar techniques for distance measurement to DME. The bearing information is accomplished by amplitude modulation achieved within the beacon which imposes 15 and 135 Hz modulated patterns and transmits this data together with 15 and 135 Hz reference pulses. The airborne equipment is therefore able to measure distance using DME interrogation techniques while using the modulated data to establish bearing.

TACAN beacons operate in the frequency band 960–1215 MHz, as opposed to the 108–118 MHz used by DME. This means that the beacons are smaller, making them suitable for shipborne and mobile tactical use. Some airborne equipment have the ability to offset to a point remote from the beacon which facilitates recovery to an airfield when the TACAN beacon is not co-located. TACAN is reportedly accurate to within ±1 per cent in azimuth and ±0.1 nm in range, so it offers accuracy improvements over VOR/DME.

VORTAC

As most military aircraft are equipped with TACAN, some countries provide VORTAC beacons which combine VOR and TACAN beacons. This allows interoperability of military and civil air traffic. Military users use the TACAN beacon while civil users use the VOR bearing and TACAN (DME) distance measuring facilities. This is especially helpful for large military aircraft, such as transport or surveillance aircraft, since they are able to use civil air lanes and operational procedures during training or in transit between theatres of operations.

Satellite navigation systems

These techniques were prevalent from the 1960s through to the 1990s when satellites became commonly available. The use of Global Navigation Satellite Systems (GNSSs), to use the generic name, offers a cheap and accurate navigational means to anyone who possesses a suitable receiver. Although the former Soviet Union developed a system called GLONASS, it is the US Global Positioning System (GPS) that is the most widely used.

GPS is a US satellite-based radio navigational, positioning, and time transfer system operated by the Department of Defense (DoD). The system provides highly accurate position and velocity information and precise time on a continuous global basis to an unlimited number of properly equipped users. The system is unaffected by weather and provides a worldwide common grid reference system based on the Earth-fixed coordinate system. For its earth model, GPS uses the world geodetic system of 1984 (WGS-84) datum.

The Department of Defense declared Initial Operational Capability (IOC) of the US GPS on 8 December 1993. The Federal Aviation Administration (FAA) has granted approval for US civil operators to use properly certified GPS equipment as a primary means of navigation in oceanic and certain remote areas. GPS equipment may also be used as a supplementary means of Instrument Flight Rules (IFR) navigation for domestic en route, terminal operations and certain instrument approaches.

The principles of satellite navigation using GPS are illustrated in Fig. 6.18. GPS comprises three major components as characterized in the figure:

- The control segment which embraces the infrastructure of ground control stations, monitor stations and ground-based satellite dishes that exercise control over the system.
- The space segment which includes the satellite constellation, presently around 25 satellites, that forms the basis of the network.
- The user segment which includes all the users: ships, trucks, automobiles, aircraft, and hand-held sets. In fact, anyone in possession of a GPS receiver is part of the user segment.

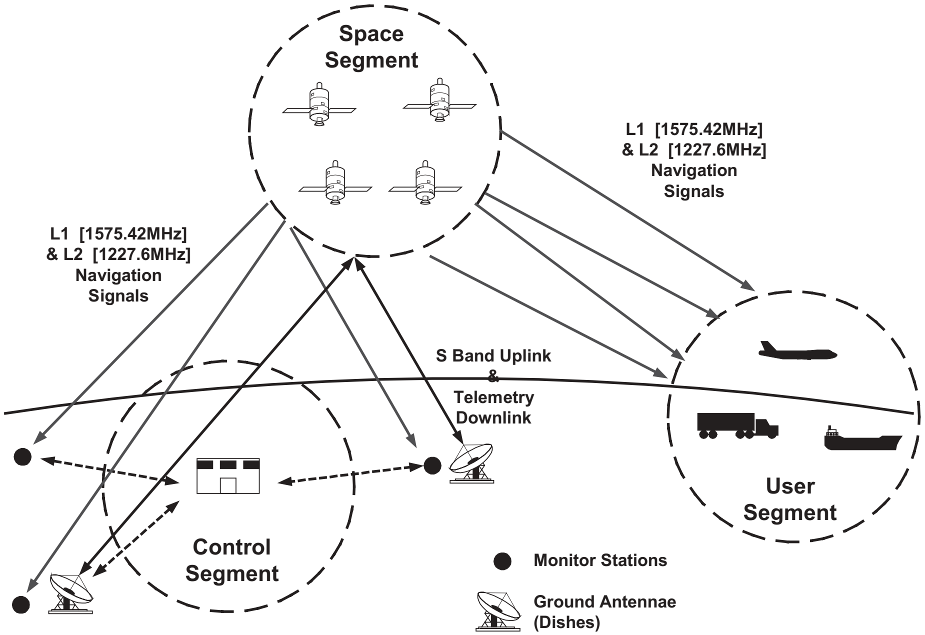


Fig. 6.18 Principles of GPS satellite navigation

The baseline satellite constellation downlinks data in two bands: L1 on 1575.42 MHz and L2 on 1227.60 MHz. A GPS modernization programme recently announced will provide a second civil signal in the L2 band for satellites launching in 2003 onwards. In addition, a third civil signal, L5, will be provided on 1176.45 MHz on satellites to be launched in 2005 and beyond. Finally, extra signals for military users (Lm) will be included in the L1 and L2 bands for satellites launched in 2005 and beyond.

GPS operation is based on the concept of ranging and triangulation from a group or constellation of satellites in space which act as precise reference points. A GPS receiver measures distance from a satellite using the travel time of a radio signal.

Each satellite transmits a specific code, called Course/Acquisition (CA), which contains information on the satellite's position, the GPS system time, and the health and accuracy of the transmitted data. Knowing the speed at which the signal travelled (approximately 186 000 mile/s) and the exact broadcast time, the distance travelled by the signal can be computed from the arrival time.

The GPS constellation of 24 satellites is designed so that a minimum of 5 are always observable by a user anywhere on Earth. The receiver uses data from a minimum of four satellites above the mask angle (the lowest angle above the horizon at which it can use a satellite).

GPS receivers match each satellite's CA code with an identical copy of the code contained in the receiver's database. By shifting its copy of the satellite's code in a matching process, and by comparing this shift with its internal clock, the receiver can calculate how long it took the signal to travel from the satellite to the receiver. The value derived from this method of computing distance is called a pseudorange because it is not a direct measurement of distance but a measurement derived from time. Pseudorange is subject to several error sources, for example ionospheric and tropospheric delays and multipath. In addition to knowing the distance to a satellite, a receiver needs to know the satellite's exact position in space; this is known as its ephemeris. Each satellite transmits information about its exact orbital location. The GPS receiver uses this information precisely to establish the position of the satellite. Using the calculated pseudorange and position information supplied by the satellite, the GPS receiver mathematically determines its position by triangulation. The GPS receiver needs at least four satellites to yield a three-dimensional position (latitude, longitude, and altitude) and time solution. The GPS receiver computes navigational values such as distance and bearing to a waypoint, ground speed, by using the aircraft's known latitude/longitude and referencing these to a database built into the receiver.

The GPS receiver verifies the integrity (usability) of the signals received from the GPS constellation through a process called Receiver Autonomous Integrity Monitoring (RAIM) to determine if a satellite is providing corrupted information. At least one satellite, in addition to those required for navigation, must be in view for the receiver to perform the RAIM function. Therefore, performance of the RAIM function needs a minimum of five satellites in view, or four satellites and a barometric altimeter (baro-aiding) to detect an integrity anomaly. For receivers capable of doing so, RAIM needs six satellites in view (or five satellites with baro-aiding) to isolate the corrupt satellite signal and remove it from the navigation solution.

RAIM messages vary somewhat between receivers; however, generally there are two types. One type indicates that there are insufficient satellites available to provide RAIM integrity monitoring. The other type indicates that the RAIM integrity monitor has detected a potential error that exceeds the limit for the current phase of flight. Without the RAIM capability, the pilot has no assurance of the accuracy of the GPS position. Areas exist where RAIM warnings apply, and these can be predicted, especially at higher latitudes, which represents one of the major shortcomings of GPS and the reason it cannot be used as a sole means of navigation.

The geometry of the GPS satellites favours accurate lateral fixes. However, because a number of the visible satellites may be low in the sky, determination of vertical position is less accurate. Baro-aiding is a method of augmenting the GPS integrity solution by using a non-satellite input source to refine the vertical (height) position estimate. GPS-derived altitude should not be relied upon to determine aircraft altitude since the vertical error can be quite large. To ensure that baro-aiding is available, the current altimeter setting must be entered into the receiver as described in the operating manual.

GPS offers two levels of service: Standard Positioning Service (SPS) and Precise Positioning Service (PPS). SPS provides, to all users, horizontal positioning accuracy of 100 m or less with a probability of 95 per cent, and 300 m with a probability of 99.99 per cent. PPS is more accurate than SPS; however, this is intended to have a selective availability function limiting access to authorized US and allied military, federal government, and civil users who can satisfy specific US requirements. At the moment, the selective availability feature is disabled, making the PPS capability available to all users pending the availability of Differential GPS (DGPS) solutions to improve the SPS accuracy. This step has been taken pending the development of differential or augmented GPS systems that will provide high accuracy to civil users while preserving the accuracy and security that military users demand.

The basic accuracy without selective availability is about ±100 m, as opposed to ±1 m when the full system is available. Developments are under way in the United

States to improve the accuracy available to civil users. These are:

- Wide Area Augmentation System (WAAS) to improve accuracy en route.
- Local Area Augmentation System (LAAS) to improve terminal guidance.

The use and limitations of GPS together with a description of the WAAS and LAAS concepts, is given in Chapter 8 – Navigation.

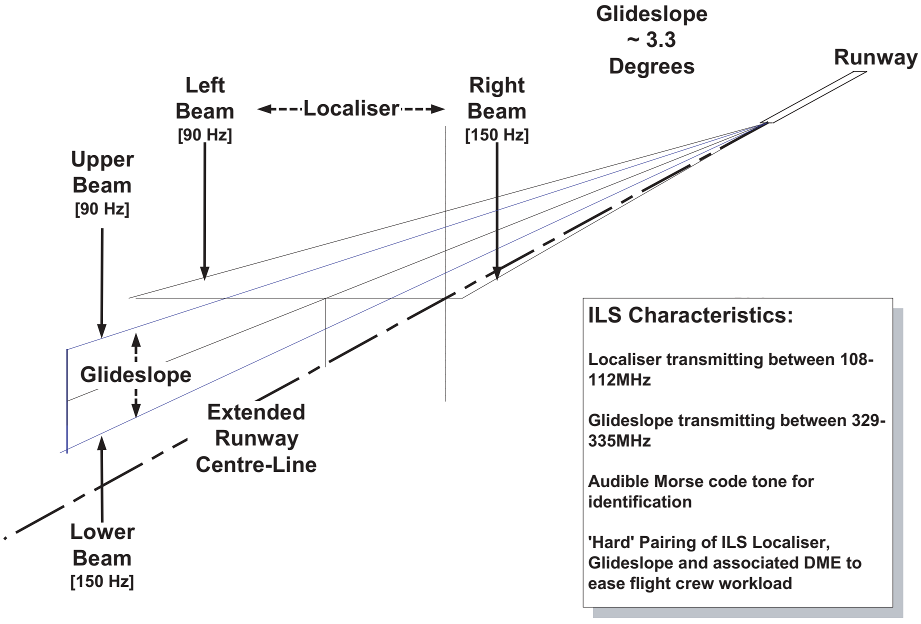
The European Community is intending to develop its own GNSS, called Galileo, so need not rely upon the US or Russian systems. Gallileo secured the agreement of funding between the Member States in early 2002.

Instrument Landing System

The ILS is an approach and landing aid that has been in widespread use since the 1960s and 1970s. The main elements of ILS include:

- A localizer antenna centred on the runway to provide lateral guidance. A total of 40 operating channels are available within the band 108–112 MHz. The localizer provides left and right lobe signals that are modulated by different frequencies (90 and 150 Hz) so that one signal or other will dominate when the aircraft is off the runway centre-line. The beams are arranged such that the 90 Hz modulated signal will predominate when the aircraft is to the left, while the 150 Hz signal will be strongest to the right. The difference in signal is used to drive a cross-pointer deviation needle so that the pilot is instructed to ‘fly right’ when the 90 Hz signal is strongest, and ‘fly left’ when the 150 Hz signal dominates. When the aircraft is on the centre-line, the cross-pointer deviation needle is positioned in the central position. This deviation signal is proportional to azimuth out to ±5º of the centre-line.
- A glide slope antenna located beside the runway threshold to provide lateral guidance. Forty operating channels are available within the frequency band 329–335 MHz. As for the localizer, two beams are located such that the null position is aligned with the desired glide slope, usually set at a nominal 3º. In the case of the glide slope, the 150 Hz modulated signal predominates below the glide slope and the 90 Hz signal is stronger above. When the signals are balanced, the aircraft is correctly positioned on the glide slope and the glide slope deviation needle is positioned in a central position. As for the localizer needle, the pilot is provided with ‘fly up’ or fly down’ guidance to help him or her acquire and maintain the glide slope. Figure 6.19 shows the general arrangement of ILS. Figure 6.20 illustrates how guidance information is portrayed for the pilot according to the aircraft position relative to the desired approach path. On older aircraft this would be shown on the compass display, but on modern aircraft, with digital cockpits, this information is displayed on the Primary Flight Display (PFD). The ILS localizer, glide slope, and DME channels are paired such that only the localizer channel needs to be tuned for all three channels to be correctly tuned.
- Marker beacons are located at various points down the approach path to give the pilot information as to what stage on the approach has been reached. These are the outer, middle, and inner markers. Location of the marker beacons are:
  - outer marker approximately 4–7 nm from the runway threshhold,
  - middle marker ~3000 ft from touchdown,
  - inner marker ~1000 ft from touchdown.

Fig. 6.19 *ILS glide slope and localizer*



ILS Characteristics:

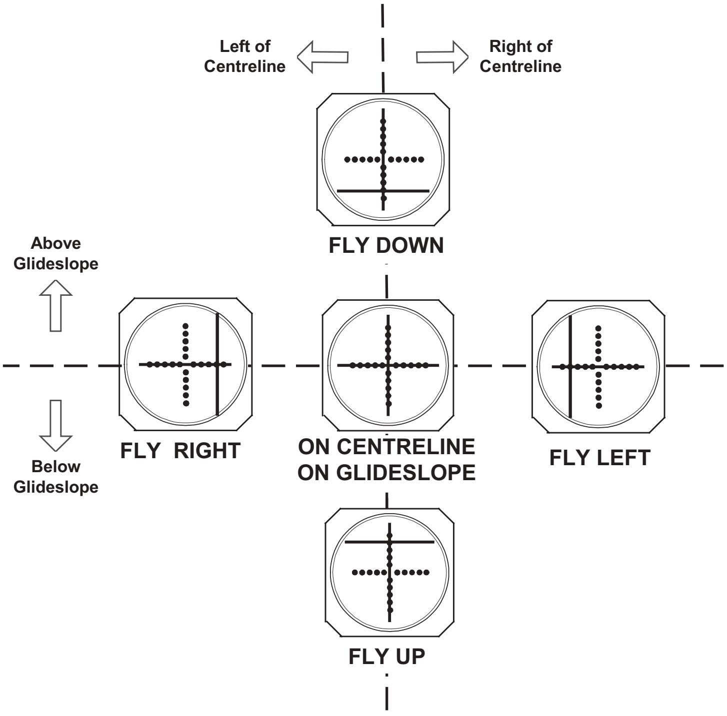
Localiser transmitting between 108-112MHz

Glideslope transmitting between 329-335MHz

Audible Morse code tone for identification

'Hard' Pairing of ILS Localiser, Glideslope and associated DME to ease flight crew workload

Fig. 6.20 *ILS guidance display*



The high approach speeds of most modern aircraft render the inner marker almost superfluous and it is seldom used.

- The marker beacons are all fan beams radiating on 75 MHz and provide different Morse code modulation tones that can be heard through the pilot's headset. The layout of the marker beacons with respect to the runway is shown in Fig. 6.21. The beam pattern is ±40° along track and ±85° across track. The overall audio effect of the marker beacons is to convey an increasing sense of urgency to the pilot as the aircraft nears the runway threshhold.

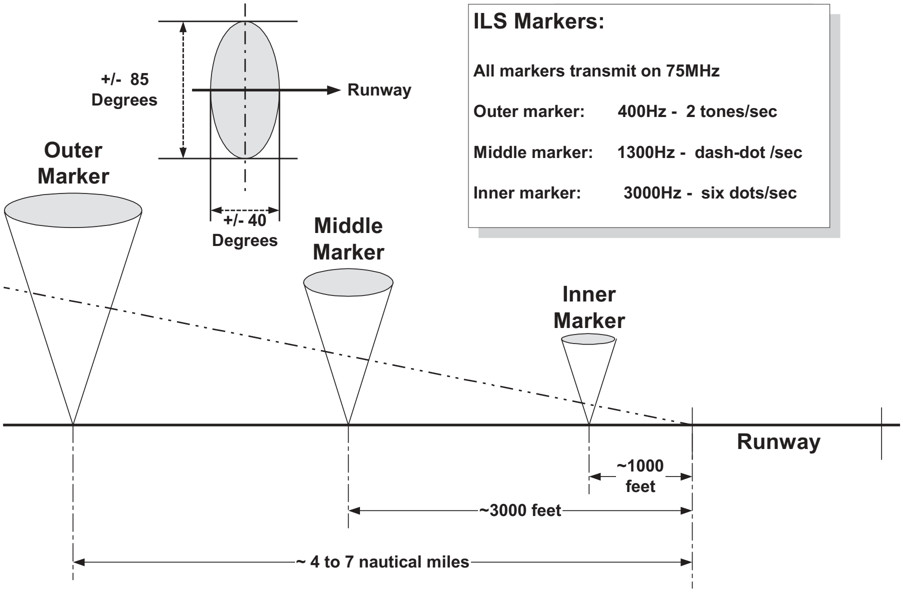


Fig. 6.21 *ILS approach markers*

A significant disadvantage of the ILS system is its susceptibility to beam distortion and multipath effects. This distortion can be caused by local terrain effects and large man-made structures, and even taxiing aircraft can cause unacceptable beam distortion, with the glide slope being the most sensitive. At times on busy airfields and during periods of limited visibility, this may preclude the movement of aircraft in sensitive areas, which in turn can lead to a reduction in airfield capacity. More recently, interference by high-power local FM radio stations has presented an additional problem, although this has been overcome by including improved discrimination circuits in the aircraft ILS receiver.

Transponder Landing System (TLS)

Developed by the Advanced Navigation and Positioning Corporation, TLS is a flexible system that appears to the pilot much like a standard ILS. The TLS employs patented and sophisticated closed-loop technology, which uses the aircraft's existing transponder to generate high-quality tracking information obtained from sensors located near the runway or landing zone. Flight guidance commands are transmitted back to the aircraft using standard ILS localizer and glide slope frequency bands. The pilot simply follows the cockpit-displayed guidance commands to the published minimum altitude. Coupled autopilot approaches can also be routinely flown.

The flight path of any aircraft is displayed in real time to the local air traffic controllers or transmitted to remote controlling agencies as desired. However, it has a major advantage over an ILS, since installation at difficult terrain sites poses no problem to its easy and rapid deployment within a 350 ft^2 area.

Unlike other existing and new precision approach system technologies (e.g. MLS, GPS, DGPS), the TLS uses standard on-board aircraft avionics such as localizer/glide slope receivers, Course Deviation Indicators (CDIs), flight directors or Flight Management Systems (FMS), and transponders. Operating on standard FAA-assigned TLS frequencies, this equipment may be coupled to autopilot. No new on-board navigation equipment or pilot training is required. Therefore, for some users TLS provides a high capability at modest outlay. TLS has been certified, and some systems are being procured by the FAA.

Microwave Landing System (MLS)

The MLS is an approach aid that was conceived to redress some of the shortcomings of ILS. The specification of a time-reference scanning beam MLS was developed through the late 1970s/early 1980s, and a transition to MLS was envisaged to begin in 1998. However, with the emergence of satellite systems such as GPS there was also a realization that both ILS and MLS could be rendered obsolete when such systems reach maturity. In the event, the US civil community is embarking upon higher-accuracy developments of the basic GPS system: Wide Area Augmentation System (WAAS) and Local Area Augmentation System (LAAS), which will be described later. In Europe, the United Kingdom, the Netherlands, and Denmark have embarked upon a modest programme of MLS installations at major airports.

MLS operates in the frequency band 5031.0–5190.7 MHz and offers some 200 channels of operation. It has a wider field of view than ILS, covering ±40º in azimuth and up to 20º in elevation, with 15º useful range coverage. Coverage is out to 20 nm for a normal approach and up to 7 nm for back azimuth/go-around. The co-location of a DME beacon permits three-dimensional positioning with regard to the runway, and the combination of higher data rates means that curved arc approaches may be made, as opposed to the straightforward linear approach offered by ILS. This offers advantages when operating into airfields with confined approach geometry and tactical approaches favoured by the military. For safe operation during go-around, precision DME (P-DME) is required for a precise back azimuth signal.

A ground-based MLS installation comprises azimuth and elevation ground stations, each of which transmits angle and data functions that are Frequency Shift Key (FSK) modulated and that are scanned within the volume of coverage already described. The

MLS scanning function is characterized by narrow beam widths of around 1–2º scanning at high slew rates. Scanning rates are extremely high at 20 000º/s which provides data rates that are around 10 times greater than is necessary to control the aircraft. These high data rates are very useful in being able to reject spurious and unwanted effects resulting from multiple reflections, etc.

Typical coverage in azimuth and elevation for an MLS installation is shown in Fig. 6.22.

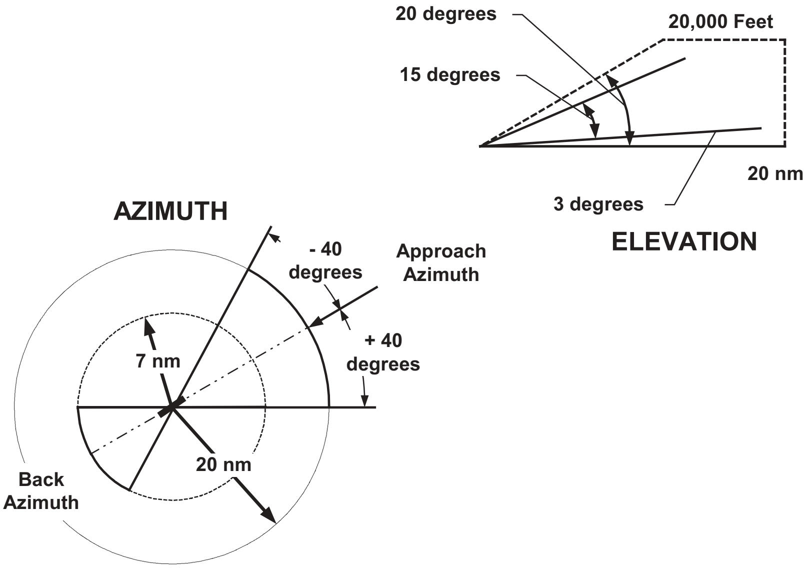


Fig. 6.22 *Microwave landing system coverage*

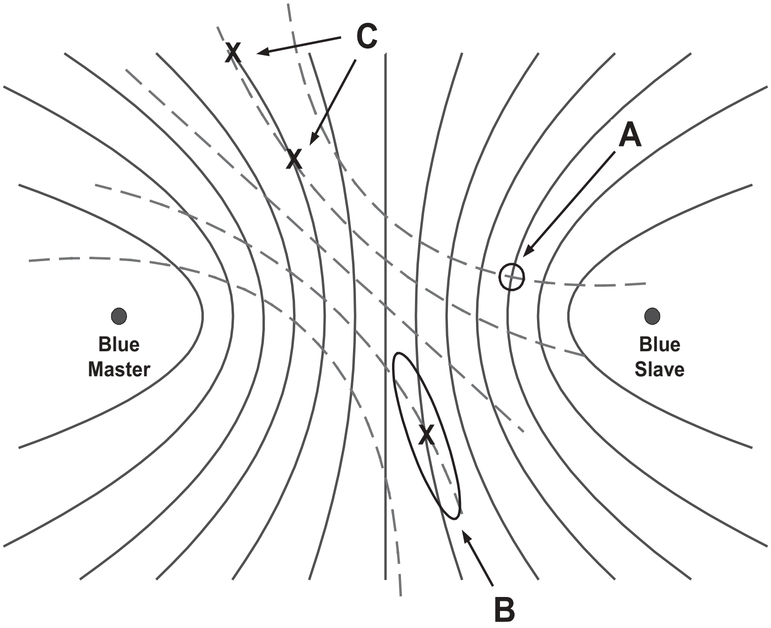
Hyberbolic navigation systems

Hyperbolic navigation systems operate upon hyperbolic lines of position rather than circles or radial lines. Figure 6.23 illustrates the principle of operation of a hyperbolic system in a very elementary manner. This shows the hyperbolic solid lines representing points that are equidistant from the two stations. These points will all have the same time difference between the arrival of signals from the blue-master and blue-slave stations (the term secondary station is probably a better and more accurate description). This in itself will not yield position, but if a second pair of stations is used – angled at approximately 45° to the first and shown as dashed lines – then position can be obtained. The relative positioning of the lines in this dual-chain example shows that three outcomes are possible:

- At point A the lines cross at almost 90°, which represents the most accurate fix.
- At point B the lines cross at a much more acute angle and the result is a larger error ellipse.
- At point C there are two possible solutions and an ambiguity exists that can only be resolved by using a further station.

LORAN-C is the hyperbolic navigation system in use today and was conceived in principle around the beginning of World War II. Worldwide coverage existed in 1996,

Fig. 6.23 *Principle of operation of a hyperbolic navigation system*



and new facilities were being planned in the late 1990s. LORAN operates in the frequency band 90–110 kHz as a pulsed system enabling the ground wave to be separated from the sky wave, the ground wave being preferred. A LORAN chain will comprise at least three stations, one being nominated as the master. The time difference of arrival between the master and slaves allows position to be determined. Each of the stations in a chain transmits unique identifiers which allow the chain to be identified. A typical example of a LORAN-C chain is shown in Fig. 6.24. which presents the north-eastern US chain.

Within the defined area of coverage of the chain, LORAN-C will provide a user with a predictable absolute accuracy of 0.25 nm. A typical chain will have over 1000 nm operating range coverage. LORAN-C is also capable of relaying GPS positional error within the transmissions. LORAN-C is expected to remain in commission until at least 2008. Reference (**8**) provides information to assist with the certification of LORAN-C navigation systems for use within the United States and Alaska.

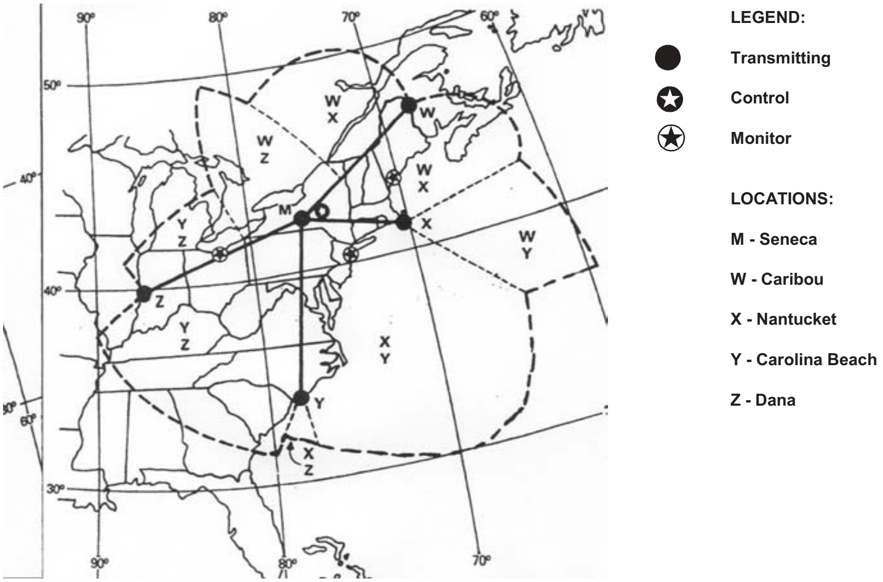


Fig. 6.24 *LORAN C Chain in North East US*

References

(1) Advisory Circular AC 20-67B (1986) Airborne VHF Communications Installations.

(2) RTCA DO-186 (1984) Minimum Operational Performance Standards (MOPS) for Radio Communications Equipment operating with the Radio Frequency Range 117.975 to 137.000 MHz.

(3) Advisory Circular 129-55A (1993) Air Carrier Operational Approval and Use of TCAS II.

(4) Advisory Circular AC 20-131A (1993) Air Worthiness Approval of Traffic Alert and Collision Avoidance Systems (TCAS II) and Mode S Transponders.

(5) RTCA DO-181 (19XX) Minimum Operational Performance Standards for Air Traffic Control Radar beacon System/Mode Select (ATCRBS/Mode S) Airborne Equipment.

(6) RTCA DO-185 (19XX) Minimum Operational Performance Standards for traffic Alert and Collision Avoidance Systems (TCAS) Airborne Equipment.

(7) Advisory Circular AC 00-31A (1982) National Aviation Standard for the Very High Frequency Omnidirectional Radio Range (VOR)/Distance Measuring Equipment (DME)/Tactical Air Navigation (TACAN) Systems.

(8) Advisory Circular AC 20-121A (1988) Airworthiness Approval of LORAN-C Navigation Systems for Use in the US National Airspace Systems (NAS) and Alaska.

CHAPTER 7

Displays

Introduction

This chapter discusses the information required and the means to display that information to the flight crew on the flight deck of a civil transport aircraft. The chapter will focus on the 'glass flight deck', but set in the context of flight instrument evolution from electromechanical instruments (**1**).

The operational requirements for the display media itself will be discussed, together with the technologies to be found on current in-service aircraft, namely electromechanical counter–pointer instruments and Cathode Ray Tube (CRT) and Active Matrix Liquid Crystal Displays (AMLCD). The advantages and disadvantages of the available technologies will be compared and indications will be given of the factors that the display equipment designer must take into account successfully to apply these technologies in the environment of the civil transport aircraft flight deck.

Finally, this chapter discusses how the information explosion from on-board and off-board systems might be collated, fused, and displayed in future generations of aircraft to improve crew situational awareness and operational flexibility and safety, particularly in the context of Controlled Flight Into Terrain (CFIT) accidents.

The electromechanical instrumented flight deck

Early flight deck instruments

The first flight of a heavier-than-air machine is generally recognized to have taken place in Kitty Hawk, USA on the 17 December 1903. The aircraft known as the Wright Flyer was built by the Wright brothers and flown by Orville Wright. The pilot lay in a prone position, facedown. There were no flight instruments, flying was by the 'seat-of-one's pants'.

By the mid-1920s, rudimentary flying instruments had found their way onto the flight deck (or cockpit as it was then more generally known in a single-seat aircraft), but there was little standardization. The cockpit shown in Fig. 7.1 had:

- Magnetic compass.
- Tachometer.
- Fuel gauge.
- Oil pressure.
- Clock.
- Turn and slip indicator.

However, there were no attitude, airspeed, altimeter, or vertical speed instruments, i.e. no flight instruments, as we would know them today.

Fig. 7.1 *Instruments in an early single seat aircraft*

The 1950s – piston engined aircraft

World War II had provided significant experience about flying aircraft for long periods in adverse conditions by day and by night. Flight instruments had become standardized to some degree.

The Handley Page Hermes designed and built at the end of that war is but one example, which encapsulates best practice of the day. The Hermes was a luxury aircraft. It was the first British aircraft to have a pressurized hull and therefore could fly above the weather and provide a smoother ride to its passengers. A typical route was from London to Sydney. The aircraft had facilities for 40 passengers plus cabin staff. Five crew – the captain, first officer, navigator, flight engineer, and radio operator – operated it.

The flight deck of the Hermes is shown in Fig. 7.2. The primary flight instruments were arranged in the following 'basic six' configuration:

• Gyro artificial horizon	top centre
• Airspeed	top left
• Vertical speed	top right
• Direction indicator	bottom centre
• Altimeter	bottom right
• Turn and bank indicator	bottom left

See also Fig. 7.4.

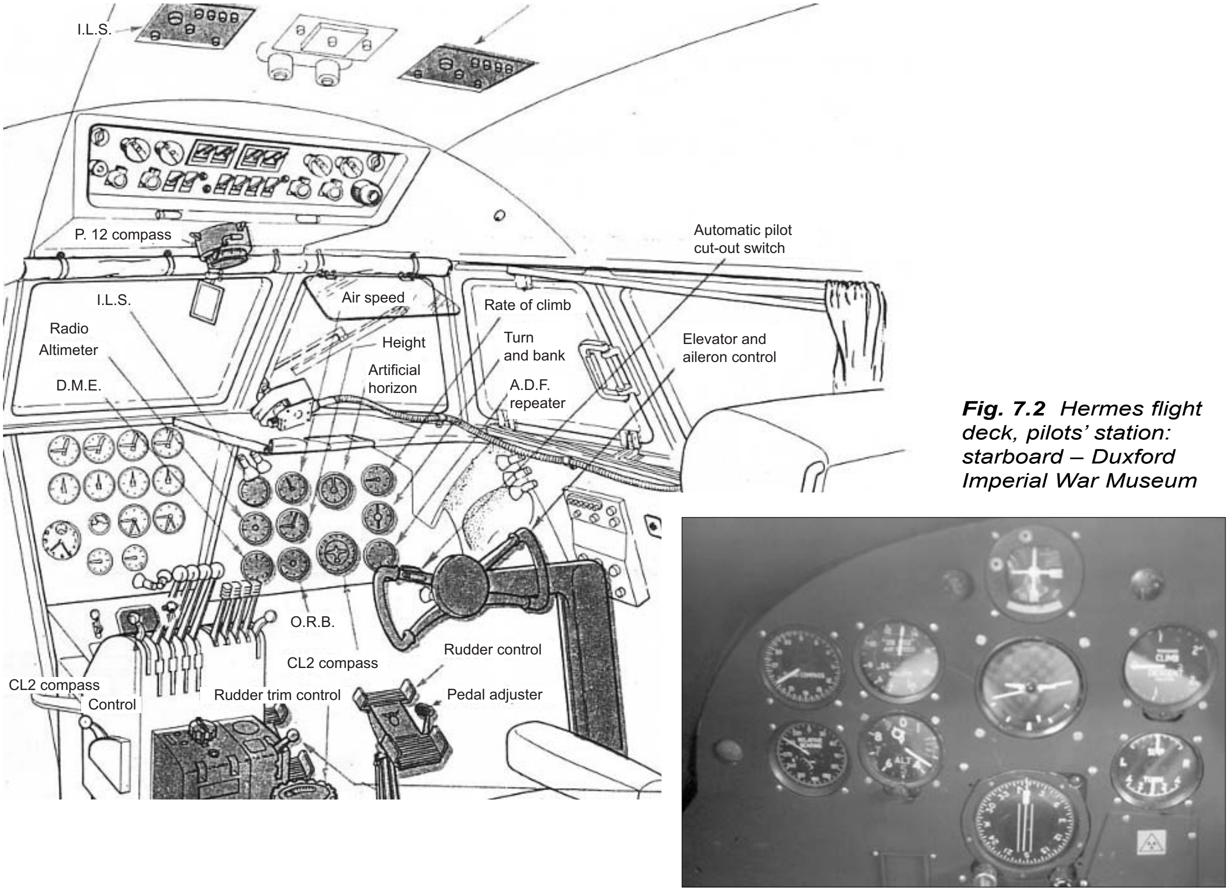


Fig. 7.2 *Hermes flight deck, pilots' station: starboard – Duxford Imperial War Museum*

Airspeed, altimeter, and rate of climb (the barometric instruments) operated from static and dynamic air pressure direct from the pitot–static probes. The artificial horizon comprised an integral two-axis gyroscope with mechanical pick-offs to drive the horizon bar to indicate aircraft pitch and roll angles with respect to the ground. The compass was a repeater instrument driven from the navigator's station, which could be slaved to one of two gyro compass units indicating aircraft heading or operate as a direction indicator to fly to a selected course. A magnetic compass was installed in the roof. As an option, an Automatic Direction Finding (ADF) repeater instrument could be fitted to indicate direction to the selected radio beacon.

On the left of the 'basic six' instrument cluster there was an option to fit an Instrument Landing System (ILS), which would indicate lateral and vertical deviation from the ILS glide slope and localizer beams, a low-range radio altimeter to provide greater height accuracy during approach and landing, and Distance Measuring Equipment (DME) to indicate distance to the ILS transmitter (positioned at the runway threshold).

The 1970s – jet aircraft

The 1970s heralded the mass market for civil transport aircraft with the introduction of the Boeing 747 Jumbo Jet. The 747 first entered service with Pan American in January 1970. The aircraft has continuously been developed throughout its operating life.

At its introduction, the flight deck of the 747 had a set of conventional electromechanical instruments and was operated by a three-man crew, the captain, the first officer and the flight engineer, as shown in Fig. 7.3. Today, the 747-400 aircraft has a fully integrated 'glass' flight deck with six AMLCD displays and is operated by a two-man crew.

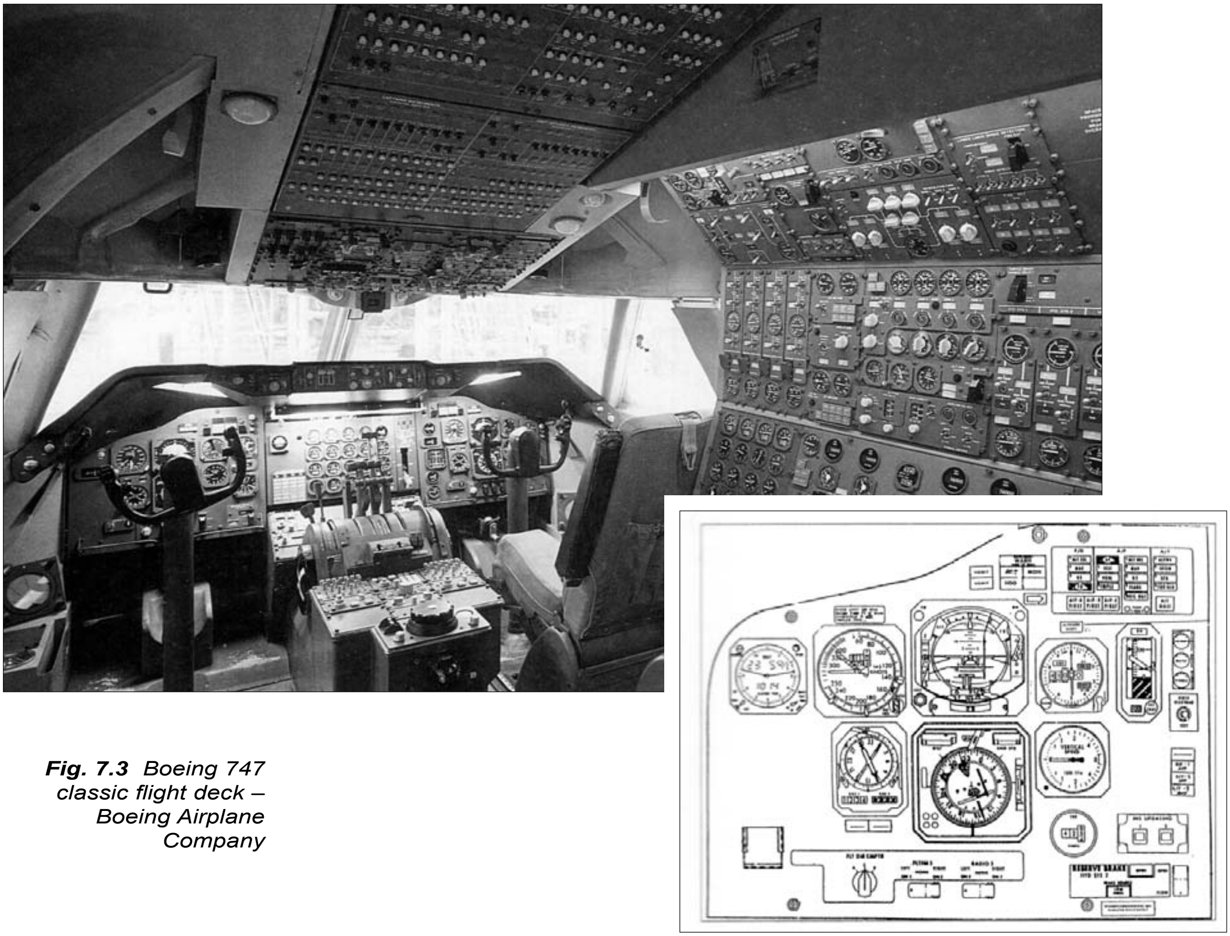


Fig. 7.3 *Boeing 747 classic flight deck – Boeing Airplane Company*

By today's standards, the flight deck of the early 'Classic' 747 might be considered to be fairly basic, but the aircraft did have some early satellite communications equipment, an automatic landing system, and an Inertial Navigation (IN) system. The IN was a direct descendant of that used on the Apollo moon landing missions, and its inclusion in the 747 allowed the aircraft to be certificated to fly anywhere in the world without a specialist navigator on-board. Refer to the Boeing 747 description (**2**).

The primary flight instruments had evolved considerably from those used on the Hermes, although their operating principles were still much the same. The arrangement of the instrument cluster had matured from the 'basic six' configuration to the 'basic T' now universally adopted on all electromechanically instrumented civil transport aircraft operating today (see Fig. 7.4).

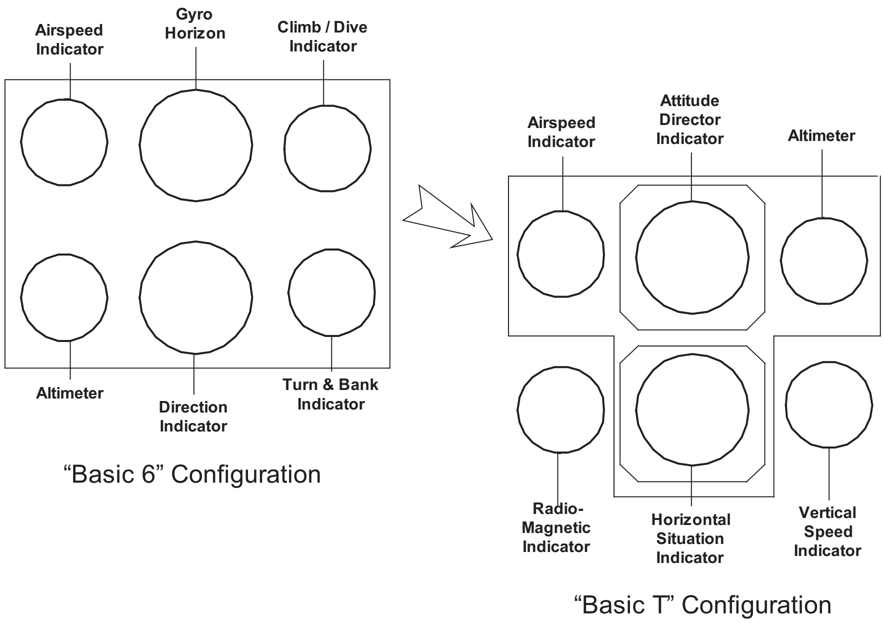


Fig. 7.4 *Migration from 'basic six' to 'basic T' configuration*

The Attitude Direction Indicator (ADI) and Horizontal Situation Indicator (HSI) are commonly found in 5ATI format, that is, with a physical form factor in accordance with the Air Transport Indicator (ATI) standard and a 5 in., square faceplate. The other instruments are commonly found in 3ATI format, that is, with a 3 in., square faceplate. The instruments are available from a number of manufacturers, have standardized functionality, and are certified to comply with Technical Standards Orders (TSOs) published by the Federal Aviation Authority (FAA) [see references (**3**)–(**7**)].

In addition, a third set of primary flight instruments, known as standby instruments, provide fully independent information to aid the flight crew to resolve discrepancies. The standby instruments (usually in 3ATI format) are totally segregated from the primary instruments and operate on separate electrical power, usually the d.c. essential bus.

The Attitude Direction Indicator

The instrument shown in Fig. 7.5 comprises:

1. An attitude ball, shaded blue for sky and brown for ground, which indicates aircraft pitch and bank angles against a fixed aircraft symbol in the centre of the instrument. The attitude ball is inscribed with pitch bar markings. Bank angle is indicated against a scale on the lower circumference of the instrument.
2. A flight director, which consists of two bars indicating lateral and vertical steering guidance to achieve a desired flight path. Different navigation aids are coupled to the flight director, depending on the phase of flight. These include radio aids (e.g.

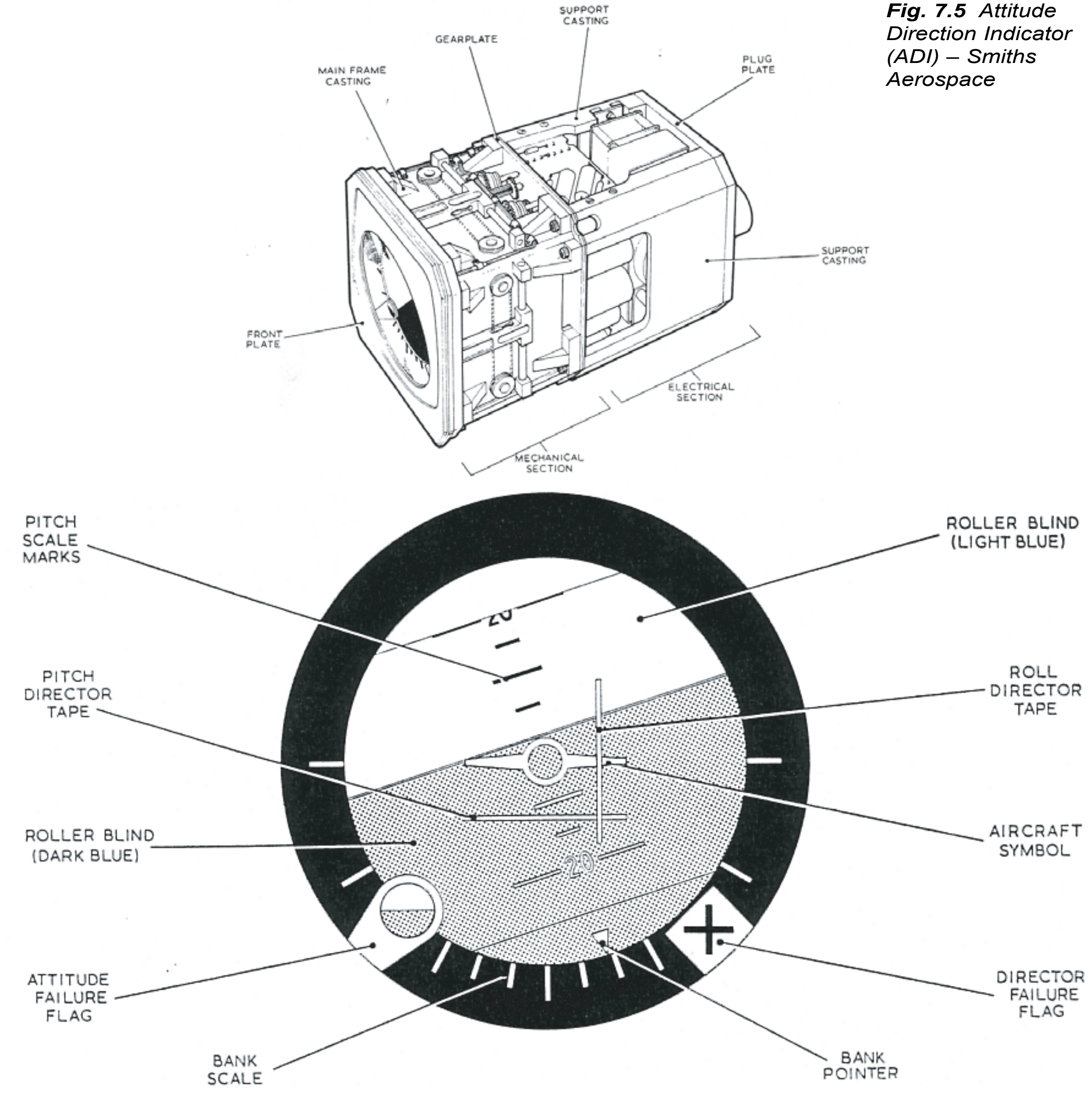


Fig. 7.5 *Attitude Direction Indicator (ADI) – Smiths Aerospace*

ADF, VOR, and TACAN), the flight management computer, and the autopilot itself as a monitor. During approach and landing, these bars are coupled to the ILS receiver to indicate glide slope and localizer deviation.

3. A turn and slip bubble, which is attached to the faceplate of the instrument.
4. Warning flags, to indicate warnings or cautions requiring crew action. The captain and first officer's instruments are fed with data from different sources to provide both redundancy and independence. Electrical cross-monitoring triggers a warning flag if a discrepancy is found between the indications on the two instruments. The crew must resolve the discrepancy.

The pitch and roll attitudes of the aircraft are represented by the position of a roller blind relative to the aircraft symbol. The blind, shown unrolled in Fig. 7. 6, consists of a two-tone band, light-blue representing the sky and brown representing the ground. A white line separating the two zones represents the horizon.

The aircraft symbol, in the form of a yellow coloured motif, is painted on the rear glass of the instrument. Movement of the blind over rollers attached to a carriage assembly depicts a change in aircraft pitch attitude, while rotation of the roller blind with the carriage assembly depicts a change in aircraft roll attitude. Pitch angle is given by pitch scale marks on the roller blind. Roll or bank angle is given by a pointer, which rotates with the roller blind, against a scale painted on the instrument bezel. Motion of the roller blind carriage is controlled by pitch and roll electromechanical servomechanisms.

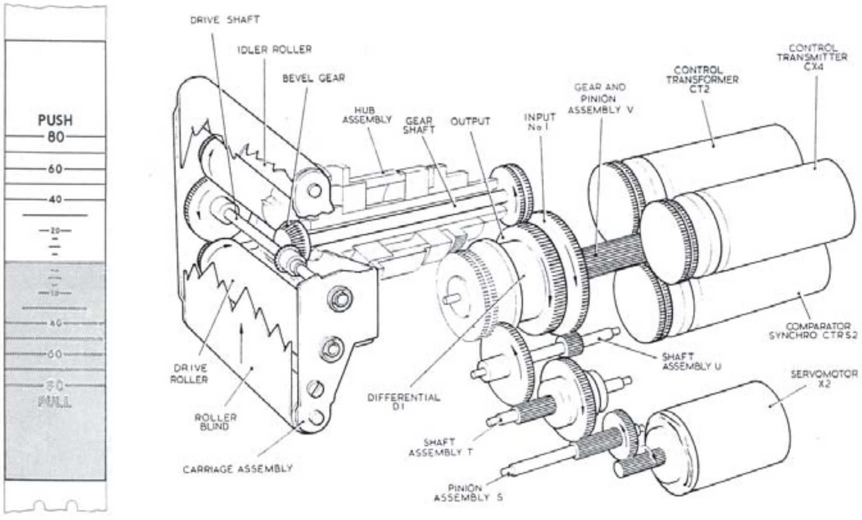


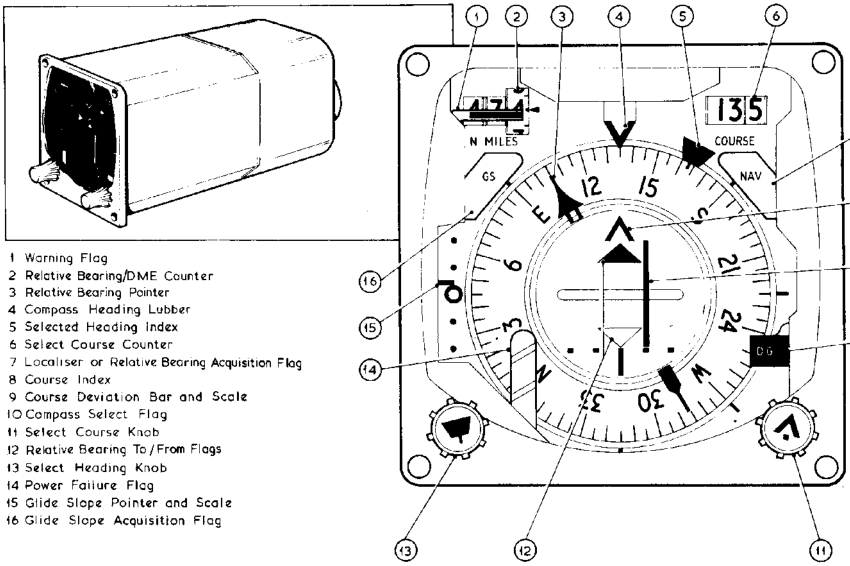
Fig. 7.6 *ADI roller blind and pitch servomechanism – Smiths Aerospace*

The Horizontal Situation Indicator

This instrument (shown in Fig. 7.7) functions as a magnetic compass or as a directional gyro indicator, depending on the mode selected. It operates as follows:

1. Aircraft heading is indicated by rotation of the compass card against a fixed lubber mark at the top centre of the instrument.
2. Aircraft course is indicated by the course index, which registers against and rotates with the compass card. The selected course may be set relative to the compass card by means of a knob at the bottom right-hand corner of the instrument. A drum counter display of selected course is also incorporated.
3. Selected heading is indicated by the heading index, which registers against and rotates with the compass card. The index may be set relative to the compass card by means of the heading selection knob.
4. Relative bearing information is displayed by the relative bearing pointer, which registers against the compass card but is driven independently of the compass card from the radio navigation equipment.
5. Deviation in azimuth from a selected relative bearing radial or ILS localizer beam is represented by the lateral displacement of the course deviation bar relative to the course deviation scale.
6. Deviation in pitch from the ILS glide path is represented by the vertical displacement of the glide slope deviation scale.
7. The DME counter displays the distance of the aircraft from a beacon. A TO/FROM flag indicates the aircraft direction relative to the beacon.

Fig. 7.7 Horizontal situtation indicator (HSI) – Smiths Aerospace



The altimeter

The primary altimeter in current-generation aircraft contains an electromechanical servomechanism driven by signals from the Air Data Computer (ADC) to indicate barometric altitude by means of a pointer and a counter (numerical readout) (see Fig. 7.8).

Early instruments and standby instruments operate directly from air pressure. By means of gears, levers, and cams, the deflection of an aneroid (sealed) capsule is translated into motion of counters and pointers against a scale indicating height (in feet). The capsule operates against aircraft outer atmosphere static pressure (derived from the pitot–static probe). The cams model the standard atmosphere chart. The pointer completes one revolution for every 1000 ft, rotating clockwise for increasing height. A bimetallic strip provides temperature compensation. A knob with a numerical readout sets the barometric pressure datum.

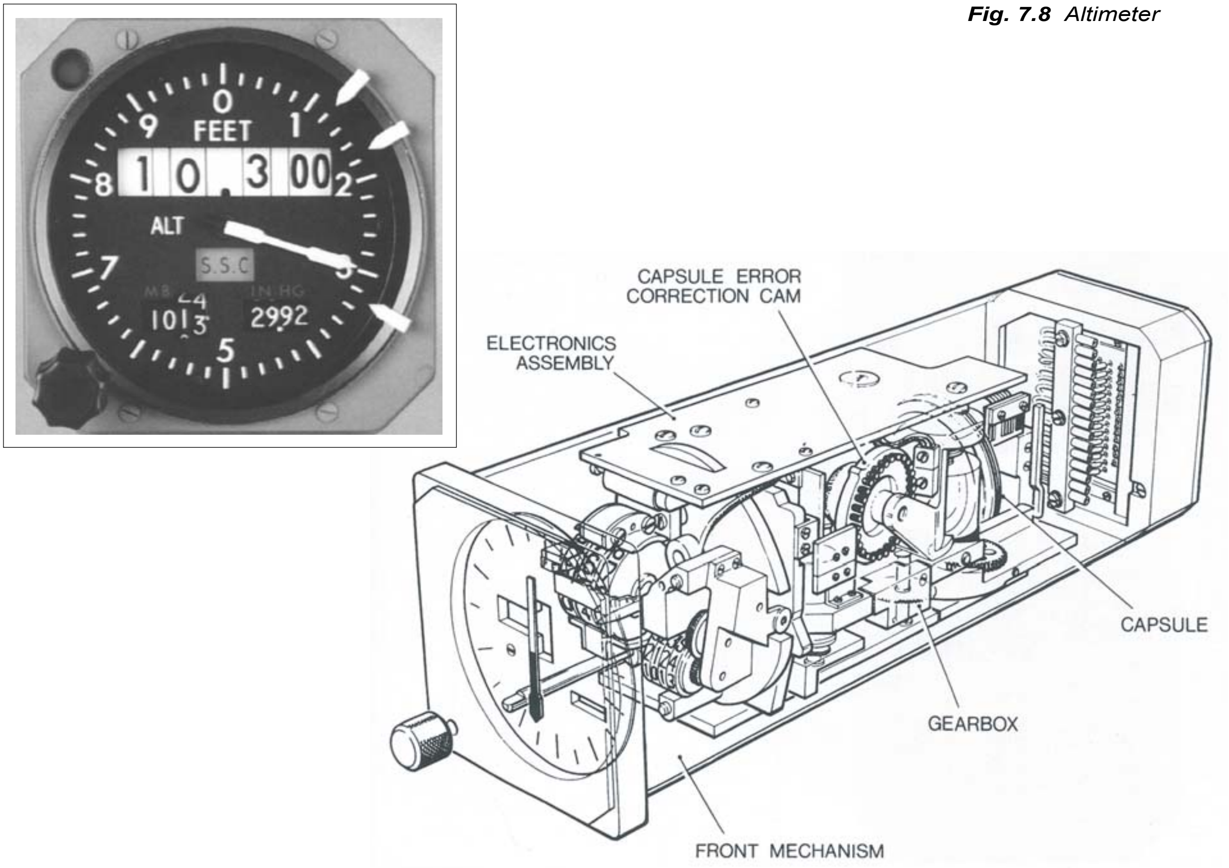


Fig. 7.8 *Altimeter*

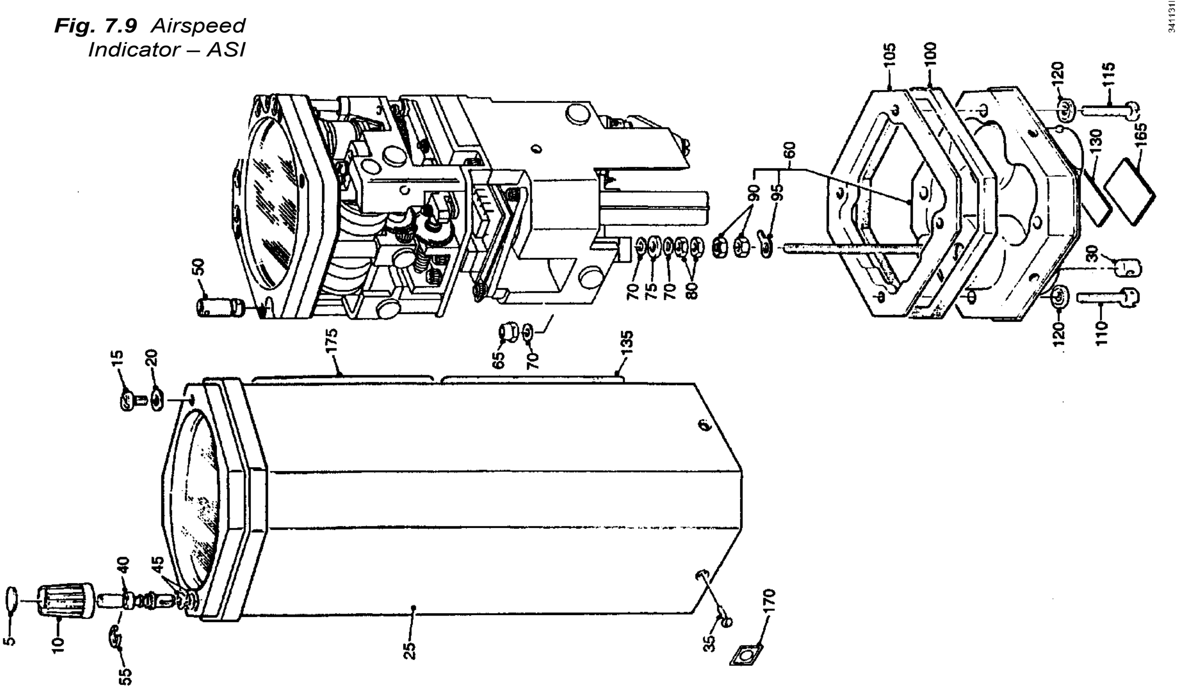
The Airspeed Indicator (ASI)

As with the primary altimeter, the primary Airspeed Indicator (ASI) in current-generation aircraft is driven by an electromechanical servomechanism slaved to the ADC for the portrayal of such an instrument, see Fig. 7.9.

Standby instruments, and early primary instruments, operate directly from air pressure. The deflection of a capsule is translated into motion of counters and pointers against a scale indicating airspeed in knots. In the ASI, the capsule inner atmosphere is airstream total pressure operating against outer atmosphere static pressure. Strictly speaking, knowledge of the air density is also required, but the simple mechanical instrument cannot compute this quantity from the information to hand and assumes a standard air density. The instrument also assumes that air is incompressible (which is true at airspeeds below Mach 1). The parameter computed with these assumptions is called calibrated airspeed (CAS).

In addition to airspeed it is not unusual also to find a numerical readout of Mach number. A warning bar indicates maximum safe speed. Bugs may be set around the circumference of the instrument to indicate placard speeds.

Fig. 7.9 *Airspeed Indicator – ASI*



Standby instruments

The availability and integrity of flight-critical information is always augmented with standby instruments. The standby instruments must operate independently of the main display suite and the avionic systems that source data to them. They are powered independently, usually by the d.c. essential bus. By these means, flight-critical data are still available on the flight deck even in the event of major avionic system failures and/or electrical power failure.

The standby instruments (Fig. 7.10) are typically a single set of miniature (2ATI and 3ATI) self-sensing instruments using integral transducers. They are the modern-day equivalents of the pitot–static barometric instruments and gyro artificial horizon found in 1950s aircraft.

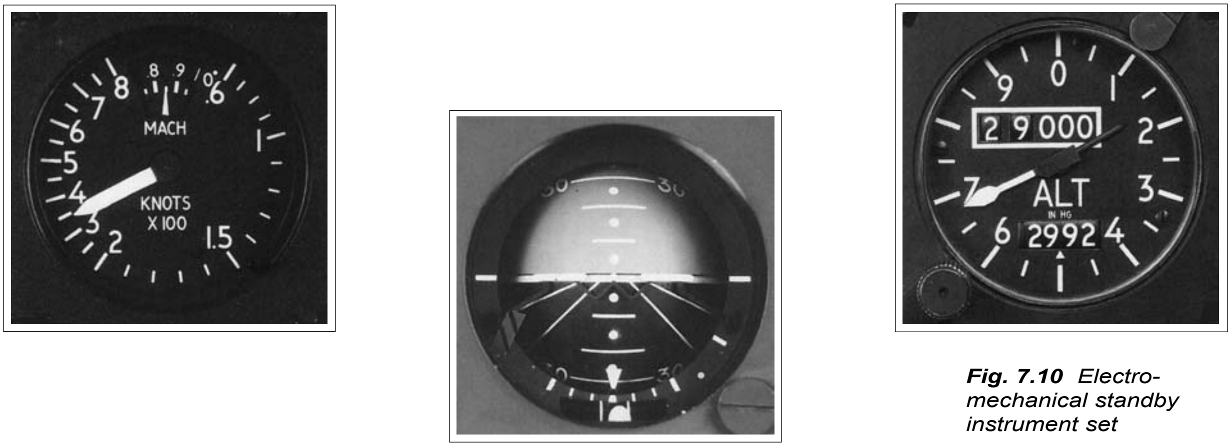


Fig. 7.10 *Electro-mechanical standby instrument set*

The ‘glass’ flight deck

This section traces some significant major milestones in the evolution of the glass flight deck from its humble beginnings. To include all the various flight deck iterations would be overwhelming, but the examples chosen address the salient points of the evolutionary process.

Advanced civil flight deck research

Research into the use of CRT technology to display crew information in civil transport aircraft began in the United Kingdom in the mid-1970s. It was carried out at BAe Weybridge with contributions from GEC and Smiths Industries. The Department of Trade and Industry (DTI) sponsored the programme, known as the Advanced Civil Flight Deck (ACFD).

The ground-based simulator (shown in Fig. 7.11) had six CRTs, each with a 6 × 4.5 in. (4:3 aspect ratio) usable screen area, arranged in landscape format in a side-by-side configuration across the flight deck. The displays were monochrome (white on black) and the images were generated in 625 line 25:50 Hz interlaced video (TV standard).

The research activities included extensive human factor evaluation by pilots and demonstrated the viability of an all-glass flight deck and the side-by-side configuration of primary flight (PFD) and navigation (ND) displays.

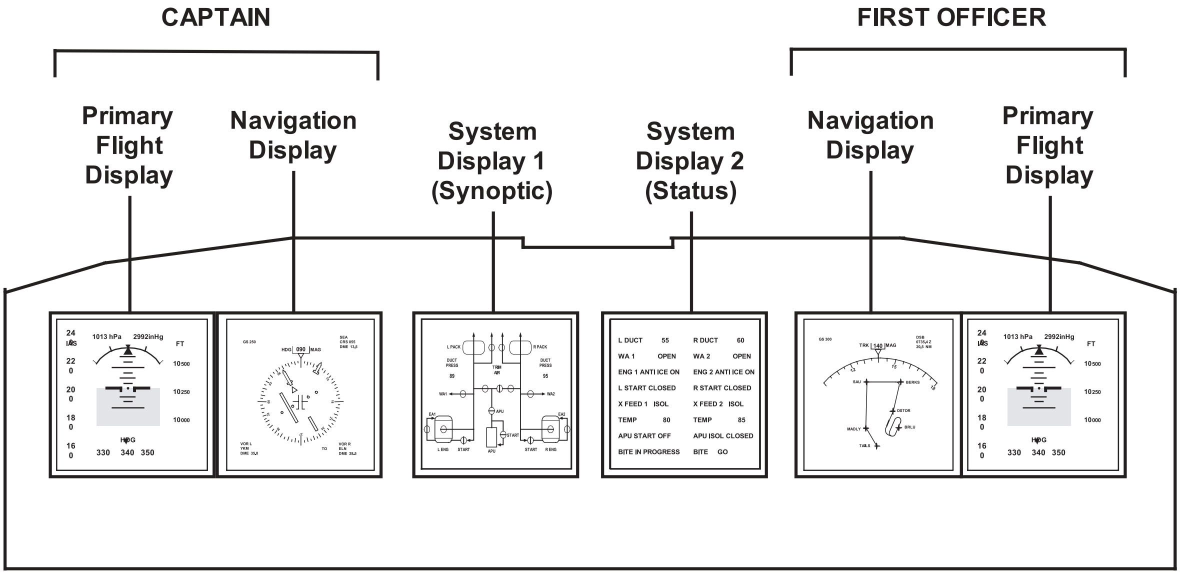


***Fig. 7.11** The advanced civil flight deck*

Although the ACFD and the BAC 1-11 demonstration established the viability of a fully integrated flight deck with side-by-side PFD and ND displays, it would be some years before the aviation world was fully confident to adopt that format, though as we shall see it has become the defacto standard today. It was necessary firstly to demonstrate with smaller display sizes that the CRT was sufficiently robust to reliably operate long term in the civil aircraft environment. In addition, and very rightly, the pilot community had also to gain experience with the medium.

BAC 1-11 technology demonstrator

In 1980 the ACFD research programme moved to a flight demonstration phase. Two displays (the PFD and ND) were installed into one crew station of the BAC 1-11 civil transport aircraft operated by the Royal Aircraft Establishment (RAE) in Bedford. The programme was launched with monochrome CRTs (green on black), but, once it was established that colour shadow mask CRTs could be ruggedized to withstand the civil air transport environment, the programme transitioned to 6.25 × 6.25 in. usable screen area square colour displays.

The display suite was fully integrated with the aircraft systems and the aircraft could be flown from the left seat, the crewmember on the right acting as the safety pilot. The aircraft first flew in the spring of 1981 and from the autumn of 1981 made an extensive series of test and demonstration flights in Europe and the United States. During a comprehensive US tour the aircraft visited nine sites, inviting guests from aircraft manufacturers, airlines, and research organizations to fly the aircraft. Some 34 sorties were flown (55 flight hours), most by guest pilots who were able to fly enroute and touch-and-go procedures to evaluate the concept. Most pilots found the displays intuitive and easy to use; they adapted quickly to the side-by-side PFD/ND configuration.

The display formats used on the BAC 1-11 aircraft drew on the experience of the ACFD ground-based programme. The formats mirrored closely the style of the

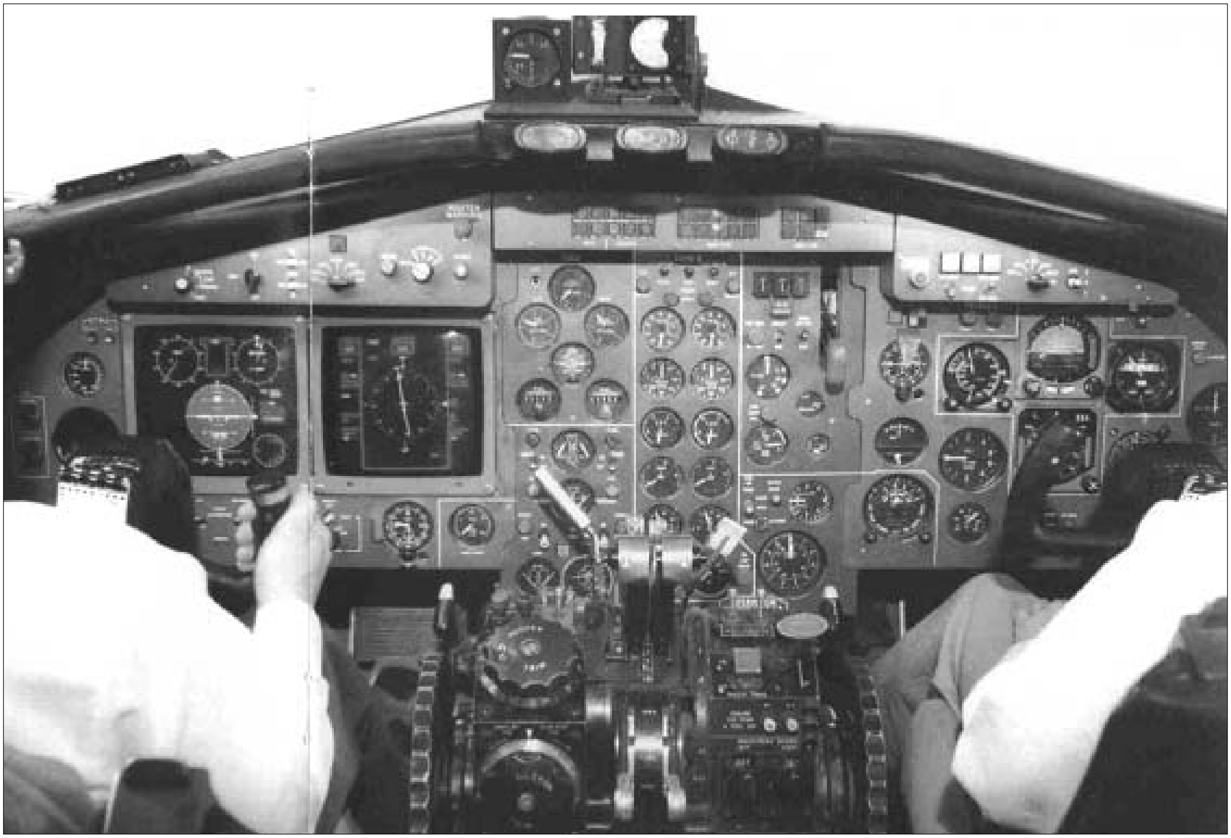


Fig. 7.12 BAC 1-11 advanced civil flight deck, Royal Aircraft Establishment

electromechanical instruments they were replacing to facilitate transition to the new media. The flight deck layout is shown in Fig. 7.12.

The PFD shown in Fig. 7.13 preserved the 'basic T' configuration of airspeed, attitude, and altitude. Counter–pointer presentations of speed and height were retained, but in addition digital readouts of speed and height were presented on the horizontal centre-line and adjacent to the attitude ball so the pilot could rapidly assess the primary flight data without having to scan the whole display.

Great care was taken to make numerical readouts emulate the rotation of 'drum counters' in mechanical instruments to aid the ability to read the digits when changing. The pointer emulates the rotation of instrument 'needles' to preserve the perception of rate information.

The ND shown in Fig. 7.14 provided two format styles: the compass rose format which preserved the original electromechanical HSI compass rose style, and the map format which presented the planned route, updated with present position, in real time. This later format demonstrated the real freedom of the media to overcome the limitations of the electromechanical instrument and present a picture of the plan situation, a picture that hitherto the pilot had had to form in his or her head.

The map could be oriented heading-up, track-up or north-up, with the aircraft position portrayed either in the lower quadrant or in the centre of the display. Map range was selectable. Information on the flight plan included waypoints and navigation aids in the local area. Weather radar data could also be added to facilitate route replanning to avoid storm centres. Down each side of the map was presented navigational information relative to the planned route, including waypoint identification, direction, distance, and time-to-go.

Fig 7.13 BAC 1-11 primary flight display – Smiths Aerospace (see colour plate section)

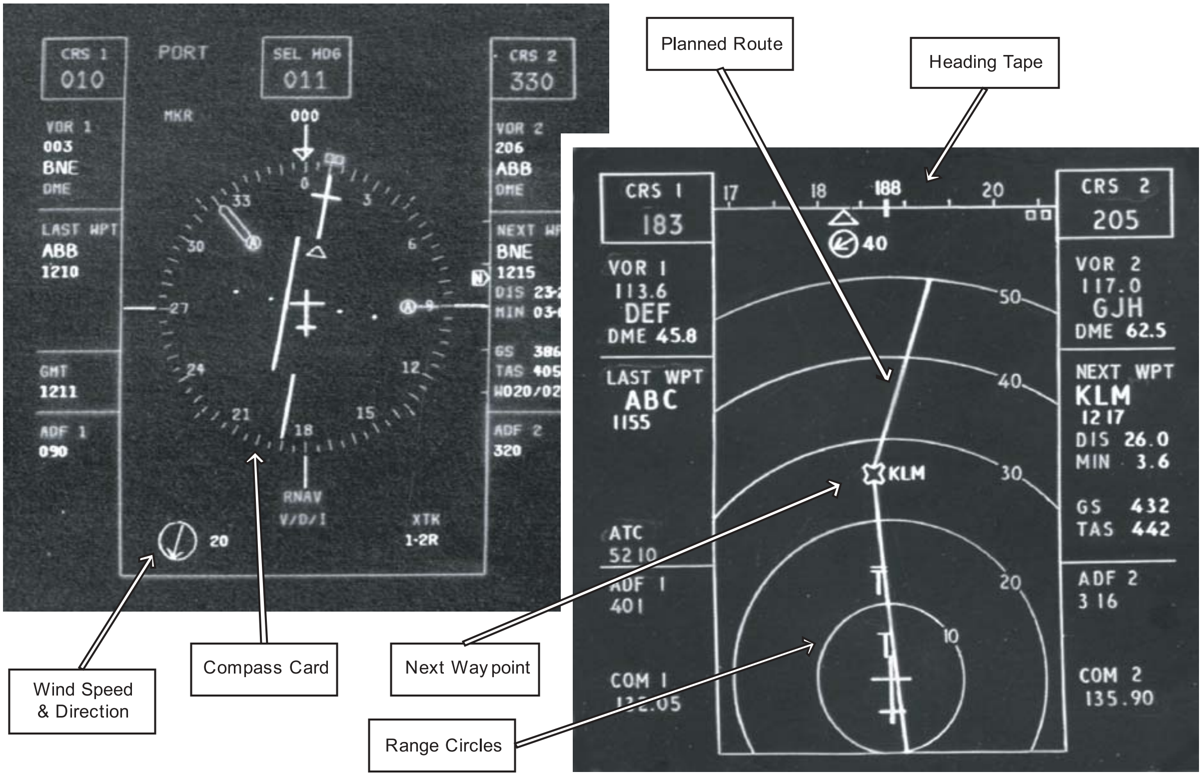


Fig 7.14 BAC 1-11 navigation display – Smiths Aerospace (see colour plate section)

Boeing 757 and 767

The Boeing 757 and 767 aircraft were the first to enter service in September 1982 with a 'glass' flight deck. (**8**). The Electronic Flight Instrument System (EFIS) comprises an Electronic ADI (EADI) with a 5 × 4 in. CRT display, in portrait mode and an electronic HSI (EHSI) with a 5 × 6 in. CRT display, in landscape mode (see Fig. 7.15). The displays are placed one above the other with a conventional electromechanical airspeed indicator and altimeter to the left and right respectively, preserving the 'basic T' configuration.

The EADI display format emulates the electromechanical instrument. The EHSI display format offers both the conventional compass and also the map format. Weather radar information can be superimposed on the map. The traditional electromechanical engine instruments and cautions are superseded by two CRT displays known as the Engine Indication and Crew Alerting System (EICAS), positioned one above the other.

Fig. 7.15 *Boeing 757 and 767 flight decks, Boeing Airplane Company*

British Aerospace advanced turbo-prop

The Advanced Turbo-Prop (ATP) followed the path of the Boeing 757/767 and placed the EADI and EHSI displays above one another as shown in Fig. 7.16. The displays have a CRT with a usable screen area of 5 × 4 in., both in landscape orientation.

The original concept had been to include both speed and height in the upper display as tape scales to the left and right respectively of the attitude 'ball'. However, it soon became apparent that the maximum display size that could be accommodated in the instrument panel would not permit the desired information to be written at a sufficiently large size to be read comfortably. Therefore, the final certificated configuration

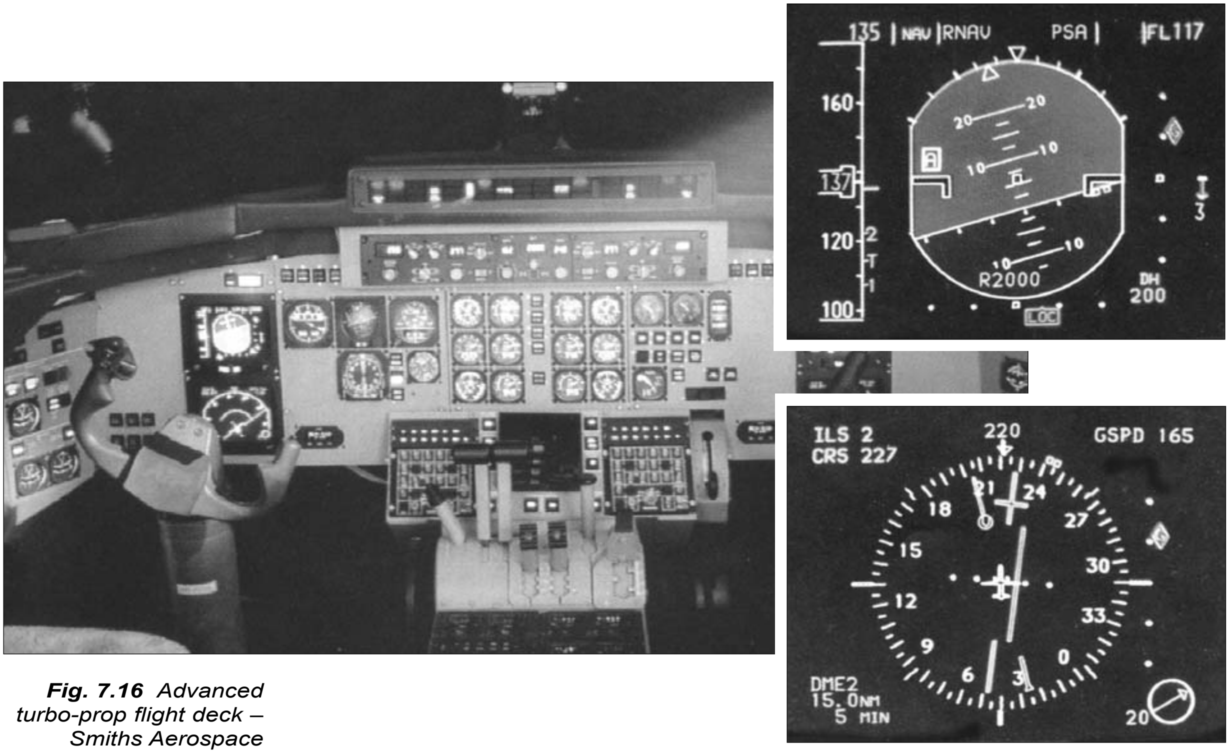


Fig. 7.16 *Advanced turbo-prop flight deck – Smiths Aerospace*

included only the speed scale on the EADI, with height presented on a conventional electromechanical altimeter to the right of the EADI.

Airbus A320/A330

The Airbus A320, which entered service in March 1988, was the first civil transport aircraft to adopt the side-by-side PFD/ND configuration with six 6.25 × 6.25 in. CRT displays installed on the flight deck. The A330 has a similar configuration (see Fig. 7.17). The two systems displays are installed one above the other in the centre of the flight deck.

The PFD shown in Fig. 7.18 introduces airspeed and altitude as two tape scales positioned either side of the attitude ‘ball’. A numerical readout is positioned as a window in the tape scales on the horizontal centre axis of the attitude ‘ball’. This concept allows the pilot quickly to acquire the key flight parameters without having to scan the whole display surface.

The tape scales facilitate the acquisition of selected speed and height cues, which are presented on and attached to the moving scales. The moving scales also offer a representation of rate of change in the parameter, although it has to be said that this is recognized as being inferior to a circular scale and pointer. A vertical speed scale therefore augments the height tape.

At the bottom of the PFD format the upper segment of a compass rose provides heading and lateral guidance cues. Autopilot-mode annunciators are incorporated along the top of the display.

Fig 7.17 *Airbus A320 flight deck, Airbus (see colour plate section)*

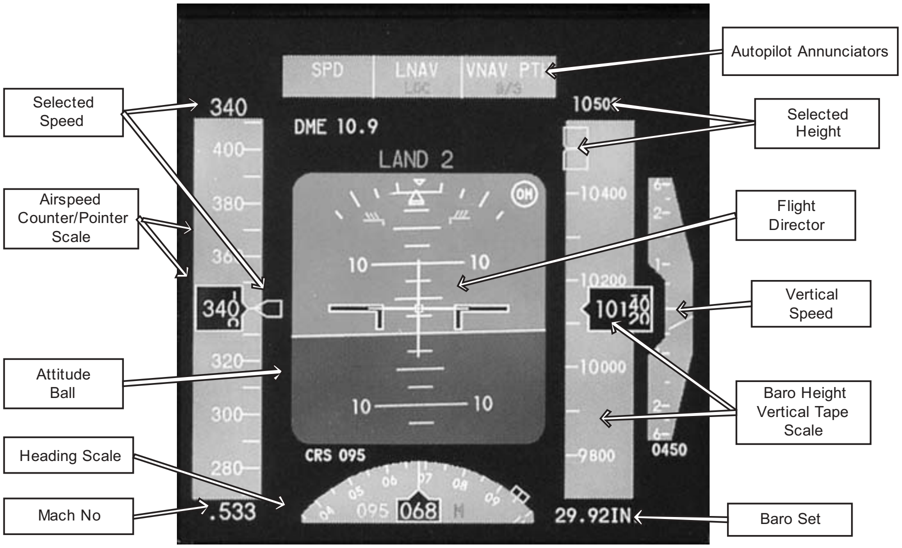


Fig. 7.18 *Airbus A320 primary flight display (see colour plate section)*

Boeing 747-400

Boeing adopted the side-by-side, six display configuration for its –400 variant of the 747 aircraft, which entered service in June 1990, and then subsequently for the Boeing 777 aircraft (see Fig. 7.19). The two systems displays are positioned one above the other in the vertically extended centre portion of the instrument panel. The 747-400 configuration is depicted in Fig. 7.20.

The displays employ CRT technology. The display formats are similar to those of the Airbus A340, with strip speed and height scales positioned either side of the attitude 'ball'.

In this configuration, the Boeing 747 operates with a two-man crew. The flight engineer's panel functions have been absorbed into the aircraft systems and are presented and controlled through the two power/systems displays.

The Boeing 777 (in-service date June 1995) adopts a similar flight deck configuration but a significantly different avionics system architecture. The displays themselves were the first in a wide-body civil transport aircraft to utilize AMLCD technology. The displays have a 6.7 × 6.7 in. useable screen area. The Boeing 777 Navigation Display (ND) shown in Fig. 7.20 provides planned and actual route, together with lateral guidance command cues optionally overlaid with weather data.

Fig. 7.19 *Boeing 747-400 flight deck, Boeing Airplane Company (see colour plate section)*

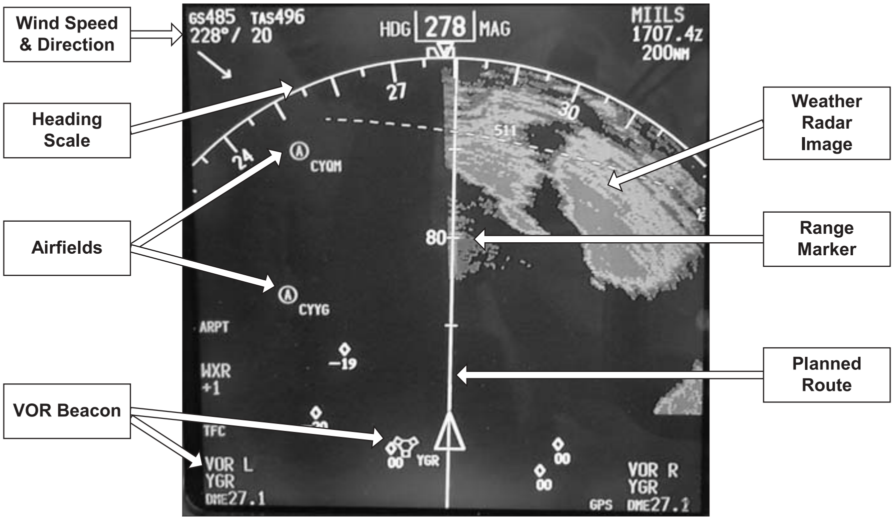


Fig 7.20 *Boeing 777 navigation display with weather radar (see colour plate section)*

Upgrade of 'Classic' aircraft flight decks

The power of the glass flight deck has been demonstrated, and new air traffic procedures are being introduced to take advantage of the improved route planning and freedoms afforded by the glass flight deck and other aids. There is a demand to upgrade the flight deck of legacy (or classic) aircraft originally fitted with electromechanical instruments to a 'glass' flight deck so that those aircraft too can enjoy more efficient route structures and operating procedures.

Complete stripping and replacement of the instrument panel and reconstruction of the aircraft systems to support a fully integrated glass flight deck is neither practical nor affordable. Replacement of the existing electromechanical instruments with form, fit but functionally enhanced 'glass' instruments is a viable option.

One such example is the upgrade to the fleet of 747-Classic aircraft operated by KLM as shown in Fig. 7.21. Seven 5ATI 'glass' instruments replace the existing 5ATI ADI and HSI instruments, together with some of the engine instruments. These instruments use AMLCD technology. The instruments are identical, their function being contained in software that configures itself when the instruments are installed, sensing their location in the aircraft by programme pins.

The EADI emulates the pre-existing electromechanical ADI but with improved flight director presentation. The EHSI emulates the pre-existing electromechanical HSI but with the addition of the map format optionally overlaid with weather radar. Two instruments combined provide engine data in tape scale format. These formats are shown in Fig. 7.22. A combined EHSI and EADI format, shown in Fig. 7.23, is provided as a standby instrument in the event of failure of either or both of the prime displays.

Fig. 7.21 *KLM 747-classic upgraded flight deck – Smiths Aerospace (see colour plate section)*

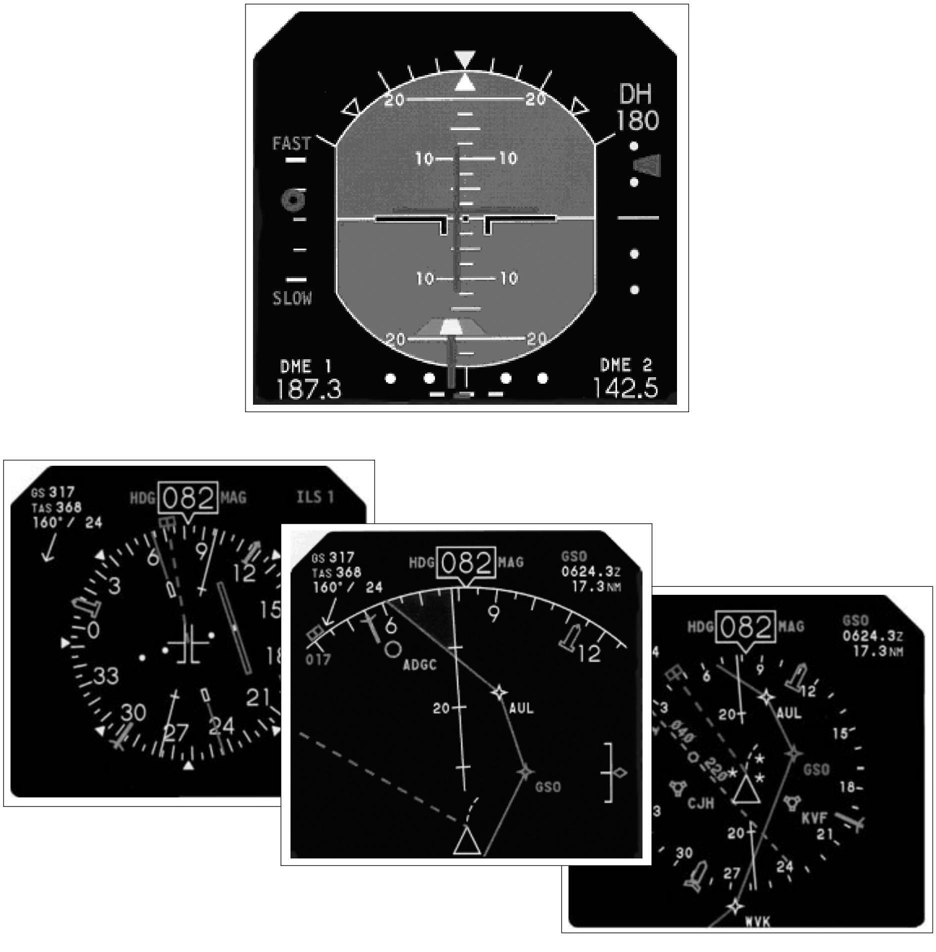


Fig. 7.22 *KLM 747-classic upgrade: EADI and EHSI formats – Smiths Aerospace (see colour plate section)*

Fig. 7.23 *KLM Classic upgrade combined ADI/HSI format – Smiths Aerospace (see colour plate section)*

Roll index pointer (white)

Bank angle markings (white)
Traditional white markings.

Heading orientation (green)
and reference (white)
The abbreviation HDG shows that the number under the pointer is a heading. The abbreviations MAG or TRU indicate the reference. When switching between TRU and MAG, a box appears around the appropriate abbreviation for 10 seconds after selection. Used in all modes.

Decision height warning (yellow and black)
Solid yellow circle with black letters 'DH'.

Navigation radio source (green)
Shows source of navigation radio data (VOR or ILS). Used in VOR/ILS and VOR/ILS centred modes.

Ground speed indication (white)
Shows current ground speed. Used in all modes

Horizon (blue and brown)
Blue sky and brown ground. The display includes a white pitch ladder and fly up/down arrows for extreme attitudes.

Decision height warning (yellow)
Displayed when RADALT is less than decision height. If the warning is not displayed, the decision height setting is displayed in green.

Aircraft symbol (black and white)
Horizontal 'L's with square centre dot.

Glideslope pointer (magenta)
and deviation scale (white)
Shows glideslope position and deviation when ILS is selected. Turns solid when glideslope alive. The pointer is not displayed when track and the front course on the ILS panel differ by more than 90°. Used in VOR/ILS and VOR/ILS centred modes.

Selected heading marker (magenta)
This symbol shows the heading set on the mode control panel. In heading mode, a dashed magenta line extends from the marker to the aircraft symbol for ease in tracking the marker. The symbol disappears in any other mode 10 seconds after heading set. Used in all modes.

Failure warnings (amber)
Failure flags are represented by an amber box and amber letters or words that are displayed at or in the area of the failed information. The affected display is removed.

Course deviation indicator (magenta)
and deviation scale (white)
Shows VOR or ILS course deviation when valid signals are present. Turns solid when radial/localiser is alive. Used in VOR/ILS and VOR/ILS centred modes.

Fast/slow indicator
A magenta 'doughnut' with a white scale and the words 'FAST' and 'SLOW' in cyan.

Aircraft symbol (white)
Represents the aircraft and indicates its position at the centre of the symbol. Used in VOR/ILS. Centred mode.

Compass rose (white)
Compass data is selected by the IRS. Used in VOR/ILS centre and map centred modes.

DME distance display (white)
Shows DME distance to the referenced navaid. Used in ADI mode.

Selected course pointer (white) **and line** (magenta)
Shows selected course as set by the VOR/ILS course selector. Used in VOR/ILS and VOR/ILS centred modes.

Glass standby instruments

As described earlier, the availability and integrity of flight-critical information is always augmented with standby instruments. This remains true in a ‘glass flight deck’.

In early glass flight decks the standbys are the traditional set of miniature (2ATI and 3ATI) dedicated electromechanical instruments. More recently these have been superseded by electronic displays using AMLCD technology with integral solid-state sensors.

Piezo-resistive pressure sensors have replaced the aneroid capsules. Micromachined rate sensors and accelerometers have replaced the rotating gyroscope. A single instrument can replace all the dedicated instruments, pin programmed to provide the desired display format. This display, shown in Fig. 7.24, is known as the Integrated Standby Instrument System, or ISIS.

The ISIS provides a display of:

- Attitude.
- Indicated airspeed and mach number.

Fig. 7.24 *3 ATI ISIS with display formats – Smiths Aerospace (see colour plate section)*

- Baro-corrected altitude and baro-set.
- Slip/skid.
- Heading.

The display formats are designed to be compatible with the primary EFIS formats to reduce pilot adaptation time and workload.

The ISIS shown is in 3ATI form factor and uses a high-resolution full-colour AMLCD with a usable screen area of 2.4 × 2.4 in. It operates from the 28 V DC essential bus and consumes 15 W at full brightness.

Airworthiness regulations

Regulatory requirements

The airworthiness regulations are expressed in the Federal Airworthiness Requirements (FARs) and Joint Airworthiness Requirements (JARs) published by the US and European Aviation Certification Authorities respectively. The most significant FARs and JARs are:

• 25.1303	Flight and navigation instruments.
• 25.1309	Equipment, systems, and installations.
• 25.1323	Airspeed indicating systems.
• 25.1333	Instrument systems.
• 25.1334	Flight director systems.

Other significant documents referenced include:

• FAA AC 25-11	Advisory circular, transport category airplane electronic display systems (**10**).
• SAE ARP 1874	Design objectives for CRT displays in transport aircraft (**11**).
• SAE AS 8034	Minimum performance standards for electronic displays (**12**).
• FAA TSO-C113	Airborne multipurpose electronic displays (**3**).

In summary, JAR 25 paragraph 1309 requires that equipment, systems, and installations must be designed to ensure that they perform their intended functions under all operating conditions. The occurrence of any failure condition that might prevent the continued safe flight and landing of the aeroplane should be extremely improbable. Also, the occurrence of any other failure condition that would reduce the capability of the aircraft or the ability of the crew to cope with adverse operating conditions should be remote. Warning information should be provided to alert the crew to any unsafe system operating conditions to enable them to take appropriate actions.

The Certification Authorities have provided extensive guidance material to facilitate the design and analysis of systems that can demonstrate compliance with the JAR requirements. This material is to be found in AMJ 25.1309 (AMJ stands for Advisory Material Joint) (**13**). The AMJ categorizes failure conditions and assigns rational probabilities of occurrence to them on the basis that an inverse relationship should exist between the probability of the loss of a function or its malfunction and the degree of hazard to the aeroplane and its occupants that arises as a consequence. Failures are classified according to their severity as minor, major, hazardous, and catastrophic and are correlated to the JAR 25 definitions as shown in Table 7.1.

Table 7.1 JAR 25 definitions of failure conditions and probabilities of occurrence

Severity	Probability	Analysis
Minor	Reasonably Probable	No more frequent than 1 in 10^{-3} per flight hour
Major	Remote	No more frequent than 1 in 10^{-5} per flight hour
Hazardous	Extremely Remote	No more frequent than 1 in 10^{-7} per flight hour
Catastrophic	Extremely Improbable	No more frequent than 1 in 10^{-9} per flight hour

Certification guidelines

Representatives from all the major stakeholders in the civil aircraft business contributed to the preparation of guidelines that would capture best practice and provide consistency of approach to aid in the certification of complex airborne systems to satisfy the regulatory requirements. This guidance material is provided in the following documents:

- SAE ARP 4754: Certification considerations for highly integrated or complex aircraft systems (**14**).
- RTCA DO 178B: Software considerations in airborne systems and equipment certification (**15**).
- RTCA DO 254: Design assurance guidance for airborne electronic hardware (**16**).

These documents provide specific guidelines for the design and production of systems, software, and hardware for airborne systems and equipment to assure that the system (in this case the display system) performs its intended function with a level of confidence in safety that complies with the airworthiness requirements above.

The guidelines are in the form of objectives for the design life cycle processes, descriptions of the type of analysis required to achieve those objectives, and descriptions of the evidence that should be provided to indicate that the objectives have been satisfied. The system, software, and hardware guidelines interact. Analysis of the system considerations will result in a system safety assessment allocating systems requirements to functional areas in hardware and software. The software and hardware analyses provide feedback into the systems analysis to complete the cycle.

Hardware and software design quality assurance levels are assigned on the basis of the potential cause or contribution to a failure of system function by their anomalous behaviour (Table 7.2).

Table 7.2 Hardware and software design quality assurance levels

Level	Contribution to resultant failure condition for the aircraft
A	Catastrophic
B	Hazardous
C	Major
D	Minor
E	No-effect

Display format guidelines

The advisory circular, FAA AC 25-11 (**10**) encapsulates perceived wisdom for the design and certification of electronic display systems. The document is advisory rather than mandatory, but it does provide guidance and outlines methods for complying with the regulatory requirements. The advisory circular provides specific display data integrity guidelines (Table 7.3). It recommends that:

- The display should convey information in a simple uncluttered manner. Colour alone should not be used as a discriminator. Symbols or messages should be logically and consistently positioned and co-located with associated information.
- That careful attention be given to symbol priority to assure easy interpretation of three-dimensional information on a two-dimensional medium.
- Display elements should be natural, intuitive, and not dependent on training or adaptation for correct interpretation.
- The 'basic T' relationship should be followed. If side-by-side formats are used, then attitude, airspeed, altitude, and heading must still reside in the 'basic T' arrangement.
- Heading and attitude must be presented on the same display.
- Airspeed and altitude should be arranged so that the present value is located as close as possible to a horizontal line extending from the centre of the attitude indicator.

The advisory circular cautions about the compelling nature of the map display and notes that there have been incidents where gross map position errors have gone undetected or unbelieved because the flight crew have falsely relied on the map instead of correct raw data. It reinforces the need for operating procedures that require one crew member still to monitor raw navigation data.

Table 7.3 Display data integrity guidelines provided by advisory circular FAA AC 25-11

		Probability of	
Parameter	**Criticality**	**Complete loss**	**Misleading data***
Engine data	Critical	Extremely improbable	Extremely improbable
Attitude, airspeed, altitude, heading	Critical	All = extremely improbable† Primary = extremely remote‡	Extremely improbable
Navigation, vertical speed, slip/skid crew alerting	Essential	Extremely remote	Extremely remote

[*] Misleading data are simultaneously misleading data on primary displays.

[†] All loss denotes loss of the parameter on primary displays and standby instruments.

[‡] Primary loss denotes loss of the parameter on primary displays.

The standardized colour coding that is advised to facilitate information discrimination is shown in Table 7.4, and that for weather radar precipitation and turbulence in Table 7.5.

Table 7.4 Standardized colour codings to facilitate information discrimination

Warnings	Red
Flight envelope and system limits	Red
Cautions, abnormal sources	Amber
Earth	Tan/brown
Sky	Cyan/blue
Scales and associated figures	White
Flight director/ILS deviation pointer	Magenta/green
Autopilot engaged modes	Green
Fixed reference symbols	White
Current data, values	White or green
Active route plan	Magenta or white

Table 7.5 Standardized colour codings for radar precipitation and turbulence

Precipitation (mm/h)	
0 – 1	Black
1 – 4	Green
4 – 12	Amber/yellow
12–50	Red
>50	Magenta
Turbulence	White or magenta

Display system architectures

Display suite components

The constituent elements of any Electronic Flight Instrument System (EFIS) (Fig. 7.25) are:

1. Data collector/concentrator which
   - (a) acquires the data to be displayed from the other on-board systems,
   - (b) selects the most appropriate data sources,
   - (c) performs data integrity checks.
2. Display management processor which
   - (a) determines the display mode, submode, and elements to be displayed,
   - (b) translates the above information into graphics data and commands.
3. Symbol/graphics generator which
   - (a) constructs the symbology and graphics comprising:
     - alpha-numeric characters in a range of font styles and sizes,
     - special symbols, pointers, and icons,
     - lines of various widths and styles,
     - circles, ellipses, and arcs of various widths and styles,
     - area shade infill.
3. Display unit which comprises
   - (a) the display device itself ,
   - (b) display device electronic support circuits.

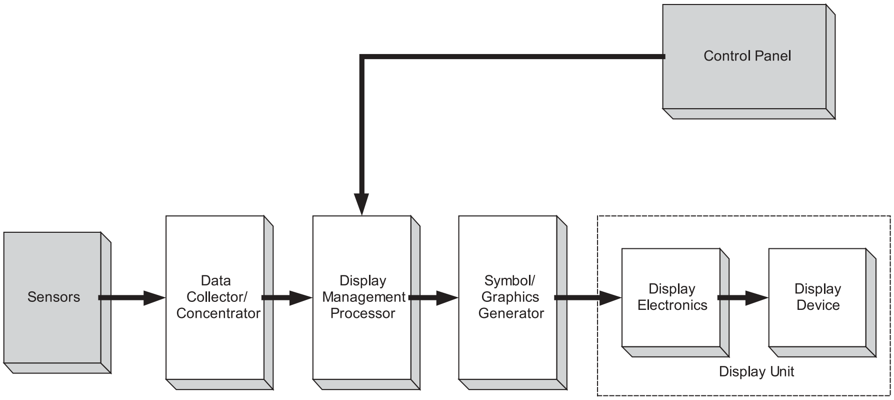


Fig. 7.25 *Basic EFIS architecture*

These display elements can be and have been combined in a variety of ways to form a display suite. The common arrangements are as follows.

Dumb display architecture

The dumb display architecture shown in Fig. 7.26 was the first to be adopted in civil avionics display systems and followed the architecture of most military CRT display systems. The display unit contains all, but only the necessary, electronics to support and drive the display device itself. For a shadow-mask CRT to provide a sufficiently bright display in an airborne environment, the image must be drawn in stroke mode. This requires complex and power-hungry electronics. The symbol generator contains all the display processing electronics.

In the situation where the display device is a CRT, the interface between the symbol generator and the display unit usually consists of specialized analogue *X* and *Y* waveforms to describe the CRT beam deflections together with digitally coded signals to describe symbol and shade infill colour. This architecture is equally applicable to AMLCD technology, in which case the interface is most likely to be a dedicated high-bandwidth digital video link encoded with red, green, and blue (RGB) information.

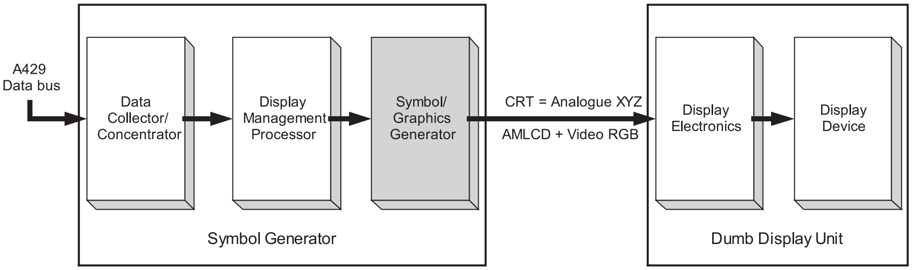


Fig. 7.26 *Dumb display architecture*

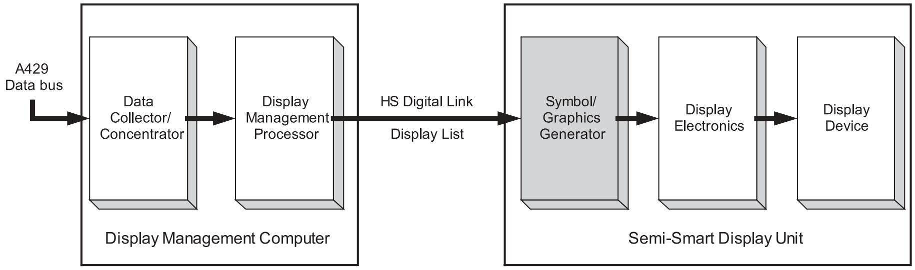
Semi-smart display architecture

The semi-smart display architecture shown in Fig. 7.27 overcomes the need for a specialized analogue/digital interface to the display unit tailored to the display device type.

The graphics generator is placed within the display unit (in addition to the display drive electronics required by the dumb display architecture). The Display Management Computer (DMC) assembles and compiles the image into a 'display list' of instructions to be executed by the symbol generator in the display unit.

The interface between the two units is digital and can use a conventional digital link such as high-speed ARINC 429. This interface is display technology independent and equally applicable to CRT, AMLCD, or any other display medium.

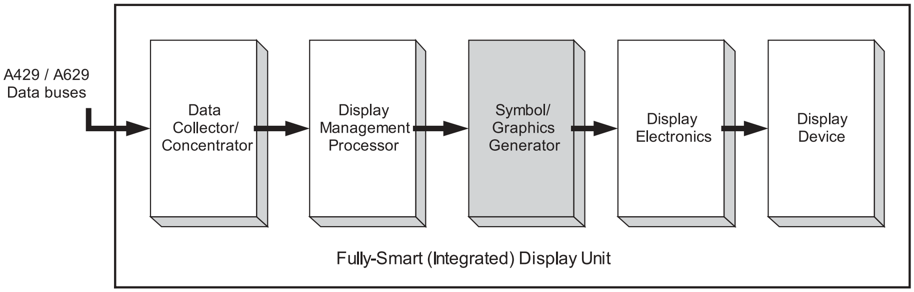
Fig. 7.27 Semi-smart display architecture



Fully smart (integrated) display architecture

In the fully integrated smart display architecture shown in Fig. 7.28, all the electronics are contained within the display unit itself. This architecture is now made practical by the physical reduction in size, weight, and power of AMLCD technology when compared with CRT. Furthermore, significant reductions have also taken place in computing and graphics technology to simplify the packaging and cooling arrangements.

Fig. 7.28 Fully smart (integrated) display architecture



Display systems

Flight deck instruments/displays can be categorized into three types:

- Primary flight instruments providing information on the ability of the aircraft to sustain safe and controlled flight.
- Navigation instruments providing information about the aircraft position with respect to its surroundings in order for the aircraft to be safely directed from its point of origin to its destination.
- Engine instruments and systems providing information about the aircraft systems to assure the continued safe function of the aircraft for the duration of the intended flight.

Electronic flight instrument systems

Systems providing EFIS PFD and ND functions typically have three symbol generators operating with four display units as shown in Fig. 7.29. In normal operation the third symbol generator is a 'hot spare'.

Each Symbol Generator (SG) is able simultaneously to produce both the PFD and ND formats. Each Display Unit (DU) sources its display format from a normal or an alternative SG source. Each symbol generator sources its inputs from both left-hand and right-hand sensors. Each display unit is able to select its display input from a normal or an alternative source.

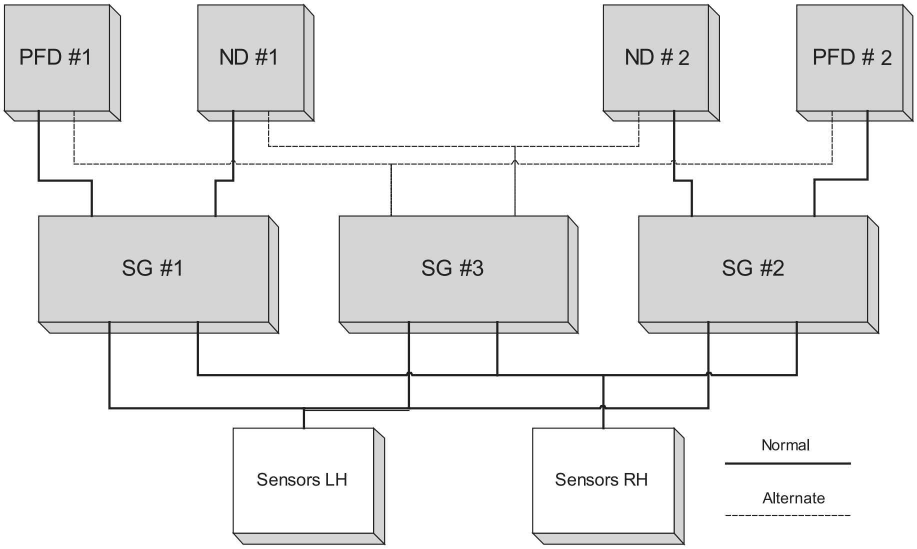


Fig. 7.29 *Typical EFIS architecture*

In the event that one symbol generator fails (say SG1), as shown in Fig. 7.30, then the affected display units (in this case PFD1 and ND1) select their inputs from SG3. The formats presented on PFD1 and ND1 remain independent of those on the other side of the flight deck.

If one DU fails (say PFD1), as shown in Fig. 7.31, then SG1 will reconfigure its output so that the image on ND1 will be a composite PFD/ND display with all requisite primary flight data including a compass.

Fig. 7.30
Reconfiguration after SG failure

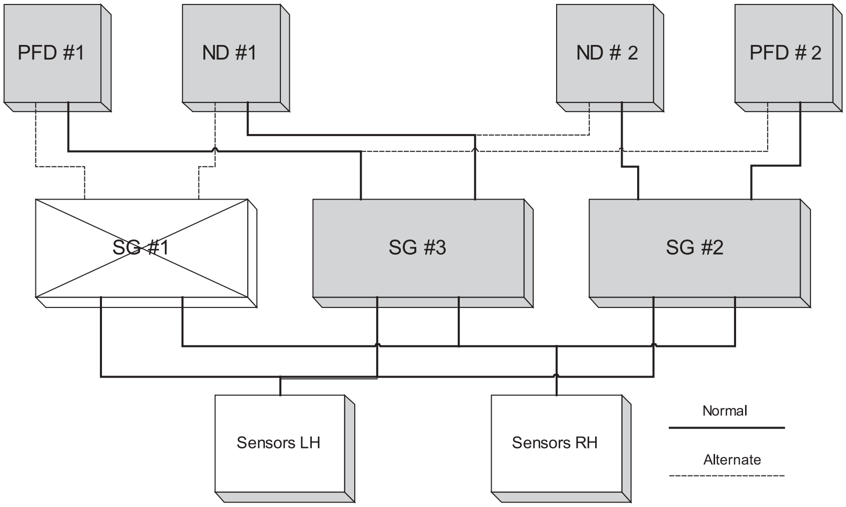
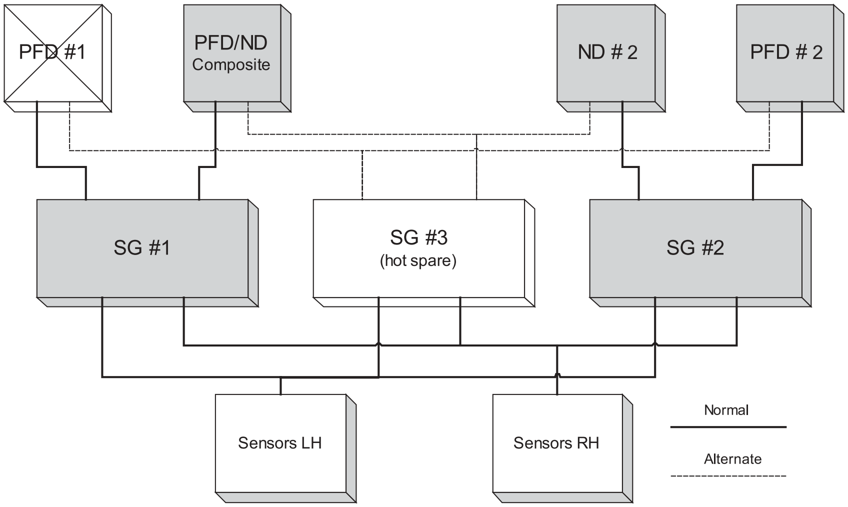


Fig. 7.31
Reconfiguration after DU failure



Independent and segregated standby ADI, altimeter, and airspeed instruments are installed in the flight deck to aid in resolving discrepancies and to provide critical flight information in the event of complete failure of the primary avionic systems.

If the failure rates of the SG and DU are λ_{SG} and λ_{DU} , then the probability of complete loss of attitude, heading, airspeed, or altitude from the primary flight displays owing to failure of the EFIS system itself is

$$(\lambda_{DU})^4 + (\lambda_{SG})^3 = \text{better than 1 in } 10^{-9} \text{ per hour (extremely improbable)}$$
$$\text{for } \lambda_{DU} = \lambda_{SG} = 4 \times 10^{-4} \text{ per hour (2500 h MTBF)}$$

Strategies to assure that hazardously misleading information is not inadvertently presented to the crew include:

- Cross-monitoring of input sensor data from duplex, independent, and segregated sensors.
- Correlation of sensor data from several input sources for reasonableness against a model of aircraft performance.
- Cross-monitoring of output data to the display.

These strategies are shown architecturally in Fig. 7.32.

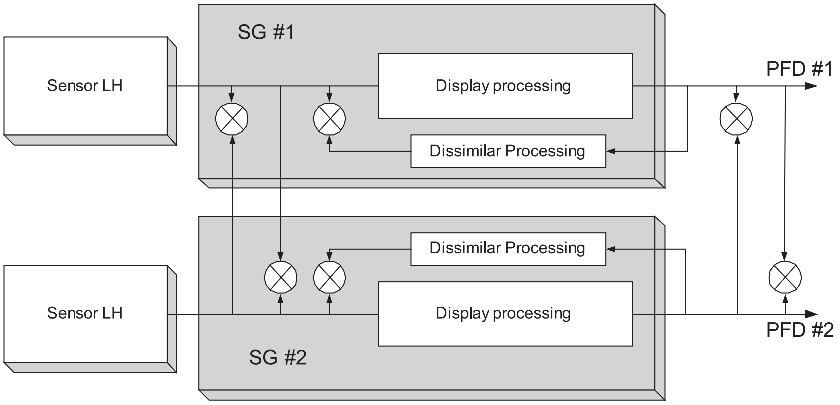


Fig. 7.32 *Cross-monitoring*

Integrity of the display system itself is continuously monitored by Built-In-Test (BIT) functions, which typically check anticipated output against known stimuli. Common mode failures of the display system itself can be addressed by a number of strategies including:

- Design quality assurance (using the design process guidelines described earlier).
- Architectural features such as multiversion dissimilar processing.

Cross-monitoring of output data is usually done at a point in the graphics generator where positional information is still available digitally. Typically, this allows the read-back of start and end points of critical lines, arcs, and characters, and read-back of character descriptors. It is usually accepted that upstream failures in the graphics channel and the display itself from this point, even if undetected by BIT, will be fail obvious and not introduce subtle undetectable errors on critical information alone. Typically:

- Alpha-numerics will be grossly distorted or illegible.
- Lines will be distorted and not join correctly with other lines to make recognizable shapes.
- Circles will be distorted.
- Shade infill will be incorrect.
- Colours will be wrong.

The frozen display is a serious potential failure mode that must be carefully analysed and addressed. The designer must include within the system secure methods to protect, warn, and remove any possibility of an undetected frozen display.

The misleading data criteria can be shown by analysis to be achieved provided that the system has been designed in accordance with the JAR guidelines and at the appropriate criticality level.

Combined EFIS and EICAS/ECAM systems

Display systems that integrate the EFIS and the EICAS (the Electronic Centralised Aircraft Monitor (ECAM) , in Airbus terminology) are typified by the architecture of the Airbus A330/340 as shown in Fig. 7.33. The A330/340 employs semi-smart Display Units (DUs) as described earlier. The Display Management Computers (DMCs) collect data from the aircraft systems and compile a 'display list' which is interpreted by the symbol generator within the display unit to generate the display image.

The avionics architecture is a traditional system. Separate Line Replaceable Units (LRUs) are provided for each major identifiable function (e.g. air data, inertial navigator, flight control, radios). In the Airbus A330/340 these systems communicate via dedicated single-source, multisink ARINC 429 serial databuses. Each data link from the DMC to the DU is a dedicated high-speed ARINC 429 databus.

Three DMCs drive the six DUs. Each DMC is able to source four display formats: the PFD and ND and the two power/systems displays. A three-way switch selects the DU display source.

In normal operation, DMC1 sources PFD1 and ND1, DMC2 sources the power/systems displays, and DMC3 sources PFD2 and ND2.

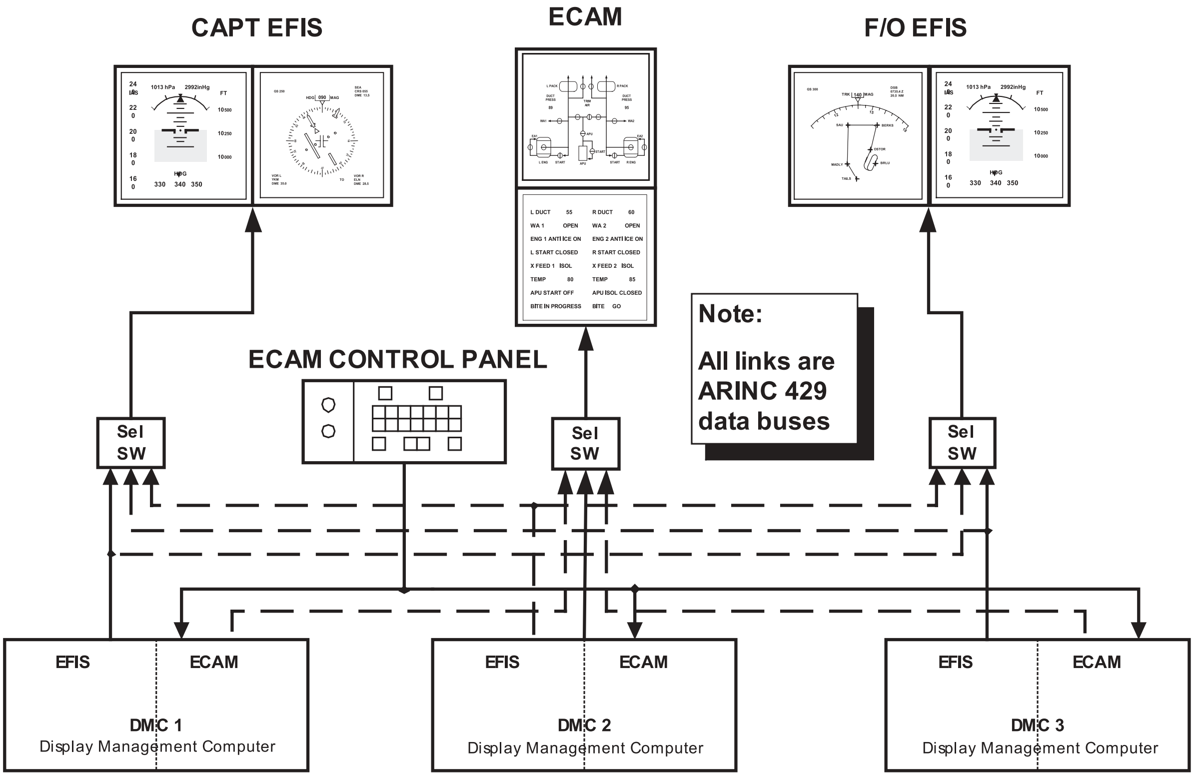
In the event of failure of DMC1, DMC2 sources PFD1 and ND1 and the power/systems displays, and DMC3 sources PFD2 and ND2.

In the event of failure of DMC2, DMC1 sources PFD1 and ND1 and the power/systems displays, and DMC3 sources PFD2 and ND2. In the event of failure of DMC3, DMC1 sources PFD1 and ND1 and the power/systems displays, and DMC2 sources PFD2 and ND2.

It is important to note that, for any single DMC failure, PFD1/ND1 remain independent of PFD2/ND2. In the event of failure of DMC2 and 3, DMC1 sources PFD1 and ND1, copied to PFD2 and ND2 and the power/systems displays.

In the event of failure of one of the pair of PFD/ND DUs, the DMC constructs a combined PFD/ND format on the remaining DU. Similarly, in the event of failure of one of the pair of power/systems DUs, the DMC constructs a combined power/systems format on the remaining DU.

By inspection it is clear that the architecture more than satisfies the JAR criteria for



parameter availability. Indeed, it can be argued that the aircraft can dispatch with only two functional DMCs.

Fig. 7.33 *A330/340 EFIS and ECAM architecture*

Modular avionics display systems

The architecture of the Boeing 777 Aircraft Information Management System (AIMS), as shown in Fig. 7.34, illustrates the integration of smart display units with a modular avionics architecture. The six fully smart integrated DUs contain all the electronics to manage, process, and generate display formats from data transferred to them from the AIMS cabinet via high-speed ARINC 629 databuses. The databus is connected to all DUs; two databuses provide redundancy.

The AIMS cabinet provides a centralized and redundant computing resource on which the avionic functions of the classic federated architecture reside as software applications. A comprehensive software operating system assures a safe and secure partitioned operating environment.

The AIMS cabinet manages failure of internal functions by reconfiguring applications to run on the remaining functional computing resources. DU failure is managed as described earlier, by presenting combined formats on remaining DUs.

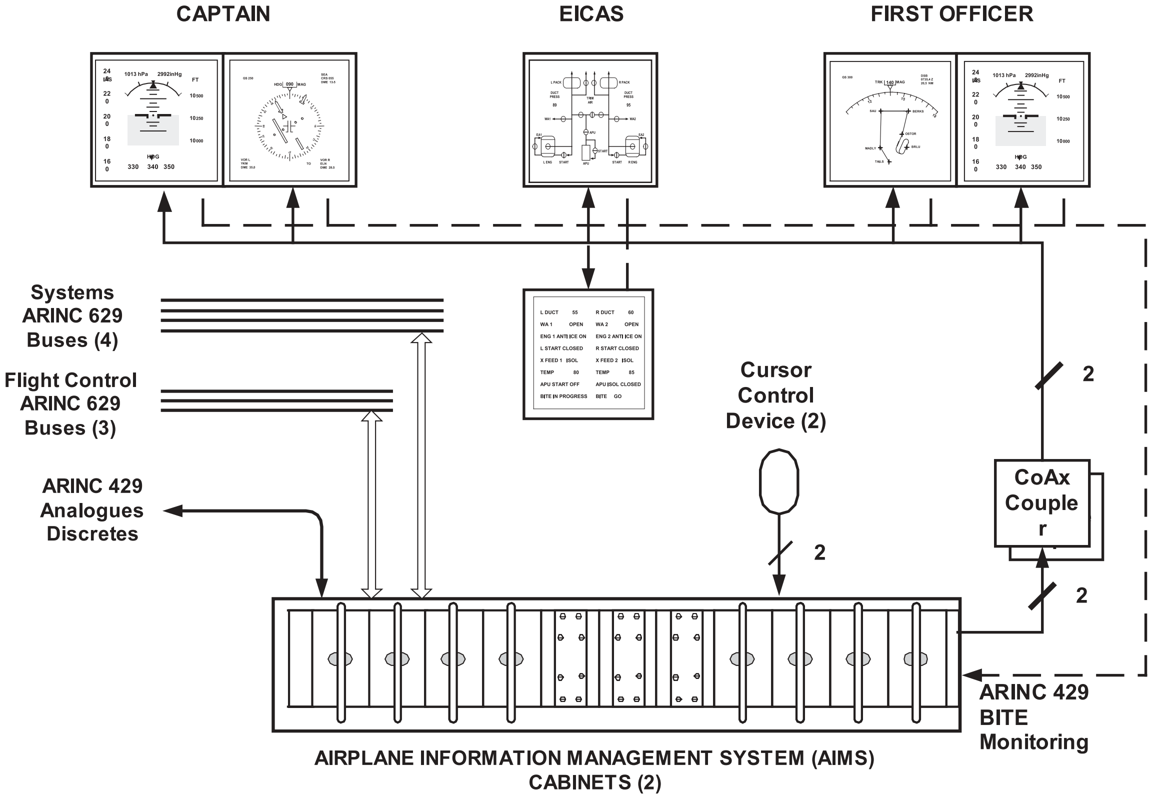


Fig. 7.34 *Boeing 777 AIMS display architecture*

Display media

Visual requirements

Four lighting conditions need to be carefully considered in the display design:

- Direct sunlight falling on the display through a side window.
- Sunlight through a front window illuminating a white shirt front that is reflected in the display.
- Sun above the forward horizon and above a cloud deck reflecting into the pilot's eyes (known as veiling glare).
- Night or dark environment in which the display brightness should be dimmable such that outside vision is not impaired.

Documents that provide guidance on the visual performance requirements include:

• FAA AC 25-11	Advisory circular, transport category airplane electronic display systems (**10**).
• SAE AS 8034	Minimum performance standards for airborne multipurpose displays (**12**).
• SAE ARP 1874	Design objectives for CRT displays for transport aircraft (**11**).
• SAE ARP 1068	Flight deck instrumentation, display criteria, and associated controls for transport aircraft (**17**).

- SAE ARP 4256 Design objectives for liquid crystal displays for Part 25 (transport) aircraft (**18**).

Chrominance and luminance

- CRT and backlight phosphors are known to age, so the start-of-life performance should take into account any degradation that will occur through the expected life.
- Shade infill used as backgrounds must not distract from the legibility of information written over the background.
- The luminance of information underlays such as weather radar should be independently adjustable relative to the luminance of the main symbology.
- Automatic brightness adjustment systems can be employed to match display luminance to the prevailing ambient lighting conditions, but experience has shown that great care needs to be taken to position light sensors appropriately.

Jitter

- Rapid small spatial movement of a symbol can cause mild fatigue.

Line width and symbol and character size

- Lines need to be of an adequate thickness to be easily assimilated. In a colour display image which is fabricated from three primary colours, care must be taken to ensure colour alignment (convergence) of the three images and that colour fringing does not occur.

Flicker

- Flicker can cause mild fatigue or headaches, and should not be perceptible day or night either in direct view or in the peripheral vision of the user.

Dynamics

- Jitter, jerkiness, or ratcheting of display elements normally in motion should not be distracting or objectionable. Conversely, smearing of moving symbols should not be discernable or objectionable.
- Screen update rates for analogue signals used in direct aircraft control tasks should be equal to or better than 15 Hz.
- Any lag between input signal and display should be consistent with the airplane control task associated with that parameter. In particular, display system lag for attitude should not exceed 100 ms.

Environmental requirements

The environmental conditions and test procedures for airborne equipment are expressed in RTCA DO-160. This document defines a minimum performance standard against a set of test conditions or categories. The onus lies with the equipment installer to be assured that the standard and test conditions are appropriate for the application, seeking additional tests and analyses if required.

Electrical power supply transient immunity

Recovery time after a loss of power can be a safety-related concern:

- The design objective should be that the displays are insensitive to power transients.
- Bus transients caused by normal load switching (hydraulic pump actuation, ovens, generator paralleling, etc.) should cause no visible effect on the display.
- Abnormal bus transients (i.e. after generator failure not caused by engine failure) should not initiate a power-up initialization or cold start process.
- The display of attitude information should not be unusable or unstable for more than 1 s after bus transients due to an engine failure, and should affect only displays on one side of the flight deck.
- Recovery from power interruptions should not cause any unsafe condition in system operation.
- In no case should power transients cause a frozen display.

Generally, these requirements have to be satisfied by a combination of display equipment performance and electrical power distribution design.

The Cathode Ray Tube

The cathode ray tube was the first fully flexible display device to be used for primary flight displays in the flight deck. Although early research used monochrome CRTs, the added dimension of colour was soon introduced once it was established that the shadow-mask colour CRT could be applied to the airborne flight deck environment.

The shadow - mask CRT – principles of operation

The shadow-mask CRT comprises an evacuated glass bulb in which an electron gun emits electrons at high velocity to impact on a phosphor screen. The electron gun comprises three cathodes. Electron emission is modulated by three grid electrodes, focused by a multistage electron lens and finally accelerated towards the faceplate through a shadow-mask. A series of red, green, and blue phosphor dots are deposited on the CRT faceplate in a fabrication process that uses the shadow-mask itself to maintain precise registration. The electron gun, shadow-mask, and phosphor dot geometry are arranged so that electrons emitted by each cathode only illuminate phosphors dots of its designated colour. The composite beam bundle is deflected in the X and Y axes by a magnetic yoke to scan the CRT display surface. Modulating the individual beams produces a full colour image comprising three registered primary colour images.

The CRT requires specialized high-voltage power supplies, typically around 20 000 V (20 kV) final anode potential and 6–8 kV focus potential. Beam currents are of the order of 600 μ amp per gun. With a shadow-mask transmission of less than 30 per cent, 20 W of power is lost in the mask itself which must be radiated to the bulb. Spring attachments maintain the mask in tension and in registration with the phosphor dots.

In order to prevent disturbance of the beam geometry by external magnetic fields, the CRT bulb is encased in a mu-metal shield. The shield also incorporates damping mechanisms to ruggedize the CRT to the airborne environment (see Fig. 7.35).

The phosphors deposited on the CRT faceplate emit light in the red, green, and blue primary colours. By suitable combination of cathode drive voltages, any colour can be produced within the triangle described by the three primary colours (see Fig. 7.36).

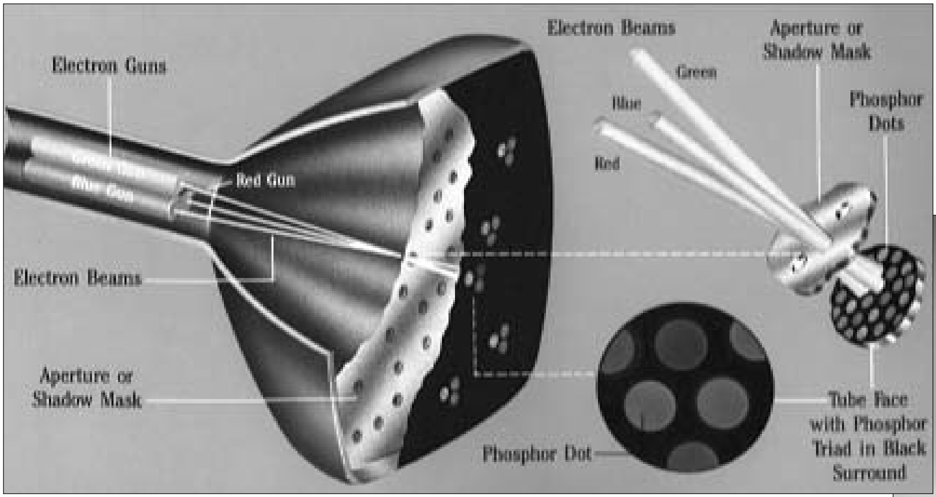


Fig. 7.35 *The shadow-mask CRT (see colour plate section)*

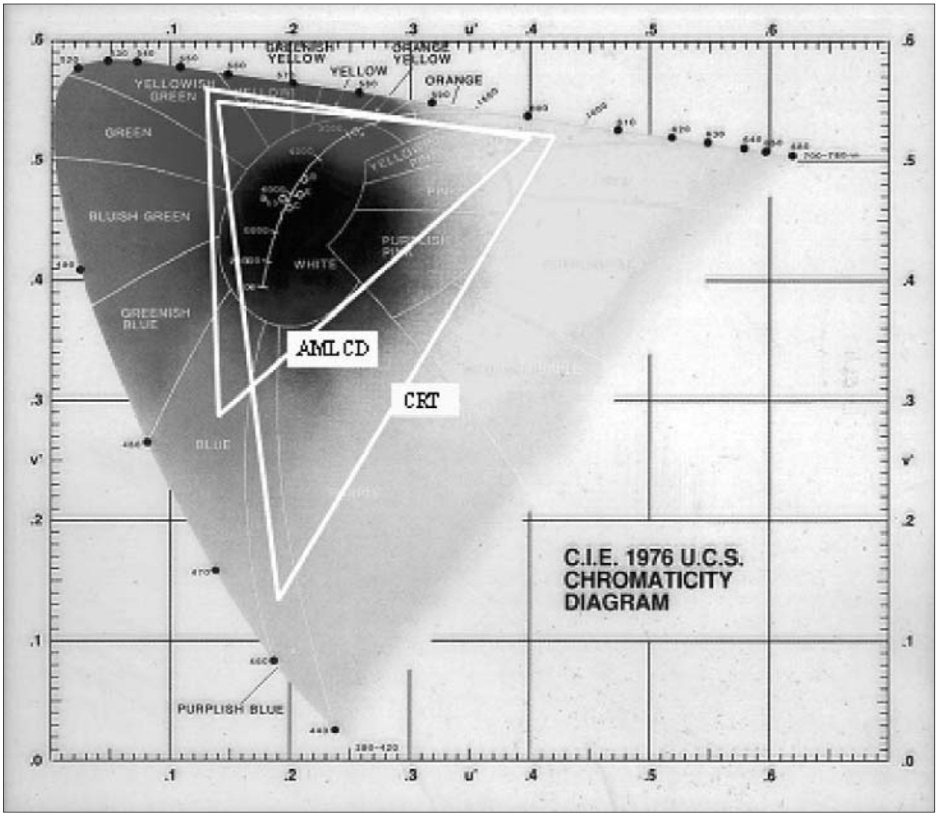


Fig. 7.36 *CRT and AMLCD colour gamut (see colour plate section)*

Shadow-mask CRT characteristics

The shadow-mask CRT has a number of characteristic features that require careful attention, if acceptable performance is to be obtained in the civil flight deck application:

1. Brightness. In video applications the CRT image is developed as a rasterscan. The information content is carried entirely in the modulation of the beam. For airborne applications, a rasterscan will simply not produce a bright enough image to be seen under the adverse lighting conditions to be found on the flight deck. For this reason, in airborne CRT displays the CRT beam is deflected to trace out the symbology like a pen plot and not waste time traversing the dark areas between. Hence, the beam can travel more slowly and impart more energy to the phosphor, producing a brighter image. The information content is thus primarily in the beam deflection, and the beam modulation merely defines the information colour. This mode of operation is called stroke (or cursive) writing.
2. Contrast. Unilluminated by the electron beam, the phosphor dots are a whitish-grey colour. A contrast enhancement filter has to be fitted to the CRT to reduce the sunlight reflections that otherwise would occur in the adverse lighting conditions of the flight deck. While the filter attenuates light emitted by the CRT, sunlight reflected by the phosphors has to pass through the filter twice and is therefore attenuated twice. Typical single-pass transmission to daylight is 30 per cent. The filter typically has a special spectral transmission characteristic, called a triple-notch filter, which favours the spectral peaks of the CRT phosphors while attenuating other frequencies.
3. Colour purity. Any errors in registration between the electron beam and the phosphor dots will produce a landing error partially exciting an adjacent phosphor dot of the wrong colour. Impure colours can be produced by small residual magnetic fields, and changes in the Earth's magnetic field are sufficient; hence the need for the mu-metal shield.
4. Convergence. It is essential to maintain the constituent parts of the image in perfect registration; that is, the three electron beams must converge at all points on the CRT faceplate. Any misconvergence of, for instance, a white line would cause it to diverge into its red, green, and blue components with irritating and potentially confusing results.
5. Line width. The symbology line width in a stroke-written CRT is a function of the electron beam spot width. At high brightness, with corresponding high electron beam current, the CRT spot width will grow with a tendency to blur or flare. Careful design of the electron lens, control of manufacturing tolerances, and electron gun alignment are required to mitigate against this effect. Conversely, at low brightness and hence low drive current, the spot size may be too small and the electron beam may have to be defocused to obtain the desired line width. Background shaded areas (such as attitude sky/ground) are generally produced by local raster infill and defocused to blur the raster line structure.
6. Greyscale/colour tracking. The display is used over a wide range of ambient lighting conditions, and the display brightness range required is of the order of 5000:1. The ratio of electron beam drive currents must be maintained across this wide dynamic range to assure that colours track correctly. Careful design is required of the cathode drive amplifiers, which must continually monitor and adjust their parameters to track cathode performance as the CRT ages.

7. Flicker/refresh rate. Colour CRT phosphors have short persistence, that is, the decay time is less than 1 ms. In order to avoid flicker in peripheral vision, it is necessary to refresh the display at rates in excess of 70 Hz. In a stroke-written display it is also essential that spatially related elements (e.g. scales with pointers) are also temporally related (i.e. written close together in time). Without this precaution, image elements may appear to separate under vibration, with the potential to misread information. The effect is created in the human brain, not in the display itself, and is due to vibration of the eyeball.

The Active Matrix Liquid Crystal Display

The AMLCD is now the accepted technology for all new applications. While it is true that the AMLCD overcomes many of the 'features' that make CRT a difficult technology for airborne applications, the AMLCD has 'features' of its own that have presented significant challenges.

The principle advantages of the AMLCD (plus its backlight) are:

- Significantly less depth than a CRT and easier to package (typically <60 mm).
- Significantly less weight than the CRT (typically <1 kg).
- Significantly less power than a CRT (typically <20 W).
- No high voltages.
- No magnetic components and no influence from external magnetic fields.
- Perfect registration (no misconvergence).
- Fixed pixel (spot) size, at all brightnesses.
- Perfect colour tracking (except, perhaps, some minor influence from the backlight at low temperatures).

The AMLCD – principles of operation

In an AMLCD, Liquid Crystal (LC) material is introduced between two glass plates. The spacing between the glass plates is critical and is around 4 μm. The following is a much-simplified explanation of the operation of the cell, a picture of which is shown in Fig. 7.37.

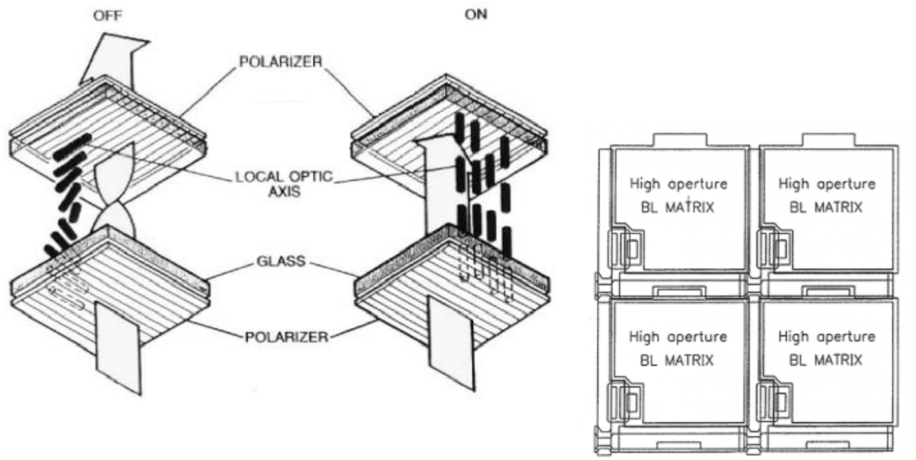
Liquid crystal material is a viscous organic fluid containing long polymer chains which have the property of rotating the plane of polarization of light according to their alignment axis. The alignment axis can be changed with electrical bias.

The plane of polarization of light alters as it passes through the cell. Placing two crossed polarizers either side of the cell (bonded to the glass plates) makes the cell into a light valve.

With the cell unbiased, light is plane polarized by the first polarizer, its plane of polarization is rotated by 90° as it passes through the LC material, and it then passes through the second polarizer. In the biased state, light is not rotated by the LC material and does not pass through the second polarizer. Greyscale is obtained by applying an intermediate voltage which partially rotates the molecules so the cell is partially transmissive.

Accurately maintaining the cell spacing is vital for correct operation. To achieve this, the LC material is filled with very accurately machined glass beads in suspension which become randomly distributed throughout the cell and are sandwiched between the glass plates under carefully controlled pressure during manufacture.

Fig. 7.37 The AMLCD cell



Row and column address lines are fabricated onto the active plate (prior to assembly) to form a matrix, and an amorphous silicon Thin-Film Transistor (TFT) is fabricated at their intersection. The row and column drive signals to the TFT control the charge applied across the local cell, defined by the TFT itself plus an associated pair of transparent electrodes placed opposite one another on the active and passive plate. This charge is the bias for the local light valve.

Red, green, and blue colour filters are deposited on the passive plate in exact register with the subpixels to form a colour group or pixel. A number of colour group arrangements have been popular, but now the accepted norm is the RGB stripe. See Fig. 7.38 for typical pixel structures.

A typical XVGA compatible panel, say 6.25 × 6.25 in., square, may have 768 × 768 RGB pixels, which is a total of 1.8 million pixels. The row and column lines are bought out to the edge of the glass plate., making a total of around 3000 connections for the XVGA panel described. The lines are driven by row and column driver Integrated Circuits (ICs).

The driver ICs are mounted on printed circuit boards which are folded back at 90° to the AMLCD glass panel. The ultrahigh-density interconnect between the drivers and the AMLCD glass panel is made with Tape Automated Bonding (TAB). The TAB process uses copper traces in a flexible polyamide base layer. Connections are made between the TAB and the glass panel using a pressure-sensitive anisotropic adhesive. The adhesive contains small silver balls which touch each other under pressure and make contact between circuit traces in the TAB and circuit traces on the glass. The balls do not touch in the axis perpendicular to the pressure axis, and isolation is maintained between adjacent traces. Once the connections are correctly made, the adhesive is heated and permanently sets (see Fig. 7.39).

The AMLCD is a light valve but with a transmission of less than 7 per cent. To obtain a usable display, the addition of a bright backlight is required (see Fig. 7.40). The fluorescent lamp is the technology of choice, but for airborne applications it has to be much brighter than the lamp in a laptop PC. Also, the light source has to be dimmed for night-time use. A dimming ratio of about 5000:1 is required.

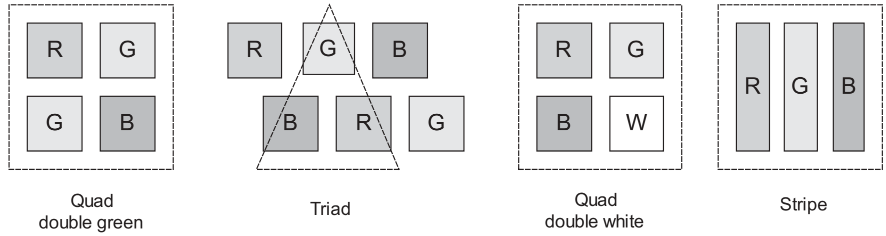


Fig. 7.38 *Colour group pixel structures (see colour plate section)*

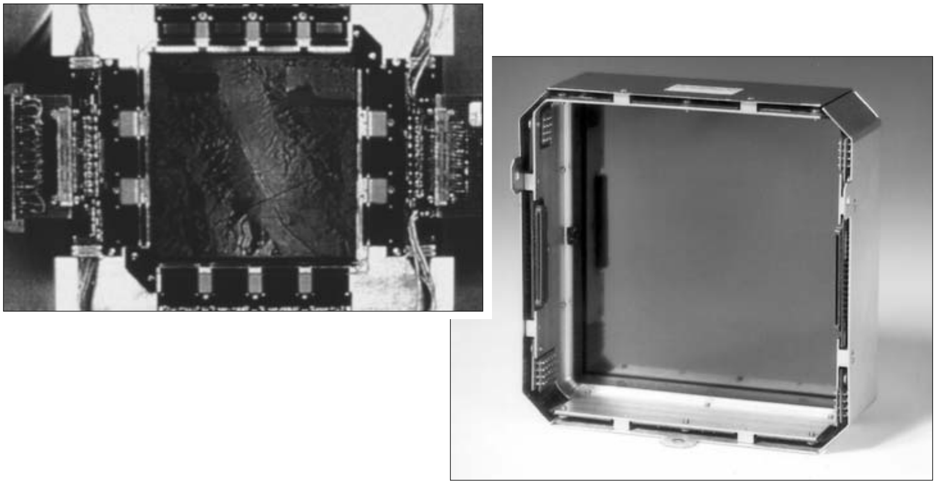


Fig. 7.39 *AMLCD panel with drivers – Korry Electronics*

Fig. 7.40 *Single cold cathode, serpentine backlight – Korry Electronics*

A number of fluorescent lamp backlight configurations have been used, hot or cold cathode, single or multiple lamps. At first there was considerable concern over the reliability and longevity of fluorescent lamp cathodes, and therefore early designs had multiple lamps since failure of the backlight would be a single-point failure that would result in complete loss of the display.

Extensive life testing coupled with materials research and development of sophisticated drive circuits has now established the fluorescent lamp as a reliable and long-lived product. The single, serpentine, cold-cathode lamp has become the configuration preferred by most designers (**19**). An optical stack is placed between the fluorescent tube and the backlight to provide a uniform diffuse light source to back-illuminate the AMLCD.

The fluorescent lamp works by striking an arc in a mercury vapour. The arc emits ultraviolet light which in turn is converted to visible light by phosphors deposited on the inside of the fluorescent tube. The lamp phosphors are a mix of the same red, green, and blue phosphors used in the CRT, but in this case the lamp produces a white light whose colour temperature is defined by the mix ratios of the three phosphors. The mix ratios are selected to match the AMLCD colour filter transmissions.

AMLCD characteristics

There are a number of characteristic features of the AMLCD, different to those of the CRT, which require careful attention if acceptable performance is to be obtained in the civil flight deck application:

1. Viewing angle. The optical performance of an AMLCD is critically dependent on the cell gap. At increasing viewing angles from the normal to the display, the optical cell gap effectively increases and the optical performance of the cell degrades. It no longer acts as a perfect light valve, and some light 'leaks' through the cell. This light leakage reduces the display contrast. Optical compensation films are often attached to the AMLCD to optimize the viewing angle for the particular flight deck installation, in particular to facilitate cross-cockpit monitoring of all the displays by both crew members.
2. The need for the crew to monitor primary flight information across the flight deck demands a horizontal viewing angle in excess of ±50°. This represents a considerable challenge for AMLCD technology. Special optical compensation films are bonded to the display to improve the viewing angle.
3. Greyscale. Greyscale is obtained by partially switching the cell between its on and off states. The greyscale transmission curve is not linear and requires temperature compensation. For the symbolic images used in primary flight instrumentation displays it has been demonstrated that 16 grey levels (4 bits) per colour are sufficient, a full colour video image requires 256 levels per colour (8 bits).
4. Black level uniformity. The display transmission in the black (driven) state is sensitive to a wide range of processing factors. Any impurities in the materials, any leaching of compounds to contaminate the LC material, any variation in cell gap introduced during the sealing process, and any non-uniform pressure applied by the bonding of cover glasses or attachment to mounting features can cause a local degradation in the black level. The result is a blotchy black background which can be most distracting. This feature, more than any other, has received adverse criticism for the AMLCD. Fortunately, improvements in materials and in process control, driven by the volume laptop PC market, are overcoming this deficiency.
5. Response time. Unlike the CRT, the response time of an AMLCD is long, typically 20–30 ms. The concern therefore is not flicker but smearing of rapidly moving characters, and careful choice has to be made of LC material to minimize smearing without degrading high-temperature performance.
6. Thermal management. Many of the parameters associated with the performance of the AMLCD are temperature sensitive, and it is necessary actively to manage the

thermal environment of the AMLCD. The LC molecules will not twist at low temperatures, and at high temperatures control is lost and the display clears. Fortunately, neither effect is irreversible. The ideal operating point is in the region 30–40 ˚C. Therefore, a rear cover glass is bonded to the AMLCD which has a transparent Indium Tin Oxide (ITO) heater deposited on it. The ITO heater forms part of a control loop that rapidly raises the temperature of the AMLCD at cold start and then maintains the operating temperature at the desired operating point.

7. The performance of the fluorescent lamp is also temperature dependent. Mercury will not vaporize at low temperatures and the arc cannot be struck. The fluorescent lamp operates most efficiently at temperatures in the range 40–50°C. Actively managed heaters likewise raise and maintain the operating temperature of the lamp.

The construction of a typical AMLCD display unit

The 5ATI display unit installed in the KLM Classic 747 aircraft is typical of current state-of-the-art fully integrated smart AMLCD instruments (see Fig. 7.41). The display unit is designed to replace electromechanical ADI, HSI, and engine instruments in ‘legacy’ or ‘classic’ aircraft. It is fully compatible with present and future aircraft systems and provides significantly improved reliability and operational flexibility when compared with the electromechanical instrument it replaces. A single part number fits all applications; the display unit detects its location in the instrument panel and configures itself accordingly.

The display unit includes interfacing and data processing to match a wide range of signal types including:

- Synchro and resolver.
- Analogue.
- Digital encoder.
- Discretes.
- Serial digital data bus (ARINC 429).
- Map (ARINC 568).

Fig. 7.41 *5 ATI display unit – Smiths Aerospace (see colour plate section)*

These interfaces can be configured to match the interfaces of the electromechanical instruments being replaced, providing form and fit interchangeability with increased functionality. A typical 5ATI unit architecture is shown in Fig. 7.42.

The integral graphics processing constructs and writes symbology onto the full colour active matrix liquid crystal display. The display formats include:

- EADI.
- EHSI – compass rose, map, plan, arc, TCAS.
- Engines – vertical tapes, round dials.
- EADI/EHSI Combined format.

Display formats mimic the presentations of the electromechanical instruments they have been designed to replace to aid pilot transition. New functionality replicates established practice for 'glass flight deck' instruments. The map display provides enhanced horizontal situation awareness and is fully compatible with present and future Global Positioning Systems (GPSs) and Flight Management Systems (FMSs). The map is a key element in the modern flight deck for the newly emerging concept of 'free flight' as part of the future air navigation system (FANS-CNS/ATM). Weather radar can optionally be overlaid on the map format.

The 5ATI subassemblies (Fig. 7.43) are housed in a chassis that is mechanically interchangeable with its electromechanical counterpart. The chassis comprises a rear card cage complete with a printed wiring interconnect harness into which the electronic modules are assembled. The AMLCD, complete with backlight, is securely attached to the front of the chassis. An optional slip indicator (passive bubble type) can be simply attached to the instrument front face if it is to be installed in the ADI location. Cardinal points of the 5ATI display unit are given in Table 7.6.

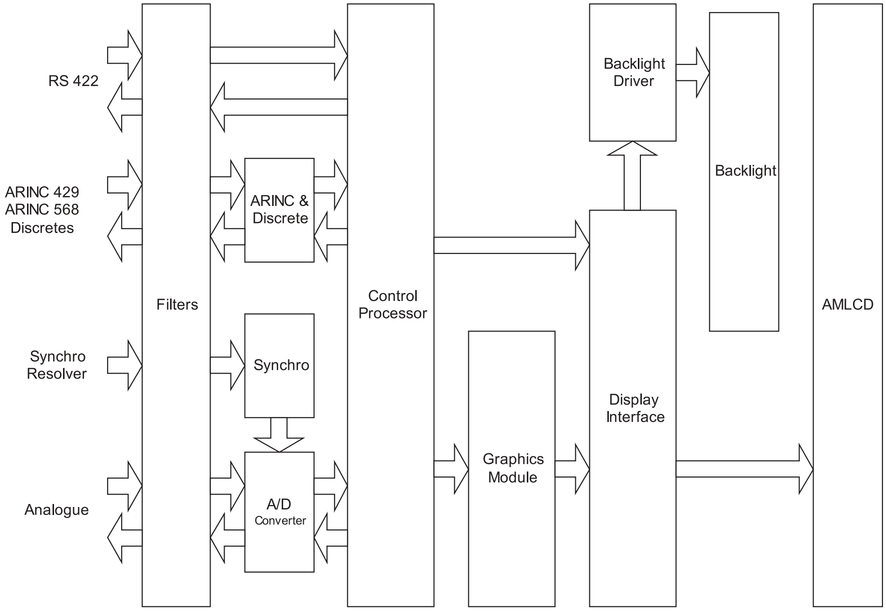


***Fig. 7.42** 5 ATI block diagram*

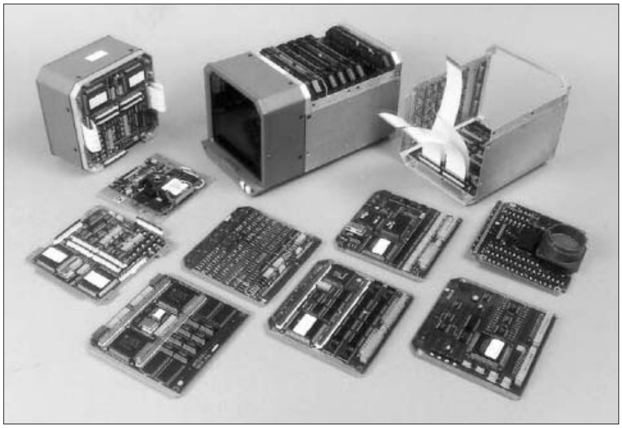


Fig. 7.43 *5 ATI sub-assemblies – Smiths Aerospace*

Table 7.6 Main characteristics of 5ATI display unit

Dimensions	127 mm × 127 mm × 229 mm deep (5 in. × 5 in. × 9 in.)
Weight	3.0 kg (8 lb)
Power	25 W typical, 35 W full brightness
Interfaces	RS422, ARINC 429/568, ARINC 453, synchro, resolver, analogue, digital encoder, discretes
Brightness	0.1–200 ft.L, continuously dimmable (including autobrightness)
Screen area	102 mm × 102 mm (4 in. × 4 in.)
Reliability	15000 h MTBF
Maintenance	On condition (no routine maintenance)
Certification	TSo-C3d, TSo-C4c, TSo-C6d, TSo-C52a, and TSo-C133

Displays for the flight deck of the future

Large area head-down displays

Side-by-side PFD and ND formats are now the accepted norm. Most recent flight deck designs have six display surfaces: a PFD and ND each for the captain and first officer and two power/systems displays. Square-format AMLCD glass is a custom product and has been difficult to source and therefore expensive. There is an emerging trend to adopt Commercial Off-The-Shelf (COTS) sizes of AMLCD glass, which, with appropriate treatment, can now achieve the performance requirements for civil air-transport applications.

Some flight decks are now using 6 × 8 in., essentially laptop PC glass, arranged in portrait mode. Future flight decks are likely to use larger-format AMLCDs as technology advances. Larger display sizes offer the potential to combine the PFD and ND onto a single composite surface. However, the display system architecture must be cognitive of the redundancy, availability, and integrity requirements demanded by the regulatory authorities. The evolution of display size is depicted in Fig. 7.44, and an example of a modern large display configuration is given in Fig. 7.45.

Fig. 7.44 *Display size evolution*

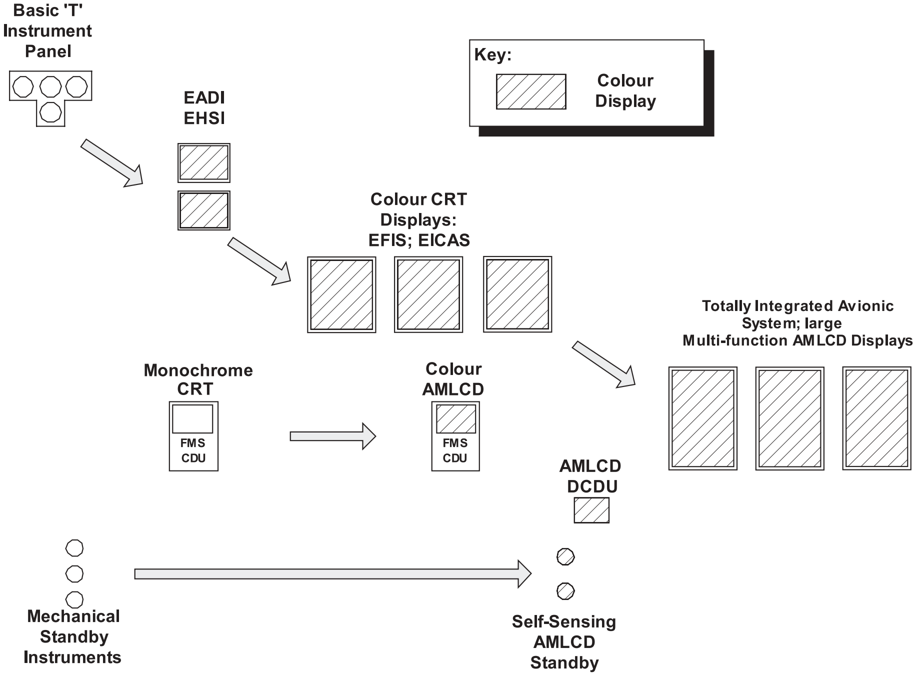


Fig. 7.45 *Conceptual flight deck with four 13 x 11 inch displays*

Three-dimensional and four-dimensional display formats

CRT and AMLCD primary flight displays have been installed and flown on civil aircraft flight decks for two decades, but the full potential of the media has yet to be realized. Display formats to date have copied or been extrapolated from the styles of the electromechanical instruments that preceded them. Those formats were very much constrained by the media and limitations of clockwork dials and pointers.

Present-day formats are still awash with numbers, letters, and symbols. These parameter descriptions are derived from tools used by scientists and engineers to abstract principles of reality for mathematical analysis. They suit that purpose well, but are an underuse of the CRT/AMLCD media to portray real-time spatial awareness. With current-generation formats, the pilot still has to perform mental gymnastics with two-dimensional information presented in two orthogonal axes on the PFD and ND displays to formulate a three-dimensional/four-dimensional picture of the real world.

The incidence of Controlled Flight Into Terrain (CFIT) accidents has not significantly improved in spite of the introduction of the 'glass flight deck'. CFIT accidents now account for more than 40 per cent of commercial air transport accidents worldwide. By most projections, air traffic is expected to double or even triple in the next 10 years. If nothing is done to improve the rate of CFIT accidents, then by the end of the decade there will be one CFIT accident every two weeks. The history of CFIT accidents shows that alpha-numeric symbology is just not compelling enough to prevent pilots from unconsciously flying into terrain.

CFIT accidents generally result from a sequence of events that link together as a chain to produce the end, often fatal, result. To prevent a CFIT accident, it is necessary only to break the chain at any one link. It has been said that, to combat CFIT, we must be more effective at managing the last moments or even seconds of an imminent event. To accomplish this, three principles must be recognized:

- There will forever be sequential combinations of events conspiring to produce the chains that lead to accidents.
- Human beings' primary reference in the world, the sense of sight, has evolved to be instantaneous, intuitive, and three-dimensional.
- CFIT accidents have historically occurred in all phases of flight and cannot be predicted.

We must therefore provide pilots with consistent, intuitive, three-dimensional real-time and predictive pictorial information, which requires no time for mental integration, in order to intervene most effectively during the last moments of a critical circumstance. Instrumentation of this kind can best be described as four-dimensional because its predictive elements account for the fourth dimension of time.

Early research into intuitive three- and four-dimensional graphical information displays coined the term 'highway-in-the-sky'. This concept was being developed at Langley Airforce Base at the time the BAC 111 first flew with colour PFD/ND displays in 1981. 'Highway-in-the-sky' itself first flew on a Calspan Laboratory C130 aircraft in 1983.

Many research programmes have further developed the 'highway-in-the-sky' concept. Modern computer graphics makes possible extremely good real-time representations of the real world. Display formats have been developed and flown experimentally that predict the next 60 s or so of flight set in the context of the real-world terrain to achieve constant, complete, naturally intuitive, and predictive spatial awareness. Simply put, the ‘highway’ tells you where you should go and the terrain tells you where you should not.

In 1995, the University of Munich, Germany, carried out a significant programme of work together with VDO Luftfahrtgerate Werk of Frankfurt. An aircraft was equipped with four-dimensional displays and executed taxi-way, take-off, approach, landing, and terrain avoidance manoeuvres with much success (**20**).

The experimental primary flight display shown in Fig. 7.46, provides a three-dimensional perspective of the surroundings. It promotes better situational awareness for the pilot and an early prediction of danger, especially the proximity of terrain. The three-dimensional presentation includes a synthetic picture of the terrain, with features such as obstacles, rivers, and towns displayed as clearly as the horizon and runway. The display changes in real time according to the aircraft direction. Surrounding the display are traditional band-scale indicator symbols for speed, height, heading, etc., to ease transition to the new format.

Special symbols for flight control are integrated into the display. The precomputed flight path (the ‘pathway-in-the-sky’) is shown as a perspective, wire-frame channel. A predictor symbol indicates the anticipated motion of the aircraft on the basis of its current manoeuvre.

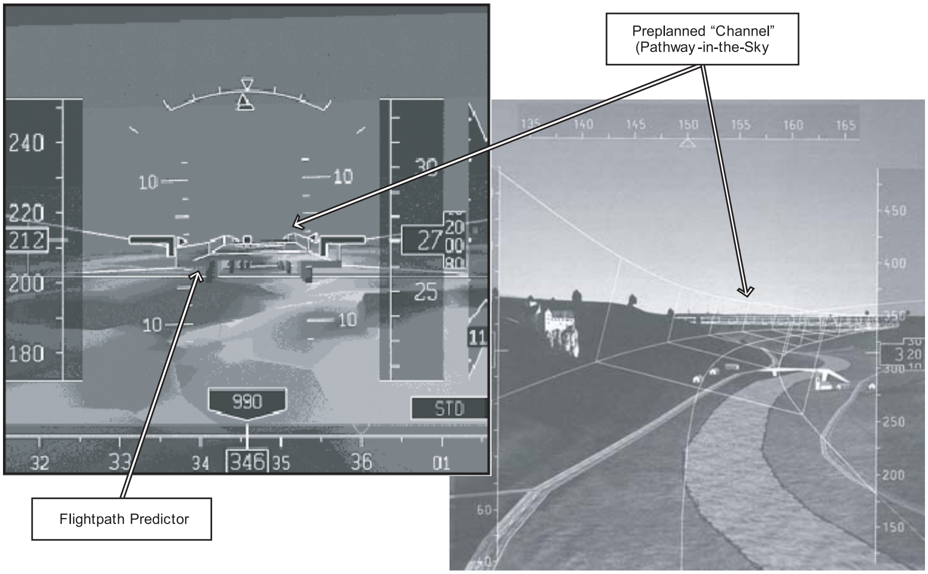


Fig. 7.46 *Four-dimensional primary flight display – VDO Luftfahrtgerate Werk (see colour plate)*

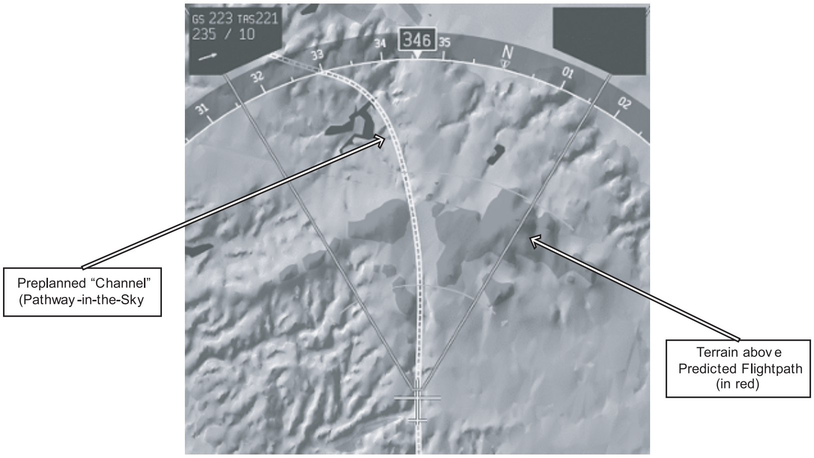


Fig. 7.47 *Four-dimensional navigation display – VDO Luftfahrtgerate Werk (see colour plate section)*

In low-level flight, terrain warning is indicated by a colour change to red of any features with heights above the predicted flight path. This provides an easy and intuitive representation of the actual situation. The channel/predictor combination provides an early warning of any deviation from the intended flight path and supports the pilot during manual approaches, facilitating precise, rational, and intuitive handling of the aircraft.

The experimental four-dimensional navigation display shown in Fig. 7.47 provides orientation of the aircraft with respect to its surroundings from a bird's eye perspective. All flight-safety-related information is displayed and coded in a manner consistent with the primary flight display. The preplanned flight path is shown overlaying the terrain data.

Review of other work in the field includes:

- A 'pathway' system flown by NASA on its Boeing 737 aircraft.
- Graphic perspective displays flown by the University of Technology in Delft, Netherlands, on its Cessna Citation II jet.
- A touch-screen navigation display developed by DASA (now EADS), Germany, which depicts an aircraft flight plan superimposed on a perspective view of the underlying terrain. The design eliminates navigation errors linked to data manipulation using alpha-numeric keypads by enabling the pilot to simply touch the desired waypoint on the screen to make a route change. Since the perspective view of the terrain is constantly underlaid on the display, it is a constant and intuitive reminder to check terrain clearance before executing a route change.

The Head-Up Display

The Head-Up Display (HUD) has long been the accepted primary flight display in military fast jet aircraft. The HUD was first introduced in the 1960s as the successor to the World War II gyro gunsight. In the military fighter the HUD is primarily used to provide targeting information for weapons; the secondary purpose is to provide guidance and flight data so the pilot can maintain complete awareness of the situation with respect to all the flight-critical parameters without having to look inside the flight deck.

The power of the head-up display comes from its collimated image, that is, an image focused at infinity. Symbology written onto the faceplate of a CRT is collimated by an optical system and then projected into the pilot's sightline by a partially reflective combiner so that the symbology appears to the pilot to be superimposed on the outside world. Being collimated, the projected image remains in perfect registration with the outside world irrespective of the pilots' head motion (provided his or her eyes remain within the HUD projection porthole). This is the principle that allows a fighter pilot to aim weapons onto targets. In terms of the human–machine interface, this property is often referred to as 'contact analogue'. Contact analogue symbology is by definition in contact with the real world and by nature intuitive. Of course, HUD contact analogue symbology is not limited to targeting information, but also includes aircraft motion and directional guidance information.

In the context of civil transport aircraft, the benefits of a HUD include:

- Display of conformal (contact analogue) information overlaid on the real world.
- Flight parameter data presented in a relatively small field of view, which improves data cross-checking.
- Improved situational awareness, especially in wind shear and terrain/traffic avoidance manoeuvres.

A typical civil HUD system (Fig. 7.48) comprises similar elements to those described earlier for any display suite architecture, namely a display guidance computer (data collection, display management, and graphics generation), an overhead projector (display device, drive electronics, and projection optics), plus the combiner. In the civil transport aircraft, the HUD is installed overhead and the combiner is attached to the roof, as shown in Fig. 7.49. The combiner can be folded (stowed) when not required.

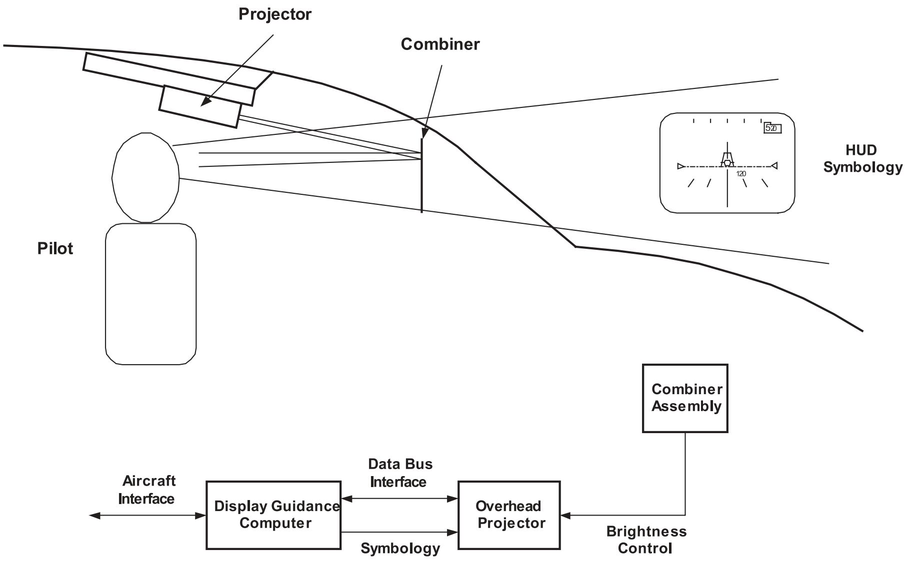


Fig. 7.48 *Civil HUD architecture*

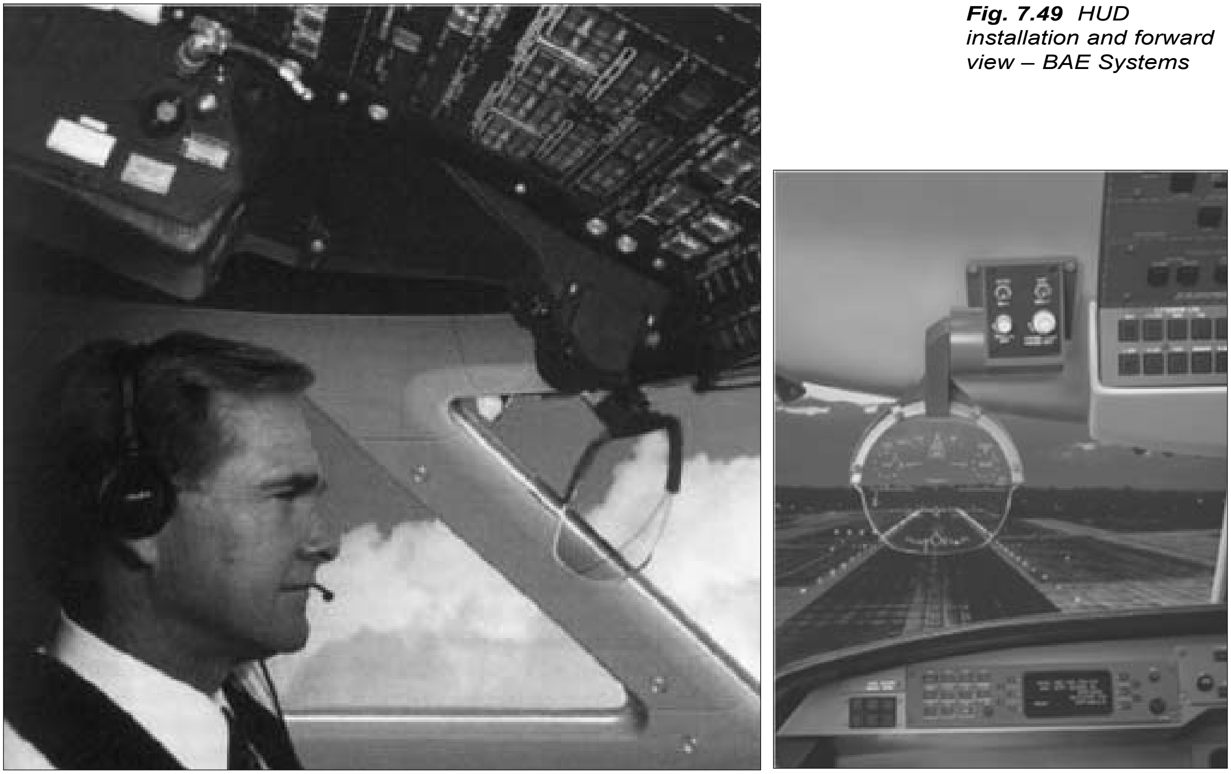


Fig. 7.49 *HUD installation and forward view – BAE Systems*

HUD symbology, shown in Fig. 7.50, provides the pilot with all the flight performance data found head-down (airspeed, altitude, attitude, heading, etc.). Attitude, heading, and directional guidance are contact analogue. Other information is presented in a similar format to that found on the PFD to aid assimilation and transition between the two instruments. For the civil airline pilot the most import feature of the HUD is its ability to display the flight path vector superimposed on the outside world. The flight path vector provides an instantaneous and intuitive indication of the direction of flight. Using the flight path vector in conjunction with other cues on the HUD, the pilot is able to control precisely the aircraft manually and routinely to maintain desired airspeed to within 2 knots, heading to within 2°, altitude to within 50 ft, and pitch to within 0.5°. This degree of control is difficult to achieve using head-down instruments.

During approach and landing, the pilot can simply fly the desired glide path angle and place the flight path vector on the intended touchdown point. The flight path vector in the HUD enables the pilot to 'spot' land the aircraft and provides important cues if drift or sideslip develop prior to touchdown (**21**).

In low-visibility approaches pilots can fly head-up using visual ILS direction cues. Their attention is directed at precisely the right point in space visually to acquire the runway at approach minima. If the minima criteria are not satisfied, they can rapidly and confidently execute a missed approach.

Ground Proximity Warning Systems (GPWSs) are a valuable tool to flight crew to avoid terrain. However, if the crew finds itself in a terrain escape manoeuvre where the terrain is visible, then the HUD provides an unambiguous presentation on whether or

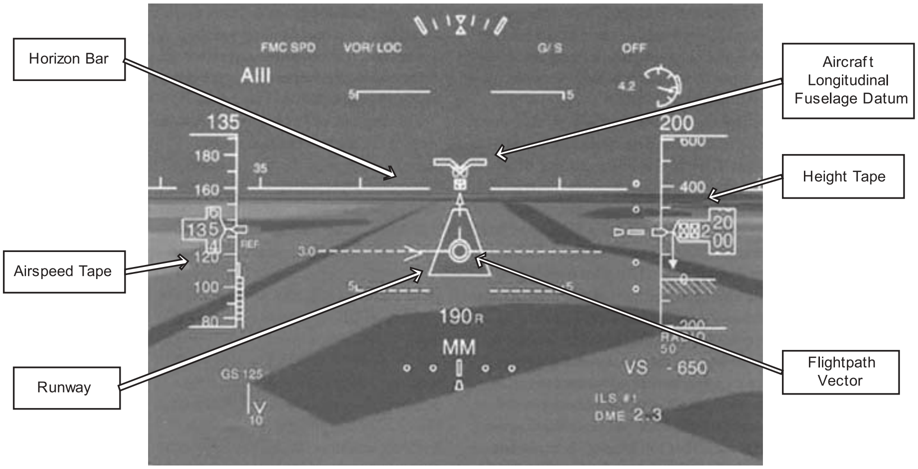


***Fig. 7.50** Typical civil HUD symbology – BAE systems*

not the terrain will be missed. If the flight path vector overlays the terrain, then the aircraft will hit it. The crew must decide on another course of action as opposed to wait and hope.

Traffic Collision and Avoidance Systems (TCASs) provide traffic conflict alerts and escape guidance. With a HUD, both the escape manoeuvre and search for traffic can be accomplished head-up.

Accidents and incidents still occur today on visual landings. Some are too short, some too long, and some not aligned with the runway. Tail strikes can occur on take-off or landing and can cause considerable damage to aircraft with consequent out-of-service time. A tail strike can occur on take-off if the pitch rate is excessive. Landing tail strikes can occur with insufficient approach speed and aggressive, last minute, flight path corrections. Having primary flight data head-up reduces the possibility that a pilot will focus on the touchdown point and allow airspeed to fall or initiate flare too high.

In the future it is probable that additional sensors will be installed in the aircraft to augment the visual acquisition of terrain in poor visibility. These systems are already in military use for the obvious reasons. Systems such as Forward Looking Infra Red (FLIR) and millimetric radar can ‘see’ through fog and rain and present a picture of the outside world on the HUD. The pilot can fly the aircraft ‘visually’ with the aid of these sensors acquiring the real world when the gloom is finally pierced. Experimental approaches have been made using sensors in this way, with much success.

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CHAPTER 8

Navigation

Navigation has been an ever-present component of man's exploitation of the capability of flight. While the principles of navigation have not changed since the early days of sail, the increased speed of flight, particularly with the advent of the jet age, has placed an increased emphasis upon accurate navigation. The increasingly busy skies and rapid technology developments have both emphasized the need for higher-accuracy navigation and the means to accomplish it. Navigation is no longer a matter of getting from A to B safely, it is about doing this in a fuel-efficient manner, keeping to tight airline schedules, and avoiding other air traffic – commercial, general aviation, leisure, and military. This section addresses some of the modern methods of navigation and leads to a later section on the Future Air Navigation System (FANS), also known as Communications, Navigation, Surveillance/Air Transport Management (CNS/ATM).

The main methods of navigation as practised today may be summarized and simplified as follows:

- Classic dead-reckoning navigation using air data, magnetic and Doppler, or LORAN-C.
- Radio navigation using navigation aids – using ground-based radio frequency beacons and airborne receiving and processing equipment.
- Barometric–inertial navigation using a combination of air data and Inertial Navigations (INs) or Doppler.
- Satellite navigation using a Global Navigation Satellite System (GNSS), more usually a Global Positioning System (GPS).
- Multiple-sensor navigation using a combination of all of the above.

Basic navigation

The basic navigation parameters are shown in Fig. 8.1 and may be briefly summarized as follows:

- An aircraft will be flying at a certain height or altitude relative to a barometric datum (barometric altitude) or terrain (radar altitude).
- The aircraft may be moving with velocity components in the aircraft X (V_x), Y (V_y), and Z (V_z) axes. Its speed through the air may be characterized as indicated airspeed (IAS) or Mach Number (M). Its speed relative to the ground is determined by true airspeed (TAS) in still air conditions.
- The aircraft will be flying on a certain heading, but the prevailing wind speed and direction will modify this to the aircraft track. The aircraft track represents the aircraft path across the terrain and will lead to the aircraft's destination or next waypoint. Wind speed and direction will modify the aircraft speed over the ground to groundspeed.
- The aircraft heading will be defined by a bearing to magnetic (compass) north or to true north relating to earth-related geographic coordinates.
- The aircraft will be flying from its present position (defined by latitude and longitude) to a waypoint also characterized by latitude and longitude.
- A series of flight legs – defined by waypoints – will determine the aircraft designated flight path from the departure airfield to the destination airfield.

As has already briefly been described, there are a number of sensors and navigation techniques that may be used solely or in combination to navigate the aircraft throughout the mission.

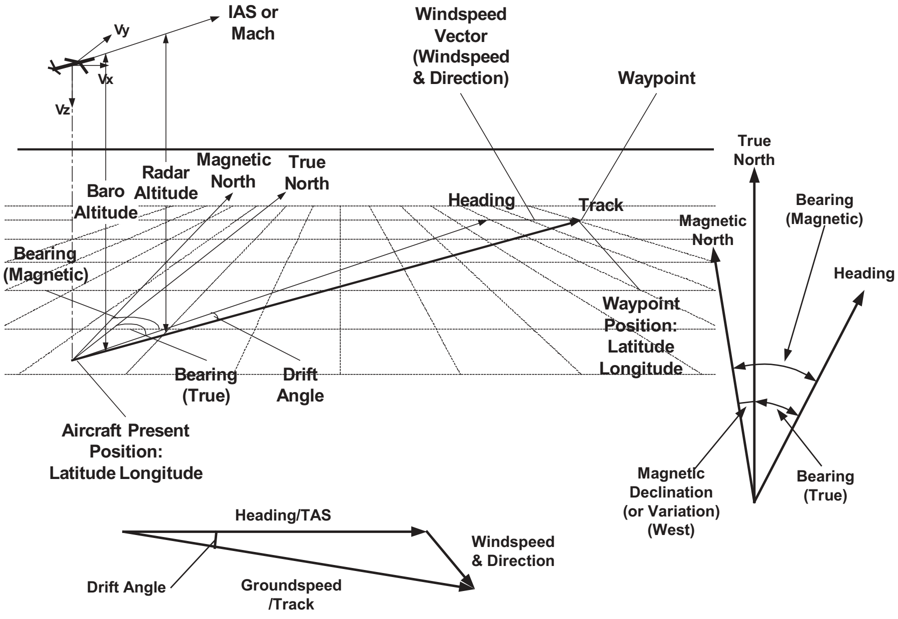


Fig. 8.1 *Basic navigation parameters*

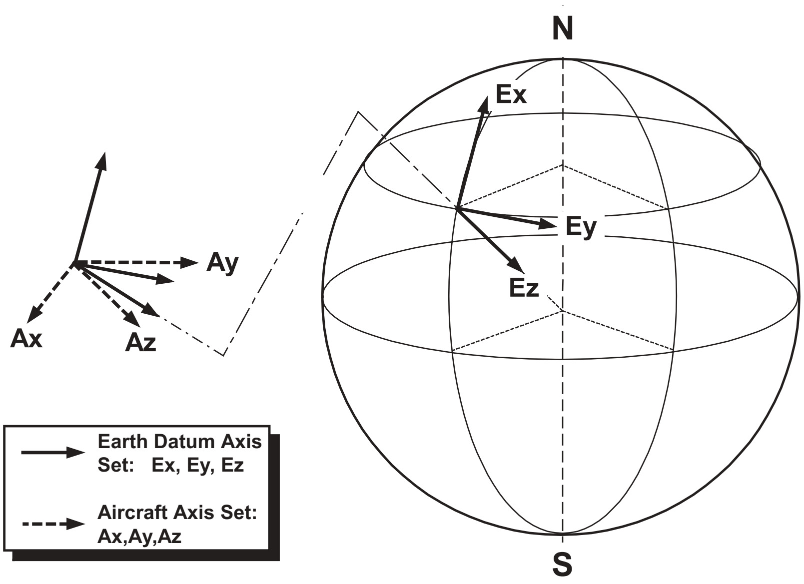


Fig. 8.2 *Earth-related coordinates*

The relationship of the different axis sets is shown in Fig. 8.2. These may be characterized as:

1. Earth datum set. As shown in Fig. 8.2, the Earth axis reference set comprises the orthogonal set *Ex*, *Ey*, *Ez* where:
   - *Ex* represents true north,
   - *Ey* represents east,
   - *Ez* represents the local gravity vector.
2. The orthogonal aircraft axis set. Here:
   - *Ax* is the aircraft longitudinal axis (corresponding to the aircraft heading),
   - *Ay* is the aircraft lateral axis,
   - *Az* is the aircraft vertical axis (corresponding to *Ez*).

For navigation purposes, the accuracy with which the aircraft attitude may be determined is a key variable for Doppler navigation systems in which the Doppler velocity components need to be resolved into aircraft axes. Similarly, attitude is used for IN axis transformations. The aircraft axes in respect of flight control will be discussed in more detail in Chapter 10.

The navigation function therefore performs the task of manoeuvring the aircraft from a known starting point to the intended destination using a variety of sensors and navigation aids.

The classic method of navigation that has been in use for many years is to use a combination of magnetic and inertial directional gyros used together with airspeed information derived from the air data sensors to navigate in accordance with the parameters shown in Fig. 8.1. This is subject to errors in both the heading system and the effects of en route winds which can cause along-track and across-track errors. In the 1930s it was recognized that the use of radio beacons and navigation aids could significantly reduce these errors by providing the flight crew with navigation assistance related to precise points on the ground.

Radio navigation

For many years the primary means of navigation over land, at least in continental Europe and the North American continent, was by means of radio navigation routes defined by VHF OmniRanging/Distance Measuring Equipment (VOR/DME) beacons as shown in Fig. 8.3. By arranging the location of these beacons at major navigation or crossing points, and in some cases airfields, it was possible to construct an entire airway network that could be used by the flight crew to define the aircraft flight from take-off to touchdown. Other radio frequency aids include Distance Measuring Equipment (DME) and Non-Directional Beacons (NDBs).

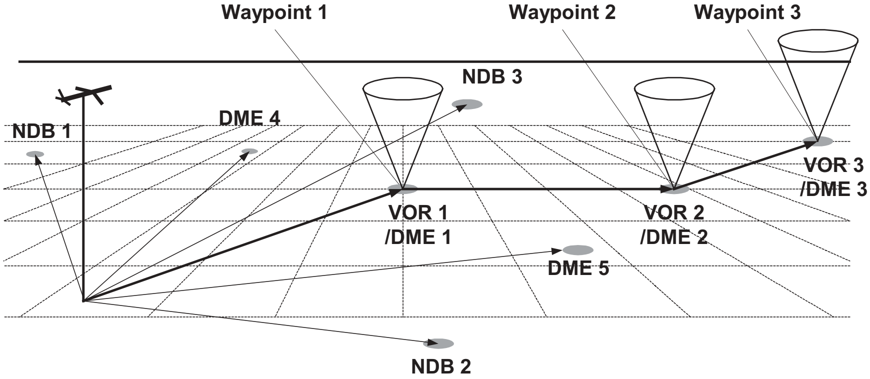


Fig. 8.3 *Radio navigation using VOR, DME and Automatic Direction Finding (ADF)*

Figure 8.3 shows:

- Three VOR/DME beacon pairs: VOR 1/DME 1, VOR 2/DME 2, and VOR 3/DME 3, which define waypoints 1–3. These beacons represent the aircraft's intended waypoints 1, 2, and 3 as it proceeds down the intended flight plan route – most likely an identified airway. When correctly tuned, the VOR/DME pairs successively present the flight crew with bearing to and distance from the next waypoint.
- Off-route DME beacons, DME 4 and DME 5, may be used as additional means to locate the aircraft position by means of the DME fix obtained where the two DME 4 and DME 5 range circles intersect. As will be seen, DME/DME fixes are a key attribute in the modern navigation system.
- Off-route NDB beacons may be used as an additional means of determining the aircraft position by obtaining a cross-fix from the intersection of the bearings from NDB 1 and NDB 2. These bearings are derived using the aircraft ADF system.

Thus, in addition to using navigation information from the 'paired' VOR/DME beacons that define the main navigation route, position fix, cross-fix, range, or bearing information may also be derived from DME or NDB beacons in the vicinity of the planned route by using automatic direction finding techniques. As has already been described in Chapter 6, a major limitation of the radio beacon navigation technique results from line-of-sight propagation limitations at the frequencies at which both VOR and DME operate. As well as the line-of-sight and terrain masking deficiencies, the reliability and accuracy of the radio beacons can also be severely affected by electrical storms. Over longer ranges, LORAN-C could be used.

Owing to the line-of-sight limitations of these radio beacons, these navigation techniques were only usable overland, where the beacon coverage was sufficiently comprehensive, or on close off-shore routes, where the beacons could be relied upon.

An example of radio navigation beacons and the associated airway structure is shown in Fig. 8.4. This example covers an area in the United Kingdom from Stafford in the West Midlands down to Yeovilton in Somerset. This shows a number of different navigation beacons:

- VOR/DME beacons at Brecon (BCN 116.3 MHz) and Honiley (HON 112.9 MHz).
- VOR/TAC beacons at Yeovilton (VLN channel 47), Boscombe Down (BDN

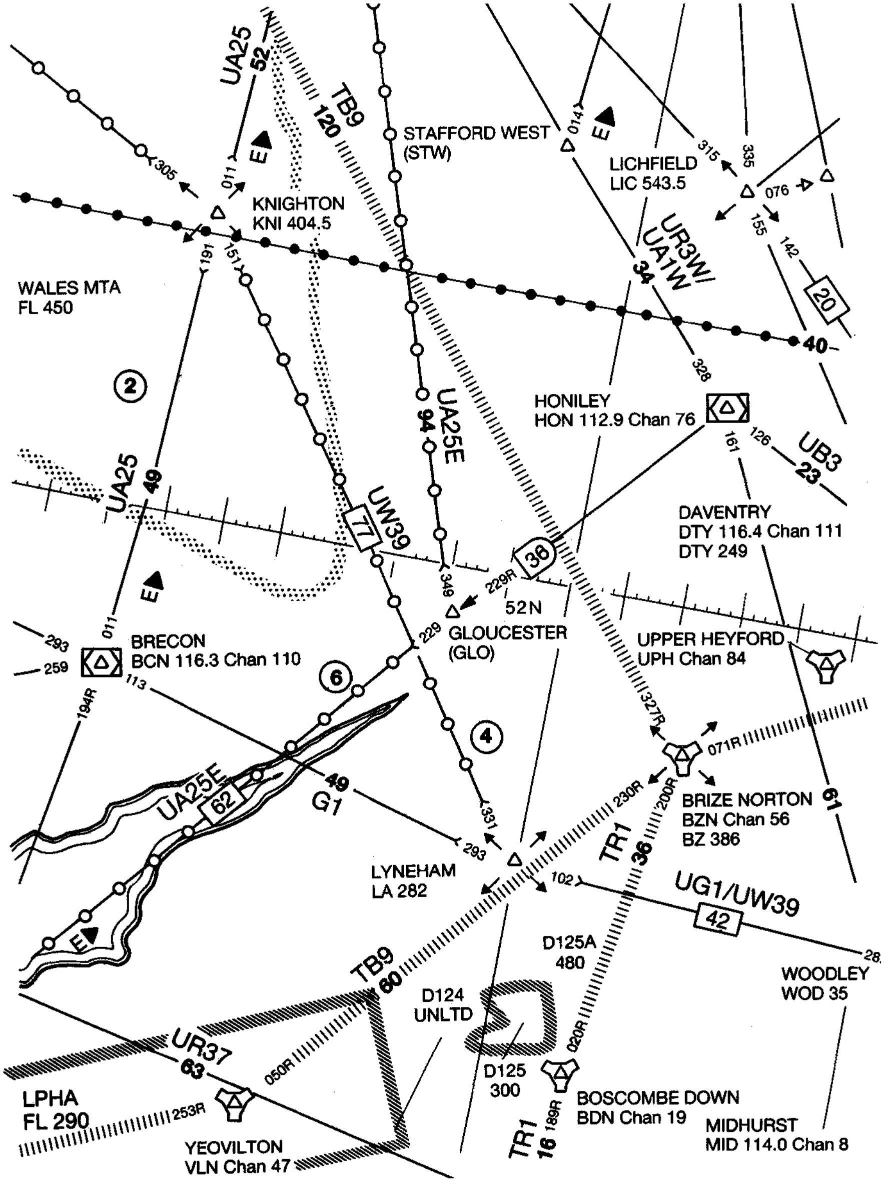


Fig. 8.4 *Example of radio navigation beacon infrastructure*

channel 19), Brize Norton (BZN channel 56), and Upper Heyford (UPH channel 84).

- NDB beacons at a variety of locations including Lichfield (LIC 543.3 kHz), Knighton (KNI 404.5 kHz), and Lyneham (LA 282 kHz).

Oceanic crossings

In 1969 the requirements were already specified for self-contained long-range navigation by reference (**1**). The appropriate document specified that self-contained navigation systems should be capable of maintaining a maximum error of ±20 nautical miles across track and ±25 nautical miles along track for 95 per cent of the flights completed. Two systems were addressed in the specification: one using Doppler radar and the other using an Inertial Navigation System (INS)

In June 1977, the North Atlantic (NAT) Minimum Navigation Performance Specifications (MNPS) were altered to reflect the improved navigation sensors [see reference (**2**)]. This defined the separation requirements for long-range navigation over the North Atlantic. The lateral separation was reduced from 120 to 60 nautical miles while retaining the previous vertical separation of 2000 ft. Statistical limits were specified as to how long an aircraft was allowed to spend 30 nautical miles off-track and between 50 and 70 nautical miles miles off-track – the latter actually representing an overlap with an adjacent track. The standard deviation of lateral track errors was specified as 6.3 nautical miles.

The Doppler radar system was specified as being an acceptable navigation means applying within certain geographical boundaries. Eastern and western entry points or ‘gateways’ were specified as entry and departure points into and out of the North Atlantic area. These gateways were identified as a number of specific named NDB or VOR beacons on both sides of the ocean. The North Atlantic transit area was specified as being the oceanic area bounded by the eastern and western gateways and lying between the latitude of 35ºN and 65ºN. By the standards of the allowable navigation routes available to today’s aviators, this represented a very restricted envelope.

The aircraft equipment requirements were also carefully specified:

- Dual Doppler and computer systems.
- Dual polar path compasses.
- ADF.
- VOR.
- One LORAN receiver capable of being operated from either pilot’s station.

Inertial navigation

The availability of INS to the civil aviation community during the late 1960s added another dimension to the navigation equation. Now, flight crew were able to navigate by autonomous means using an on-board INS with inertial sensors. By aligning the platform to Earth-referenced coordinates and present position during initialization, it was now possible to fly for long distances without relying upon VOR/DME beacons overland or hyperbolic navigation systems elsewhere. Waypoints could be specified in terms of latitude and longitude as arbitrary points on the globe, more suited to the aircraft’s intended flight path rather than a specific geographic feature or point in a radio beacon network (see Fig. 8.5).

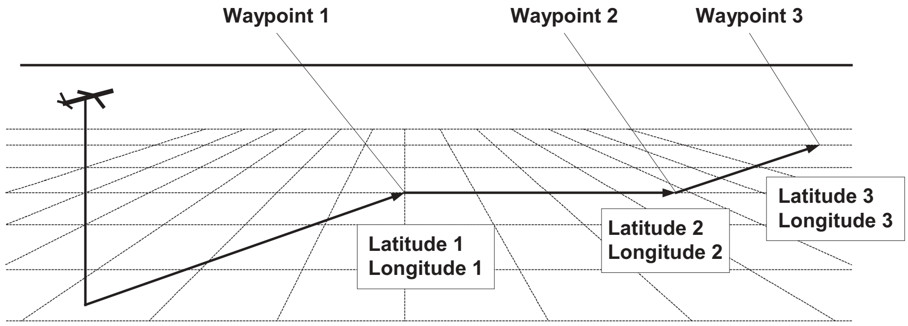


Fig. 8.5 *Principles of inertial navigation*

The specifications in force at this time also offered an INS solution to North Atlantic crossings as well as the dual-Doppler solution previously described. The inertial solution required serviceable dual INS and associated computers to be able to undertake the crossing. There were also limitations on the latitudes at which the ground alignment could be performed – 76º north or south – as attaining satisfactory alignment becomes progressively more difficult as the INS approaches the poles.

Reference (**3**) sets out requirements for operating an INS as sole means of navigation for a significant portion of the flight. These requirements may be summarized as:

- The ability to provide the following functions:
  - valid ground alignment at all latitudes appropriate for the intended use of the INS;
  - the display of alignment status to the crew;
  - provision of the present position of the aircraft in suitable coordinates, usually latitude from +90º (north) to -90º (south) and longitude from +180º (east) to -180º (west);
  - provision of information on destinations or waypoints;
  - provision of data to acquire and maintain the desired track and the ability to determine deviation from the desired track (across-track error);
  - provision of information needed to determine the Estimated Time of Arrival (ETA).
- The ability to comply with the following requirements:
  - ±20 nautical miles across track and ±25 nautical miles along track in accordance with reference (**1**);
  - the ability to maintain this accuracy on a 95 per cent probability basis at representative speeds and altitudes and over the desired latitude range;
  - the ability to compare the INS position with visual fixes or by using LORAN, TACAN, VOR, DME, or ground radar (air traffic control).
- The provision of a memory or in-flight alignment means. Alternatively, the provision of a separate electrical power source – usually a dedicated stand-alone battery – able to support the INS with full capability for at least 5 min in the event of an interruption of the normal power supply.

For reasons of availability and accuracy, systems were developed with dual and triple INS installations. A typical triple-INS installation is presented in Fig. 8.6, showing three INS units integrated with the other major systems units. This type of system would be representative of an INS installation before the availability of satellite sensors in the 1980s. By this time the gimballed IN platform would have been replaced by a more reliable strapdown system similar to the Litton LTN-92 system described in Chapter 5.

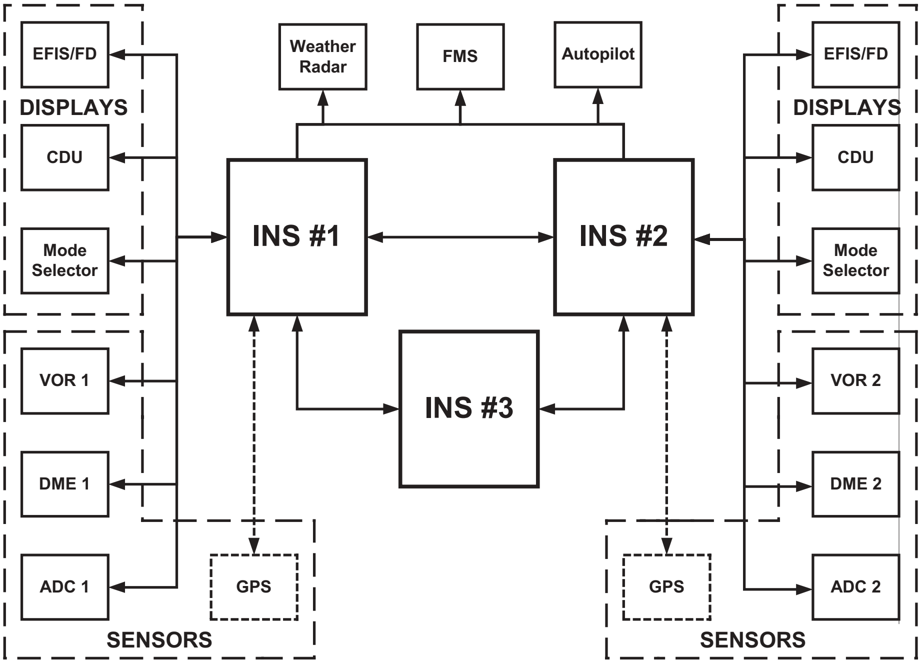
This integrated system comprised the following units:

- Dual sensors:
  - VOR for bearing information,
  - DME for range information,
  - Air Data Computer (ADC) for air data,
  - Provision for a dual GPS interface.
- Controls and displays:
  - Control and Display Unit (CDU),
  - Electronic Flight Instrument System/Flight Director (EFIS/FD),
  - mode selector unit.
- Other major systems receiving INS data for stabilization or computation:
  - Weather radar,
  - Flight Management System (FMS),
  - Autopilot.

The weight of this system comprising three LTN-92 platforms with back-up battery power supplies, two CDUs, and two mode selector units was in the region of 234 lb.

By integrating the air data information with the inertially derived flight information, the best features of barometric and inertial systems can be combined. Means of taking external fixes were evolved so that longer-term inaccuracies could be corrected by updating the INS position during long flights. Some fighter aircraft systems such as Tornado also added a Doppler radar so that Doppler-derived data could be included in

Fig. 8.6 *Typical triple INS system*



the navigation process. The availability of on-board digital computers enabled statistical Kalman filtering techniques to be used to calculate the best estimated position using all of the sensors available. The fundamental problem with the INS is the long-term and progressive accrual of navigation error as the flight proceeds.

Air Data and Inertial Reference Systems (ADIRS)

The system illustrated in Fig. 8.6. utilizes stand-alone ADCs, but, as already described in Chapter 5, the use of ADCs was superseded in many new systems by the introduction of Air Data Modules (ADMs) in the late 1980s. The new integrated ADIRS developed in the early 1980s combined the computation for air data and inertial parameters in one multichannel unit. Taking the B777 as an example, the primary unit is an Air Data and Inertial Reference Unit (ADIRU) which provides the main source of air data and inertial information. This unit is supported by an Attitude and Heading Reference System (AHRS) which on the B777 is called the Secondary Attitude Air data Reference Unit (SAARU). This provides secondary attitude and air data information should the primary source, the ADIRU, become totally unusable.

The B777 ADIRU is shown in Fig. 8.7. There are six Ring Laser Gyros (RLGs) and six accelerometers included in the unit. It can be seen that both sets of sensors are arranged in a hexad-skew redundant set in relation to an orthogonal axis set. This means that, by resolving the output of each of the six sensors in the direction of the axis set, each sensor is able to measure an element of the relevant inertial parameter – body rate or acceleration – in each axis. This provides a redundant multichannel sensor set with the prospect of achieving higher levels of accuracy by scaling and combining sensor outputs. Additionally, the output of erroneous sensors may be detected and 'voted out' by the remaining good sensors. This multiple-sensor arrangement greatly

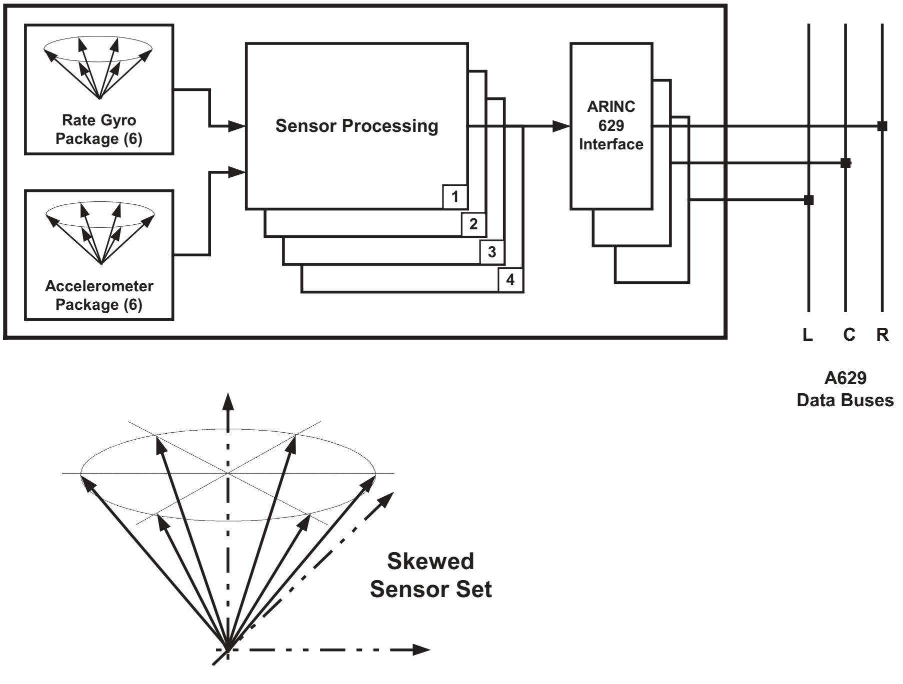


Fig. 8.7 *B777 ADIRU*

increases the availability of the ADIRU as the performance of the unit will degrade gracefully following the failure of one or more sensors. The ADIRU may still be used with an acceptable level of degradation until a replacement unit is available or the aircraft returns to base, and only has to be replaced following the second failure of a like sensor (e.g. second rate or accelerometer sensor). By contrast, the failure of a sensor in an earlier three-sensor, orthogonally oriented set would lead to a sudden loss of the INS.

Coupled with the dual-hexagonal sensor arrangement, there are four independent lanes of processing within the ADIRU and three interfaces with the A629 data buses (L – left, C – centre, R – right). Each processor lane computes a wide range of navigation parameters:

North velocity
East velocity
Groundspeed
Latitude
Longitude
Wind speed
Wind direction
True heading
Magnetic heading
True track angle
Magnetic track angle
Drift angle
Flight path angle
Inertial altitude
Computed airspeed
Mach number
Roll attitude
Pitch attitude
Track angle rate
Corrected Angle of Attack (AoA)
Altitude rate
Altitude
Total air temperature (TAT)
Static air temperature (SAT)
True airspeed
Static pressure (corrected)
Impact pressure
Corrected computed airspeed
Corrected Mach number
Corrected total pressure
Corrected static pressure
C of G longitudinal acceleration
C of G lateral acceleration
C of G normal acceleration
Flight path acceleration
Vertical speed
Roll, pitch, and yaw attitude rates
Body pitch, roll, and yaw rates
Body longitudinal, lateral, and normal accelerations

Finally, the ADIRU interfaces with the remainder of the aircraft systems by means of triple flight control ARINC 629 digital data buses: left, centre, and right. The unit is provided with electrical power from a number of independent sources.

In the B777 ADIRS the secondary unit is the SAARU which is depicted in Fig. 8.8. The SAARU also has a degree of in-built redundancy, but not to the same extent as the ADIRU. The rate gyro and accelerometer sensors are arranged in a tetrad-skew redundant set, and the sensor processing is dual redundant. As for the ADIRU, there are three ARINC 629 data bus interfaces. The SAARU provides a comprehensive list of air data and inertial parameters to the aircraft systems, though not as extensive a list as the ADIRU.

Figure 8.9 shows the entire B777 ADIRS. The main computation is provided by the ADIRU and SAARU, as already described. There are six ADMs which interface with the left, right, and centre pitot and static probes and with the ADIRU and SAARU via the ARINC 629 buses. The Airplane Information Management System (AIMS) left and right cabinets interface directly with the left and right AoA and TAT sensors. The

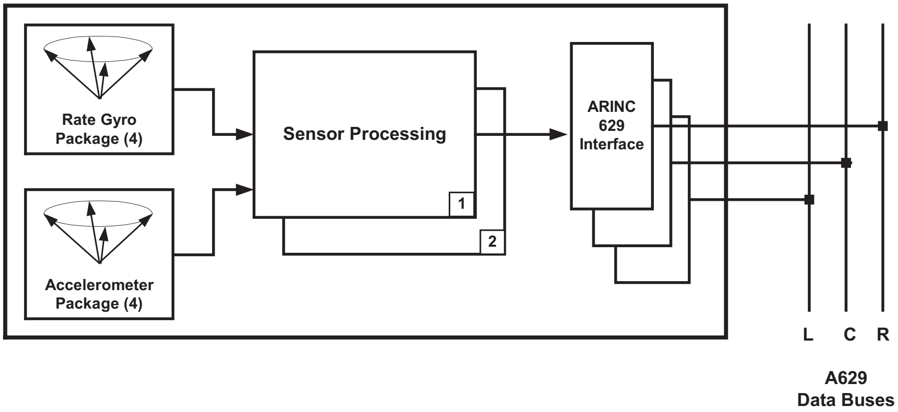


Fig. 8.8 *B777 Secondary Attitude and Air Data Reference Unit*

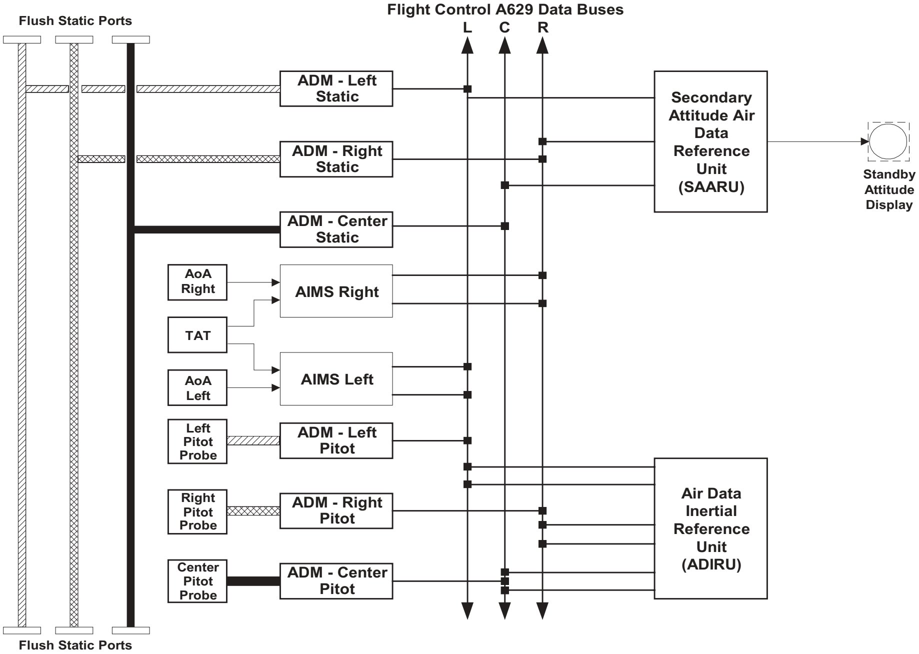


Fig. 8.9 *B777 air data and inertial reference system*

SAARU provides attitude information to the standby attitude display. The AIMS cabinets also receive navigation data from the ADIRU and SAARU and perform the flight management function. To give an idea of the pace of development in air data and INS sensor and computing technology, the multichannel B777 ADIRU and SAARU (10 MCUs) combined have a total weight in the region of 100 lb. This compares with the 234 lb of the triple stand-alone INS unit described earlier and illustrated in Fig. 8.6.

The B777 represents the most integrated solution available today, and the AIMS will be more fully described in Chapter 11. The Airbus approach is very similar, although ARINC 429 data buses are used. In the Airbus approach, many functions, including flight management are integrated into a unit called the Flight Management Guidance and Envelope Computer (FMGEC). On a number of current aircraft a 4MCU ADIRU is used – albeit this is not fault tolerant to the same extent as the B777 unit. These aircraft include the Fokker 100, a number of business jets, and the Airbus A320 and A330/A340 families.

Typical performance details associated with the latest-generation ADIRUs are given in Table 8.1.

Table 8.1 Performance data for the latest-generation ADIRUs

Parameter	Characteristic
MTBF	12 000 operating hours
NAV accuracy	2 nautical miles per radial, 95% probability
Velocity accuracy	8 knots, 2 sigma per axis
Attitude accuracy	0.05º, 2 sigma
Heading accuracy	0.4º, 2 sigma
Alignment time:	
above 70º latitude	15 min
between 60º and 70º latitude	10 min
zero latitude	Decreasing to less than 2.5 min at zero latitude

Satellite navigation

The foregoing techniques were prevalent from the 1960s through until the 1990s when satellite navigation became commonly available. The employment of Global Navigation Satellite Systems (GNSSs), to use the generic name, offers a cheap and accurate navigational means to anyone possessing a suitable receiver. Although the former Soviet Union developed a system called GLONASS, it is the US Global Positioning System (GPS) that is the most widely used. The principles of satellite navigation using GPS have already been described in Chapter 6.

GPS receivers may be provided for the airborne equipment in a number of ways:

- Stand-alone GPS receivers, most likely to be used for GPS upgrades to an existing system.
- Multichannel (typically 12-channel) GNSS receivers – the B777 utilizes this approach.
- GPS receivers integrated into a multifunction receiver unit called a MultiMode Receiver (MMR) where the GPS receiver function is integrated into one LRU along with VOR and ILS receivers.

GPS error

GPS sources of error without selective availability are commonly ascribed to the causes listed in Table 8.2, with a typical error budget mentioned in each case.

Table 8.2 Causes of GPS errors and typical error budgets

Error source	Typical value (m)
Selective availability enabled	±100
Atmospheric errors	±5
Horizontal dilution of precision	±5
Ephemeris error	±2
Satellite error	±2
Receiver error	±1
Root mean square error:	
Selective availability enabled	±100
Selective availability disabled [present situation depending, commissioning of Wide Area Augmentation System (WAAS) and Local Area Augmentation System (LAAS)]	**±7.7**

Note: The precise errors will be determined by a number of factors; the figures quoted are typical and may be lower in specific implementations.

Differential GPS

One way of overcoming the problems of selective availability is to employ a technique called Differential GPS (DGPS). Differential techniques involve the transmission of a corrected message that is derived from users located on the ground. The correction information is sent to the user who can apply the corrections and reduce the satellite ranging error. The two main techniques are:

1. Local area DGPS. The corrections are derived locally at a ground reference site. As the position of the site is accurately known, the satellite inaccuracies can be determined and transmitted locally to the user, in this case by line-of-sight VHF data link. The local area DGPS system under development in the United States is called the Local Area Augmentation System (LAAS) and is described below.
2. Wide area DGPS. The wide area correction technique involves networks of data collection ground stations. Information is collected at several ground stations which are usually located more than 500 miles apart. The correction information derived by each station is transmitted to a central location where the satellite corrections are determined. Corrections are sent to the user by geostationary satellites or other appropriate means. The system being developed in the United States is called the Wide Area Augmentation System (WAAS) and is outlined below.

Note that differential techniques may be applied to any satellite system. For GPS the basic accuracy without selective availability is about ±100 m, as opposed to ±8 m when the full system is available. The DGPS developments underway in the United States are intended to improve the accuracy available to civil users.

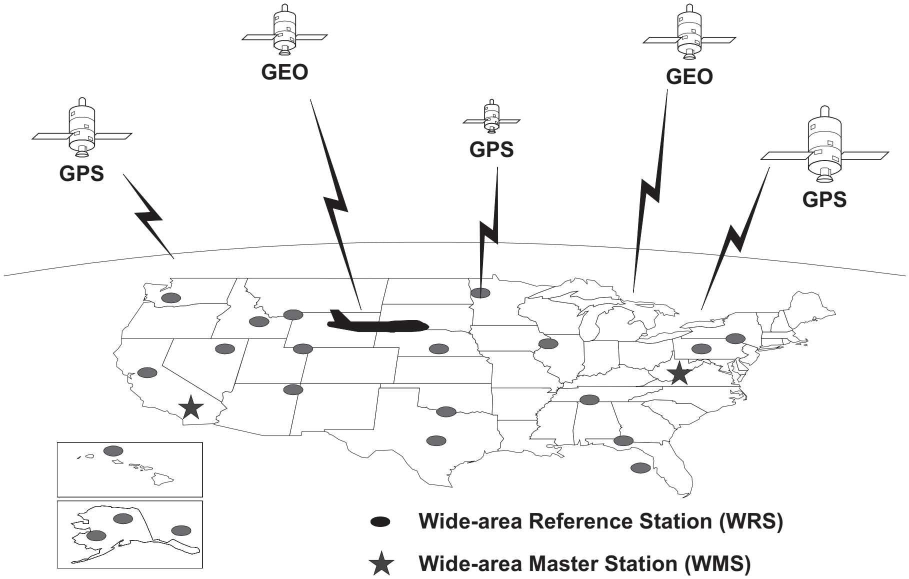
Wide Area Augmentation System

The operation of WAAS, shown in Fig. 8.10, is described as follows:

- WAAS is a safety-critical system that augments basic GPS and will be deployed in the contiguous United States, Hawaii, Alaska, and parts of Canada.
- WAAS has multiple wide area reference stations that are precisely surveyed and monitor the outputs from the GPS constellation.
- These reference stations are linked to wide area master stations where corrections are calculated and the system integrity assessed. Correction messages are uplinked to Geostationary Earth Orbit (GEO) satellites that transmit the corrected data on the communications L1 band to aircraft flying within the WAAS area of coverage. Effectively, the GEO satellites act as surrogate GPS satellites.
- WAAS improves the GPS accuracy to around ±7 m, which is a considerable improvement on the 'raw' signal. This level of accuracy is sufficient for Cat I approach guidance.

Some problems were experienced in initial system tests during 2000. Commissioning of the WAAS requires extensive testing to assure integrity levels, accuracy, etc., and it now appears unlikely to become operational before 2003 at the earliest in some parts of the United States.

Fig. 8.10 *Wide Area Augmentation System*



Local Area Augmentation System

The operation of LAAS, as shown in Fig. 8.11, is described below:

- LAAS is intended to complement WAAS but at a local level.
- LAAS works on similar principles except that local reference stations transmit correction data direct to user aircraft on VHF. As such, the LAAS coverage is limited by VHF line-of-sight and terrain masking limitations.
- LAAS improves the GPS accuracy to about ±1 m, close to the higher GPS level of accuracy. This level of accuracy is sufficient to permit Cat II and Cat III approaches which are described more fully in Chapter 9.

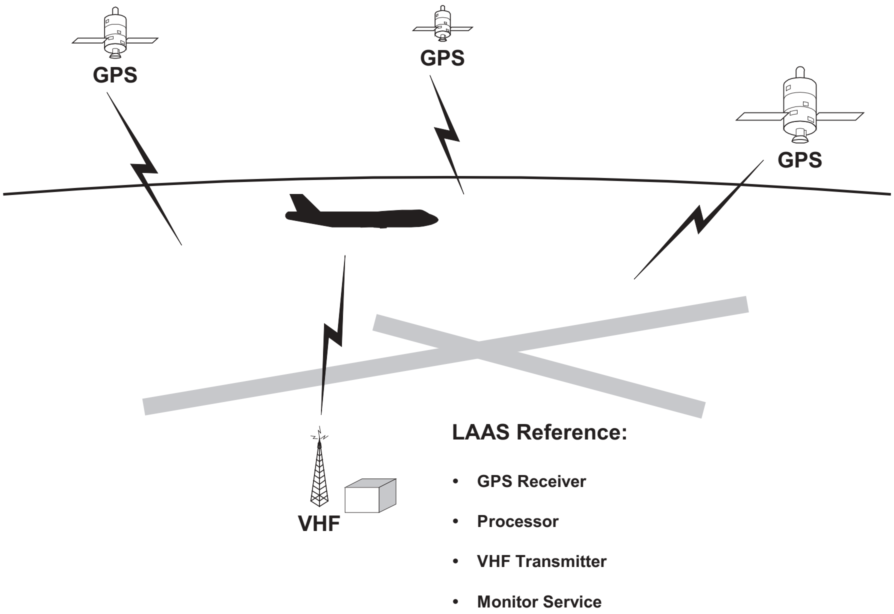


Fig. 8.11 *Local area augmentation system*

Implementation is expected to begin in 2003, with final deployment in 2006, though these time-scales are apt to slip, as with the implementation of WAAS. According to present plans it is expected that LAAS will be deployed at up to 143 airfields throughout the United States. It was the anticipation of LAAS implementation that caused the United States to modify its stance on the implementation of MLS as an approach aid successor to ILS, the space-based GPS system being seen as more flexible than ground-based MLS.

Integrated navigation

Integrated navigation, as the name suggests, employs all of the features and systems described so far. An integrated navigation solution using a multisensor approach blends the performance of all the navigation techniques already described, together with GPS, to form a totally integrated system. Such an integrated system is a precursor to the introduction of the advanced navigation capabilities that will comprise the Future Air Navigation System (FANS). FANS is designed to make more efficient use of the existing airspace such that future air traffic increases may be accommodated. Some elements of FANS have already been implemented, others will take several years to attain maturity. A key prerequisite to achieving a multisensor system is the installation of a high-grade Flight Management System (FMS) to perform the integration of all the necessary functions and provide a suitable interface with the flight crew.

Sensor usage – phases of flight

When assessing which navigation sensors to use for various phases of flight, the navigation accuracy, equipment availability and reliability, and operational constraints all need to be taken into account. The advent of GPS, with its worldwide coverage at high levels of accuracy, has given a tremendous impetus to the navigation capabilities of modern aircraft. However, system integrity concerns have meant that the certification authorities have stopped short of relying solely on GPS.

Bearing in mind physical and radio propagation factors, and the relative traffic densities for various phases of flight, a number of requirements are specified for the use of GPS. Reference (**4**) specifies the considerations that apply for the use of GPS as a sole or supplementary method of navigation.

These considerations apply for the following phases of flight:

1. Oceanic en route. Operation over long oceanic routes means that the aircraft will be denied the availability of most of the line-of-sight radio navigation aids such as NDB, VOR, TACAN, etc. LORAN-C may be available in some circumstances. The aircraft will need to depend upon an approved primary long-range method of navigation. For most modern transport aircraft, that means equipping the aircraft with a dual- or triple-channel INS or ADIRS. Supplementary means such as GPS may be used to update the primary method of navigation. Aircraft using GPS under Instrument Flight Rules (IFR) must be equipped with another approved long-range navigation system: GPS is not certified as a primary and sole means of navigation. Certain categories of GPS equipment (see Table 8.3) may be used as one of the approved long-range navigation means where two systems are required. The availability of a functioning Receiver Autonomous Integrity Monitor (RAIM) capability is also important in view of the impact that this has upon GPS integrity. Providing RAIM is available the flight crew need not actively monitor the alternative long-range navigation system.
2. Domestic en route. Once overland, most of the conventional navigation aids may be available, unless the aircraft is transiting a wilderness area such as Siberia. For the most part, NDB, VOR, TACAN, and LORAN-C will be operational and available to supplement GPS. These ground-based systems do not have to be used to monitor GPS unless RAIM failure occurs. Within the United States, Alaska, Hawaii, and surrounding coastal waters, IFR operation may be met with independent NDB, VOR, TACAN, or LORAN-C equipment. This may not necessarily be the case outside the US National Airspace System (NAS).
3. Terminal. GPS IFR operations for the terminal phases of flight should be conducted as for normal area navigation (RNAV) operations using the standard procedures:
   - Standard Instrument Departures (SIDs),
   - Standard Terminal Arrival Routes (STARs),
   - Standard Instrument Approach Procedure (SIAP) .

Table 8.3 GPS classes of equipment (TSO-129a)

Equipment class	RAIM	Integrated navigation system to provide RAIM	Oceanic	En route	Terminal	Non-precision approach capable
Class A – GPS sensor and navigation capability						
A1	yes		yes	yes	yes	yes
A2	yes		yes	yes	yes	no
Class B – GPS sensor data to an integrated navigation system (e.g. FMS, multisensor navigation system, etc.)						
B1	yes		yes	yes	yes	yes
B2	yes		yes	yes	yes	no
B3		yes	yes	yes	yes	yes
B4		yes	yes	yes	yes	no
Class C – GPS sensor data to an integrated system (as in Class B) which provides enhanced guidance to autopilot or flight director to reduce flight technical errors						
C1	yes		yes	yes	yes	yes
C2	yes		yes	yes	yes	no
C3		yes	yes	yes	yes	yes
C4		yes	yes	yes	yes	no

The normal ground-based equipment appropriate to the phase of flight must be available, but as before, they do not need to be used to monitor GPS unless RAIM fails.

4. Approach. In the United States an approach overlay programme has been introduced by the FAA to facilitate the introduction of instrument approaches using GPS. The key features of the GPS overlay programme are described below.

GPS overlay programme

The GPS overlay programme allows pilots to use GPS equipment to fly existing VOR, VOR/DME, NDB, NDB/DME, TACAN, and RNAV non-precision instrument approach procedures. This facility only applies in US airspace and was introduced in February 1994. The approach aid appropriate to the type of approach being flown must be available for use, but need not be monitored provided RAIM is available. In April 1994, 'Phase III' approaches introduced the first GPS-specific approaches with GPS specifically included in the title. For these approaches the traditional avionics need not be available – either ground-based or airborne equipment – provided RAIM is available. For aircraft fitted with GPS without a RAIM capability, these navigation aids must be available.

Categories of GPS receiver

The different types of GPS are mandated in reference (**5**), Technical Standing Order (TSO) C-129a. This categorizes the different GPS receivers by three major classes:

1. Class A. This equipment incorporates a GPS receiver and the navigation capability to support it.
2. Class B. This consists of GPS equipment providing data to an integrated navigation system such as an FMS or an integrated multisensor navigation system.
3. Class C. This includes equipment comprising GPS sensors which provide data to an autopilot or flight director in order to reduced flight technical errors.

This classification therefore categorizes the GPS equipment types according to function. TSO C-129a specifies which class of equipment may be used for the typical flight phases described above. It also specifies whether the RAIM function is to be provided by the GPS or the integrated system. Table 8.3 details the provisions of TSO C-129a.

Flight Management System (FMS)

It is clear from the foregoing description of the aircraft navigation functions that navigation is a complex task and becoming more so all the while. FMS functionality has increased rapidly over the last decade, and many more enhancements are in prospect as the future features required by FANS are added. A typical FMS will embrace dual computers and dual Multifunction Control and Display Units (MCDUs), as shown in Fig. 8.12. This diagram is key to depicting the integration of the navigation functions described above. Inputs, usually dual for reasons of availability and integrity, are shown on the left. These are:

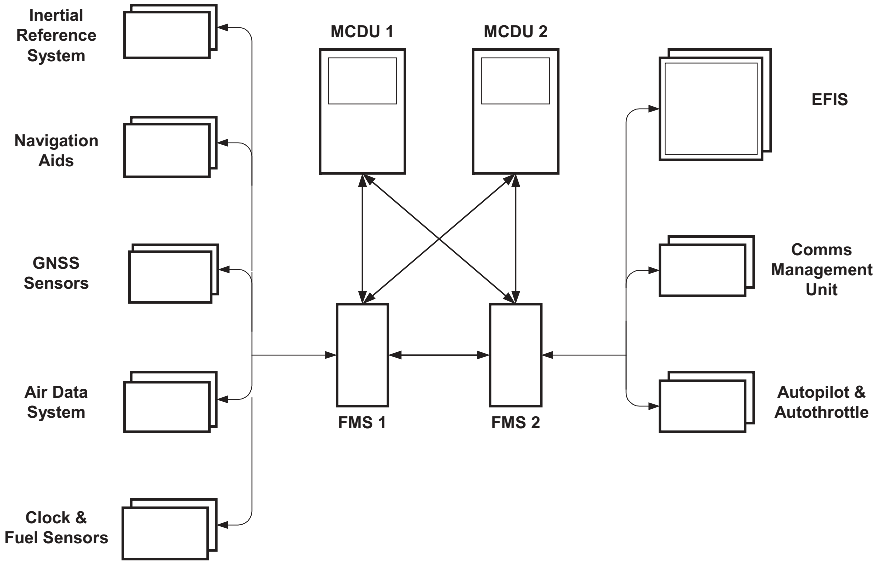


Fig. 8.12 *Typical Flight Management System (FMS)*

- Dual INS/IRS.
- Dual navigation sensors: VOR/DME, DME/DME, etc.
- Dual GNSS sensors – usually GPS.
- Dual air data sensors.
- Dual inputs from on-board sensors relating to fuel on-board and time.

These inputs are used by the FMS to perform the necessary navigation calculations and provide information to the flight crew via a range of display units:

- Electronic flight instrument system (EFIS).
- Communications control system.
- Interface with the autopilot/flight director system to provide the flight crew with flight direction or automatic flight control in a number of predefined modes (see Chapter 9).

The FMS/crew interface is shown in Fig. 8.13. The key interface with the flight crew is via the following displays:

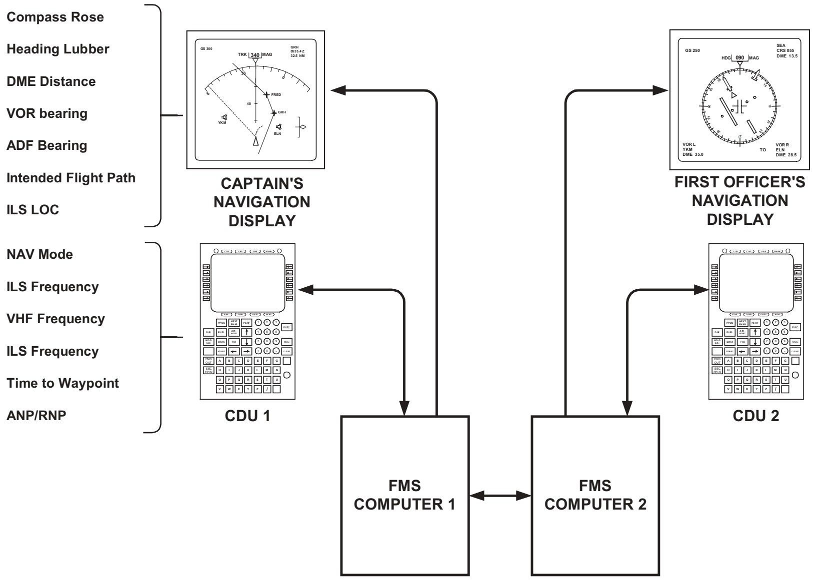
- Captain and first officer's Navigation Displays (ND), part of the EFIS. The navigation displays may show information in a variety of different ways.
- Control and display units 1 and 2, part of the FMS. The CDUs both display information and act as a means for the flight crew manually to enter data.

The FMS computers perform all the necessary computations and show the appropriate navigation parameters on the appropriate display.

The navigation displayS show the navigation and steering information necessary to fly the intended route. These are colour displays and can operate in a number of different formats, depending upon the phase of flight. These are:

1. Expanded approach mode. This display shows the selected runway heading and the lateral deviation from the runway centre-line. The expanded approach mode

Fig 8.13 *FMS control and display interface*



displays 80º (±40º) of the compass rose, with the aircraft symbol and localizer deviation bar at the bottom. The display is oriented with heading at the 12 o'clock position. The glide slope deviation is shown at the right of the display (Fig. 8.14, left).

2. Centre approach mode. The centre approach mode shows 360º of the compass rose, with the aircraft symbol in the centre. Otherwise, the display is heading oriented, as before, and the ILS deviation symbology is the same as for the expanded mode (Fig. 8.14, right).
3. Expanded VOR mode. This display shows the VOR course and lateral deviation from that course. The expanded VOR mode displays 80º of the compass rose, with the aircraft symbol and the lateral deviation bar at the bottom. The display is oriented with heading at the 12 o'clock position (Fig. 8.15, left). The VOR information is only portrayed in this manner if the VOR is manually tuned. The VOR beacon details show in the lower corners and a bearing pointer on the compass card when selected on the EFIS panel.
4. Centre VOR mode. The Centre VOR mode shows 360º of the compass rose, with the aircraft symbol and lateral deviation bar in the centre. As before the heading is oriented to the 12 o'clock position (Fig. 8.15, right).
5. Expanded map mode. This display shows the portion of the flight plan within the selected range (up to 640 nautical miles). The expanded portion of the map displays 80º of the compass rose, with the aircraft symbol at the bottom. The display is oriented with track at the 12 o'clock position. The display shows elements of the flight plan indicating intended track between waypoints and also displays the relevant radio beacons – in this example VOR and TACAN (Fig. 8.16 left).
6. Centre map mode. This shows 360º of the compass rose, with the aircraft symbol in the centre. The display is track oriented, as for the expanded map mode and also shows elements of the flight plan track and relevant beacons (Fig. 8.16. right).

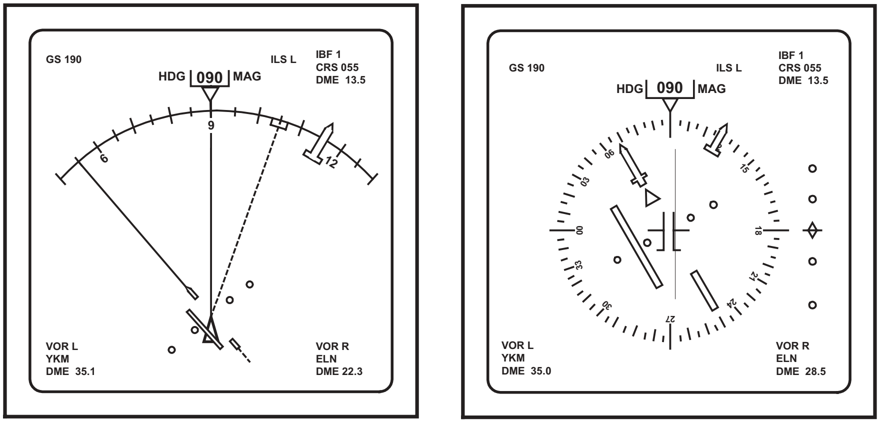


Fig 8.14 *Approach displays – expanded and centre modes*

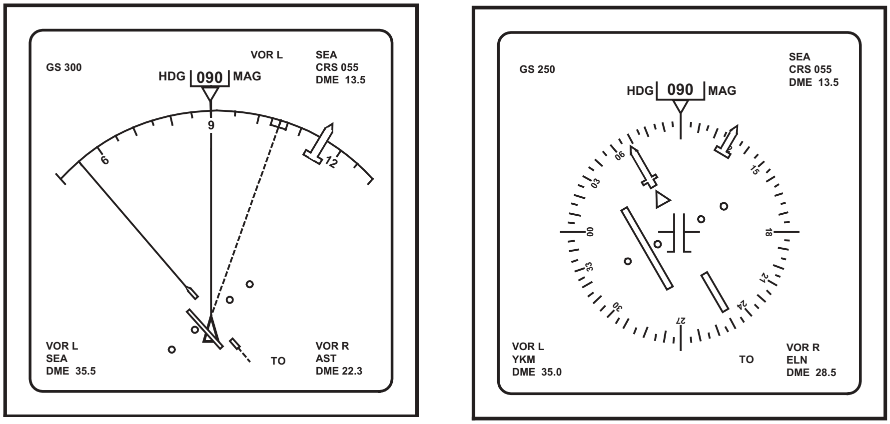


Fig. 8.15 *VOR displays – expanded and centre modes*

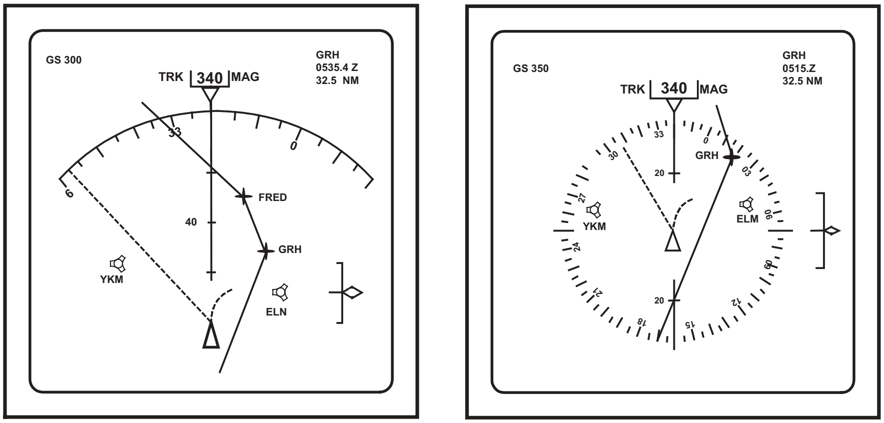
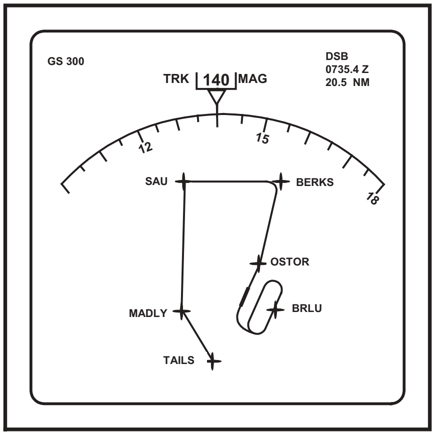


Fig. 8.16 *Map display – expanded and centre modes*

7. Plan Mode. This display (Fig. 8.17) is used for three purposes:
   - to view a flight plan stored in the FMS computers,
   - to amend a flight plan stored in the FMS computers,
   - to create a new flight plan.

   The plan mode displays 80º of the compass rose. The plan mode is oriented with true north at the 12 o'clock position. This mode effectively represents a 'birds eye' view of the intended flight plan, and its representation of information is more like that displayed in the published terminal procedures: SIDS, STARS, etc., examples of which are given later. Colour is used extensively to accentuate the visual impact of the displays. The certification guidance regarding the use of colour is described in Chapter 7.

Fig 8.17 *Plan mode*



FMS Control and Display Unit (FMS CDU)

The FMS CDU is the key flight crew interface with the navigation system, allowing the flight crew to enter data as well as having vital navigation information displayed. A typical FMS CDU is shown in Fig. 8.18.

The CDU has a small screen on which alpha-numeric information is displayed, in contrast to the pictorial information displayed on the EFIS navigation displays. This screen is a Cathode Ray Tube (CRT) monochrome display in early systems; later systems use colour Active Matrix Liquid Crystal Display (AMLCD) (see Chapter 7). The tactile keyboard has alpha-numeric keys in order to allow manual entry of navigation data (perhaps inserting final alterations to the flight plan), as well as various function keys by which specific navigation modes may be selected. The line keys at the sides of the display are soft keys that allow the flight crew to enter a menu-driven system of subdisplays to access more detailed information. On many aircraft the CDU is used to portray maintenance status and to execute test procedures using the soft keys and the menu-driven feature. Finally, there are various annunciator lights and a lighting control system.

An example of the type of data displayed on the CDU is presented in Fig. 8.19. The example shown displays:

- An ETA waypoint window that shows the Estimated Time of Arrival (ETA) at the waypoint, in this case waypoint 15.

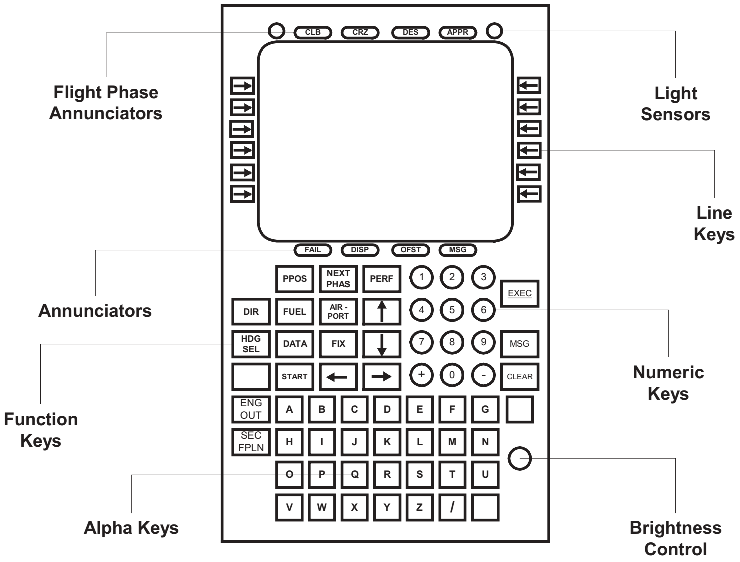


Fig. 8.18 *Typical FMS Control and Display Unit*

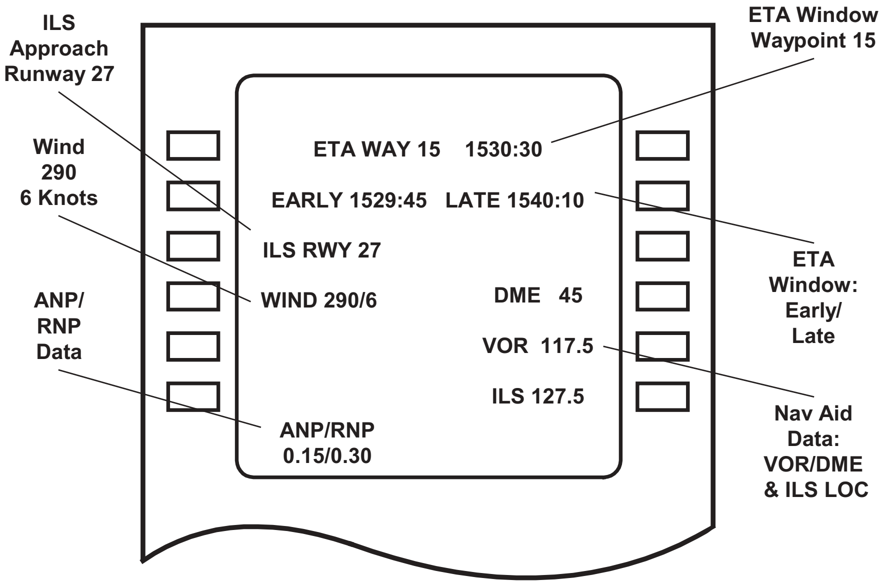
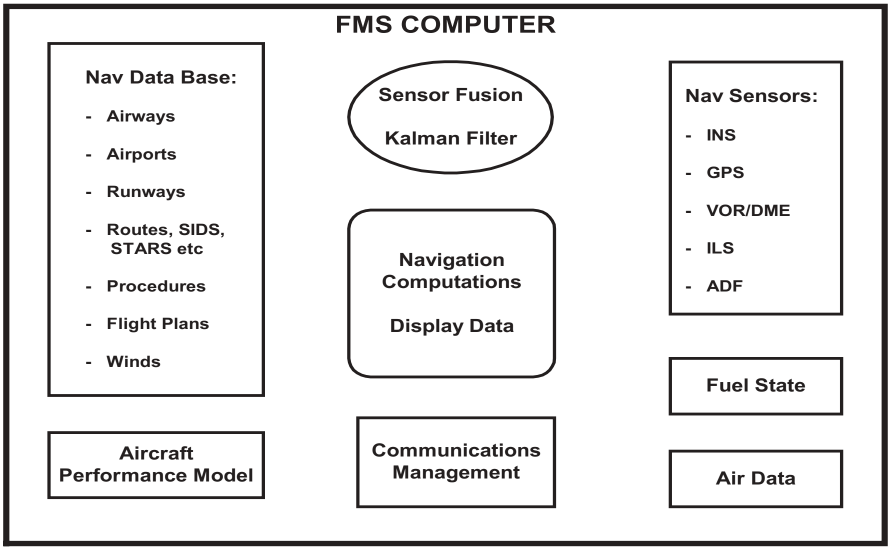


Fig. 8.19 *Typical FMS CDU display data*

- Early/late timing information which represents the earliest and latest times the aircraft can reach the waypoint given its performance characteristics.
- Information on the runway – an ILS approach to runway 27.
- Wind information for the approach – wind bearing 290 at 6 knots.
- Information on the navigation aids being used: VOR, DME, and ILS/LOC.
- ANP/RNP window. This compares the Actual Navigation Performance (ANP) of the system against the Required Navigation Performance (RNP) for the flight phase and navigation guidance being flown. In this case the ANP is 0.15 nautical miles, against an RNP of 0.3 nautical miles, and the system is operating well within limits.

Fig. 8.20 *Top-level FMS functions*



The functions of the FMS at a top level are shown in Fig. 8.20. This diagram gives an overview of the tasks performed by the FMS computers. These may be summarized as follows:

1. Navigation computations and display data. All the necessary navigation computations are undertaken to derive the navigation or guidance information according to the phase of flight and the sensors utilized. This information is displayed on the EFIS navigation display or the FMS CDU. Flight director and steering commands are sent to the autopilot for the flight director with the pilot in the loop or for the engagement automatic flight control modes.
2. Navigation sensors. INS, GPS, VOR, ILS, ADF, TACAN, and other navigation aids provide dual sensor information to be used for various navigation modes.
3. Air data. The ADC or ADIRS provides the FMS with high-grade corrected air data parameters and attitude information for use in the navigation computations.
4. Fuel state. The fuel quantity measurement system and the engine mounted fuel flow-meters provide information on the aircraft fuel quantity and engine fuel flow. The calculations of fuel use and total fuel consumption are used to derive aircraft and engine performance during the flight. When used together with a full aircraft performance model, optimum flight guidance may be derived which minimizes fuel consumed.
5. Sensor fusion and Kalman filter. The sensor information is fused and validated against other sources to determine the validity and degree of fidelity of the data. By using a sophisticated Kalman filter, the computer is able to determine the accuracy and integrity of the navigation sensor and navigation computations and determine the Actual Navigation Performance (ANP) of the system in real time.
6. Communications management. The system passes information to the communication control system regarding communication and navigation aid channel selections that have been initiated by the FMS in accordance with the requirements of the flight plan.
7. Navigation database. The navigation base contains a wide range of data that are

relevant to the flight legs and routes the aircraft may expect to use. This database will include the normal flight plan information for standard routes that the aircraft will fly, together with normal diversions. It will be regularly updated and maintained. A comprehensive list of these items is:

- airways;
- airports, including approach and departure information, airport and runway lighting, obstructions, limitations, airport layout, gates, etc.;
- runways, including approach data, approach aids, category of approach (Cat I or Cat II/III), and decision altitudes;
- routes, clearance altitudes, SIDS, STARS, and other defined navigation data;
- procedures, including notification of short-term airspace restrictions or special requirements;
- flight plans with standard diversions;
- wind data: forecast winds and actual winds derived throughout flight.

8. Aircraft performance model. The inclusion of a full performance model adds to the ability of the system to compute four-dimensional (x, y, z, time) flight profiles, and at the same time make optimum use of the aircraft energy to optimize fuel use.

The FMS provides the essential integration of all of these functions to ensure that the overall function of controlling the navigation of the aircraft is attained. As may be imagined, this does not merely include steering information to direct the aircraft from waypoint to waypoint. The FMS also controls the tuning of all of the appropriate aircraft receivers to navigation beacons and communications frequencies via the communications control units, and has many other functions besides. The flight plan that resides within the FMS memory will be programmed for the entire route profile, taking into account all eventualities, including emergencies. More advanced capabilities include three-dimensional navigation and the ability to adjust the aircraft speed to reach a waypoint within a very small time window (typically ±6 s). The various levels of performance and sophistication are summarized in Table 8.4. These capabilities will be examined in a little more detail.

LNAV

Lateral navigation, or LNAV, relates to the ability of the aircraft to navigate in two dimensions, in other words, the lateral plane. LNAV was the first navigation feature to be implemented and involved navigating aircraft to their intended destination without any other considerations. LNAV comprises two major implementations:

- Airway navigation.
- Area navigation, or RNAV.

Airway navigation

Airway navigation is defined by a predetermined set of airways that are based primarily on VOR stations although some use NDB stations. In the United States these airways are further categorized depending upon the height of the airway:

- Airways based on VOR from 120 ft above the surface to 18 000 ft Above Mean Sea Level (AMSL) carry the V prefix and are called *Victor Airways.*
- Airways using VOR from 18 000 ft MSL to 45 000 ft AMSL are referred to as *Jet Routes*.

Each VOR used in the route system is called either a terminal VOR or a low- or high-altitude en route VOR. Terminal VORs are used in the terminal area to support approach and departure procedures and are usable up to ~25 nautical miles; they are not to be used for en route navigation. Low-altitude en route VORs have service volumes out to a range of 40 nautical miles and are used up to 18 000 ft on victor airways. High-altitude VORs support navigation on jet routes and their service volume may extend to a range ~200 nautical miles from the ground station. Clearly, the range of the VOR beacons is limited by line-of-sight propagation considerations, as has already been described.

Table 8.4 Summary of FMS capabilities

Function	Capability
Lateral navigation (LNAV)	The ability to navigate laterally in two dimensions
Vertical navigation (VNAV)	The ability to navigate laterally in two dimensions, plus the ability to navigate in the vertical plane. When combined with LNAV this provides three-dimensional navigation
Four-dimensional navigation	The ability to navigate in three dimensions, plus the addition of time constraints for the satisfaction of time of arrival at a waypoint
Full performance-based navigation	The capability of four-dimensional navigation together with the addition of an aircraft-specific performance model. By using cost indexing techniques, full account may be taken of the aircraft performance in real time during flight, allowing optimum use of fuel and aircraft energy to achieve the necessary flight path
Future Air Navigation System (FANS)	The combination of the full performance model together with all the advantages that FANS will confer, eventually enabling the concept of 'free flight'

Airway width is determined by the navigation system performance and depends upon error in the ground station equipment and in the airborne receiver and display system, and upon flight technical error, as has been described in Chapter 6. These accuracy requirements will be revisited in Chapter 12, where the existing airway navigation requirements will be compared with the more demanding FANS requirements.

Area navigation

Many aircraft possess an area navigation (RNAV) capability. The on-board navigation together with the FMS can navigate along a flight path containing a series of waypoints that are not defined by the airways. Navigation in these situations is not confined to VOR beacons but may use a combination of VOR, DME, LORAN-C, GPS, and/or INS. Random routes have the advantage that they may be more direct than the airway system, and also that they tend geographically to disperse the aircraft away from the airway route structure. Reference (**6**) defines the regulations that apply to aircraft flying two-dimensional RNAV in IFR conditions in the US national airspace system.

Reference (**6**) is the original guidance on the use of RNAV within the United States while Reference (**7**) is a more recent publication on navigation or FMS systems using multiple sensors – including GPS – and is probably more relevant to the sophisticated FMS in use today. A summary of the accuracy requirements for multisensor (i.e. not single co-located VOR/DME) operation specified in Reference (**7**) is given in Table 8.5.

Table 8.5 Summary of multisensor accuracy requirements

Error type	Oceanic and remote (nautical miles)	En route domestic (nautical miles)	Terminal (nautical miles)	Non-precision approach (nautical miles)
Two-dimensional accuracy requirements – equipment not incorporating GPS (95%)				
Position fixing error	12.0	2.8	1.7	0.3 (0.5 if navigation data derived from a single VOR/DME station)
CDI centring error	0.2	0.2	0.2	0.1
Two-dimensional accuracy requirements – equipment incorporating class B or C GPS sensor (95%)				
Position fixing error	0.124	0.124	0.124	0.056
CDI centring error	0.2	0.2	0.2	0.2
Note: As the flight technical error is outside the control of equipment manufacturers and is not included in these error budgets, it should not exceed the following on a 95% probability basis				
Flight Technical Error (FTE)		1.0	1.00	0.25 (GPS) 0.5 (non-GPS)

VNAV

Following on from the LNAV and RNAV capabilities, vertical navigation (VNAV) procedures were developed to provide three-dimensional guidance. Present VNAV systems use barometric altitude as it will be recalled that the GPS satellite geometry does not generally provide accurate information in the vertical direction. Whereas DGPS systems such as WAAS will address and overcome this issue, these systems will not be available for some time. Reference (**8**) provides details of the use of VNAV guidance in association with RNAV instrument approaches with a VNAV Decision Altitude (DA). One disadvantage of using barometric means to provide the VNAV guidance function is the non-standard nature of the atmosphere. Therefore, VNAV approaches embrace a temperature limit below which the use of VNAV decision height is not permitted. If the temperature on a particular day falls below this limit, then the flight crew must instead respect the published LNAV Minimum Decision Altitude. The accuracy requirements for VNAV are addressed in reference (**9**) and summarized in Table 8.6.

Table 8.6 Summary of VNAV accuracy requirements

Altitude region	Level flight segments and climb/descent, intercept of specified altitudes (ft)	Climb/descent along specified vertical profile (angle) (ft)
System accuracy for en route, terminal, and approach IFR operation in NAS (excluding altimetry) on 99.7% basis		
At or below 5000 ft	50	100
5000 – 10 000 ft	50	150
Above 10 000 ft	50	220
Flight technical (pilotage errors) – 99.7 (3 sigma)		
At or below 5000 ft	150	200
5000 to 10 000 ft	240	300
Above 10 000 ft	240	300

Four-dimensional navigation

The combination of LNAV and VNAV provides a three-dimensional navigation capability. However, in a busy air traffic management situation the element of time is equally important. A typical modern FMS will have the capability to calculate the ETA to a specific waypoint and ensure that the aircraft passes through that point in space within ±6 s of the desired time. Furthermore, calculations can be made in response to an air traffic control enquiry as to when the aircraft can reach an upcoming waypoint. By using information regarding the aircraft performance envelope, the FMS can perform calculations that determine the earliest and the latest possible time within which the aircraft can reach the waypoint. The ability to determine this time window can be of great use in helping the air traffic controller to maintain steady traffic flow during periods of high air traffic density.

Full performance based navigation

If the FMS contains a full performance model provided by the aircraft manufacturer, then even more detailed calculations may be performed. By using the aircraft velocity and other dynamic parameters, it is possible to compute the performance of the aircraft over very small time increments. By using this technique, and provided that the sensor data is sufficiently accurate, the future dynamic behaviour of the aircraft may be accurately predicted. Using this feature, and knowing the four-dimensional trajectory and gate speeds that are detailed in the flight plan, the aircraft can calculate the optimum trajectory to meet all these requirements while conserving energy and momentum and assuring minimum fuel burn. When this capability is combined with the increasing flexibility that FANS will provide, then further economies will be possible. Today, most FMS systems are being developed with these emerging requirements in mind, such that future implementation will depend upon system software changes and upgrades rather than aircraft equipment or architecture modifications.

FMS procedures

Although the foregoing explanations have concentrated on performance enhancements, the assistance that the FMS provides the flight crew in terms of procedural displays cannot be forgotten. Several typical scenarios are included in this section that illustrate the importance of the procedural information. These examples include:

- Standard Instrument Departure (SID).
- En route procedures.
- Standard Terminal Approach Routes (STARs).
- ILS approach.

Standard Instrument Departure (SID)

The SID shown in Fig. 8.21 is a simplified version of the Manchester airport Pole Hill departure to the north-west. There are VOR/DME beacons at the airport runway – 06/24 (MCT) and Pole Hill (POL) – that, together with the waypoint at XUMAT, form the basis of the various instrument departures, of which there are four in all: POL 12Y (runway 24L) and POL 5R (runway 24R), departing to the west, and POL 4S (runway 06L) and POL 1Z (runway 06R), departing to the east. The waypoints at XUMAT and Pole Hill also have latitude and longitude specified. The SID also specifies (not shown in Fig. 8.21 for reasons of clarity) height gates at various points on the departure path to ensure that departing aircraft climb in a structured and orderly manner. Departure speeds are also regulated – on this chart all aircraft are limited to a maximum of 250 knots below FL100 unless otherwise advised.

It may be seen that there are a large number of options available to the flight crew within even this relatively simple example. For instance, an aircraft climbing out on

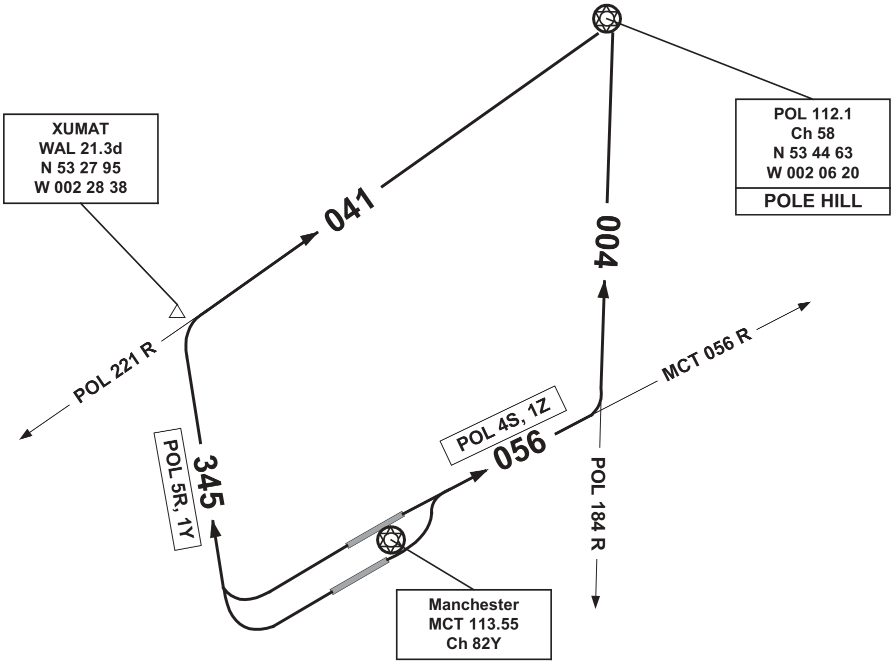


Fig. 8.21 *Typical standard instrument departure (SID)*

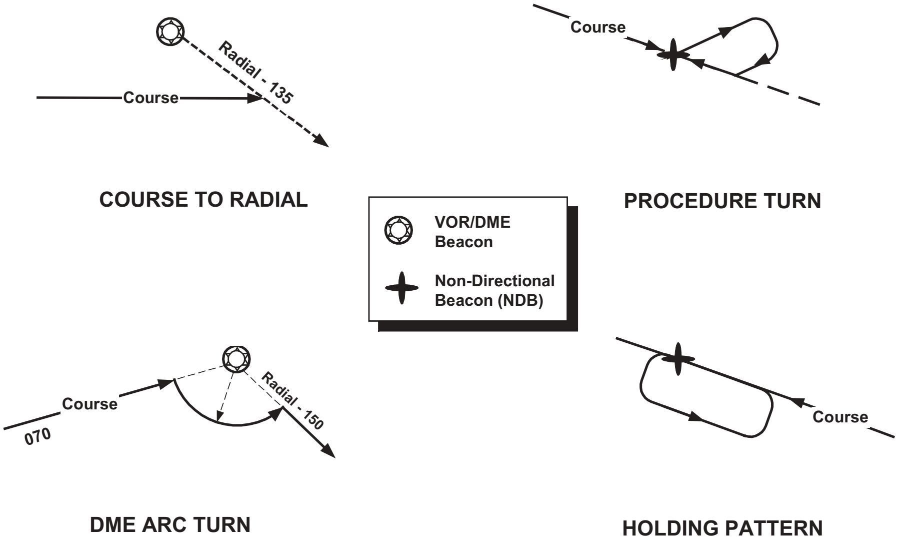
POL5R/1Y from runways 24L or 24R could fly to intercept the POL 221 radial before following the Pole Hill VOR/DME, or could elect to fly to inertial latitude/longitude waypoints at XUMAT and then Pole Hill. An FMS will therefore contain all the SIDs for all the airports that the aircraft may expect to use, both for normal and alternative departures, and also including procedural information for possible diversion airfields.

En route procedures

There are a number of FMS en route procedures – in all more than 20 different procedures are specified in ARINC 424-15 (**10**). These vary from the simplest procedure, following a course to a radio beacon, to some of the more complex, associated with holding patterns in terminal areas. Several examples of these procedures, which are self-explanatory, are depicted in Fig. 8.22. The en route FMS procedures as shown in Fig. 8.22 are:

- Course to radial.
- DME arc turn.
- Procedure turn.
- Holding pattern.

Fig. 8.22 *Several examples of en route FMS procedures*



Standard Terminal Arrival Routes

Standard Terminal Approach Routes (STARs) are defined for all airfields for approaches from various directions to accommodate aircraft arriving from different departure airfields and arriving at different runways, e.g. to allow for wind changes. Alternative procedures are also published to allow for airfield and terminal area equipment unserviceabilities.

The example shown in Fig. 8.23 is a simplified version of the STAR for Manchester airport for occasions when the airport VOR is not operational. All arrivals are routed via various waypoints to the Pole Hill VOR/DME beacon described in the SID in

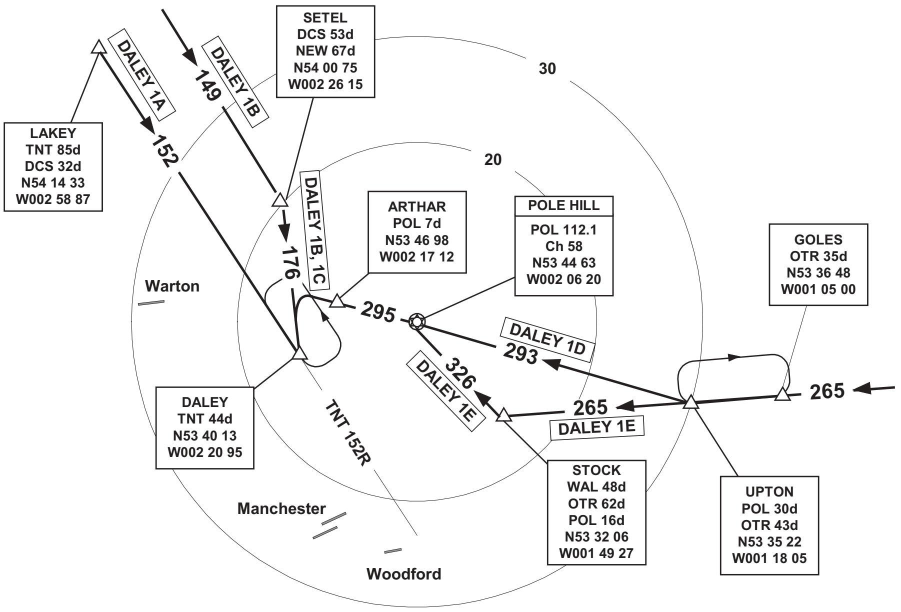


Fig. 8.23 *Typical Standard Terminal Arrival Routes (STAR)*

Fig. 8.21. However, whereas all the departures via Pole Hill were at 5000 ft the arrivals are all at FL80 (8000 ft) or above. There are four main routes to Pole Hill:

- DALEY 1A from the north-west for flights at FL150 or above. High-altitude arrivals are told to expect clearance to FL200 by 10 nautical miles before LAKEY.
- DALEY 1B, also from the north-west for flights at FL140 or below.
- DALEY 1D from the east for flights at FL90 and above. High-altitude arrivals are told to expect clearance to FL200 by 10 nautical miles before GOLES.
- DALEY 1E from the east for aircraft at FL80.

There is an additional holding area between GOLES and UPTON for aircraft approaching from the east. Aircraft arriving from these main arrival routes are spatially separated both laterally and vertically.

If traffic is busy, aircraft arriving at Pole Hill may be directed into a holding pattern at the DALEY waypoint to the west, which is located about 20 nautical miles north-west of the airfield. At this point, aircraft will be flying at an altitude of between FL70 and FL140 with a speed not exceeding 230 knots. If clearance is available, traffic will be advised to fly directly from Pole Hill to the airport holding area waypoint AMLET before being cleared for approach (see Fig. 8.24).

ILS approach

Having negotiated a passage through one of the STARs, the aircraft will be ready to perform an approach to the active runway. A simplified example of an ILS/DME approach to runway 06L is shown in Fig. 8.24.

Aircraft from Pole Hill will arrive at AMLET on the 166º radial and take up a position at 6000 ft in the terminal holding pattern defined by the AMLET and DAYNE waypoints. When cleared, the aircraft will be instructed to descend to 3000 ft on the 236º outbound course from the airfield (MCH/MCT). At around 11 miles from the

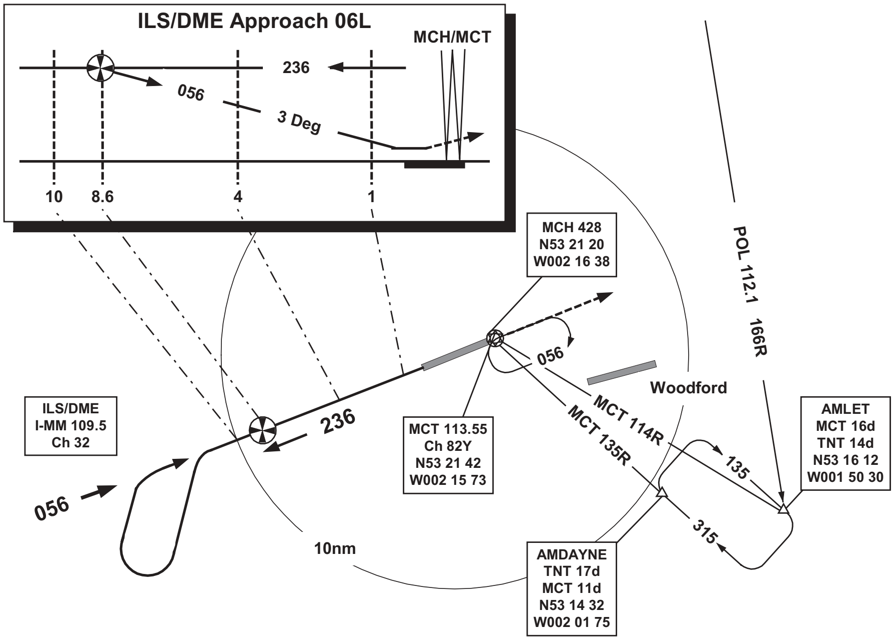


Fig. 8.24 *Typical ILS approach procedure*

airfield it will perform a procedural manoeuvre to position itself inbound on the 056º heading, by which time it should have acquired the airport DME/ILS localizer. At a point ~ 8.6 nautical miles from the runway threshold, the aircraft should capture the glide slope and descend on a 3º glide slope to the touchdown point. Should the aircraft need to execute a missed approach, it will be vectored back to the terminal area holding zone via the MCT 114º radial to AMLET where it will hold at 5000 ft to avoid conflict with incoming traffic while awaiting clearance for another approach.

FANS

Future Air Navigation System (FANS) is the term used to describe future developments of aircraft communications and navigation systems and the ground and satellite infrastructure. FANS will be an evolutionary process that will progressively improve the capabilities of systems in three key areas:

- Communications.
- Navigation.
- Surveillance.

Certain elements of FANS are partly implemented today and others are planned for the medium and longer term. The whole subject of FANS is addressed as a separate topic in Chapter 12.

Terrain Awareness and Warning System (TAWS)

TAWS embraces the overall concept of providing the flight crew with prediction of a potential controlled flight into terrain. The new term is a generic one, since Ground Proximity Warning System (GPWS) and Enhanced GPWS (EGPWS) became associated mainly with the Allied Signal (now Honeywell) implementation. The latest manifestation is designed to provide the crew with an improved prediction compared with previous systems. The FAA is presently in the process of specifying that turbine-equipped aircraft with six seats or more will be required to be equipped with TAWS by 2003. Reference (**11**) addresses the airworthiness requirements associated with TAWS.

GPWS and EGPWS

While TCAS is designed to prevent air-to-air collisions, the GPWS is intended to prevent unintentional flight into the ground. Controlled Flight Into Terrain (CFIT) is the cause of many accidents. The term describes conditions where the crew are in control of the aircraft, but owing to a misplaced sense of situational awareness, they are unaware that they are about to crash into the terrain. GPWS takes data from various sources and generates a series of audio warnings when a hazardous situation is developing.

GPWS uses radar altimeter information together with other information relating to the aircraft flight path. Warnings are generated when the following scenarios are unfolding:

- Flight below the specified descent angle during an instrument approach.
- Excessive bank angle at low altitude.
- Excessive descent rate.
- Insufficient terrain clearance.
- Inadvertent descent after take-off.
- Excessive closure rate to terrain – the aircraft is descending too quickly or approaching higher terrain.

Inputs are taken from a variety of aircraft sensors and compared with a number of algorithms that define the safe envelope within which the aircraft is flying. When key aircraft dynamic parameters deviate from the values defined by the appropriate guidance algorithms, then appropriate warnings are generated.

The installation of GPWS equipment for all airliners flying in US airspace was mandated by the FAA in 1974, since when the number of CFIT accidents has dramatically decreased.

More recently enhanced versions have become available. EGPWS offers a much greater situational awareness to the flight crew as more quantitative information is provided, together with earlier warning of the situation arising. It uses a worldwide terrain database which is compared with the aircraft's present position and altitude. Within the terrain database the Earth's surface is divided into a grid matrix with a specific altitude assigned to each square within the grid, representing the terrain at that point.

The aircraft intended flight path and manoeuvre envelope for the prevailing flight conditions are compared with the terrain matrix and the result is graded according to the proximity of the terrain, as shown in Fig. 8.25. Terrain responses are graded as follows:

- No display for terrain more than 2000 ft below the aircraft.
- Light-green dot pattern for terrain between 1000 and 2000 ft below the aircraft.
- Medium-green dot pattern for terrain between 500 and 1000 ft below the aircraft.
- Medium-yellow dot pattern for terrain between 1000 ft above and 500 ft below the aircraft.

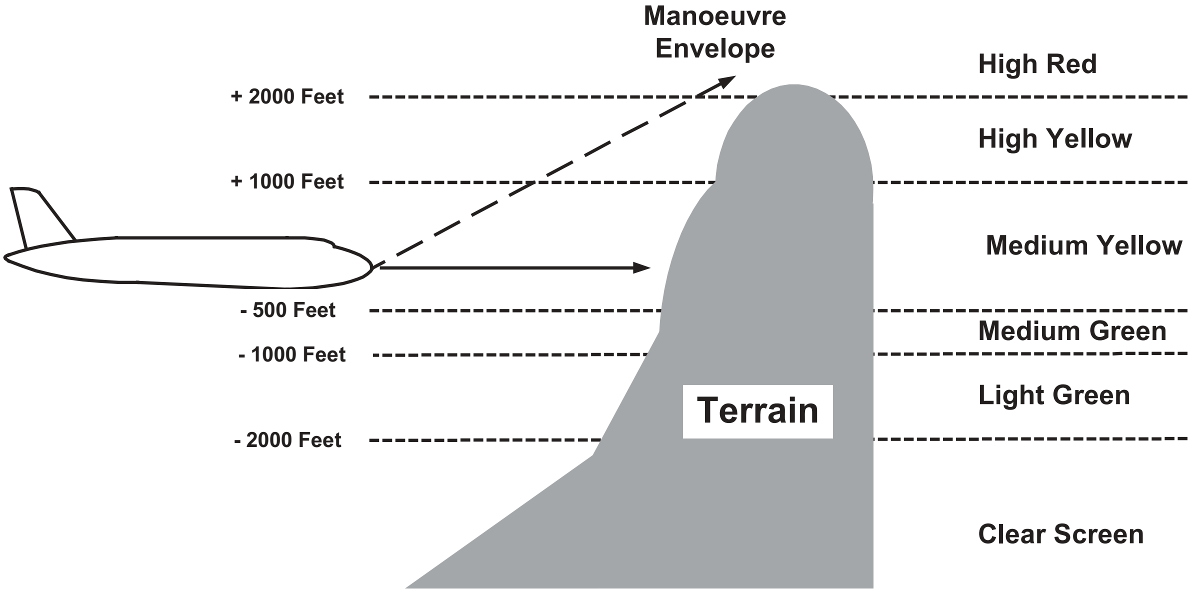


Fig. 8.25 *Principle of operation of EGPWS*

- Heavy-yellow display for terrain between 1000 and 2000 ft above the aircraft.
- Heavy-red display for terrain more than 2000 ft above the aircraft.

This type of portrayal using coloured imagery is very similar to that for the weather radar and is usually shown on the navigation display. It is far more informative than the audio warnings given by earlier versions of GPWS. The EGPWS also gives audio warnings, but much earlier than those given by the GPWS. The earlier warnings, together with the quantitative colour display, give the flight crew a much better overall situational awareness in respect of terrain and more time to react positively to their predicament than did previous systems.

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(3) Advisory Circular AC 25-4 (1966) Inertial Navigation Systems (INS).
(4) Advisory Circular AC 90-94 (1994) Guidelines for using Global Positioning System Equipment for IFR En-Route and Terminal Operations and for Non-Precision Approaches in the US National Airspace System.
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(8) Advisory Circular AC 90-97 (2000) Use of Barometric Vertical Navigation (VNAV) for Instrument Approach Operations using Decision Altitude.

(9) Advisory Circular AC 20-129 (1998) Airworthiness Approval of Vertical Navigation (VNAV) Systems for use in the National Airspace System (NAS) and Alaska.

(10) ARINC 424-15 (2000) Navigation System Data Base.

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CHAPTER 9

Flight Control Systems

The task of flying and navigating the modern commercial aircraft has become more difficult and stressful with crowded skies and busy airline schedules. To ease the pilot's task, the functional complexity of flight control and guidance has increased**.** Whereas Concorde was the first civil aircraft to have a fly-by-wire system with mechanical back-up, Airbus introduced a fly-by-wire system on to the A320 family and a similar system has been carried forward to the A330/340. Boeing's first fly-by-wire system on the Boeing 777 was widely believed to be a response to the Airbus technology development. This chapter on flight control systems examines some of the key differences between the Airbus and Boeing philosophies and implementations and also examines the systems that will be introduced on the Airbus A380. The impact of advanced autopilot and flight management functions is also considered.

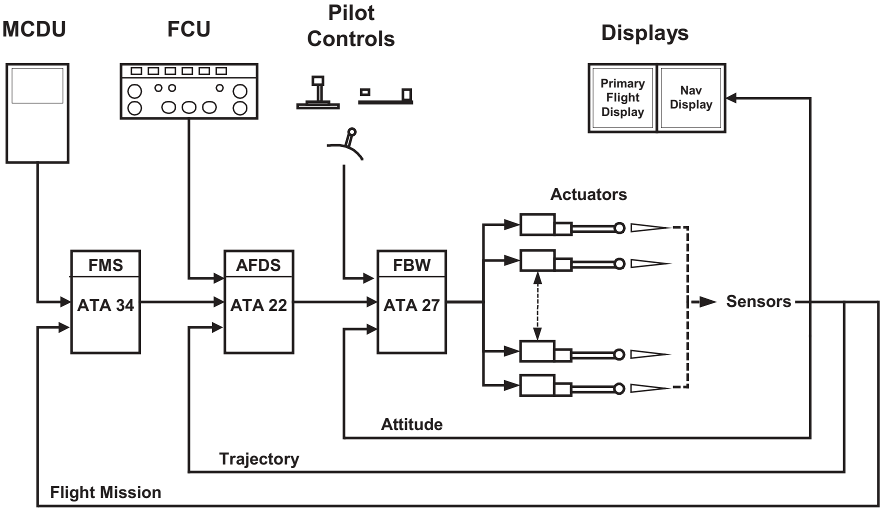
Inter-relationship of flight control functions

There is sometimes some confusion regarding the interrelationship of primary flight control, autopilot/flight director system, and flight management system functions. Figure 9.1 provides clarification. These separate but intertwined functions may be described as three nested control loops, each with their own distinct tasks. These are:

- Primary flight control or Fly-By-Wire (FBW) – ATA Chapter 27, Flight Control.
- Autopilot/Flight Director System (AFDS) – ATA Chapter 22, Autoflight Control.
- Flight Management System (FMS) – ATA Chapter 34, Navigation.

Information regarding the basic flight control system – push/pull rods, cable and pulleys, and artificial feel systems – is contained within a companion volume **(1)**.

Fig. 9.1 Inter-relationship of flight control functions



The FBW system comprising the inner loop is concerned with controlling the attitude of the aircraft. Inputs from the pilot's controls – control column (or, in Airbus systems, sidestick), rudders, and throttles – determine, via the aircraft dynamics, how the aircraft will respond at various speeds and altitudes throughout the flight envelope. Inertial and air data sensors determine the aircraft response and close the pitch, roll, and yaw control loops to ensure that the aircraft possesses well-harmonized control characteristics throughout the flight regime. In some aircraft, relaxed stability modes of operation may be invoked by using the fuel system to modify the aircraft centre of gravity, reducing trim drag and reducing aerodynamic loads on the tailplane or stabilizer. The aircraft pitch, roll, and yaw (azimuth) attitude or heading are presented on the primary flight display and navigation display.

The autopilot flight director system performs additional control loop closure to control the aircraft trajectory. The AFDS controls the speed, height, and heading at which the aircraft flies. Navigation functions associated with specific operations such as heading hold and heading acquire are also included. Approach and landing guidance is provided by coupling the autopilot to the ILS or MLS approach systems. The control and indication associated with these multiple autopilot modes is provided by a Flight Mode Selector Panel (FMSP) which enables the selection of the principal modes and also provides information confirming that the various modes are correctly engaged and functioning properly.

The final outer loop closure is that undertaken by the FMS which performs the navigation or mission function, ensuring that the FBW and AFDS systems position aircraft at the correct point in the sky to coincide with the multiple way-points that characterize the aircraft route from departure to destination airfield. The pilot interface with the FMS to initiate and monitor the aircraft progress is via a Multifunction Control and Display Unit (MCDU), also known more loosely as the Control and Display Unit (CDU).

Flight control – frames of reference

The frames of reference for the motion are referenced to the aircraft and comprise the axis set shown in Fig. 9.2.

- The *X* axis represents the direction of motion of the aircraft. Axial forces due to aircraft thrust or drag operate in this direction. Differences in axial force will result in axial acceleration or deceleration, causing an increase or decrease in aircraft forward (axial) velocity. Rotation around the *X* axis relates to the aircraft roll or bank angle and rates of change such as aircraft body roll rate which will be important in flight control.
- The *Y* axis portrays the direction of lateral movement and forces. Normally on a civil aircraft lateral forces and accelerations are not large, though they need to be taken into account. Rotation around the *Y* axis results in changes in pitch angle which results in the aircraft climbing or descending. Body pitch rate may also be an important consideration in some modes of flight control operation.
- The *Z* axis portrays the direction of normal force and acceleration. Lift and aircraft weight are major forces that act in this vertical direction. Changes in normal force lead to normal accelerations and velocities; vertical velocity is an important parameter to be considered during climb and descent. Rotation around the *Z* axis results in changes in yaw in the direction the aircraft is pointing, usually portrayed in navigation terms as changes in angle in azimuth or heading. Body yaw rates will be taken into account in flight control laws.

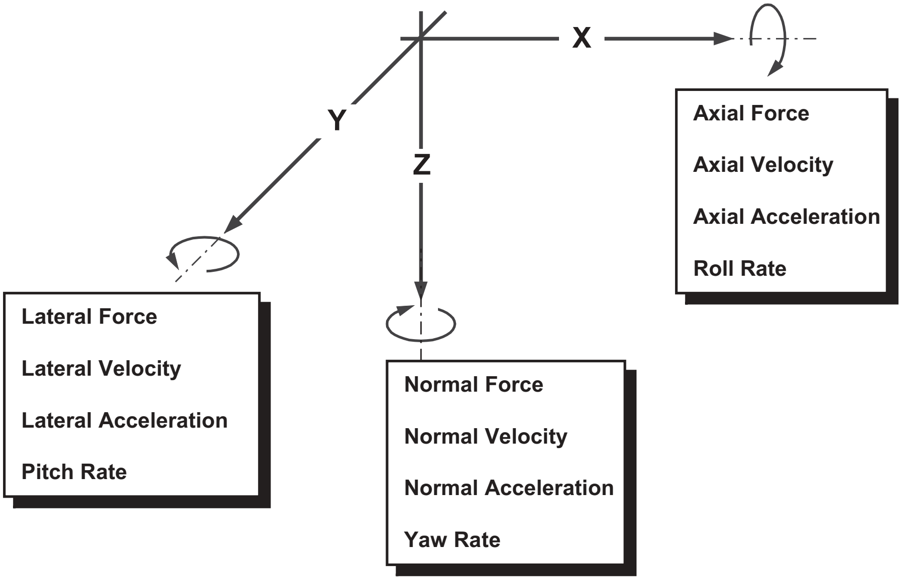


Fig. 9.2 *Flight control – frames of reference*

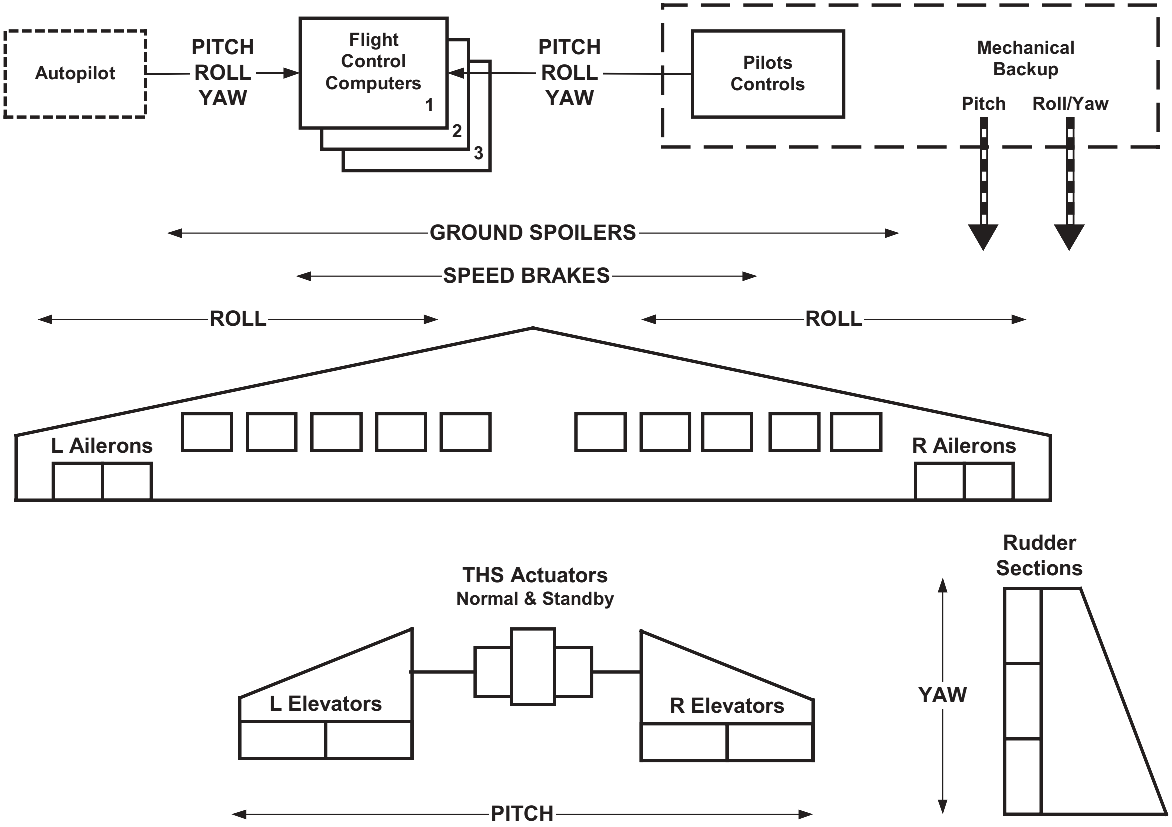
Flight control systems

A generic primary flight control system for a civil transport aircraft is shown in Fig. 9.3. Although the precise number of control surfaces and means of flight control computer implementation may vary from aircraft to aircraft, all modern FBW systems accord to this generic form.

The key control surfaces are as follows:

- Pitch control is usually effected by four powered flight control actuators powering four elevator sections.
- Pitch trim is undertaken by means of two Tailplane Horizontal Stabilizer (THS) actuators – operating as normal and standby systems – which move the entire horizontal tailplane surface or stabilizer (or stabilator in US parlance).
- Roll control is invoked by using the left and right ailerons, augmented as required by the extension of a number of spoilers on the inboard wing sections.
- Yaw control by means of two or three rudder sections.
- Both wing sets of spoilers may be extended together to perform the following functions:
  - Inboard spoiler sections to provide a speed brake function in flight, allowing the aircraft to be rapidly slowed to the desired airspeed, usually during descent.
  - Use of all spoilers in a ground spoiler or lift dump function during the landing

Fig. 9.3 Generic flight control system – civil aircraft



roll, enabling the aircraft rapidly to reduce lift during the early portions of the landing run.

Direct inputs from the pilot's controls or inputs from the autopilot feed the necessary guidance signals into a number of flight control computers depending upon the system architecture. These computers modify the flight control demands according to a number of aerodynamic and other parameters such that effective and harmonized handling characteristics are achieved.

- In addition, secondary flight control or high lift augmentation is provided by leading-edge slats and trailing-edge flaps which are extended for take-off and landing as appropriate. On the Boeing 777, the aileron and flap functions are combined by the use of two inboard flaperons, whereas conventional ailerons are used outboard. Operation of the speed brakes, flaps, and slats is initiated by dedicated control levers located on the flight deck central console.

Both the Airbus and Boeing implementations have the ability to fly the aircraft in a manual back-up mode as outlined below.

Mechanical back-up

All the modern civil aircraft using FBW systems employ some form of direct mechanical link as a back-up system:

- In the case of Airbus, the mechanical trim wheel can alter the position of the tailplane surface for pitch control. Inputs from the rudder pedals can alter the inputs to the three rudder actuators. A combination of pitch trim and rudder pedals therefore allows the aircraft to be controlled by manual means in an emergency situation.
- In the Boeing 777 system, alternate pitch trim levers are mechanically connected directly to the horizontal stabilator. A direct mechanical link from the rudder to one pair of spoilers allows roll control to be maintained in a standby mode.

Flight control actuation

The nature and use of the different forms of flight control actuation are worthy of examination, and their typical use in a flight control system will be described. The main types of actuation are:

1. Mechanically signalled, hydraulically powered ram (see Fig. 9.4). This is the most conventional arrangement by which the pilot's demands are mechanically relayed to the actuator by means of push/pull control rods or cable and pulley control runs. An electrically operated solenoid valve initiates the flow of hydraulic power to the actuator, thereby providing the muscle to operate the actuator ram and move the flight control surface. On most modern civil aircraft the aircraft hydraulic systems work at or close to the main system operational pressure of 3000 psi. In this scheme, mechanical feedback levers provide feedback so that the actuator is nulled when it reaches the desired position. This is the most common method of actuation provided for most primary flight control surfaces.
2. Mechanically signalled, hydraulically powered screw jack (see Fig. 9.5). As before, the signalling is mechanical, but the motive load is provided to a screw-jack actuator that moves the ram to a predetermined position. This type of actuation is used for

Fig. 9.4 *Mechanical signalling/hydraulically powered ram*

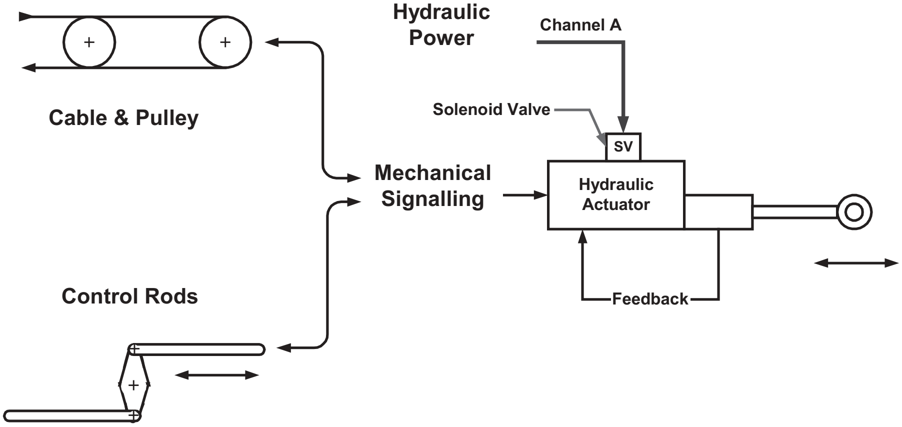
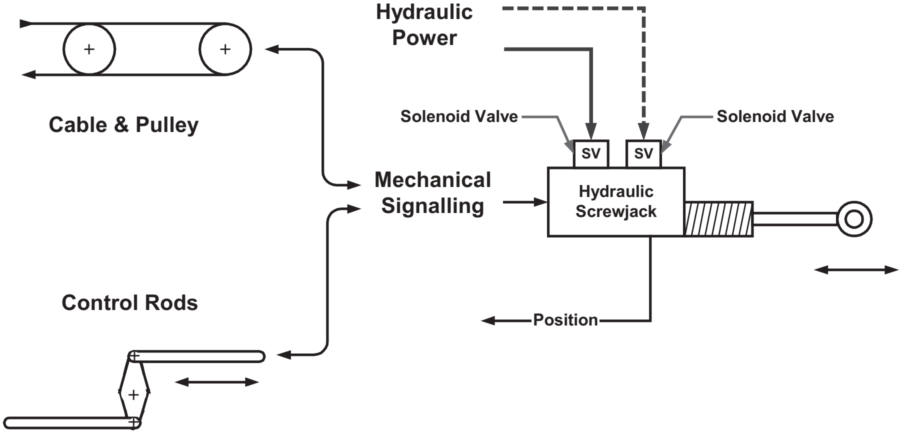


Fig. 9.5 *Mechanical signalled/hydraulically powered screw jack*



relatively slow moving surfaces such as the secondary flight control flap and slat surfaces, or for high load bearing applications such as the Tailplane Horizontal Stabilizer (THS). In some applications there may be a need for dual supplies of aircraft hydraulic power for availability or integrity reasons. The position feedback for the screwjack actuator will be fed to the controller for the particular function: flap or slat controller or horizontal trim controller in the case of the THS.

3. Mechanically signalled, electrically powered ram, as shown in Fig. 9.6. There are some similarities with the scheme shown in Fig. 9.4, with the exception that the actuation power is provided by the three-phase 400 Hz electrical power as opposed to the central aircraft hydraulic systems. Such an arrangement is also called a ‘power-by-wire’ system, and similar actuation types are being examined for use in modern aircraft, as will be seen. Early examples of this approach – using actuators called Integrated Actuator Packages (IAPs) – were used for primary flight control actuation on the Vickers VC10, which first flew in the early 1960s. In these Integrated Actuator Packages, a continuously running three-phase electrical motor powered an integral hydraulic pump that continuously pressurized a local hydraulic system providing power to the flight control actuator. The VC10 still flies today as an airborne refuelling tanker in the Royal Air Force, and recently gave sterling

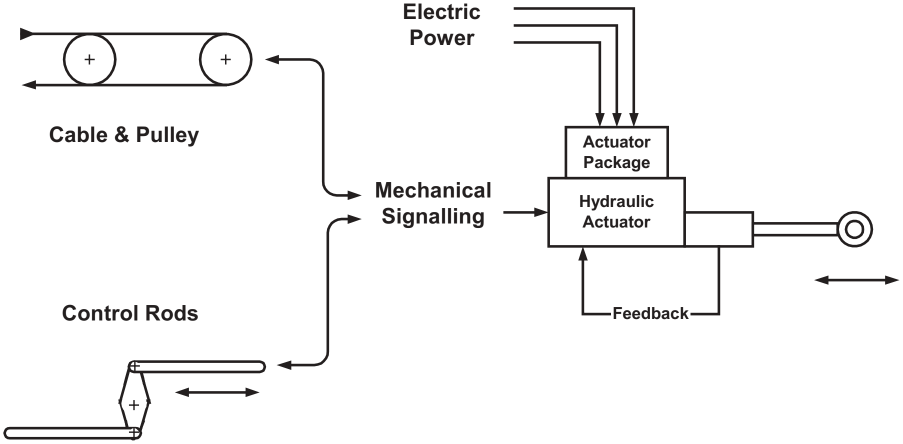


Fig. 9.6 *Mechanically signalled/electrically powered ram*

support in this role to US forces during the Afghan crisis.

4. Electrically signalled, hydraulically powered ram as depicted in Fig. 9.7. For the first time the possibility of signalling the actuator primary demand by electrical means is considered – either by an FBW command from a Flight Control Computer (FCC) or directly signalled by an electrical link. In the former example, more complex guidance and control laws will be implemented, probably with some envelope protection in the form of hard or 'soft' limits within the flight control algorithms. In the direct link, the control surface will move directly in proportion to the pilot's control inputs, with little or no change due to other influences, and will often but not exclusively be used in a back-up or reversionary mode of control. As before, the actuation power is provided by one or more central hydraulic systems. In both applications, feedback will be provided to the control entity – the FCC – closing the direct link control loop.
5. Electrically signalled, electrically powered ram, as shown in Fig. 9.8. This has been known more recently as an ElectroHydrostatic Actuator (EHA) with a localized electrohydraulic system similar to the IAP previously described. While this may appear to be somewhat similar to the IAP, there are substantive differences. Whereas the IAP runs with a continuously powered and pressurized localized system, the EHA contains sophisticated control and power electronics that ensure that the

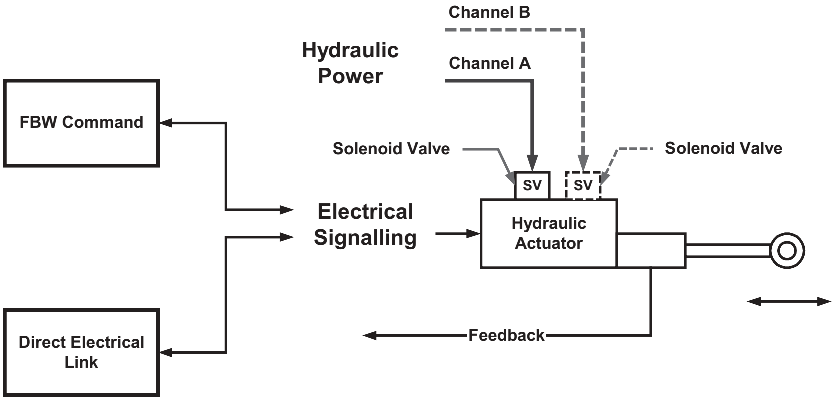
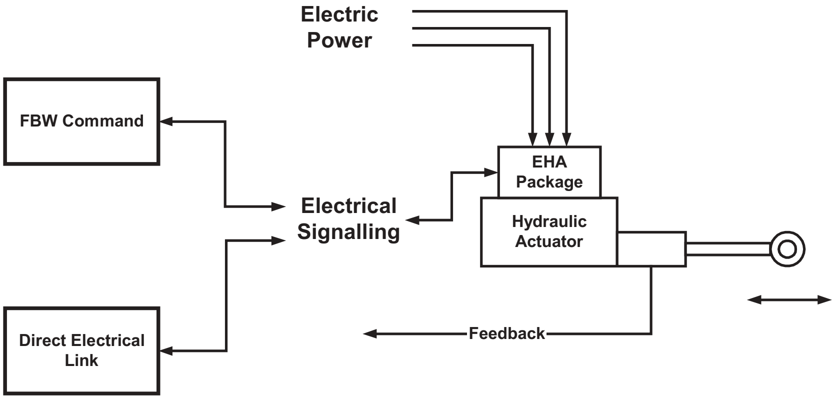


Fig. 9.7 *Electrically signalled/hydraulically powered ram*

Fig. 9.8 Electrically signalled/electrically powered ram



actuator only acts upon demand. Therefore, while the IAP may draw substantial power even under no-load conditions, the EHA will draw very low levels of power until a change in demand is required. This combination of low steady state loads and relatively high but transient loads results in a lower mean loading of the system. After being under development and flight demonstration for a number of years, EHAs are being introduced into the most modern systems such as the Joint Strike Fighter (JSF) and Airbus A380 flight control systems. These aircraft are moving towards the More-Electric Aircraft (MEA) concept, where electrical power is used more extensively across the aircraft – not just for flight control actuation but for other applications – with significant claimed savings in overall power useage and improved efficiency.

6. Electrically signalled, electrically powered screw jack, as shown in Fig. 9.9. This is called an ElectroMagnetic Actuator (EMA). This actuator has some similarities to the EHA in that it receives inputs from the FBW computer or direct electrical link. However, as for the previous screwjack implementation, this example is more likely to be used for flap, slat, and THS applications where relatively low rates of operation and high load applications are required. One similarity to the EHA is that it also operates in an ‘on-demand’ mode of operation, thereby reducing steady state power requirements.

While it is not possible to be categorical regarding the application of these actuator types, most will be used according to the summary in Table 9.1.

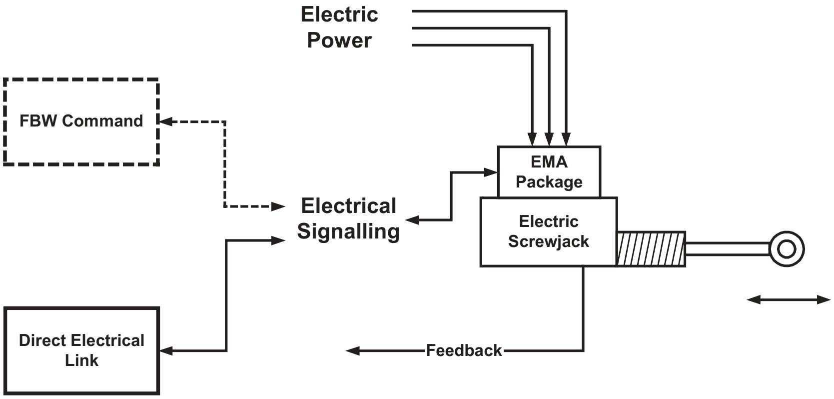


Fig. 9.9 *Electrically signalled/electrically powered screw jack*

Table 9.1 Summary of flight control actuator types

Actuator type	Primary flight control: Elevators, ailerons, rudder	Spoilers	Tailplane horizontal stabilizer	Flaps and slats
Hydraulic actuator – conventional	◆	◆		
Hydraulic actuator – screw jack			◆	◆
Integrated Actuator Package (IAP)	◆			
Electrically signalled hydraulic actuator	◆			
Electrohydrostatic actuator (EHA)	◆	◆		
Electromechanical actuator (EMA)			◆	◆

Flight control and monitoring requirements

The importance and integrity aspects of flight control lead to some form of monitoring function to ensure the safe operation of the control loop. Also, for integrity and availability reasons, some form of redundancy is usually required. Figure 9.10 shows a top-level comparison between the Boeing and Airbus FBW implementations.

In the Boeing philosophy, shown in simplified form on the left of Fig. 9.10, the system comprises three Primary Flight Computers (PFCs), each of which has three similar lanes with dissimilar hardware but the same software. Each lane has a separate role during an operating period, and the roles are cycled after power-up. Voting techniques are used to detect discrepancies or disagreements between lanes, and the comparison techniques used vary for different types of data. Communication with the four Actuator Control Electronics (ACE) units is by multiple A629 flight control data buses. The ACE units directly drive the flight control actuators. A separate flight control DC electrical system is provided to power the flight control system. The schemes used on the Boeing 777 will be described in more detail later in this chapter.

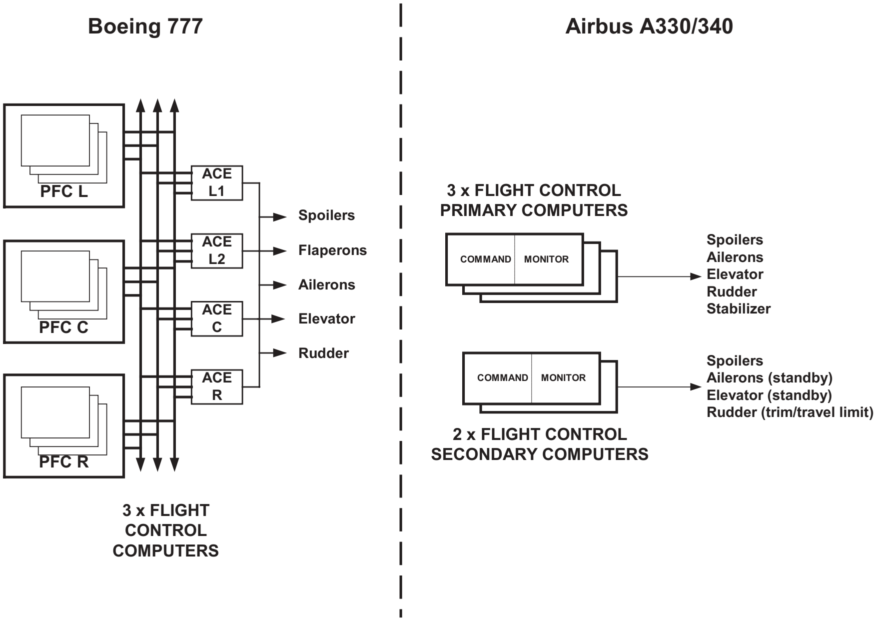


Fig. 9.10 *Top-level Boeing and airbus comparison of control philosophy*

The Airbus approach is shown on the right of Fig. 9.10. Five main computers are used: three Flight Control Primary Computers (FCPCs) and two Flight Control Secondary Computers (FCSCs). Each computer comprises command and monitor elements with different software. The primary and secondary computers have different architectures and dissimilar hardware and software. Command outputs from the FCSCs to ailerons, elevators, and rudder are for standby use only. Power sources and signalling lanes are segregated. The A320 and A330/340 flight control systems are described more fully later in this module.

Airbus FBW philosophy

In recent times, Airbus was the first aircraft manufacturer to introduce FBW to civil transport aircraft. The original aircraft to utilize FBW was the A320, and the system has been used throughout the A319/320/321 family and more recently on the A330/340. The A320 and A330/340 systems will be described in this section, the latter in more detail.

A320 FBW system

The A320 FBW system is shown in Fig. 9.11. This system uses a total of seven flight control computers, as follows:

- Two Elevator/Aileron Computers (ELACs). The ELACs control the aileron and elevator actuators according to the notation in Fig. 9.11.
- Three Spoiler/Elevator Computers (SECs). The SECs control all of the spoilers and in addition provide secondary control to the elevator actuators. The various spoiler sections have different functions, namely:
  - ground spoiler mode: all spoilers;
  - speed brake mode: inboard three spoiler sections;

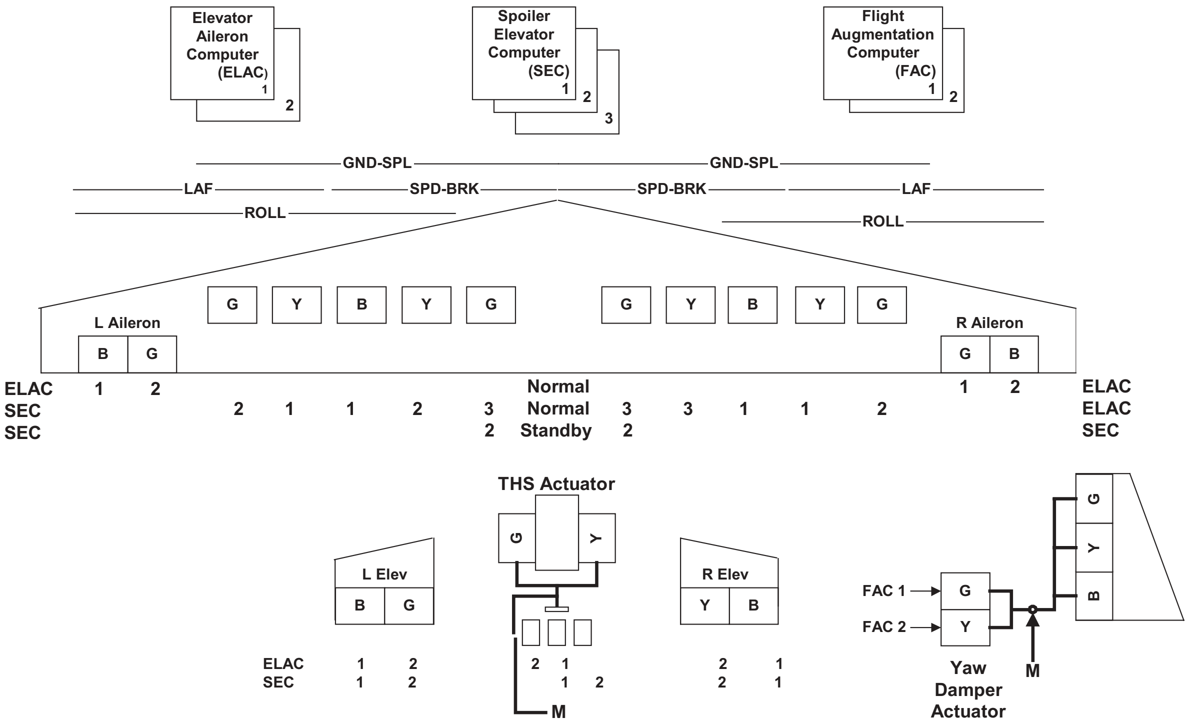


Fig. 9.11 A320 flight control system

  - load alleviation mode: outboard two spoiler sections (plus ailerons) (this function was later disabled);
  - roll augmentation: outboard four spoiler sections.
- Two Flight Augmentation Computers (FACs). These provide a conventional yaw damper function, interfacing only with the yaw damper actuators.

The three aircraft hydraulic systems [green (G), blue (B), and yellow (Y)] provide hydraulic power to the flight control actuators according to the notation shown in Fig 9.11.

In the very unlikely event of the failure of all computers, it is still possible to fly and land the aircraft manually using direct mechanical links – this has been demonstrated during certification and during a recent emergency landing. In this case the THS and rudder sections are controlled directly by mechanical trim inputs – shown as M in the diagram – which allow pitch and lateral control of the aircraft to be maintained.

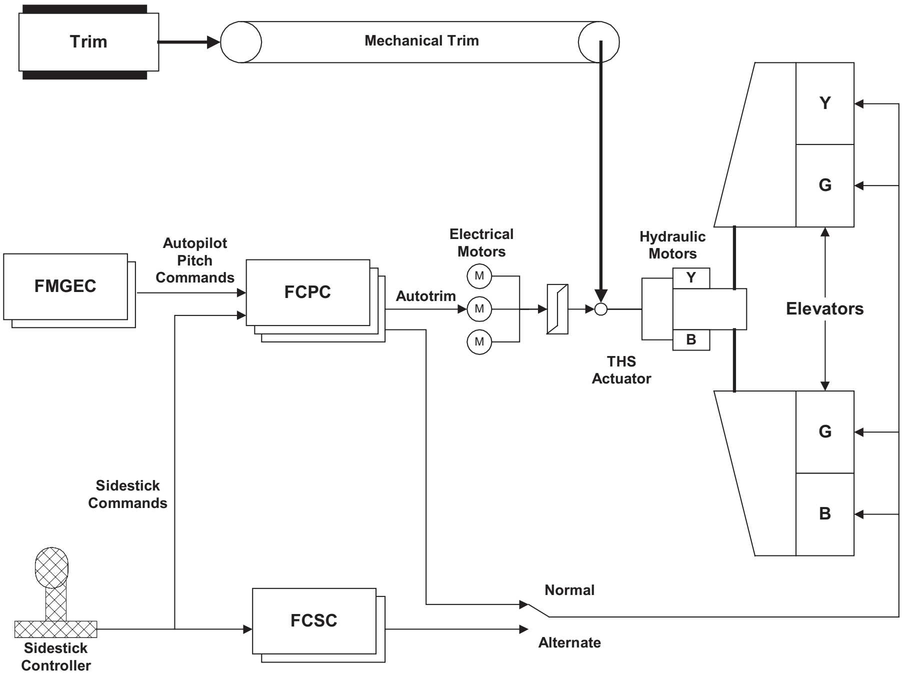
Another noteworthy feature of the Airbus FBW systems is that they do not use the conventional pitch and roll yoke. The pilot's pitch and roll inputs to the system are by means of a sidestick controller, and this now seems to be widely accepted by the international airline community.

A330/340 FBW system

The A330/340 FBW system shows many similarities to the A320 heritage as might be expected. There are, however, some differences, and it is also worth examining the basic pitch and yaw control loops, including the back-up modes.

The A330/340 pitch control is shown in Fig. 9.12. The pilot's input to the FCPCs and FCSCs are by means of the sidestick controller. The Flight Management Guidance

Fig. 9.12 *A330/340 pitch control*



and Envelope Computers (FMGECs) provide autopilot pitch commands to the FCPC. The normal method of commanding the elevator actuators is via the FCPC, although they can be controlled by the FCSC in a standby mode. Three autotrim motors may be engaged via a clutch to drive the mechanical input to the THS.

A direct mechanical link to the THS is also possible by means of the pitch trim wheel in the cockpit. The three hydraulic systems power the THS and elevator actuators as shown.

The A330/340 yaw control is shown in Fig. 9.13. As for the pitch channel, the FCPCs provide primary control, and the FCSCs the back-up. The pilot's inputs are via the rudder pedals directly, or, in the case of rudder trim, via the FCSC to the rudder trim motors.

The yaw damper function resides within the FCPCs rather than the separate FACs used on the A320 family. Autopilot yaw demands are fed from the FMGECs to the FCPCs.

There is a variable travel limitation unit to limit the travel of the rudder input at various stages of flight. As before, the three hydraulic systems feed the rudder actuators and two yaw damper actuators, as annotated in Fig. 9.13.

Therefore, although the implementation and notation of the flight control computers differs between the A320 and A330/340, a common philosophy can be identified between the two families.

The overall flight control system layout for the A330/340 is shown in Fig. 9.14 and includes terminology that by now should be becoming familiar. The major computing elements are:

- Three Flight Control Primary Computers (FCPCs). The function of the FCPCs has already been described.
- Two Flight Control Secondary Computers (FCSCs). Similarly, the function of the

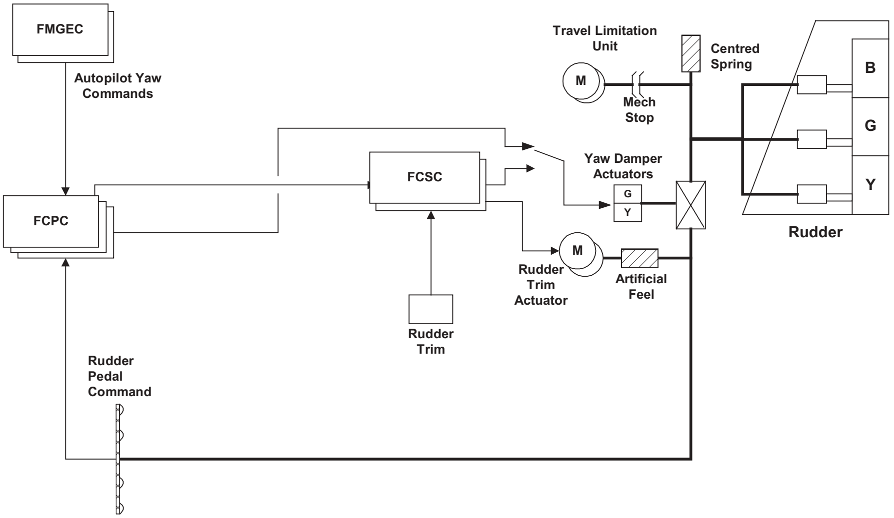
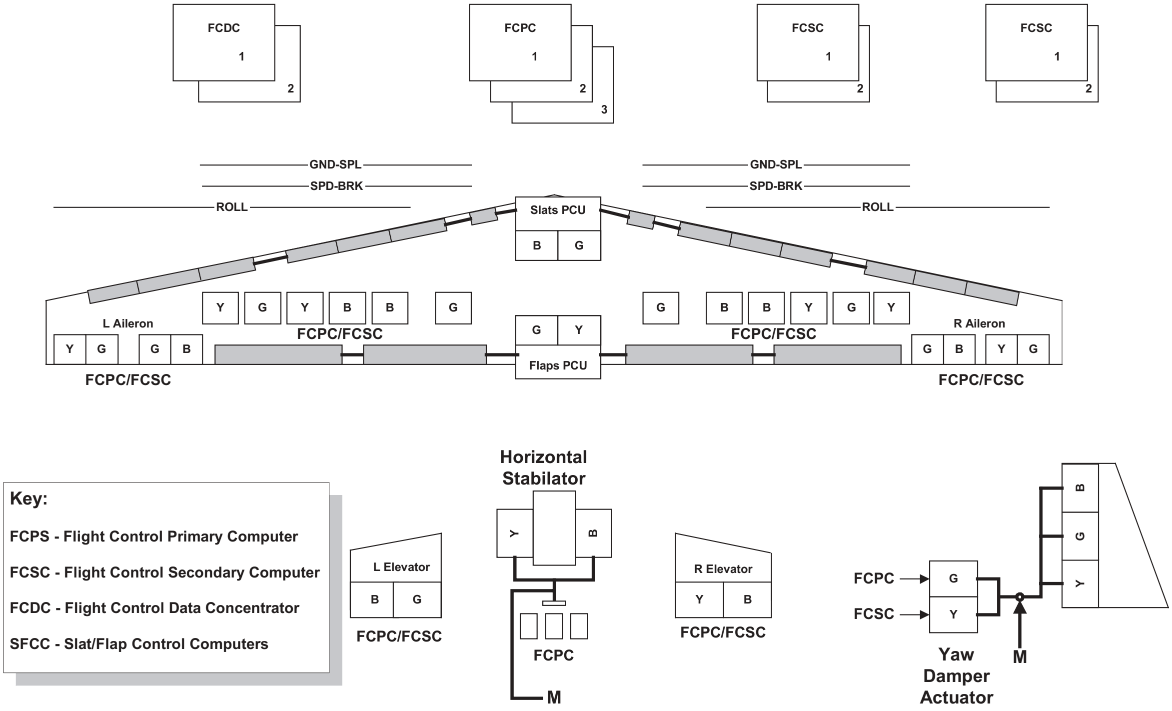


Fig 9.13 *A330/340 yaw control*

Fig 9.14 *A330/340 flight control system*



secondary computers has been described.

- Two Flight Control Data Concentrators (FCDCs).
- Two Slat/Flap Control Computers (SFCCs).

The FCDCs provide data from the primary and secondary flight computers for indication, recording, and maintenance purposes. The FSCCs are each able to control the full-span leading-edge slats and trailing-edge flaps via the hydraulically driven slat and flap motors.

Spoiler usage on the A330/340 differs from that on the A320. There is no load alleviation function, and there are six pairs of spoilers as against the five pairs on the A320. Also, in the A330/340 system the functions of the various spoiler pairs differ slightly from the A320 implementation, as close inspection of Figs 9.11 and 9.14 shows. Overall, the philosophy is the same.

Finally, Fig. 9.14 shows how the various flight control actuators, flap and slat motors and THS actuator are supplied with hydraulic power from the green, blue, and yellow hydraulic systems.

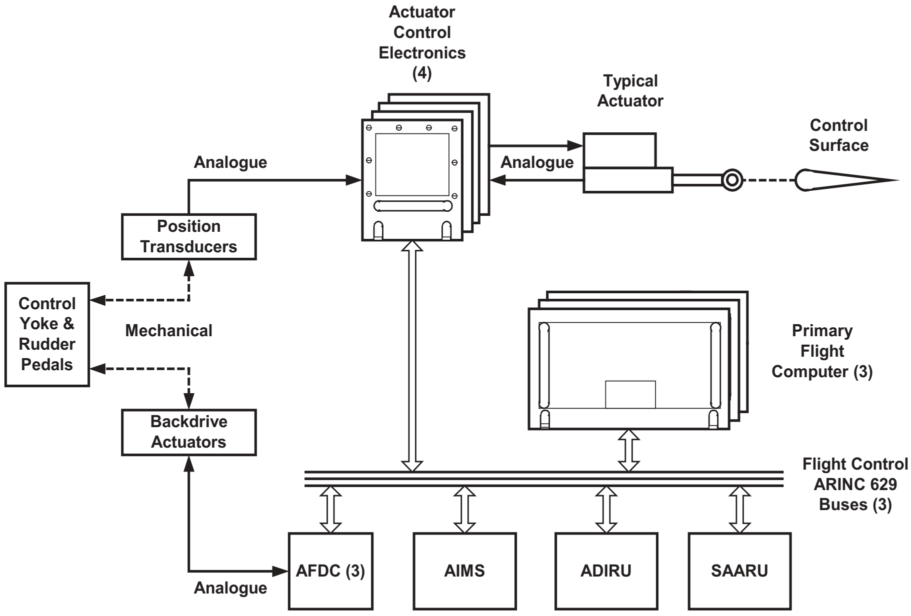
Boeing 777 flight control system

The Boeing Commercial Aircraft Group (BCAG) ventured into the FBW field with the Boeing 777, partly, it has been said, to counter the technology lead established by Airbus with the A320. Whatever the reason, Boeing have approached the job with precision and professionalism and have developed a solution quite different to the Airbus philosophy, as has been outlined earlier. This section explores the Boeing 777 Primary Flight Control System (PFCS) in more detail.

At the heart of the Boeing PFCS are three Primary Flight Computers (PFCs) in which the necessary computations are made; these computers interface with the three A629 flight control data buses, as shown in Fig. 9.15.

The PFCS comprises the following control surface actuators and feel actuators:

Fig. 9.15 *B777 Boeing PFCS overview*



- Four elevators: left and right inboard and outboard.
- Elevator feel: left and right.
- Two rudder: upper and lower.
- Four ailerons: left and right inboard and outboard.
- Four flaperons: left and right inboard and outboard.
- Fourteen spoilers: seven left and seven right.

The flight control actuators are interfaced to the three A629 flight control data buses by means of four Actuator Control Electronics (ACE) units. These are:

- ACE left 1.
- ACE left 2.
- ACE centre.
- ACE right.

The PFCS interfaces with the AIMS via A629 data buses. This permits interchange of information between the PFCS and AIMS systems and also provides a route for flight control system data to be interfaced with the flight deck.

The Air Data and Inertial Reference Unit (ADIRU), which provides the primary source of air data, and inertial and attitude data, and the Secondary Air Data and Attitude Reference Unit (SAARU), which is the secondary source of air data and inertial data, both interface with the PFCS via the three flight control system A629 data buses.

The ACE units contain the digital-to-analogue and analogue-to-digital elements of the system. A simplified schematic for an ACE is shown in Fig. 9.16. Each ACE has a single interface with each of the A629 flight control data buses, and the unit contains the signal conversion electronics to interface the 'digital' and 'analogue' worlds.

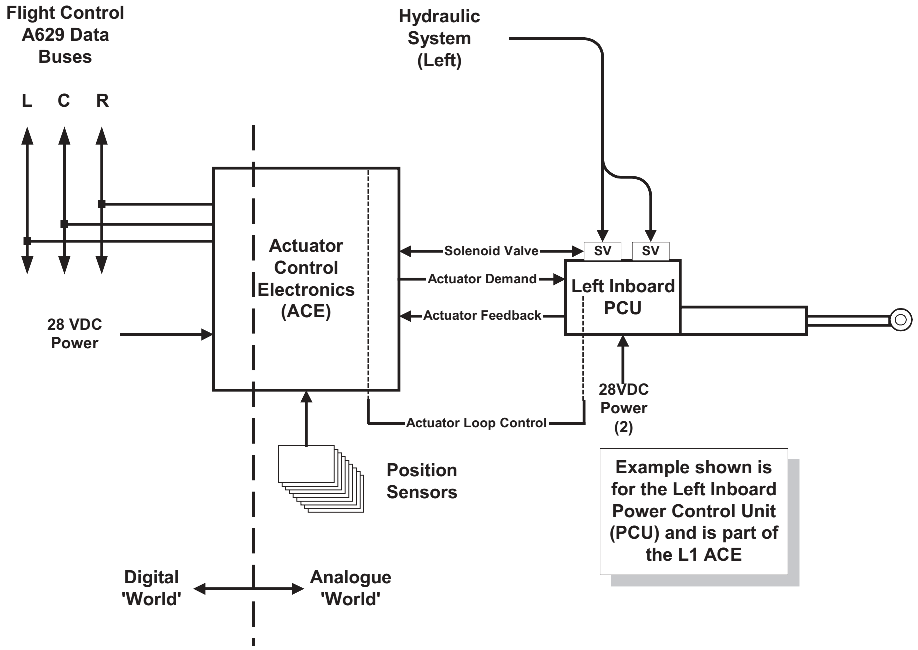


Fig. 9.16 *Actuator Control Electronics (ACE) unit*

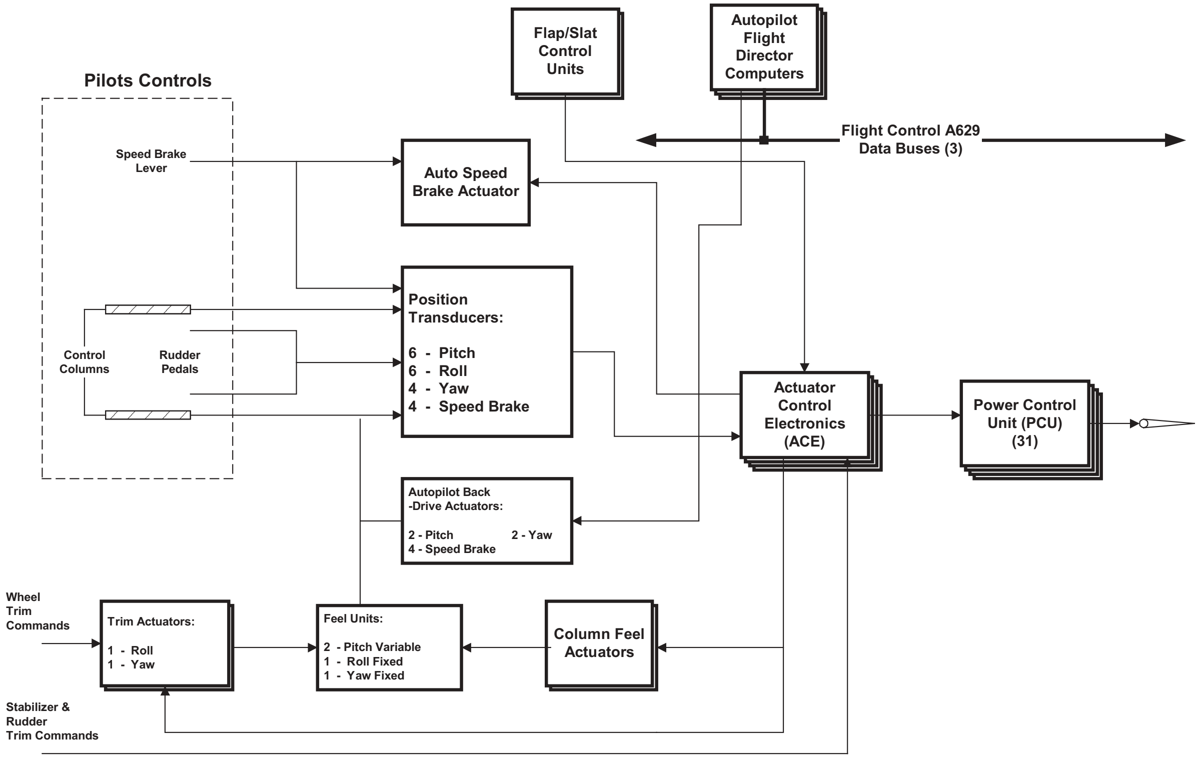
The actuator control loop is shown in the centre-right of the diagram. The actuator demand is signalled to the Power Control Unit (PCU) that moves the actuator ram in accordance with the control demand and feeds back a ram position signal to the ACE, thereby closing the actuator control loop. The ACE also interfaces to the solenoid valve with a command to energize the solenoid valves in order to allow – in this example – the left hydraulic system to supply the actuator with motive power, and at this point the control surface becomes ‘live’.

A simplified overview of the system, showing all the relevant units, is depicted in Fig. 9.17. This diagram de-emphasizes the A629 buses so that the other interfaces may be better understood. The main pilot’s controls are shown in the dotted box at the left of the diagram. The control columns are conventional, unlike the Airbus sidestick approach, and there are a total of six pitch, six roll, four yaw and four speed brake position transducer inputs.

There are roll and yaw trim actuators, two variable pitch feel units, and one roll and one yaw feel unit (both fixed). In the centre of the diagram the four ACEs can be seen driving the 31 actuators described above. At the top of the diagram, the flap/slat control units interface with the ACEs. The autopilot back-drive interfaces with the pilot’s controls are also shown. Fully to understand the operation of the PFCS it is convenient to divide the system into several discrete modes of operation. These are:

- Pitch elevator control – see Fig. 9.18.
- Pitch stabilizer control – see Fig. 9.19.
- Roll control – see Fig. 9.20.

Fig 9.17 *Simplified Boeing PFCS*



- Yaw control – see Fig. 9.21.
- Mechanical link – see Fig. 9.22.

Boeing 777 pitch elevator control

The pitch elevator control laws exercise control over the four PCUs that drive the left and right elevators to provide the aircraft with primary pitch control. The pitch portion of the control column is mechanically connected to position transducers (2), elevator feel units/actuators (2), and force transducers (2), as well as the back-drive actuator units (2).

Pitch demands are therefore fed into the ACE units either from the pilot transducers or as FCC or Autopilot Flight Director Computer (AFDC) commands from the A629 data buses, as appropriate, and are converted into analogue demands for the four elevator PCUs. Position feedback from each of the PCUs closes the control loop for each actuator.

The AFDCs interface with two back-drive actuators to align the mechanical transducers with the autopilot demands when the autopilot is engaged to ensure that no disagreement between pilot and autopilot inputs may persist. In this way, no out-of-trim conditions exist when the autopilot is disengaged or becomes disconnected, for example after a detected fault.

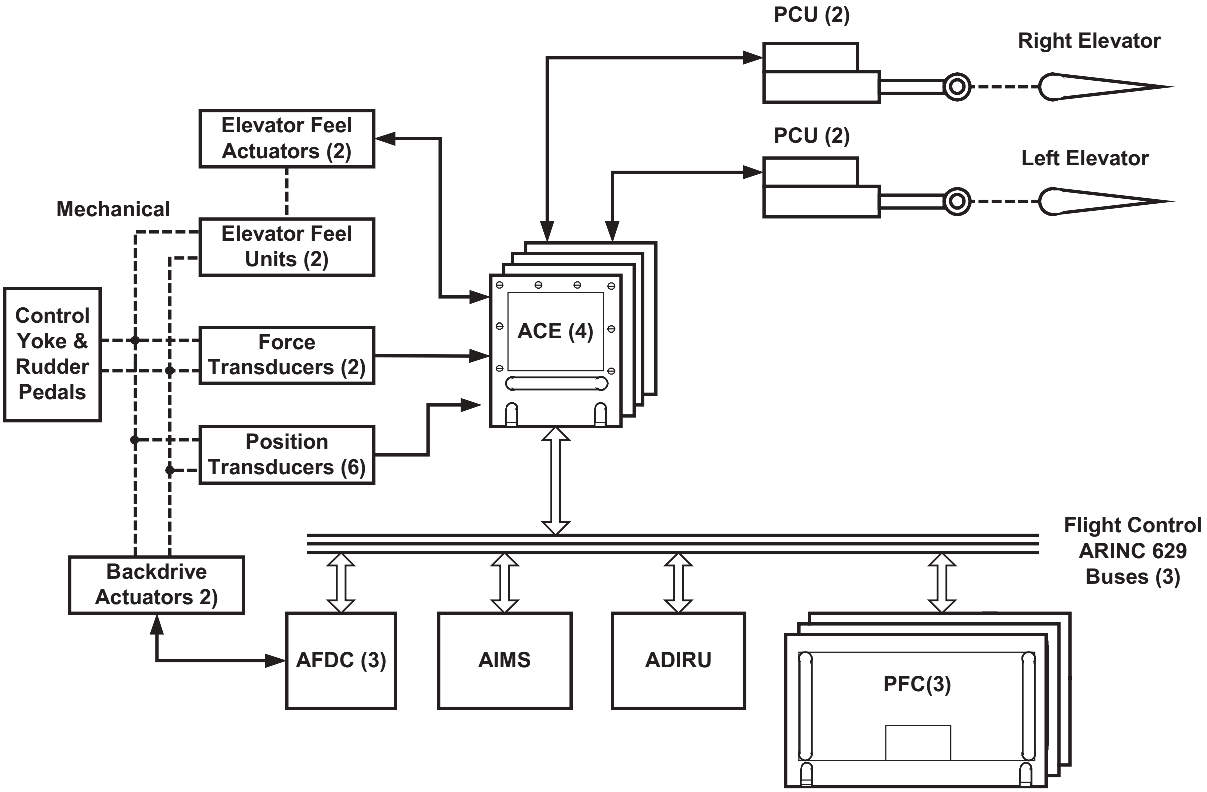


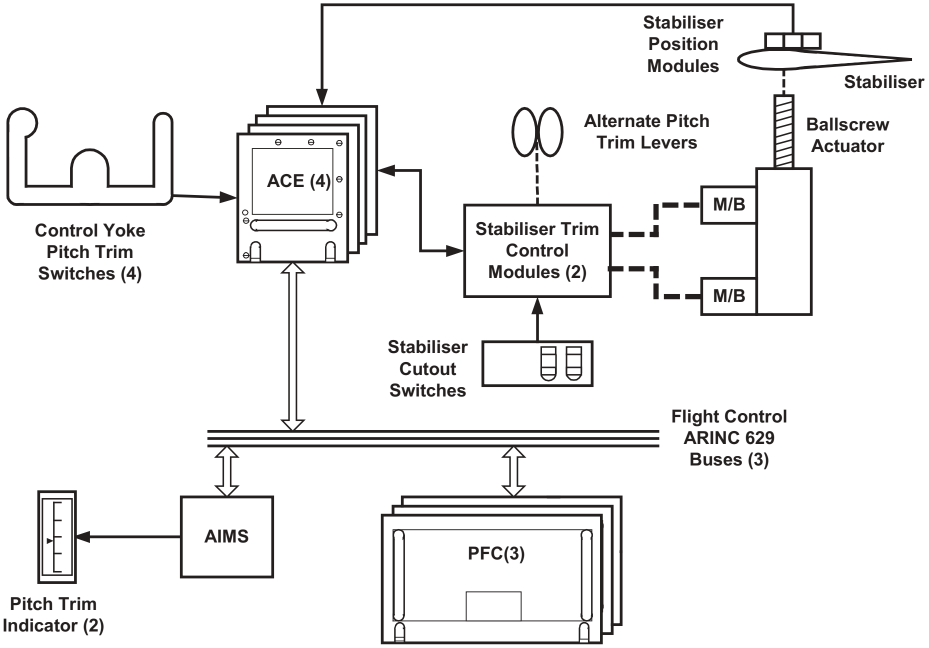
Fig. 9.18 *Boeing 777 – pitch elevator control*

Boeing 777 pitch stabilizer control

The pitch stabilizer is controlled independently of the elevator, though commands are still routed via the ACE units. In this case, pitch trim switches (4) located on the control yoke permit the pilot to select pitch trim inputs into the system. Outputs from the ACEs pass to the Stabilizer Trim Control Modules (STCMs) (2) which control hydraulically powered motor/brakes fed from two different hydraulic systems. These hydraulic units drive the stabilizer screw-jack or ball-screw actuator which repositions the stabilizer. Stabilizer position modules (3) feed back the stabilizer position to the ACEs.

Alternate pitch trim levers (2) and stabilizer cut-out switches (2) also interface with the STCMs, enabling the flight crew to isolate and overcome runaway pitch trim conditions. The alternate pitch trim levers are mechanically connected. The aircraft pitch trim is fed via the AIMS to be displayed on two pitch trim indicators on the flight deck.

Fig. 9.19 *Boeing 777 – pitch stabilizer control*



Boeing 777 roll control

The control yoke is mechanically connected to roll force transducers (2), roll position transducers (2), and aileron trim and feel and centring units. The control yoke is also mechanically connected to spoiler PCUs 4 and 11 to provide a direct roll control mechanical link. The AFDCs can back-drive the mechanical assembly, as before, via two back-drive actuators to ensure that autopilot and pilot demands are harmonized.

Pilot or system roll demands are fed to the PCUs via the ACE units. In total, the roll control channel inputs are provided to the following PCUs:

- Left and right flaperons (performing aileron functions) (4).

- Left and right ailerons (4).
- All spoilers excluding spoilers 4 and 11 (12).

In addition to providing roll control by operating left and right wing spoilers differentially, the collective spoiler control for speed brake and ground spoiler functions is effected by this control channel. The speed brake selections are made using a separate control lever located on the centre console that is mechanically connected to the speed brake lever sensors (4), and these sensors connect directly to the ACEs.

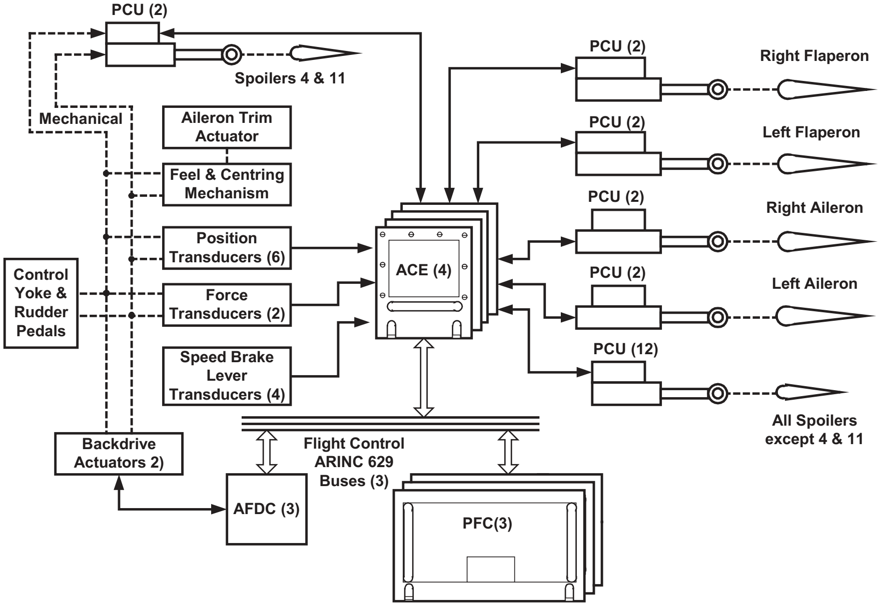


Fig. 9.20 *Boeing 777 – roll control*

Boeing 777 yaw control

The yaw command from the pilot's rudder pedals are mechanically connected to the pedal position transmitters (4) and via a feel and centring unit to the rudder feel unit. Pilot yaw demands are therefore fed into the ACEs and thence to the three rudder PCUs. As for the other channels, two back-drive units permit the mechanical assemblies to be backed off by the yaw channel AFDC commands.

A rudder trim selector allows the flight crew to apply rudder trim via the ACE units. Rudder trim indication is fed via the AIMS to the rudder trim indicator on the flight deck.

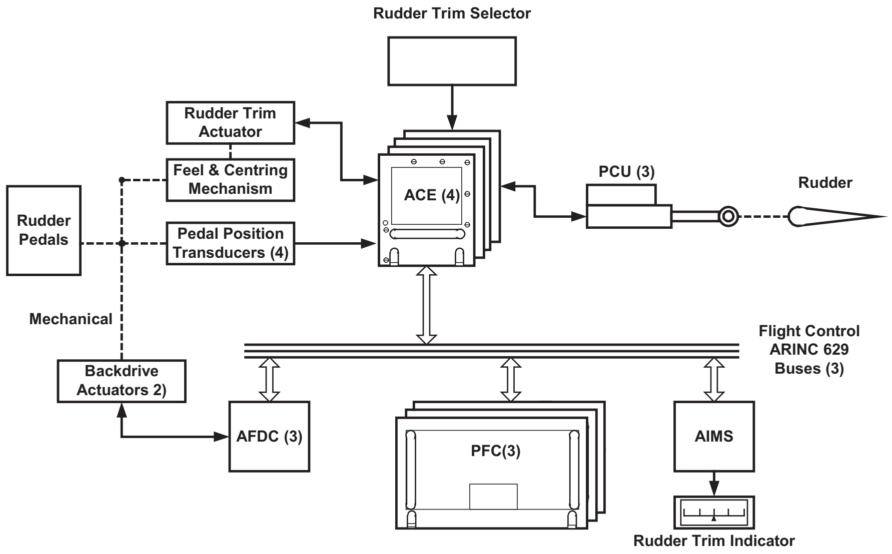


Fig. 9.21 *Boeing 777 – yaw control*

Boeing 777 mechanical reversion

In the unlikely event, given all the other safeguards, that the normal secondary and direct electrically controlled means are not available to the pilot, then the aircraft can be flown in a direct mechanical link mode. In the mechanical link mode, rudimentary pitch and roll control is available as shown in Fig. 9.22.

For mechanical link roll control, the control yoke is connected via a conventional cable and pulley system to provide mechanical inputs to the PCUs for spoilers 4 and 11,

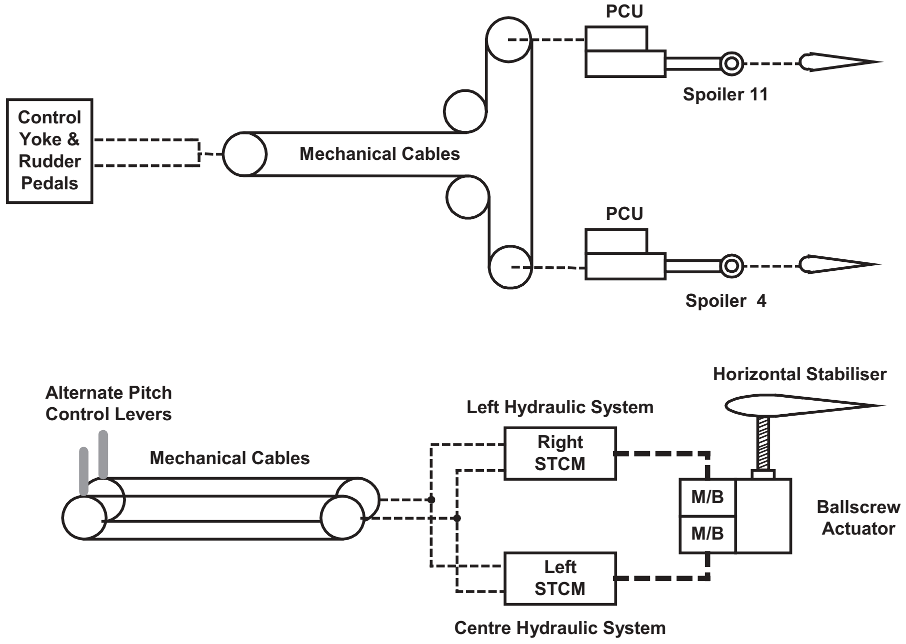


Fig. 9.22 *Boeing 777 – mechanical link*

the mid-spoilers for left and right wings respectively. Mechanical link for the pitch channel is achieved by alternate pitch control levers (2) connected by cables and pulleys to provide dual inputs into each of the STCMs. Each STCM controls a separate hydraulic power supply (left and centre) to the ball-screw actuator motor/brake assemblies.

Top-level Boeing 777 PFCS overview

The B777 PFCS is outlined at a system level in Fig. 9.23. The drawing shows the PFCS along the top, together with the three CDUs. Most of the sensors are shown along the bottom of the diagram, and most of these have already been described in detail in Chapter 6. The PFCS system units are interconnected by three A629 flight control data buses: left, centre, and right. In total there are 76 A629 couplers on the flight control buses.

These units interface in turn with the flight control and feel actuators in accordance with the scheme shown in the centre of Fig. 9.23 – in all a total of 31 actuators. References (**2**) and (**3**) are technical papers explaining the operation of the B777 PFCS in more detail.

The flight control computations are carried out in the primary flight computers (PFCs) shown in Fig. 9.23. The operation of the PFCs has been briefly described earlier in the chapter but will be recounted and amplified in this section.

Each PFC has three A629 interfaces with each of the A629 flight control buses, giving a total of nine data bus connections in all. These data bus interfaces and how they are connected and used form part of the overall Boeing 777 PFCS philosophy. The three

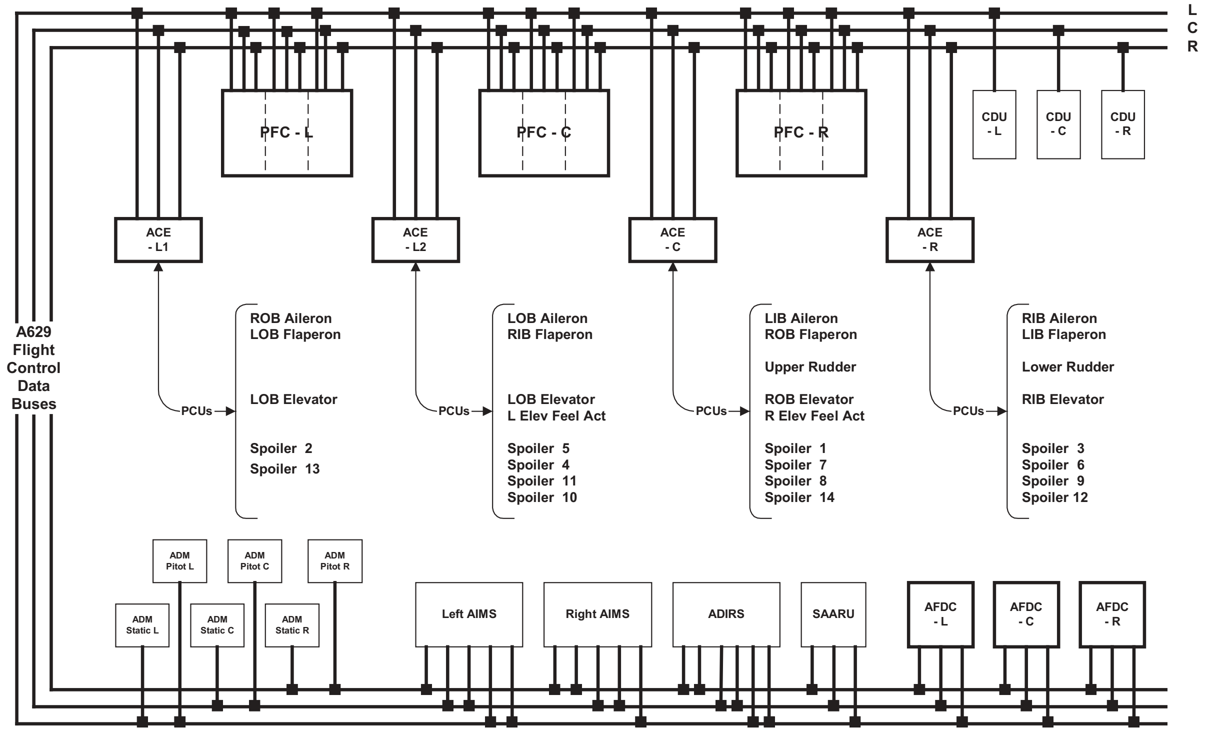


Fig. 9.23 *B777 simplified Primary Flight Control System (PFCS)*

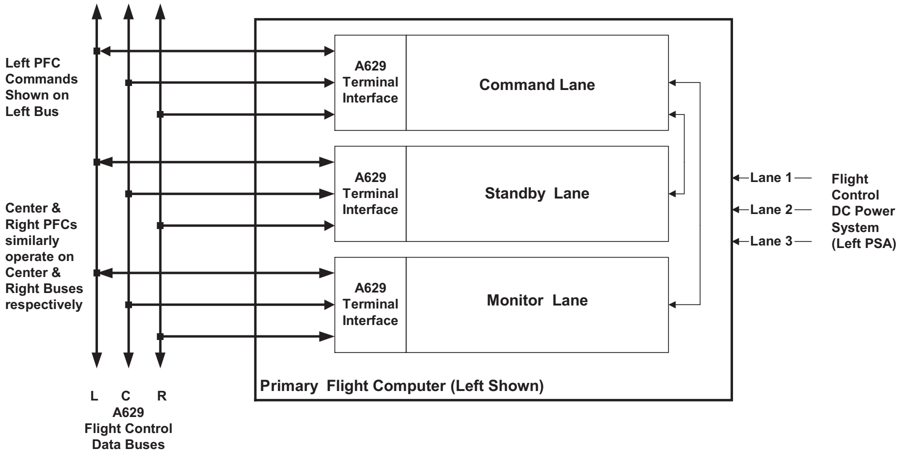


Fig. 9.24 *B777 Primary Flight Computers (PFCs)*

active lanes within each PFC are embodied in dissimilar hardware. Each of the three lanes is allocated a different function as follows:

1. PFC command lane. The command lane is effectively the channel in control. This lane will output the flight control commands on the appropriate A629 bus; e.g. PFC left will output commands on the left A629 bus.
2. PFC standby lane. The standby lane performs the same calculations as the command lane but does not output the commands on to the A629 bus. In effect, the standby lane is a ‘hot standby’, ready to take command in the event that the command lane fails. The standby lane only transmits cross-lane and cross-channel data on the A629 data bus.
3. PFC monitor lane. The monitor lane also performs the same calculations as the command lane. The monitor lane operates in this way for both the command lane and the standby lane. Like the standby lane, it only transmits cross-lane and cross-channel data on the A629 data bus.

Figure 9.24 indicates that, on the data bus, each PFC will only transmit aircraft control data on the appropriate left, centre, or right A629 data bus. Within each PFC the command, standby, and monitor lanes will operate as previously described, and only the command channel – shown as the upper channel in Fig. 9.24 (left command channel for left PFC) – will actually transmit command data.

Within this PFC and A629 architecture:

- Cross-lane comparisons are conducted via the like bus (in this case the left bus).
- Cross-channel comparisons are conducted via the unlike buses (in this case the centre and right buses).

This use of standard A629 data buses to implement the flight control integration and to host the cross-lane and cross-channel monitoring is believed to be unique in flight control. There are effectively nine lanes available to conduct the flight control function. In the event that a single lane fails, then only that lane will be shut down. Subsequent loss of a second lane within that channel will cause that channel to shut down, as simplex control is not permitted.

The aircraft may be operated indefinitely with one lane out of nine failed. The aircraft may be dispatched with two out of nine lanes failed for 10 days. The aircraft may be operated for 1 day with one PFC channel wholly inoperative.

Autopilot Flight Director Systems (AFDSs)

The workload in modern flight decks, together with the precision with which the aircraft needs to be flown, makes an autopilot an essential part of the avionics system. The generic portrayal of an autopilot channel is shown in Fig. 9.25. This comprises a forward loop where autopilot inputs are processed according to a defined set of control laws. The autopilot demands derived from the control laws drive servomotors or actuators which impart these demands to the flight control surfaces. Inner loop feedback is provided which ensures that the servomotors achieve and maintain the demanded position with the desired response rate and stability. To enhance performance, position, rate, and acceleration, loop feedback may be required, and many control systems will utilize all forms of feedback.

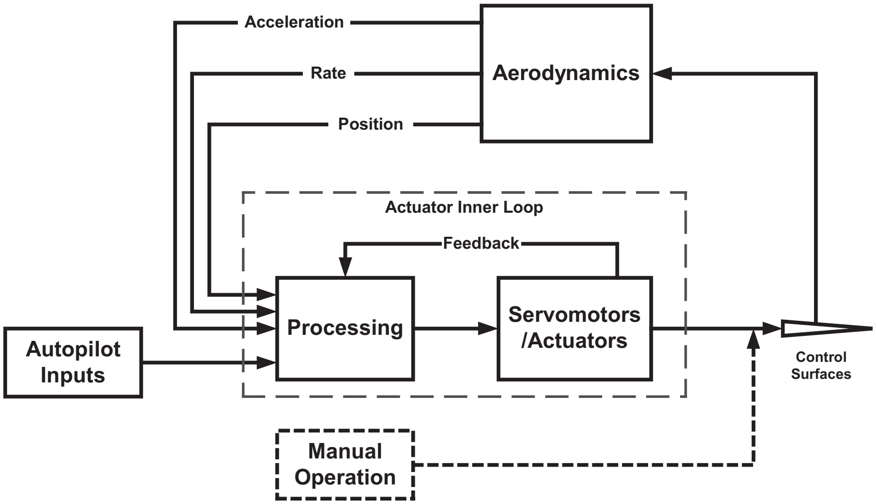


Fig. 9.25 *Typical AFDS control loop*

A typical digital autopilot system is depicted in Fig. 9.26. This figure shows a system with two digital flight control computers, each of which has dual lanes of operation (lane A and lane B). Relevant sensor and input data are fed from the units shown on the left side of the diagram. These include:

- Captain and first officer's FMS Multifunction Control and Display Units (MCDUs).
- Dual Central Air Data Computers (CADC).
- Dual vertical gyros providing a pitch attitude reference.
- Dual directional gyros furnishing a directional attitude reference.
- Dual VOR/ILS units to provide radio navigational and approach guidance.
- Dual radar altimeters.
- Two accelerometer units to provide aircraft acceleration data for advanced control loop closure.

The functional modes of the two digital computers are selected by means of a flight guidance control panel which enables autopilot modes to be engaged and various speed, altitude, and heading datums to be adjusted and displayed.

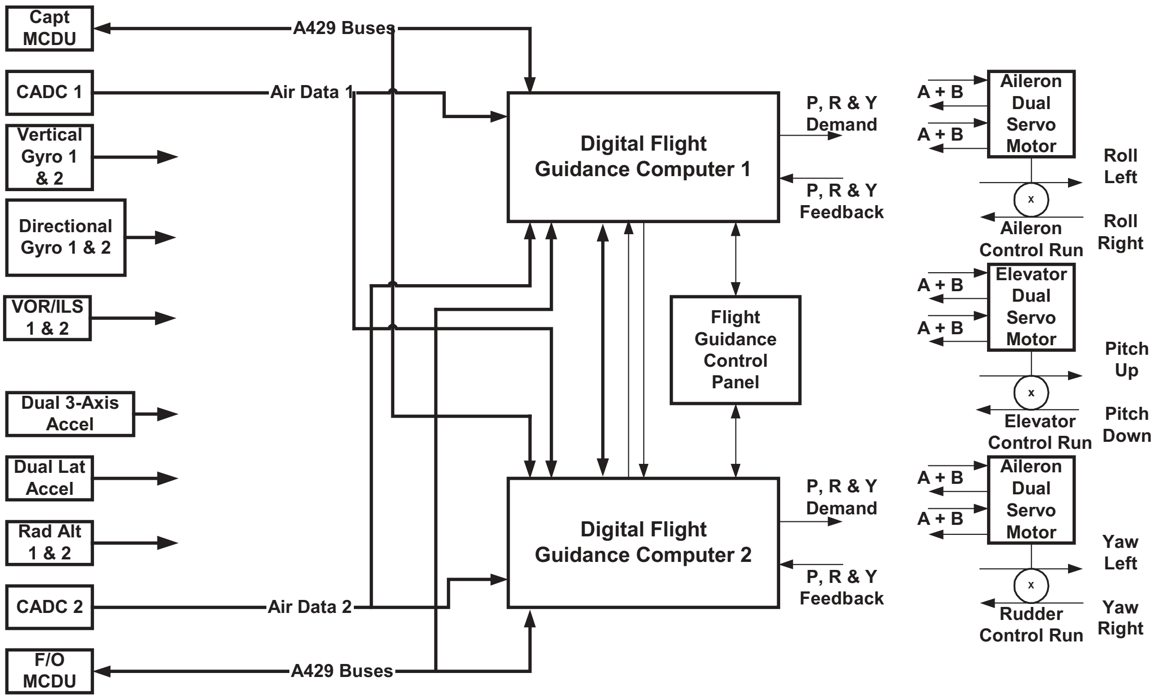


Fig. 9.26 Typical autopilot sytem

Both computers output two lanes of autopilot demand in the pitch, roll, and yaw channels according to the autopilot mode(s) selected. These demands are fed to an appropriate dual servomotor which enables the autopilot demands to be coupled into the appropriate cable and pulley flight control run in pitch, roll, and yaw. Feedback from the servomotor enables the autopilot control laws to ensure that the demands are being fully satisfied.

Figure 9.27 illustrates how autopilot demands and position and rate feedback are duplicated for lanes A and B and fed to a dual servomotor. Each lane uses position and rate feedback, and both flight guidance computers can drive both lanes. These dual demands are summed via a series of gearboxes to a single rotary demand shown at the bottom of the servomotor. The output from the dual servomotor to the appropriate flight control run drives through two clutch mechanisms. The first is the engage clutch which closes and engages the autopilot when the necessary engagement logic conditions have been satisfied. The second is a slip clutch which enables the control run to be moved if the servomotor jams, once an established breakout force has been exceeded.

In order to describe the operation of various autopilot channels, the following examples are outlined:

- Pitch channel autopilot control.
- Yaw channel autopilot control.
- Autopilot speed control (autothrottle).

Understanding the operation of these channels forms the basis of many of the more complex modes of operation, where two or more modes may be used to control the aircraft trajectory.

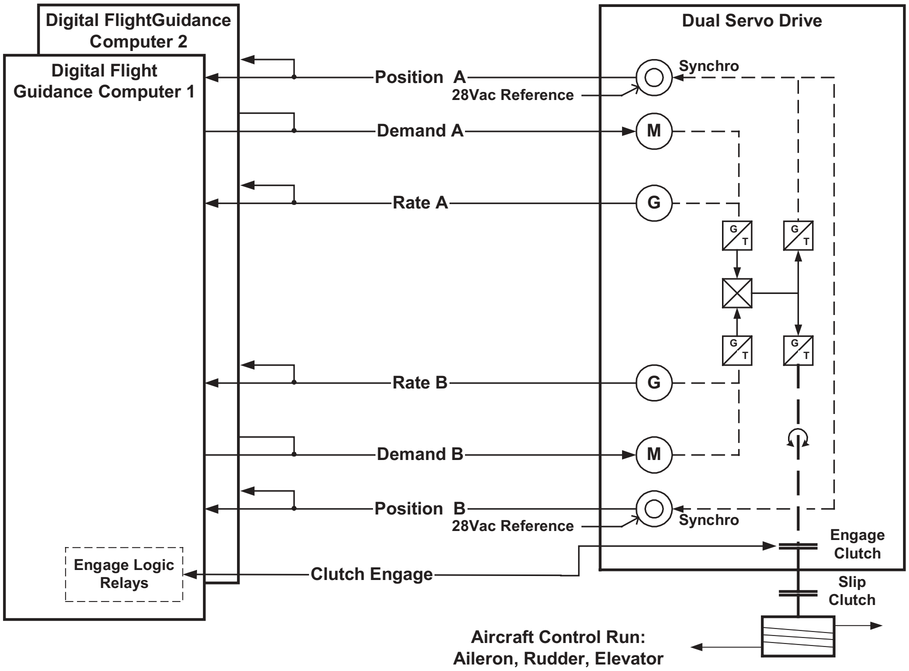


***Fig. 9.27** Dual servomotor operation*

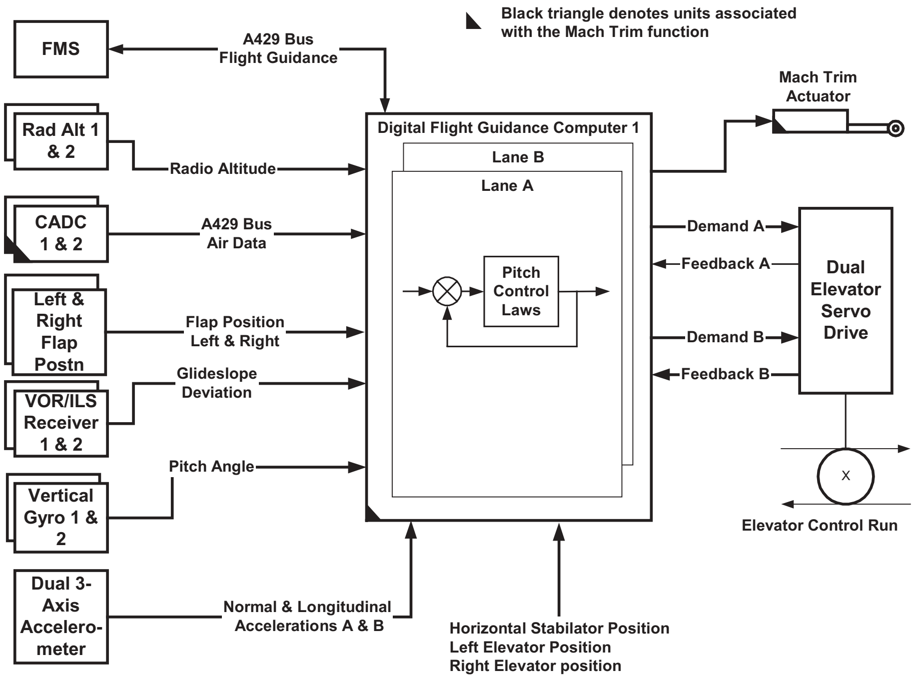
Autopilot pitch control

A typical autopilot pitch control is shown in Fig. 9.28. This is typical of the Boeing Super 80 (previously McDonnell Douglas MD-80) family of aircraft and many other aircraft designed in the 1980s. Each of the dual-redundant Digital Flight Guidance Computers (DFGCs) contains the pitch control laws, and the sensors at the right provide information for the execution of these control laws. These sensors are:

- Radar altimeters 1 and 2 providing radar altitude.
- Central Air Data Computers (CADCs) 1 and 2 providing air data for the Mach trim function.
- Left and right flap position sensors providing flap position information.
- VOR/ILS receivers 1 and 2 for ILS glide path guidance.
- Vertical gyros 1 and 2 providing pitch attitude.
- A dual three-axis accelerometer package providing normal acceleration.
- Left and right elevator and stabilator (stabilizer) position.
- FMS guidance is also provided via A429 data links.

The outputs from the pitch control laws are fed to the pitch channel dual servomotor which inserts commands into the pitch channel flight control run and therefore causes the required change in pitch attitude. The channel also feeds commands to the Mach trim actuator which allows pitch trim changes due to increase in Mach to be automatically counteracted.

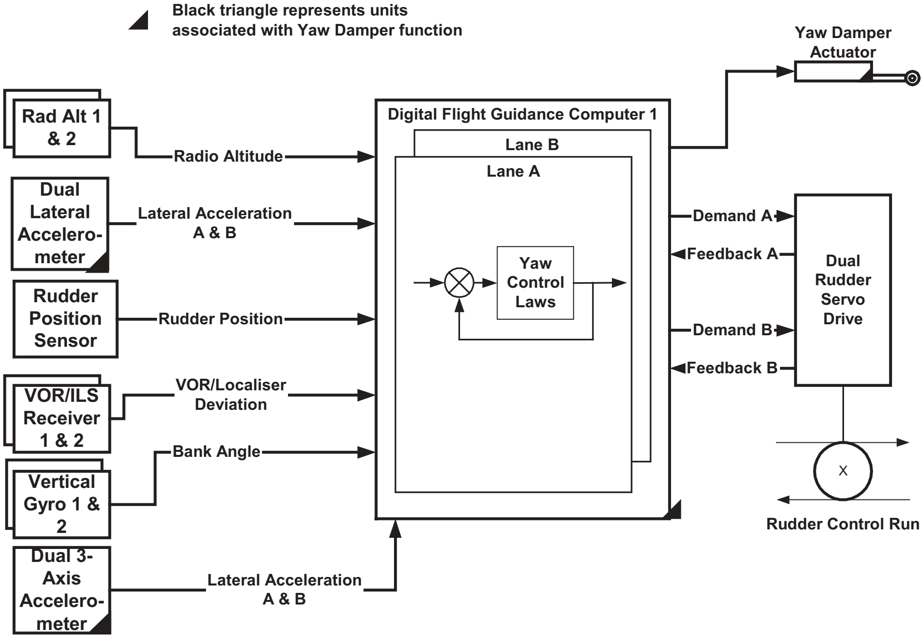
Fig. 9.28 *Autopilot pitch control*



Autopilot yaw control

A similar arrangement exists for the yaw autopilot shown in Fig. 9.29. In this channel the dual lateral accelerometer and dual three-axis accelerometers provide information for the yaw damper actuator. As for the pitch channel, yaw demands are fed to the dual

Fig. 9.29 *Autopilot yaw control*



rudder servomotor which injects commands into the rudder flight control run.

In the yaw channel, VOR and ILS localizer deviation information can be coupled into the autopilot to provide heading steering to a VOR beacon or the ILS localizer as appropriate.

Autopilot speed control

As well as controlling the aircraft pitch, roll, and yaw motion by coupling into the aircraft flight control runs, most aircraft have an autothrottle capability such that the aircraft speed may also be maintained at a fixed datum as required. Figure 9.30 shows all those sensors whose inputs are necessary for the operation of a speed control mode.

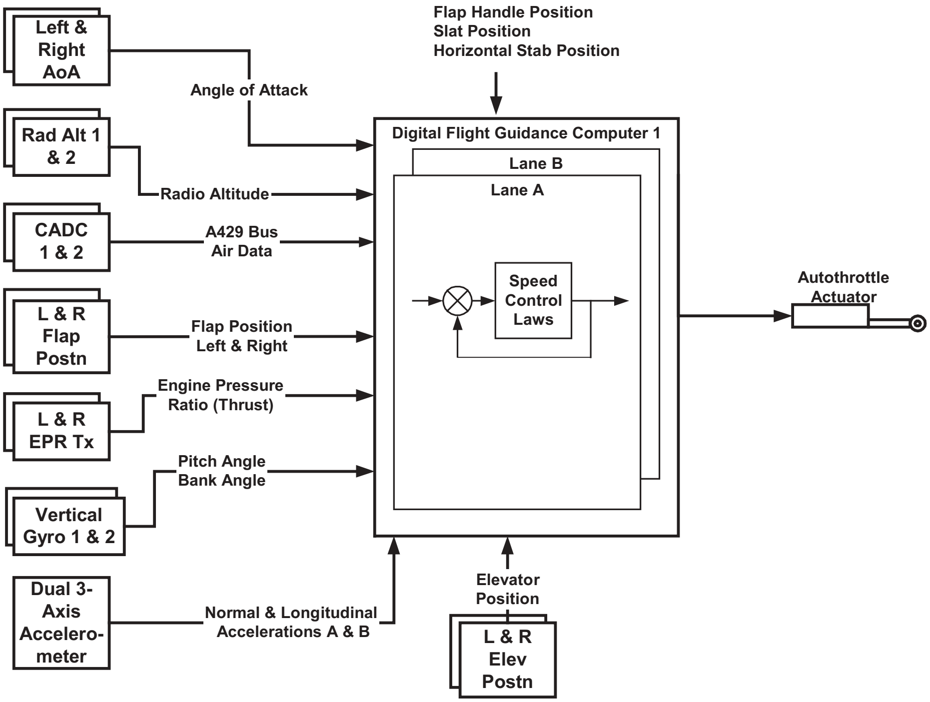


Fig. 9.30 Autothrottle operation

The calculations associated with the autothrottle control laws are performed within lanes A and B of each digital flight guidance computer, but in this case the outputs are fed to an autothrottle actuator which is able to alter the position of the engine throttle linkage. The resultant changes in engine power enable the autopilot to control the aircraft speed to the required speed datum.

Autopilot modes

There are many autopilot modes that can be used singly or in combination – a modern autopilot will possess some 20–30 modes of operation. Autopilot modes are usually selected by means of a Flight Mode Selector Panel (FMSP) located centrally on the glare-shield (See Fig. 9.31). The function of the Flight Mode Selector Panel is to:

- Provide a means by which autopilot or flight director may be selected.
- Enable engagement of the major autopilot modes by providing mode selection

buttons for key modes of operation.
- Provide display windows for important autopilot parameters and datums.
- Enable reference speed or datum adjustment.

The example given has the following features: speed window (knots or Mach); heading window (degrees), vertical speed window (ft/s), altitude window (ft). A number of mode selection buttons are shown that allow the pilot to select these and other modes in conjunction to provide automatic control as required throughout various phases of flight. As has already been mentioned, the FMS may be coupled into the autopilot to provide three-dimensional LNAV plus VNAV guidance to enable the aircraft to fly the necessary procedural profiles as required in busy terminal areas (see Chapter 8).

An Instinctive Cut-Out (ICO) button will also be provided among the pilot's controls. The purpose of this device is to allow the pilot rapidly to disengage the autopilot in an emergency such as an uncommanded pitch movement or trim runaway.

The different modes of autopilot operation each require a suite of sensors providing the necessary data such that the relevant autopilot control laws may be performed. In several modes of operation, dual sensors may be required. Table 9.2 is a simplified matrix of the sensors required for various modes of autopilot operation.

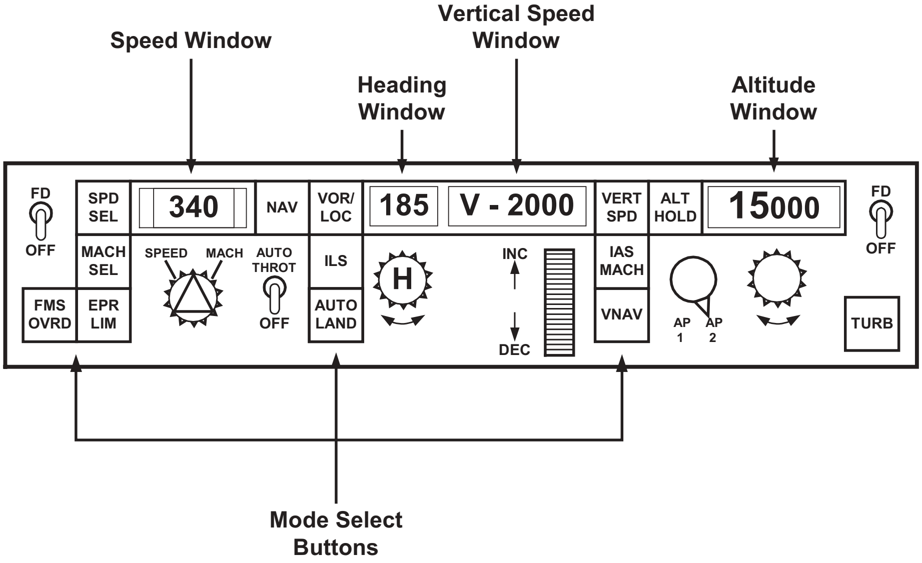


Fig. 9.31 Typical Flight Mode Selector Panel

Table 9.2 Simplified autopilot modes and sensor matrix

	Roll/yaw modes						Pitch modes						Land modes	
	Heading hold	Heading select	VOR	Localizer	Take-off	Go-around	Glide slope	Mach hold	IAS hold	Altitude preselect	Altitude hold	Vertical speed	Localizer	Glide slope
Dual three-axis accelerator A					◆	◆		◆	◆					
Dual three-axis accelerator B					◆	◆		◆	◆					
L slat valid					◆	◆								
R slat valid					◆	◆								
Altitude (corrected)					◆	◆		◆	◆	◆	◆	◆		
Altitude (uncorrected)					◆	◆				◆				
Mach					◆	◆		◆	◆			◆		
Indicated airspeed					◆	◆			◆					
Altitude rate										◆	◆	◆		
Compass 1	◆	◆	◆	◆										
Compass 2	◆	◆	◆	◆										
VG 1 (roll and pitch)	◆	◆	◆	◆	◆	◆	◆	◆	◆	◆	◆	◆	◆	◆
VG 2 (roll and pitch)	◆	◆	◆	◆	◆	◆	◆	◆	◆	◆	◆	◆	◆	◆
GS 1 or GS 2							◆							◆

In this table the following modes are addressed:

- Roll/yaw modes: heading hold, heading select, VOR, localizer.
- Pitch modes: take-off, go-around, glideslope, Mach hold, IAS hold, altitude preselect, attitude hold, vertical speed.
- Land modes: localizer, glide slope.

For each autopilot mode the required sensors are also shown. In the event that any of these sensors are unavailable, the autopilot engagement logic will prevent the selection of the associated mode(s) that require that sensor information. In the cruise while flying airway routes using VOR beacons it is likely that the flight crew will have VOR selected, together with altitude hold. This combination of modes will ensure that the aircraft flies directly to the next VOR (waypoint) at the altitude for which clearance has been obtained.

Integrated autopilot system – Boeing 777 AFDS

The system previously described is typical of the type of autopilot system introduced in the mid-1980s. Since then, more integrated solutions have been introduced. The Boeing 777 AFDS is typical of a more integrated system that entered service in the mid-1990s.

The autopilot function of the B777 PFCS is undertaken by the three AFDCs: left, centre, and right, as shown in Fig. 9.32. The AFDCs have A629 interfaces to the respective aircraft systems and flight control data buses. In other words, the left AFDC

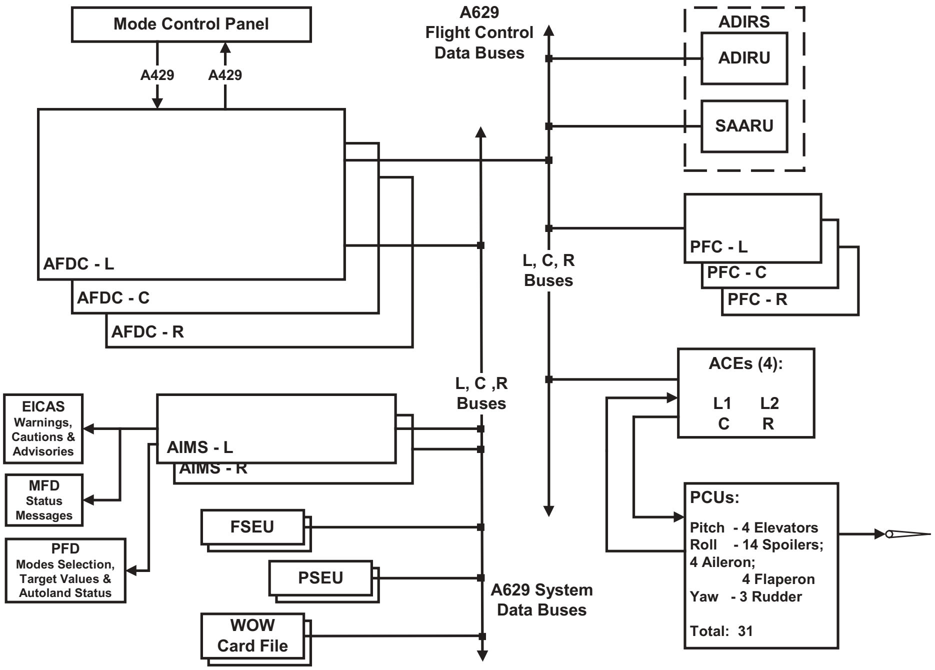


Fig. 9.32 *B777 Autopilot Flight Director Computers (AFDCs)*

will interface to the left A629 buses, the centre AFDC to the centre buses, and so on.

The AFDCs are connected to the aircraft system buses to exchange data with:

- The left and right AIMS cabinets.
- The Proximity Switch Electronic Units (PSEUs) in the landing gear system.
- The Flap/Slats Electronic Units (FSEUs).
- The Weight-On-Wheel (WOW) card files.

All of these units provide data to the AFDCs about the aircraft configuration that is needed for autoflight computations.

Information relating to the AFDS is displayed to the flight crew after being processed by the Airplane Information Management System (AIMS). Key displays are:

- Primary flight display.
- Navigation display.
- Engine Indication and Crew Alerting System (EICAS) displays.

Refer to Chapter 9.

On the flight control data buses the AFDCs interface to:

- ADIRS and SAARU for attitude and air data information.
- PFCs to input autoflight demands.

The other functions on the flight control buses have been described earlier.

Autoland

As the importance of maintaining airline schedules in adverse weather conditions has increased, so too has the importance of autoland. At its most refined, autoland allows the aircraft to land in virtually zero-visibility conditions. The various stages of autoland are defined by criteria summarized in Fig. 9.33. These criteria may be summarized as:

- Category I (Cat I). This relates to a Decision Height (DH) of not less than 200 ft and a visibility of not less than 2600 ft [or not less than 1800 ft Runway Visual Range (RVR)], where the airfield/runway is equipped with dedicated measuring devices).
- Category II (Cat II). A DH of not less than 100 ft and an RVR of not less than 1200 ft.
- Category IIIA (Cat IIIA). A DH of less than 100 ft and an RVR of not less than 700 ft. Also described as ‘see to land’.
- Category IIIB (Cat IIIB). A DH of less than 50 ft and an RVR of not less than 150 ft. Also described as ‘see to taxi’.

The introduction of a Cat III autoland capability poses further constraints for the aircraft systems. Simply put, these are:

- Three independent AC and DC electrical power channels.
- Three independent flight control channels.
- High-integrity approach guidance – ILS or MLS. The categorization of the ground -based equipment is equally as important as the aircraft mounted equipment.

It should also be noted that the GPS augmentation systems will also be capable of providing autoland quality flight guidance once the systems are fully operational. It is intended that the Local Area Augmentation System (LAAS) and Wide Area Augmentation System (WAAS) GPS augmentation systems will be capable of providing Cat II/III and Cat I guidance respectively (see Chapter 8). References (**4**–**6**) are those Advisory Circulars (ACs) that address automatic flight control issues.

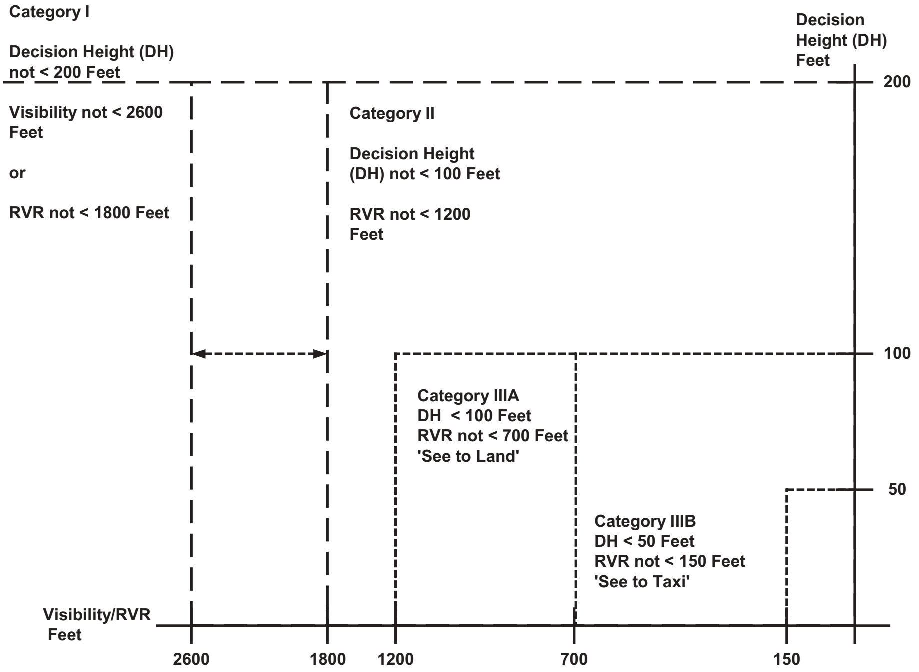


Fig. 9.33
Categorization of automatic approaches

Flight Management System

The final element in the overall flight guidance equation is the FMS. The function of the FMS is fully described in Chapters 8 and 12. To understand the function of the FMS without reference to those chapters, a brief overview is given below.

This system comprises dual-redundant FMS computers and CDUs as shown in Fig. 9.34. Whereas the FMS computers are shown as stand-alone four Modular Concept Units (MCUs), most modern applications could now be hosted on a single card, such has been the degree of integration of microelectronics.

The FMS receives inputs from the following systems:

- INS.
- Navigation aids.
- GPS.
- Air data system.
- Fuel sensors and clock.

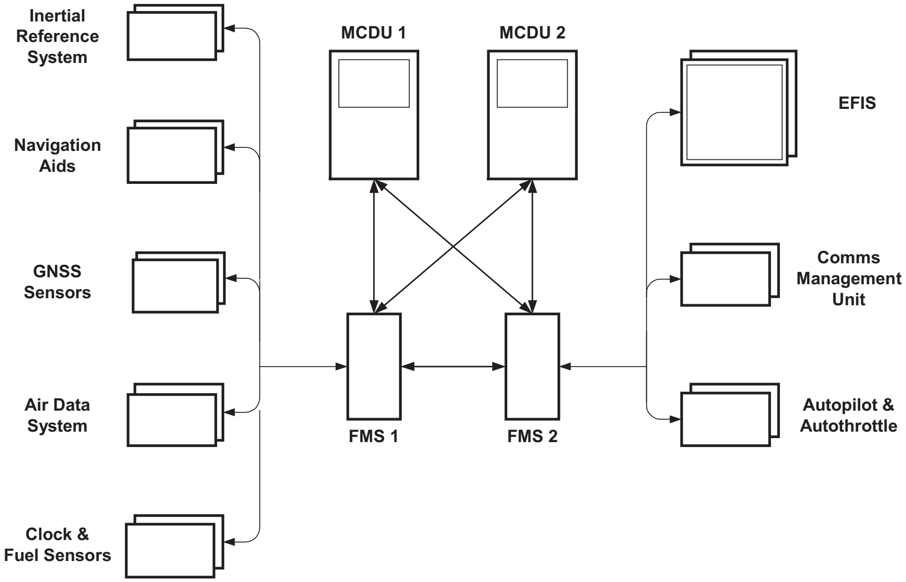
In a modern high-performance system, most of these sensors will be dual redundant.

The FMS outputs data to:

- EFIS displays.
- Communications management unit.
- Autopilot and autothrottle.

The FMS provides the essential integration function of all these units to provide the overall function of controlling the flight path of the aircraft. This does not merely include steering the aircraft from waypoint to waypoint. The FMS also controls the tuning of all the appropriate aircraft receivers to navigation beacons and communications frequencies via the communications control units. The flight plan that resides within the FMS memory will be programmed for the entire route profile, taking

Fig. 9.34 *Typical FMS architecture*



into account all eventualities, including aircraft diversion. More advanced capabilities include three-dimensional navigation and the ability to adjust the aircraft speed to reach a waypoint within a very small time window (typically ±6s). The various levels of performance and sophistication are summarized in Table 9.3.

Table 9.3 Summary of FMS capabilities

Function	Capability
LNAV	The ability to navigate laterally in two dimensions
VNAV	The ability to navigate laterally in two dimensions, plus the ability to navigate in the vertical plane. When combined with LNAV this provides three-dimensional navigation
Four-dimensional navigation	The ability to navigate in three-dimension plus the addition of time constraints for the satisfaction of time of arrival at a waypoint
Full performance based navigation	The capability of four-dimensional navigation together with the addition of an aircraft-specific performance model. By using cost indexing techniques, full account may be taken of the aircraft performance in real time during flight, allowing optimum use of fuel and aircraft energy to achieve the necessary flight path
Future Air Navigation System (FANS)	The combination of the full performance model together with all the advantages that FANS will confer, eventually enabling the concept of 'free flight'

The essential crew-to-FMS interface is achieved through the CDU. This provides a very powerful interface tool using mode and function keys, soft keys, and alphanumeric keys which enable the crew to use a powerful menu-driven system. The CDU also displays key information relating to navigation, radio navigation, and communications frequencies, time to waypoints and destination, and navigation accuracy. As the functional capabilities of the FMS increase, as shown in Table 9.3, so the display of the relevant data becomes more important.

Future systems – Airbus A380 FBW

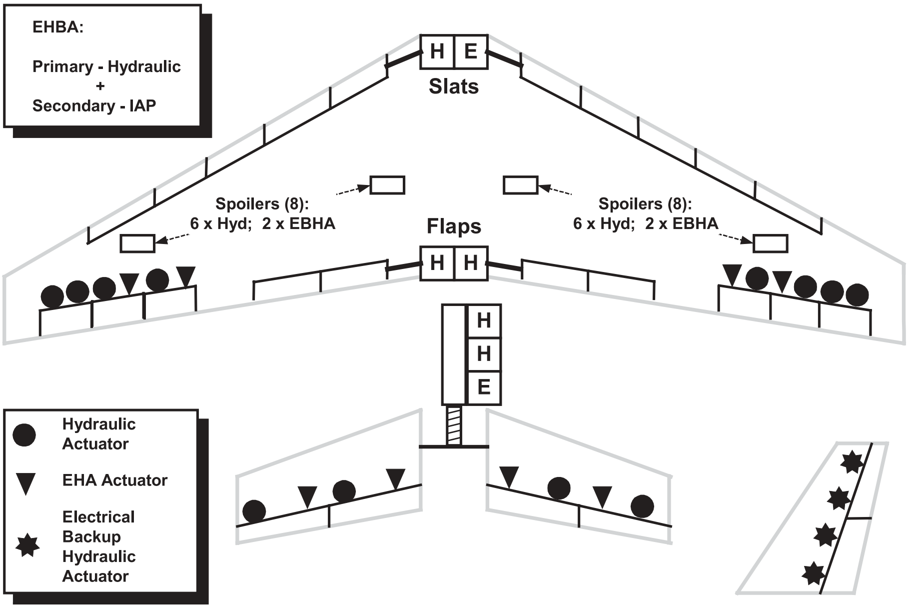
The new Airbus A380 will have a FBW system that is more advanced than the previous A320 and A330/340 families. The mechanical reversion used in previous systems will be replaced by a direct electrical link. The aircraft is shown in Fig. 9.35.

The top-level architecture of the A380 flight control system is shown in Fig. 9.36. The aircraft primary flight control surfaces are powered by a range of different actuator types:

- Conventional hydraulic ram actuators: used for ailerons (8) and elevators (4) and 12 spoiler sections.
- ElectroHydrostatic Actuators (EHAs): used for ailerons (4) and elevators (4).
- Electrical Back-Up Hydraulic Actuators (EBHAs). These actuators operate in two modes:
  - in a primary mode as a conventional hydraulic actuator,
  - in a secondary mode as a locally electrically powered hydraulic actuator similar to an Integrated Actuator Package (IAP). EBHAs are used for the rudder actuators (4) and four spoiler sections.

Fig. 9.35 Airbus A380 in Singapore Airlines livery (Airbus)

Fig. 9.36 A380 flight control surfaces



- The THS is powered by two hydraulic motors with an electrical back-up.

Secondary flight controls are powered as follows:

- Flaps by dual hydraulic supplies.
- Slats by hydraulic or electrical means.

The main differences of this arrangement from previous Airbus FBW aircraft are:

- The increased number of roll control surfaces – ailerons and spoilers.

- Replacement of the direct mechanical links by electrically signalled EBHAs which are also used for the rudder (4) and spoiler sections (2 per side).

The motive power for this architecture is also novel – two aircraft level hydraulic and electrical systems provide power, as shown in Fig. 9.37.

Firstly, the basic flight control system power architecture contains:

- Hydraulic power channel 1 with four Engine-Driven Pumps (EDPs) driven by engines 1 and 2. This is augmented by two Electrical Motor Pumps (EMPs).
- Hydraulic power channel 2 with four EDPs driven by engines 3 and 4 and also augmented by two EMPs.

This represents a departure from normal practice which is usually to have three (occasionally four) aircraft hydraulic systems. The other novel feature is that the working pressure of the hydraulic systems is 5000 psi against the more usual working pressure of around 3000 psi. These hydraulic channels power all the conventional hydraulic actuators shown in Fig. 9.37. The EMPs may be powered by either the APU generator or a ground power supply.

Secondly, the basic flight control system power architecture contains:

- Electrical power channel 1 powered by three electrical generators, two of which are Variable-Frequency (VF) machines.
- Electrical power channel 2 operating in a similar fashion to channel 1.

The VF generators are novel in that they are rated at a higher power than has conventionally been used for transport aircraft, being rated at 150 kVA per generator as opposed to the previous highest of 120 kVA (Boeing 777 and Boeing 767-400). These electrical channels power all the EHAs and EBHAs shown in Fig. 9.37.

There is no direct mechanical link for reversionary flight control, and the power architecture has been revised compared with previous systems. Nevertheless, the A380

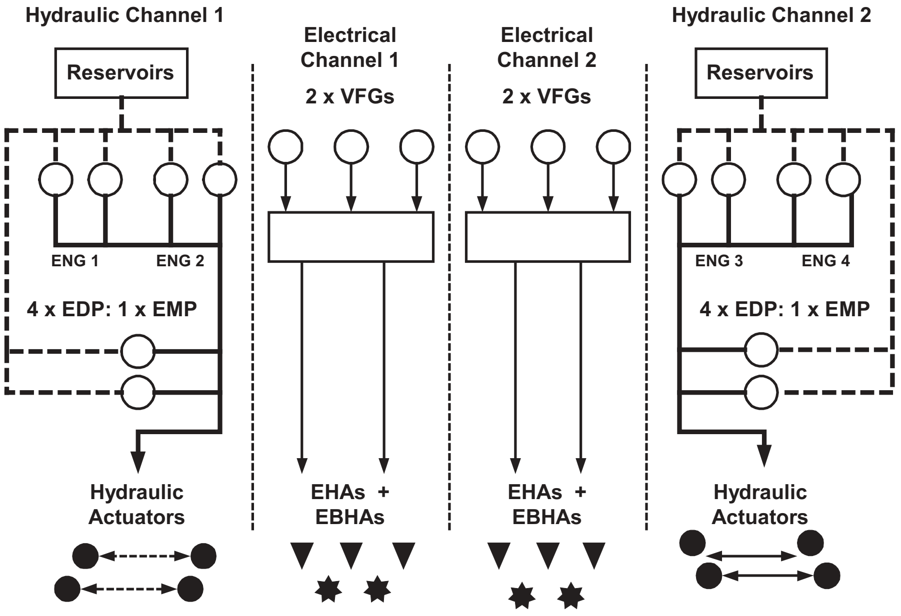


Fig. 9.37 *A380 flight control power*

system provides a considerable degree of dissimilar redundancy, enabling flight criticality requirements to be achieved.

From the control viewpoint, it is expected that the flight control computations will follow similar philosophies to those proven on the A320 and A330/340 families.

Flight Data Recorders

All aircraft of a certain size are mandated to carry a flight data recording system, and the requirements for these systems are becoming steadily more rigorous. Older Flight Data Recorders (FDRs) recorded various aircraft functions, such as vertical acceleration, heading, airspeed, altitude, etc; in analogue form using a stylus and moving oscillographic foil medium comprised of steel or steel alloy. In modern systems the recording medium is ruggedized to withstand shock, fire, and long-term immersion in seawater using digital solid-state electronics memory. In the United States present regulations now dictate that a Digital Flight Data Recorder (DFDR) be used. A Cockpit Voice Recorder (CVR) is used to record crew and ATC conversations using a 'hot mike' located on the flight deck.

The overall system is known as a Digital Flight Data Recording System (DFDRS); this comprises the equipment, sensors, wiring, and other installed equipment necessary to perform the function. Where a dedicated sensor has to be installed to provide the DFDR function, then it forms part of the DFDRS. Where a sensor is already installed for another purpose, e.g. a lateral accelerometer for automatic flight control, it does not comprise part of the system.

The main elements of a DFDRS are:

- DFDR.
- Flight Data Acquisition Unit (FDAU) or a Digital Flight Data Acquisition Unit (DFDAU). The FDAU has the ability to collect, sample, condition, and digitize analogue signals and provide the data to the DFDR in a digital data stream according to the requirements of ARINC 573. The DFDAU has the ability to receive both analogue parameters and digital data streams and convert them to the DFDR digital data format in accordance with ARINC 717.
- Underwater Locating Device (ULD) in the form of a sonar locator beacon to aid the location of the unit underwater.

The number of parameters that the DFDRS is required to record has progressively increased in recent years. In essence, all aircraft with provision for carrying ten passengers or more have to have a DFDRS fitted in accordance with reference (**6**). The key provisions of reference (**7**) relate to the standard of DFDRS to be fitted to a compliance schedule depending upon the date of aircraft manufacture:

- All aircraft manufactured before 11 October 1991 had to be retrofitted with a system capable of recording 34 parameters by 20 August 2000. All aircraft manufactured between 11 October 1991 and 20 August 2000 had to be similarly equipped.
- All aircraft manufactured between 20 August 2000 and 19 August 2002 were mandated to be fitted with a system capable of recording 57 parameters.
- All aircraft manufactured after 19 August 2002 must have a system capable of recording 88 parameters.

Refer to ED 55 (**8**).

DFDRs, known colloquially in the media as 'black box recorders', are actually painted bright orange to aid recovery at a crash scene. The reason for increasing the number of recorded parameters was because the causes of many accidents were not being identified owing to the paucity of data, and the schedule outlined above is an attempt to redress this shortcoming. There is still a problem relating to powering of the DFDRS; where electrical power is lost to part or all of the system, so too are the recorded data, and consideration is being given to alternative electrical power sources to prevent this happening.

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CHAPTER 10

Engine and Utility Systems

There are a number of aircraft systems that contribute to the safe passage of the aircraft and to the comfort of the passengers. The continuous performance of these systems without failure is essential to the 'image' of the aircraft type and the airline operator. These are systems that, by their failure to perform, lead to delayed or cancelled flights which damage the operator in cost of repair and customer management. This can lead to customers choosing an alternative airline. Some aircraft types have been unsuccessful because of their perceived unreliability or lack of comfort and passenger facilities. The success of many Boeing and Airbus types has been achieved by customer acceptance. This chapter outlines the function of these systems and the avionic control systems associated with them. These systems are:

- Engine systems.
- Air and environmental systems.
- Fuel systems.
- Hydraulic systems.
- Central maintenance systems.
- In-Flight Entertainment (IFE) systems.

Engine systems

In spite of the fact that engine control systems have become very comprehensive in maintaining operating conditions at the most economic or the highest performance, depending on the application, there is still a need to provide the pilot with an indication of certain engine parameters. Under normal conditions the pilot is interested in engine condition only at engine start or when something goes wrong. The engine control system, with its monitoring and warning capability, should inform the pilot when something untoward does happen. However, there may be circumstances when human intuition wins the day.

During engine start, the pilot monitors (and checks with his copilot in a multicrew aircraft) that start progresses satisfactorily with no observed sluggish accelerations and no low oil pressures or overtemperatures. Much of this monitoring involves pilot familiarity with the aircraft type and engine type, obtained over many starts. The crew may accept certain criteria that an automatic system would not.

During normal operation the control system should provide sufficient high-integrity observation by self-monitoring and by checking certain parameters against preset values. In this way the system can monitor accelerations, rates of change, value exceedances, and changes in state and issue the necessary warning using experience and judgement.

Until recently, all aircraft had at least one panel dedicated to engine instruments. These were in view at all times and took the form of circular pointer instruments, or occasionally vertical strip scales, reading such parameters as:

- Engine speed – NH and NL.
- Engine Gas Temperature (EGT).
- Engine Pressure Ratio (EPR).
- Engine vibration.
- Thrust (or torque).

In modern aircraft cockpits the individual indicator has largely given way to an integrated engine display, either an Engine Indication and Crew Alerting System (EICAS) or, on Airbus aircraft, Electronic Centralised Aircraft Monitor (ECAM). With such a system, any information can be shown in any format, in full colour, at any time. This facility is often exploited to ensure that the pilot is only given the information that is essential for a particular phase of flight. This means that engine displays may occur on a single screen or page that is automatically presented to the pilot at certain times, say starting, take-off, and landing, but may be hidden at all other times. Provision is made for the pilot to select any page so that the engine can be checked from time to time, and an engine warning will automatically trigger the engine page to appear.

Engine indications are obtained from the same type of sensors and transducers that provide the inputs to the control system, as described earlier. However, for integrity reasons at least two independent sources of signal are required – one (or more) for control, another for the indicator. For example, the engine speed signal will be obtained from two separate coils of a common speed sensor. This guards against a common mode failure that would otherwise affect both the control system and the indication system.

Engine control on a modern civil aircraft

A typical civil engine is shown in Fig. 10.1. Most are twin-shaft engines with Low-Pressure (LP) and High-Pressure (HP) shafts. Some Rolls-Royce engines such as the RB211 and Trent family are triple-shaft engines with LP, Intermediate-Pressure (IP), and HP shafts. A high proportion of air bypasses the engine core on a modern gas turbine engine; the ratio of bypass air to engine core air is called the bypass ratio. The bypass ratio for most civil engines is in the region of 4:1–5:1.

Most modern civil engines use a Full-Authority Digital Engine Control system (FADEC) mounted on the fan casing to perform all the functions of power-plant management and control. Effective control of engine parameters is a contributory factor to achieving stress-free operation of the engine. This, together with major advances in materials technology and engine build quality, has led to high availability of the commercial aircraft turbofan engine. A highly simplified diagram showing all the functions to be performed on the aircraft's large, high-bypass engines is illustrated in Fig. 10.2.

The key areas of monitoring and control are:

- Various speed probes (N1, N2) and temperature and pressure sensors (P2/T2, P2.5/T2.5, and T3). Exhaust Gas Temperature (EGT), Engine Pressure Ratio (EPR), and oil temperature and pressure sensors are shown.
- The turbine case cooling loops – HP and LP.
- Engine start.
- Fuel control for control of engine speed and, therefore, thrust.
- The engine Permanent Magnet Alternators (PMAs) are small dedicated generators that supply primary power on the engine for critical control functions.

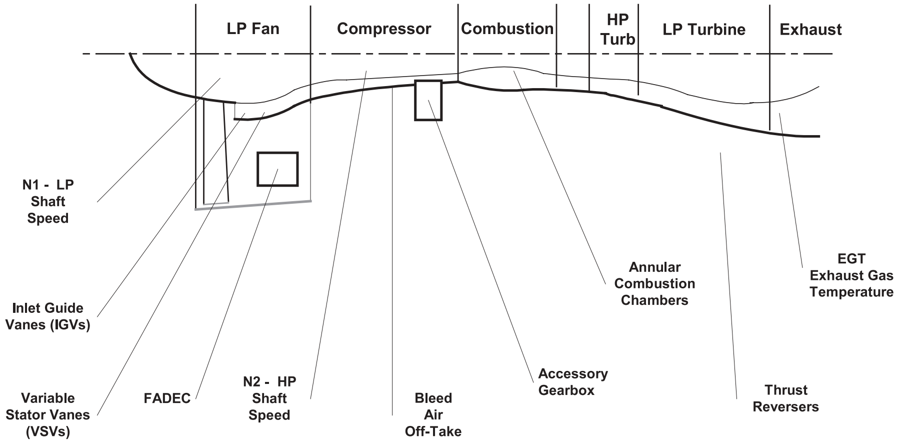


Fig. 10.1 *Typical civil engine components*

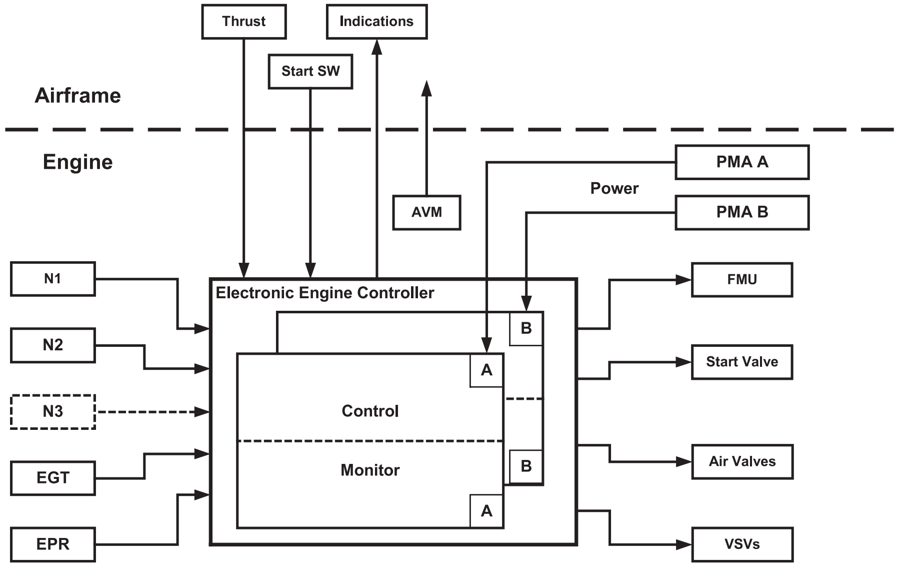


Fig. 10.2 *Simplified engine control and monitoring system*

- Various turbine blade cooling, Inlet Guide Vanes (IGVs), Variable Stator Vanes (VSVs) and bleed air controls.

The engine supplies bleed air for a variety of functions, as described in Chapter 7 of the sister publication 'Aircraft Systems'. Bleed air provides the actuator motive power for some of the controls on the engine as well as supplying medium-pressure air to the airframe for a variety of functions such as anti-icing, cabin pressurization, and cabin air conditioning, among other functions.

An idea of the complexity of other engine off-takes may be gained from Fig. 10.3 which shows a typical engine accessory gearbox. It can be seen that many of the drives off the accessory gearbox are for the use of the engine:

- LP and HP fuel pumps.
- Oil scavenge pumps; oil is used to cool the electrical generator as well as lubricate the engine.
- PMAs to supply 28 V DC power for the dual-channel FADEC.
- Oil breather.

Interfaces with the aircraft include:

- Supply of three-phase 115 V AC, 400 Hz electrical power – rated in the range from 40 to 90 kV A per channel on most civil transport aircraft; 120 kV A per channel on B777 and B767-400 (150 kV A per channel on the A380).
- Supply of 3000 psi hydraulic power (5000 psi on the A380).
- Engine tachometer and other engine indications.
- Input of bleed air from a suitable air source to start the engine. This can be a ground power cart, the APU, or air from the other engine if that has already been started.

An important feature of commercial aircraft operations is the increasing use of two-engine aircraft flying Extended Range Twin OperationS (ETOPS) routes across

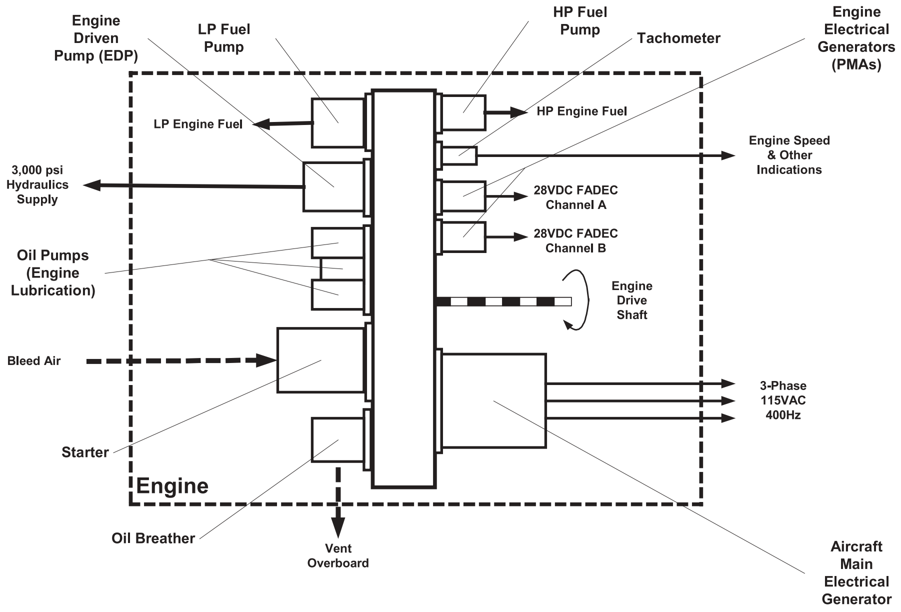


Fig. 10.3 *Typical engine accessory gearbox*

transoceanic or wilderness terrain. The majority of transatlantic flights today are ETOPS operations. The integrity of the engines and related systems is clearly vital for these operations, and the engine In-Flight ShutDown (IFSD) rate is central to determining whether 120 min or 180 min ETOPS approval may be granted by the certification authorities. Reference (**1**) is consulted for ETOPS clearance. It mandates that the engine IFSD needed for ETOPS approval is <50 per million flight hours and <20 per million flight hours for 120 min and 180 min respectively. The actual rate achieved in service today is well below these minima.

Recently, efforts have been made by Boeing to extend this to 208 min (180 min plus 15 per cent) to take full account of the reliability growth of systems and the extended range of later versions of the B777.

Air and environmental systems

Throughout the operation of an aircraft, whether on the ground or in the air, the crew and passengers must be kept in an adequate environment. They must be neither too hot nor too cold, they must have air to breathe, and they must be kept in comfortable atmospheric humidity and pressure conditions. The environmental control system must cope with widely differing temperature conditions, must extract moisture and provide air with optimum humidity, and must ensure that the air in the aircraft always contains a sufficient concentration of oxygen. In addition to these essentially comfort-related tasks, environmental control systems provide demisting, anti-icing, and rain dispersal services. Systems must be designed to provide all these services during extended single-engine operations resulting from ETOPS.

Controlled environment for crew, passengers, and equipment

In the early days of flight, pilots and passengers were prepared to brave the elements for the thrill of flying. However, as aircraft performance has improved and the operational role of both civil and military aircraft has developed, requirements for Environmental Control Systems (ECSs) have arisen. They provide a favourable environment for the instruments and equipment to operate accurately and efficiently, to enable the pilot and crew to work comfortably, and to provide safe and comfortable conditions for the fare-paying passengers.

In the past, large heating systems were necessary at low speeds to make up for the losses to the cold air outside the aircraft. With today's aircraft operating at higher speeds, the emphasis is more towards the provision of cooling systems, although heating is still sometimes required. Providing sufficient heat for the aircraft air conditioning system is never a problem, since hot air is bled from the engines to provide the source of conditioning air. Rather, the issue on a modern civil jet transport is that of heat rejection.

The key elements of an ECS are:

- Provision of bleed air.
- Environmental control.
- Cabin pressurization.

Bleed air

The main source of conditioning air for both civil and military aircraft is engine bleed from the engine intermediate- or high-pressure compressors. This provides a source of hot air whenever the engines are running. The conditioning air is also used to provide cabin pressurization.

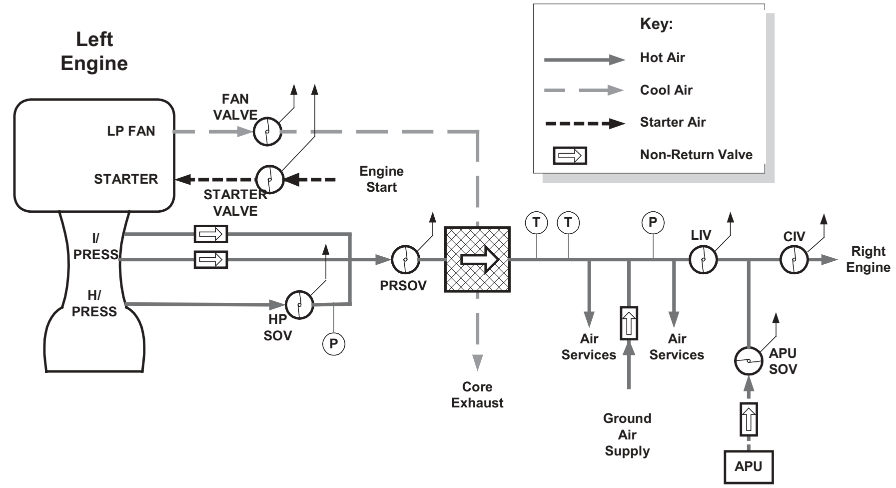
There are two types of bleed air system: open loop and closed loop. Open-loop environmental control systems continually bleed large amounts of air from the engines, refrigerate it, and then use it to cool the passengers and crew, as well as equipment, before dumping the air overboard. Closed-loop systems collect the air once it has been used for cabin conditioning, refrigerate it, and recycle it to be used again. In this way, bleed air is used only to provide pressurization, a low venting air supply, and sufficient flow to compensate for leaks in the closed-loop system. This means that such a system uses considerably less engine bleed air than an open-loop system and therefore has a correspondingly reduced effect on the engine. Civil aircraft invariably use the open-loop method, as this also allows a continuous change of air circulating through the cabin which is important for the health and comfort of the passengers, reducing the risk of discomfort and infection from airborne viruses.

Bleed flow and temperature control

A bleed air system for a modern transport aircraft is shown in Fig. 10.4. Typically, air at a workable pressure of about 650 kPa absolute (6.5 atm) and a temperature of about 100 °C is needed to provide sufficient system flow and a temperature high enough for such services as rapid demisting and anti-icing. However, the air tapped from the engine intermediate- and high-pressure compressor stages is often at higher pressures and temperatures than required. Tapping air at lower pressures and temperatures from a lower compressor stage would be detrimental to engine performance. On many civil aircraft, different bleed tappings can be selected according to engine speed, as shown in Fig. 10.4 where intermediate- and high-pressure tappings are illustrated.

A Pressure Reducing Shut-Off Valve (PRSOV) is used to reduce the pressure of the engine bleed air. This valve controls its downstream pressure to a constant value, no matter what the upstream pressure. The maintenance of this downstream pressure

Fig. 10.4 *Typical bleed air system*



controls the amount of flow from the engines through the environmental control system.

Once the air pressure has been reduced to reasonable working values, the air temperature needs to be reduced to about 100 °C for such services as de-icing and demisting. Heat exchangers are used to reject unwanted heat to a cooling medium, generally using low-pressure air from the core exhaust. Beyond this point the air is controlled by a number of devices, which control the valves in Fig. 10.4 to modulate airflow according to:

- Engine start requirements.
- APU start requirements.
- Wing anti-ice.
- ECS requirements.

Environmental Control System

The top-level overview ECS is shown in Fig. 10.5, which emphasizes the bleed air system from the environmental control system viewpoint. Engine bleed air is processed through two air conditioning packs that cool the bleed air before feeding it to a mixing chamber or plenum chamber. The usual method of operation of the air conditioning packs to cool the air is to use ram air from outside the aircraft together with an Air Cycle Machine (ACM). Air from the mixing chamber is fed to a series of zones around the aircraft where the temperature is individually controlled to the necessary level. Cooling air is also supplied to cool the aircraft avionics equipment as well as maintaining the passengers in a comfortable environment.

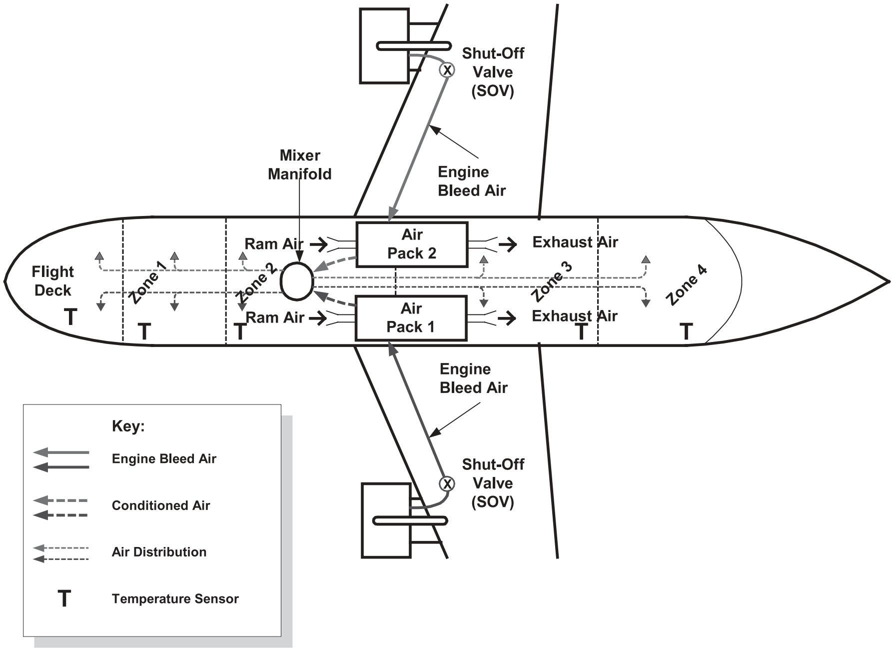


Fig. 10.5 *Top-level ECS overview*

A typical air-conditioning pack with a three-rotor ACM as used on the majority of passenger aircraft is shown in Fig. 10.6. Air enters the system through the primary heat exchanger which is being cooled by means of ram air. This air gains heat when compressed in the ACM compressor and loses heat when passing through the secondary heat exchanger, also cooled by ram air. Part of the resulting air is fed through a reheater and condenser before being applied to the first of the ACM turbines. Eventually, the air passes to the secondary ACM turbine where further heat is extracted before passing to the ECS mixing chamber.

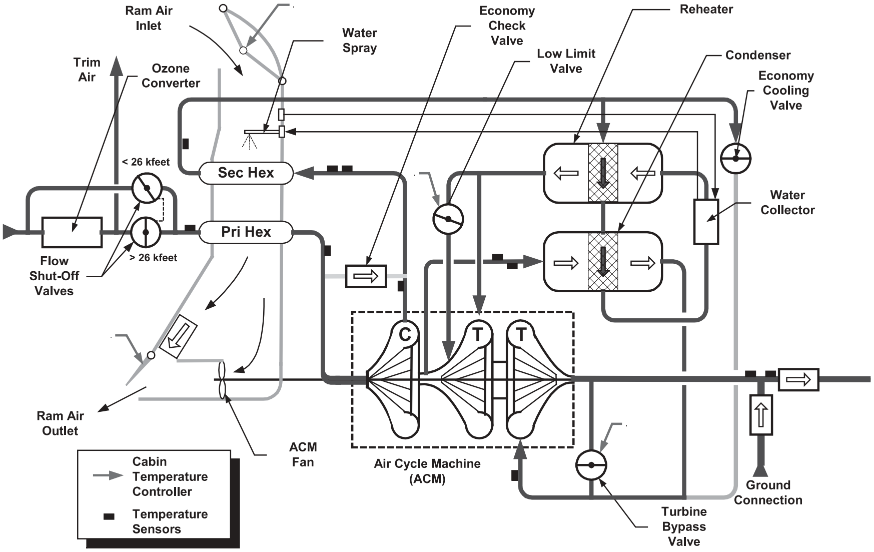


Fig. 10.6 *Typical air cycle cooling system*

In the normal mode of operation, output temperature is controlled by the combination of a series of actuators and valves that operate under the control of the pack temperature controller:

- Ram air throughput is controlled by modulating the position of the ram air inlet and outlet actuators, thereby controlling the flow of air through the primary and secondary heat exchangers and hence the amount of heat rejected overboard.
- The low limit valve controls the air flow to the first ACM turbine, thereby modifying the ACM speed.
- Air passing to the second ACM turbine may be allowed instead to flow through the turbine bypass valve, thereby imparting less energy to the ACM and retaining more heat.

There are a number of temperature sensors located around the air conditioning pack to assist the temperature controller to maintain sufficient control of air temperature at key locations. The example shown has a total of 11 temperature sensors, some of which are duplicated for reasons of redundancy. In addition, the air conditioning pack has the capability of running in reversionary modes of operation should key components fail. In these reversionary modes, cooling will still be possible but not to the same degree as when operating in the primary mode.

The cooled air is fed via a distribution system and series of valves and fans to the passengers and avionics equipment.

The ECS for a large aircraft is shown in Fig. 10.7. Air enters the system from the bleed air system on the left and right and enters the mixing manifold after passing through the respective air conditioning pack. Air also enters the system via upper (into the passenger zones) and lower (into the mix manifold) recirculation fans which recirculate a portion of air around the aircraft. Air is fed to a number of zones – in the example the flight deck plus five passenger cabin zones are shown – which will vary from aircraft to aircraft. Warm air supplied from upstream of the air conditioning packs may also be passed into the various zones via a number of regulating and shut-off valves and trim valves should additional heat be required.

As before, a number of temperature and pressure sensors are located at a number of locations around the system so that the Cabin Temperature Controllers (CTCs) can control the temperature appropriately. In the example shown there are a total of almost 30 temperature sensors – many of them dual redundant – and two pressure sensors.

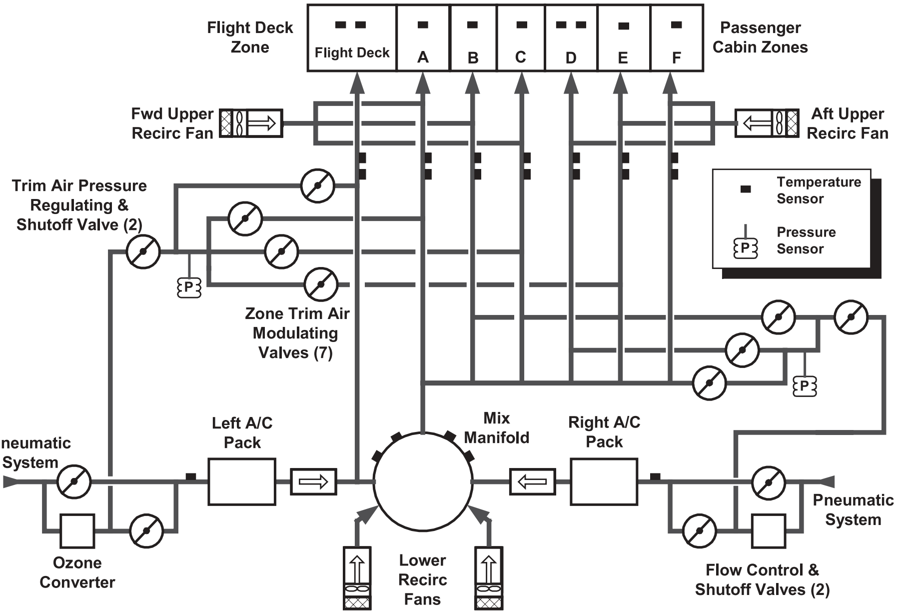


Fig. 10.7 *Typical large aircraft cabin temperature control*

The example given in Fig. 10.8 shows how the various controllers are integrated on the Boeing 777 using ARINC 629 data buses; the right system units are shown, similar units control the left system. However, although specific to the Boeing 777, this is similar in concept to many other systems that use ARINC 429 buses to pass data between the ECS equipment.

In this example, control is effected by:

- CTCs interfacing with the air conditioning pack.
- Card file interfacing with the cabin temperature sensors.
- Electrical Load Management System (ELMS) controlling the distribution system fans, valves, and heaters.
- Connection via the interface cards to the Overhead Panel (OP).
- Connection via the AIMS cabinets to the flight crew displays/EICAS.

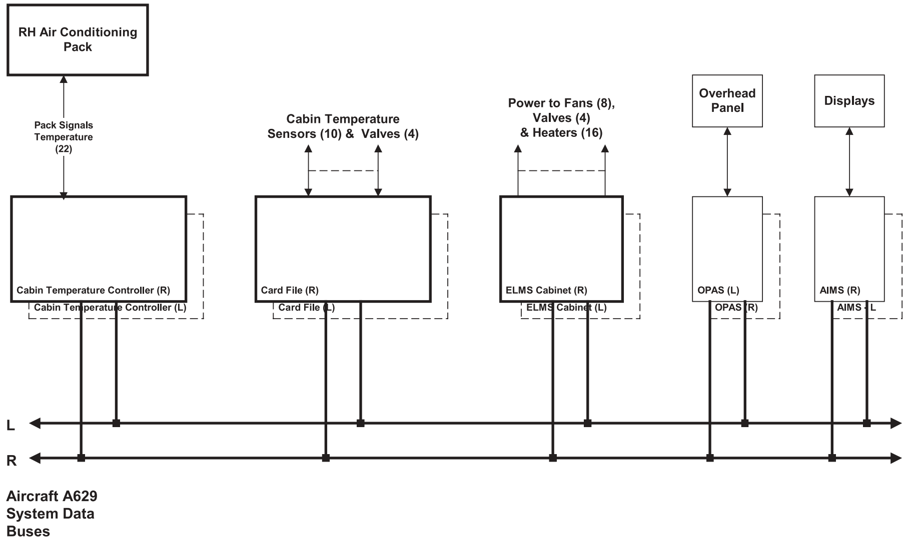


Fig. 10.8 *Avionics integration of ECS controllers*

The need for avionics conditioning

Most aircraft equipment that generates heat will operate quite satisfactorily at a much higher ambient air temperature than can be tolerated by a human. The maximum temperatures at which semiconductor components can safely operate is above 100 °C, although prolonged operation at this level will seriously affect reliability and increase the probability of equipment failure in service.

Air conditioning systems are typically designed to provide a maximum conditioned bay (an area of the aircraft dedicated to the installation of LRUs) temperature of 70 °C, which is considered low enough to avoid significantly affecting the reliability of components. The minimum design equipment operating temperature for worldwide use tends to be about –30 °C. Equipment must also be designed to remain undamaged over a wider temperature range, typically from –40 °C to +90 °C for worldwide use. These figures define the maximum temperature range to which the equipment may be subjected depending on the storage conditions, or in the event that the aircraft is allowed to remain outside for long durations in extreme hot or cold conditions – hot or cold soak respectively. Alternatively, the location of avionics units on the aircraft may subject them to harsh environmental conditions in flight if they are placed outside a conditioned bay.

The cooling problem brought about by the heat sources described above must be solved successfully to cool the aircraft systems and passengers in flight. For ground operations, some form of ground cooling system is also required.

Heat must be transferred from these sources to a heat sink and rejected from the aircraft. Heat sinks easily available are the outside air and the internal fuel. The outside air is used either directly as ram air or indirectly as air bled from the engines. Since the available heat sinks are usually at a higher temperature than that required for cooling the systems and passengers, some form of heat pump is usually necessary.

Cabin pressurization

Cabin pressurization is achieved by cabin pressure control valves that are installed in the cabin wall to control cabin pressure to the required value depending on the aircraft altitude by regulating the flow of air from the cabin. A top-level overview of the pressurization system is shown in Fig. 10.9.

After passing through the air conditioning packs, the control valves – more correctly known as outflow valves – are modulated to control the pressure of the cabin and hence the environment. For aircraft where oxygen is not used routinely, and where the crew and passengers are free to move around, as in a long-range passenger airliner, the cabin will be pressurized so that a cabin altitude of about 8000 ft is never exceeded. This leads to a high differential pressure between the cabin and the external environment. Typically for an airliner cruising at 35 000 ft with a cabin altitude of 8 000 ft there will be a differential pressure of about 50 kPa (0.5 atm) across the cabin wall. The crew is able to select a desired cabin altitude from the cockpit, and cabin pressurization will begin when the aircraft reaches this altitude. This will be maintained until the maximum design cabin differential pressure is reached.

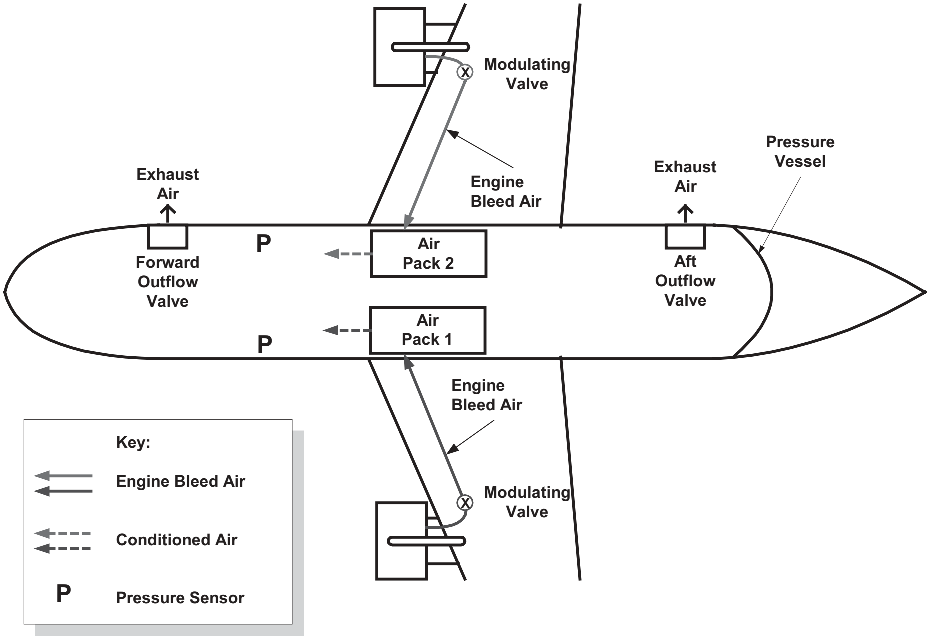


Fig. 10.9 *Top-level view of pressurization system*

Figure 10.10 illustrates a typical pressurization profile for a modern transport, with the right-hand chart showing the variation in ambient pressure (in psi) versus altitude. Therefore, the cabin must be designed as a pressure vessel with minimum leakage. In the event of loss of pressurization, the cabin pressure control valve will close and the only leakage will be through the structure. Non-return valves are installed in the air distribution pipes where they pass through the cabin wall, so that when the air supply fails the air already in the cabin cannot leak back out through the pipes. A safety valve is installed in the cabin wall to relieve internal pressure if it increases above a certain value in the event of failure of the pressure control valve.

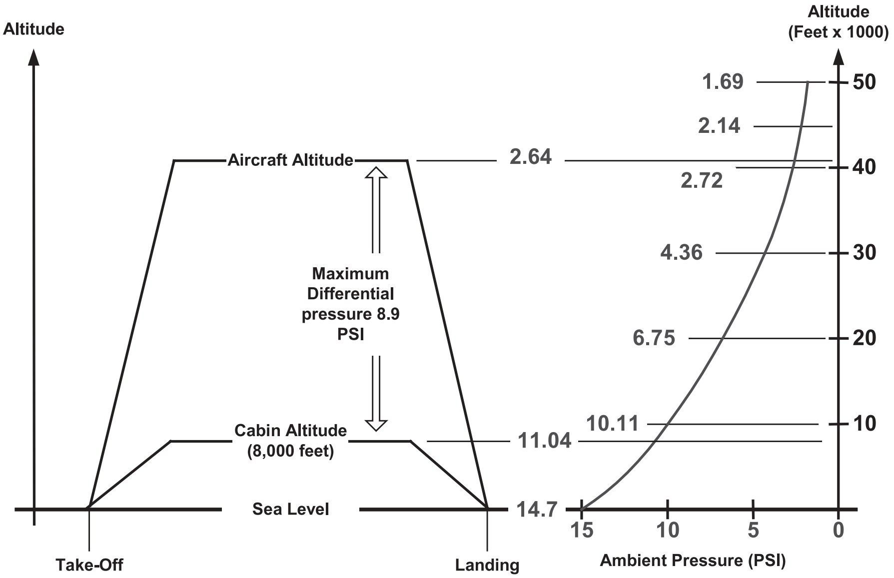


Fig. 10.10 Typical pressurization schedule

Oxygen system

Oxygen is essential for the maintenance of life. If the oxygen supply to the brain is cut off, unconsciousness soon follows, and brain death is likely to occur within 4–5 min. Breathing air at reduced atmospheric pressure results in a reduction in alveolar oxygen pressure which in turn results in an oxygen supply deficiency to the body and brain tissues. This condition is termed hypoxia.

In the event of a decompression following pressurization failure, oxygen masks automatically deploy to provide the passengers with oxygen while the aircraft executes an emergency descent. The flight crew don emergency oxygen masks until the aircraft reaches a safe altitude below 10 000 ft.

Fuel systems

At the onset of aviation, aircraft fuel systems were remarkably simple affairs. Fuel was gravity fed to the engine in most cases, though higher-performance engines would have an engine-mounted fuel pump. Tank configurations were extremely simple and fuel contents were visible float-driven indications. In the case of the Tiger Moth, fuel indication was by means of a simple sight glass located on top of the fuel tank between the two upper wing sections.

Higher performance gave rise to more complexity within the fuel system. The need for transfer and booster pumps accompanied the arrival of high-performance aircraft. More complex tank configurations introduced the need for multivalve systems such that the flight crew could move fuel around the fuel tanks according to the needs at the time.

The arrival of jet turbine powered aircraft brought a range of engines that were much thirstier than their piston-engined predecessors: the early jet aircraft in general had a very short sortie length. More accurate fuel gauging systems were required to give the pilot advanced and accurate information regarding the aircraft fuel state in order that recovery to an airfield could be accomplished before running out of fuel. The higher-performance jet engine also required considerably greater fuel delivery pressures to avoid cavitation and flame-out.

Fuel gauging systems became more complex as greater gauging accuracies were sought and achieved. Most systems are based upon capacitance measurement of the fuel level within the aircraft, using fuel probes placed at various locations within the fuel tanks. A large system may require up to 150 probes or more to measure the contents accurately, particularly if the tank shapes are irregular. Typical figures for the airliners of today are in the region of 1–2 per cent accuracy, depending upon the sophistication of the systems, some of which can compensate for fuel temperature and density, aircraft attitude, fuel height, and a variety of other variables.

Modern aircraft fuel management and gauging systems are based on a plethora of valves, pumps, probes, level sensors, switches, etc., controlled by microprocessor-based systems. This has led to more capable and more reliable systems needed for the aircraft to meet the exacting demands placed upon them.

Characteristics of fuel systems

The purpose of an aircraft fuel system is primarily to provide a reliable supply of fuel to the engines. Therefore, the fuel system is an essential element in the overall suite of systems required to assure safe flight. Modern aircraft fuels are hydrocarbon fuels similar to those used in the automobile. Piston-engined aircraft use a higher octane fuel called AVGAS in aviation parlance. Jet engines use a cruder fuel with a wider distillation cut and with a lower flash point. AVTAG and AVTUR are typical jet engine fuels. The specific gravity of aviation fuels is around 0.8, that is, about eight-tenths of the density of water. Therefore fuel may be quantified by reference to either volume (gallons or litres) or mass (pounds or kilograms). As the density of fuel varies according to temperature, both may be used. The volume of an aircraft fuel tankage is fixed, and therefore it will not be able to accommodate the same mass of fuel at high temperature when the fuel density is lower.

The essential characteristics of a modern aircraft fuel management system may embrace some or all of the following modes of operation:

- Engine feed.
- Fuel transfer.
- Pressurized refuel/defuel.
- Vent systems.
- Use of fuel as heat sink.
- Fuel jettison.
- In-flight refuelling (military application).

Before describing the operation of these typical modes of operation it is worth outlining the primary components that comprise such a system.

Fuel system components

Fuel transfer pumps

Fuel transfer pumps perform the task of transferring fuel between the aircraft fuel tanks to ensure that the engine fuel feed requirement is satisfied. Transfer pumps may also be required to transfer fuel around the aircraft to maintain pitch or lateral trim. In the case of pitch trim, this requirement is becoming more critical for unstable control configured aircraft where the task of active Centre of Gravity (CG) control may be placed upon the fuel management system. Similarly, on civil aircraft there is a requirement to transfer fuel from wing tanks to the fuselage centre tank where fuel may typically be consolidated before engine feed. However, there are FAR/JAR regulations that require independent engine feed systems. On more recent civil aircraft such as the Airbus A340, as will be seen, the horizontal stabilizer may contain fuel that has to be transferred to maintain the aircraft CG within acceptable limits.

However, some aircraft also contain fuel in the empennage, in this case the fin, to increase fuel capacity. In these cases, pumps are also required to transfer fuel forward to a centre tank for consolidation. A typical aircraft system will have a number of transfer pumps for the purposes of redundancy, as will be seen in the examples given later in this chapter.

Fuel booster pumps

Fuel booster pumps, sometimes called engine feed pumps, are used to boost the fuel flow from the aircraft fuel system to the engine. One of the reasons for this is to prevent aeration (i.e. air in the fuel lines that could cause an engine ‘flame-out’ with consequent loss of power). Another reason in the case of military aircraft is to prevent ‘cavitation’ at high altitudes. Cavitation is a process in which the combination of high altitude, relatively high fuel temperature, and high engine demand produces a set of circumstances where the fuel is inclined to vaporize. Vaporization is a result of the combination of low fuel vapour pressure and high temperature. The effect is drastically to reduce the flow of fuel to the engine, which can cause a flame-out in the same way as aeration. An aircraft system will possess a number of transfer pumps, as will be illustrated later in the chapter.

Fuel quantity indication

The ability to measure fuel quantity accurately is a vital function in a modern fuel system. This is achieved by using a number of fuel probes in each tank, the outputs of which are aggregated to sum the fuel in the tank. The number of probes required in a system depends greatly on the fuel tank geometry and on the levels of accuracy and redundancy specified. A long slender wing tank with complex tank boundaries determined by the wing structural members will require considerably more probes than a regular box structure that may encompass the centre tank.

Accuracy also depends upon measuring fuel density and in some cases temperature since these factors can affect the accuracy. Most fuel probes are either AC or DC capacitive probes, although some examples such as the Boeing 777 use an ultrasonic measurement technique.

Integrated civil aircraft fuel systems

The integration of aircraft civil fuel systems has become more prevalent over the last decade or so, using digital data buses and the supply of hardware from one or more manufacturers. Most civil aircraft have a fuel tank configuration as shown in Fig. 10.11. This configuration comprises left and right wing tanks and a centre tank.

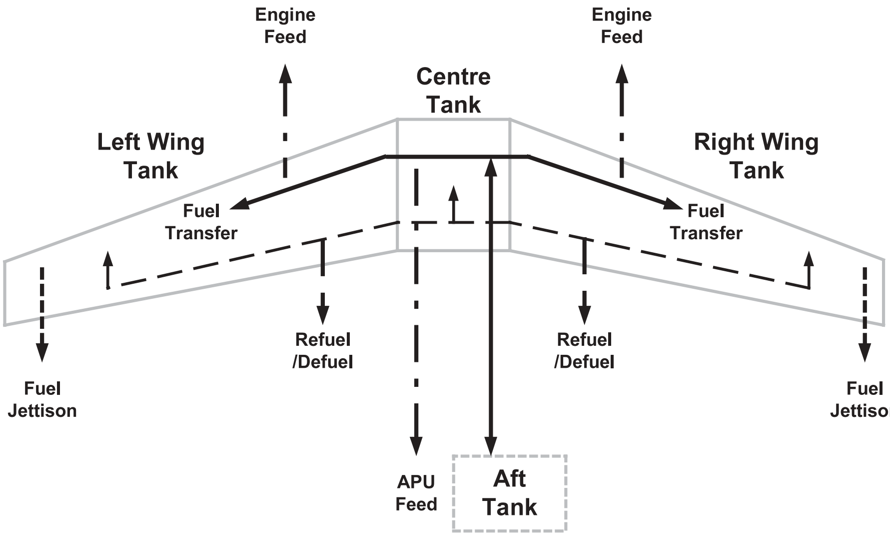


Fig. 10.11 *Typical civil aircraft fuel tank configuration*

However, it is also possible for aircraft to have an aft or trim tank. The major transfer modes are:

- Engine and APU feed.
- Fuel transfer.
- Refuel/defuel.
- Fuel jettison.

Depending on the aircraft configuration and the degree of control, the aft or trim tank may be used as a means of controlling the aircraft CG. Altering the contents of a trim tank can reduce trim drag and improve aircraft range; it is also possible to reduce the structural weight of the tailplane. Most aircraft have variations on this basic topology, although the number of wing tanks may also be dictated by the wing structure, the number of engines, or the need to partition fuel to cater for engine turbine disc burst zones.

This section addresses three examples of highly integrated fuel systems:

- Bombardier Global Express.
- Boeing 777.
- Airbus A340-500/600.

Bombardier Global Express

The Fuel Management and Quantity Gauging System (FMQGS) developed by Parker Aerospace for the Bombardier Global Express is typical of a family of systems that may be found fitted to regional aircraft and business jets. The Global Express (Fig. 10.12) has a true intercontinental range capability of well over 6 000 miles and is cleared to operate at altitudes of up to 51 000 ft.

Fig. 10.12 *Bombardier global 5000 intercontinental business jet – Bombardier Aerospace*

The system has interfaces to:

- Engine Indication and Crew Alerting System (EICAS) and ground crew via A429 data buses.
- Cockpit control panel for APU and engine selector switches and fire handles.
- Cockpit fuel panel for fuel system mode selections.
- Electrical load management system for supplying power to the electrically powered pumps and valves. The system receives status discretes from fuel pumps and valves.
- Cockpit and wing Refuel/Defuel Control Panels (RDCPs).

Refer to Fig. 10.13.

The heart of the system is the dual-channel FMQGS which embraces the following functions:

- Fuel management
  - control, status, and Built-In Test (BIT) of all system pumps, valves, and pressure sensors;
  - fuel transfer – burn sequence and lateral balance;
  - flight crew and ground crew interface;
  - automatic/manual refuel/defuel operation;
  - BIT fault detection and annunciation.
- Optional thermal management. The operation of the aircraft for long periods at altitude provides extreme cold soak conditions. The system provides control of the return of warm fuel from the engine oil coolers to the wing tanks when extremely low-temperature operation might be encountered.

- Fuel quantity gauging. Fuel quantity gauging using the following sensors:
  - linear AC capacitance fuel probes (34);
  - level sensors – software adjustable (6);
  - fuel compensators (2);
  - self-calibrating densitometers (2);
  - temperature sensors (10).

The FMQGS is an ARINC 600 LRU designed to meet the DO160C environment. The unit contains a dual-channel microprocessor architecture hosting software to DO178B level B. On this system, Parker Aerospace performed the role of systems integrator, taking responsibility for design and development, controlling configuration, and certifying the system (**2**).

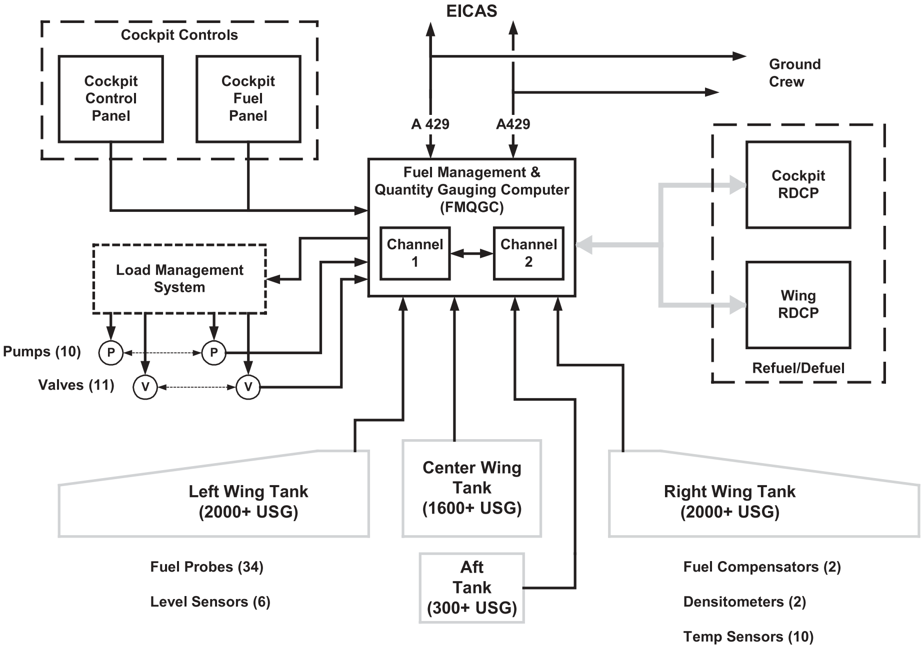


Fig. 10.13 *Simplified global express fuel system – Parker Aerospace*

Boeing 777

In contrast, the Boeing 777 uses an integrated architecture based upon A429 and A629 data buses as shown in Fig. 10.14. This diagram emphasizes the refuel function which is controlled via the ELMS P310 standby power management panel in association with the Integrated Refuel Panel and the Fuel Quantity Processor Unit (FQPU).

There are six refuelling valves, marked as R on the diagram, two in each of the left-wing, centre, and right-wing tanks. The P310 standby power management panel provides power to the FQPU and IRP and controls the operation of the refuelling valves. The FQPU and IRP communicate by means of dual A429 data links. The top-level integration of the FQPU and ELMS P310 panel is via the aircraft systems left and right A629 data buses. This system permits the automatic refuelling of the aircraft to a preset value – as the FQPU senses the fuel tank quantities reaching their assigned value, messages are sent to the ELMS to shut off the refuelling valves until all three tanks have

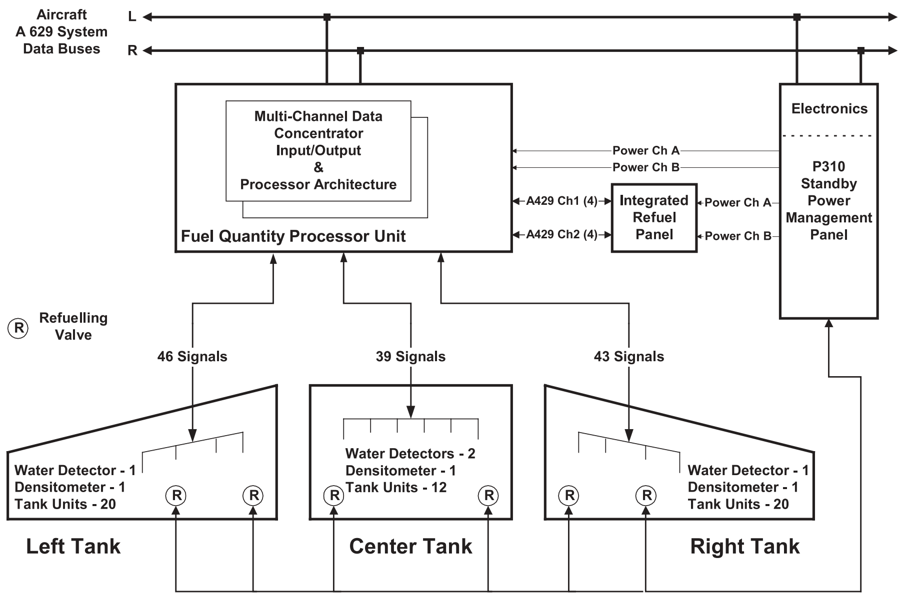


Fig. 10.14 *Simplified portrayal of B777 fuel guaging/fuel management – Smiths Aerospace*

attained the correct fuel quantity or mass.

The function of the B777 ELMS is described in Chapter 4. In this mode of operation the ELMS is able to power up the necessary components of the fuel system to accomplish refuelling during ground maintenance operations without the need to power the entire aircraft.

The FQPU is a multichannel, multiprocessor controller that processes the fuel quantity information provided by a total of 52 tank units (probes), four water detectors, and three densitometers located in the three fuel tanks. The B777 uses ultrasonic fuel probes, the first civil airliner to do so.

The ELMS, FQPU, IRP are supplied by Smiths Aerospace.

Airbus A340-500/600

The A340-500/600 models represent one of the longest-range civil air transport aircraft. The A340-600 is the longest civil transport aircraft in the world – even longer than the Boeing 777-300. Apart from the obvious characteristics of all fuel systems, this aircraft makes great use of fuel to minimize the trim drag and stabilizer weight by trimming the aircraft much closer to the neutral point. Also, fuel is transferred into the outer tanks during flight to reduce the wing bending moment. The result of seeking these performance improvements is a more sophisticated system than is usually the case for a transport aircraft, with more fuel system capabilities and components. Reference (**3**) provides an extensive description of the A340-500/600 fuel system.

One of the main reasons for redesign of the earlier A340-200/300 fuel system was the change in wing design which increased wing sweep. This rendered the previous fuel system unusable owing to the effects of debris from a turbine disc burst. Additional changes had to be made on account of the requirement for higher refuel rates of 400 000 l/h to enable more rapid turnrounds. A top-level portrayal of the A340-500/600 fuel tank arrangement is shown in Fig. 10.15.

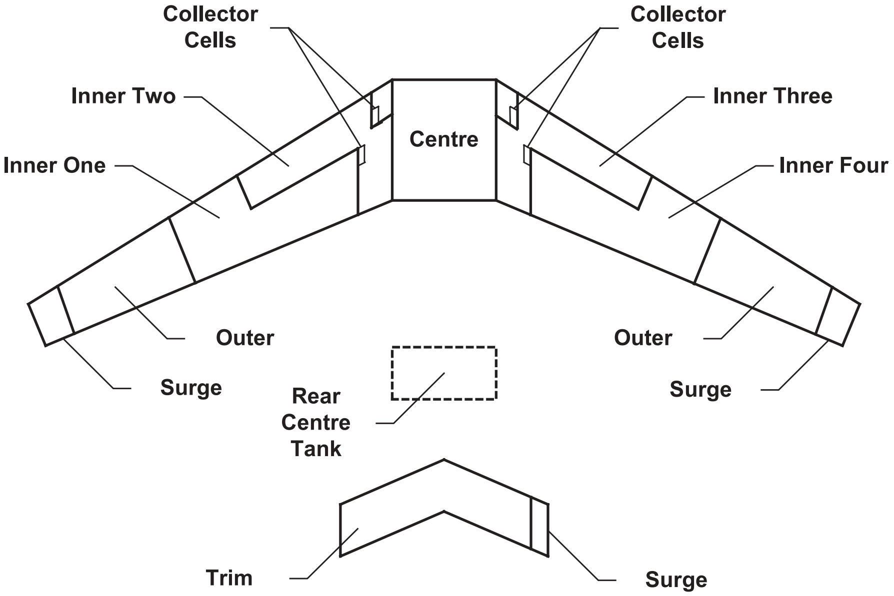


Fig. 10.15 *A340-500/600 fuel tank configuration*

The total number of fuel tanks is 9 for the A340-500 with the Rear Centre Tank (RCT) fitted, and 8 for the A340-600. These are:

- Outer, inner 1, and inner 2 tanks in the left wing.
- Centre tank.
- Inner 3, inner 4, and outer tanks in the right wing.
- RCT (A340-500 only).
- Trim.

This compares with six tanks for the A340-200/300 models.

The total number of fuel valves and pumps is given in Table 10.1 – a total of 36 (40) valves and 22 (24) pumps for the A340-600 (-500). This compares with 30 valves and 13 pumps for the A340-200/300 model. The increase of ~33 per cent (fuel valves) and ~ 45 per cent (pumps) is indicative of the effect that changes in basic requirements can have on a major system.

The control of the fuel system is carried out by a Fuel Control and Monitoring System (FCMS) supplied by Parker Aeropace that comprises four units:

- Two Fuel Control Management Computers (FCMCs) which perform the Fuel Quantity Indication (FQI), fuel level indication, refuel control, CG calculations, and control general fuel transfer. Automatic compensation is applied for changes in fuel density, fuel permittivity, and aircraft attitude.
- Two Fuel Data Concentrators (FDCs) which perform the fuel probe excitation and signal processing.

Table 10.1 A340-500/600 fuel system – tally of fuel valves and fuel pumps

Location	Refuel/defuel valves	Transfer valves	Jettison valves	Engine/APU feed valves	Transfer pumps	Jettison/transfer pumps	Engine/APU feed pumps
Aircraft Level	4		2	12			12
Left Outer	1	1					
Inner One	2					1	
Inner Two	2					1	
Centre	2				2	2	
Inner Three	2					1	
Inner Four	2					1	
Right Outer	1	1					
Rear Centre Tank (A340-500 only)	(2)	(2)			(2)		
Trim	2	2			2		
Total	18 (20)	4 (6)	2	12	4 (6)	6	12
Grand Total	36 (40)				22 (24)		

Hydraulic systems

The introduction of powered flying controls was an obvious application for hydraulic power by which the pilot was able to move the control surfaces with ever-increasing speeds and demands for manoeuvrability. This application brought hydraulics into the area of safety-critical systems in which single failures could not be allowed to hazard the aircraft. The system developed to take account of this using multiple pumps, accumulators to store energy, and methods for isolating leaks.

The hydraulic system today remains a most effective source of power for both primary and secondary flying controls, and for undercarriage, braking, and antiskid systems. The last decade has seen the ever-accelerating introduction of electronics and microprocessors, both for monitoring system performance and to perform control functions. This has proved to be a major step forward, permitting some previous shortcomings to be overcome and opening the way to so-called ‘smart’ pumps and valves.

Hydraulic system services

The majority of aircraft in use today need hydraulic power for a number of tasks. Many of the functions to be performed affect the safe operation of the aircraft and must not operate incorrectly, i.e. must operate when commanded, must not operate when not commanded, and must not fail totally under single-failure conditions. A typical list of functions included in the hydraulics systems today include:

- Primary flight controls:
  - elevators,
  - rudders,
  - ailerons,
  - canards.
- Secondary flight controls:
  - flaps,
  - slats,
  - spoilers,
  - airbrakes.
- Utility systems:
  - undercarriage retraction and extension – including doors,
  - wheelbrakes and antiskid,
  - nosewheel steering,
  - cargo doors,
  - loading ramp,
  - passenger stairs.

The different types of flight control actuation and the associated controls were addressed in detail in Chapter 9 – Flight Control Systems. The remainder of this section will address the nature of hydraulic power generation and control of some of the major services.

Other functions, commonly known as ‘services’ or ‘utilities’, may be considered expendable after a failure, or may be needed to operate in just one direction after a positive emergency selection by the pilot. In this case the designer must provide for the emergency movement to take place in the correct direction, for example, undercarriages must go down when selected. It is not essential for the landing gear to return to their previous position in an emergency, since the aircraft can land once the legs are extended.

Wheelbrakes tend to be a special case where power is frequently provided automatically, or on selection, from three sources. One of these is a stored energy source that also allows a parking brake function to be provided.

A hydraulic system in its simplest form is shown in Fig. 10.16. The primary source of power on an aircraft is the engine, and a hydraulic pump, known as the Engine-Driven Pump (EDP), is connected to the engine gearbox. The pump causes a flow of fluid, at a certain pressure, through stainless steel pipes to various actuating devices. A reservoir ensures that sufficient fluid is available under all conditions of demand. On most civil aircraft, provision may be made for an AC powered Electrical Motor Pump (EMP) to augment the EDP during periods of high demand or to provide a back-up.

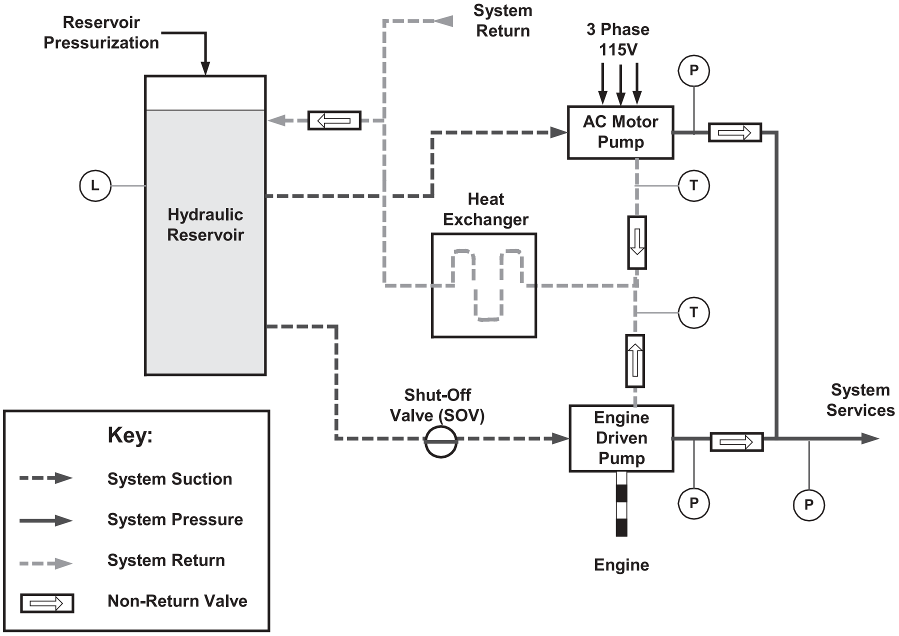


Fig. 10.16 *Typical hydraulic system*

It can be seen that control and indications are required to control the basic hydraulic power generation:

- Control of a Shut-Off Valve (SOV) to control the EDP output.
- Control of the EMP by means of an electrical relay or contactor.
- Monitoring of the key parameters:
  - – hydraulic reservoir fluid level,
  - – hydraulic delivery pressure,
  - – hydraulic return temperature.

This simple system is unsatisfactory from an integrity viewpoint and in practice most aircraft contain multiple pumps and connections of pipes to ensure that single failures and leaks do not deplete the whole system of power. The degree of redundancy necessary is very largely controlled by specifications and mandatory regulations issued by the national and international bodies charged with air safety. Civil aircraft invariably have three or four channels. In both types, additional auxiliary power units and means of transferring power from one system to another are usually provided by mean of a Power Transfer Unit (PTU). More recent developments such as the Airbus A380 have adopted dual hydraulic systems, albeit with the introduction of more-electric systems.

Most transport systems today have hydraulic systems that have become standardized at 3000 psi. A recent exception is the Airbus A380 which will be the first civil aircraft to use 5000 psi systems.

Emergency hydraulic power sources

All hydraulic systems have some form of emergency power source. In its simplest form this will be an accumulator. It is mandatory for wheelbrake systems to have a standby accumulator capable of supplying power for a predetermined number of brake applications when all other sources of power are inoperative.

To supply emergency power for longer periods, an electric motor driven pump may be provided. Frequently it is also possible to operate at some pressure below nominal system pressure. Even so, it is unlikely that an acceptable installation can be achieved that will provide power for more than 5 or 6 min. The use of directly powered electrical flight control actuators was addressed in Chapter 9.

For continuous emergency supply, a Ram Air Turbine (RAT) may be used. This carries with it several disadvantages. Space must be found to stow the turbine and carriage assembly, a small accumulator is needed to deploy the turbine in emergency, and, because speed governing and blade feathering are employed, the assembly is complicated. Hydraulic pumps and/or emergency electrical generators can be mounted immediately behind the turbine on the same shaft. It is, however, more common to mount them at the bottom of the carriage arm close to the deployment hinge axis. This involves the use of driveshafts and gears. To keep the turbine blade swept diameter at a reasonable figure, the power developed must be kept low, and it may be difficult to mount the assembly on the airframe so that the airflow is not impeded by the fuselage at peculiar aircraft attitudes. Deployment of the RAT is as for the electric motor driven pump.

In spite of these drawbacks, ram air turbines have several times proved their worth, particularly on civil aircraft, providing the only means of hydraulic power until an emergency has been dealt with and the aircraft has been recovered to a safe altitude.

Civil transport comparison

However, as a way of examining different hydraulic system philosophies, a comparison is made between an Airbus narrow body (the A320 family) and a Boeing wide body (the B767). It is usual for three independent hydraulic systems to be employed, since the hydraulic power is needed for flight control system actuation. Hydraulic power is produced by pumps driven by one of the following methods of motive power:

- Engine driven – EDP as already described.
- Electrically driven – EMP as already indicated.
- Air turbine/bleed air driven pumps are used on Boeing wide-body aircraft.
- Ram air turbine driven.

Airbus A320

The aircraft is equipped with three continuously operating hydraulic systems called blue, green, and yellow. Each system has its own hydraulic reservoir as a source of hydraulic fluid:

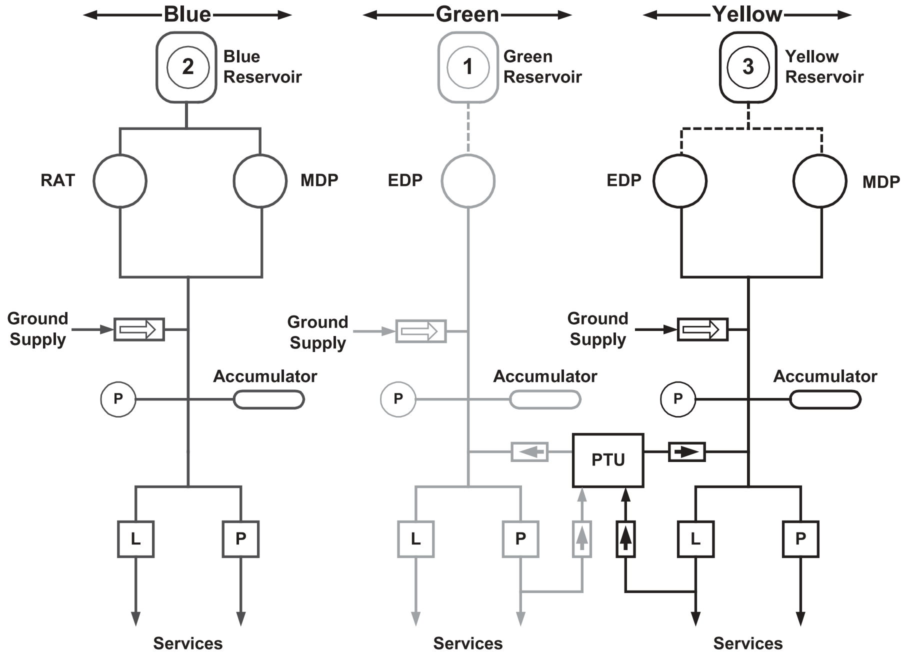
- The green system (system 1) is pressurized by an EDP located on No. 1 engine which may deliver 37 US gal/min or 140 l/min.
- The blue system (system 2) is pressurized by an electric motor driven pump capable of delivering 6.1 US gal/min or 23 l/min. A RAT can provide up to 20.6 US gal/min or 78 l /min at 2175 psi in emergency conditions.
- The yellow system (system 3) is pressurized by an EDP driven by No. 2 engine. An electric motor driven pump is provided which is capable of delivering 6.1 US gal/min or 23 l/min for ground servicing operations. This system also has a handpump to pressurize the system for cargo door operation when the aircraft is on the ground with electrical power unavailable.

Each channel has the provision for the supply of ground-based hydraulic pressure during maintenance operations. Each main system has a hydraulic accumulator to maintain system pressure in the event of transients (see Fig. 10.17).

Each system includes a leak measurement valve (shown as L in a square on the diagram) and a priority valve (shown as P in a square):

- The leak measurement valves are positioned upstream of the primary flight controls and are used for the measurement of leakage in each flight control system circuit. They are operated from the ground maintenance panel.
- In the event of a low hydraulic pressure, the priority valve maintains pressure supply to essential systems by cutting off the supply to heavy load users.

Fig. 10.17 *Simplified A320 family hydraulic system*



The bidirectional Power Transfer Unit (PTU) enables the green or the yellow systems to power each other without the transfer of fluid. In flight, in the event that only one engine is running, the PTU will automatically operate when the differential pressure between the systems is greater than 500 psi. On the ground, while operating the yellow system using the electric motor driven pump, the PTU will also allow the green system to be pressurized.

The RAT extends automatically in flight in the event of failure of both engines and the APU. In the event of an engine fire, a fire valve in the suction line between the EDP and the appropriate hydraulic reservoir may be closed, isolating the supply of hydraulic fluid to the affected engine.

Pressure and status readings are taken at various points around the systems which allows the composition of a hydraulic system display to be shown on the Electronic Centralized Aircraft Monitor (ECAM). Control and monitoring of the hydraulic system on Airbus aircraft is carried out by a Hydraulic Systems Monitoring Unit (HSMU), a multichannel stand-alone LRU.

Boeing 767

The B767 also has three full-time independent hydraulic systems to assure the supply of hydraulic pressure to the flight controls and other users. These are the left, right, and centre systems serviced by a total of eight hydraulic pumps:

- The left system (red system) is pressurized by an EDP capable of delivering 37.5 US gal/min or 142 l/min. A secondary or demand electric motor driven pump capable of delivering 7 US gal/min or 26.5 l/min is turned on automatically in the event that the primary pump cannot maintain pressure.
- The right system (green system) has a similar configuration to the left system.
- The centre system (blue system) uses two electrically driven motor pumps, each with the capability of delivering 7 US gal/min or 26.5 l/min as the primary supply. An Air-Driven Pump (ADP) with a capacity of 37 US gal/min or 140.2 l/min is used as a secondary or demand pump for the centre system. The centre system also has an emergency RAT rated at 11.3 US gal/min or 42.8l/min at 2140 psi.

Figure 10.18 presents a simplified diagram of the B767 hydraulic system. Primary flight control actuators, autopilot servovalves, and spoilers receive hydraulic power from each of the three independent hydraulic systems. The stabilizer, yaw dampers, elevator feel units, and the brakes are operated from two systems. A PTU between the left and right systems provides a third source of power to the horizontal stabilizer.

A motorized valve (shown as M on the figure) located between the delivery of ACMP 1 and ACMP 2 may be closed to act as an isolation valve between the ACMP 1 and ACMP 2/ADP delivery outputs.

Hydraulic system status and a synoptic display may be portrayed on the EICAS displays situated between the captain and first officer on the instrument console. A number of maintenance pages may also be displayed. The supply schedule for the different pumps is given in Table 10.2.

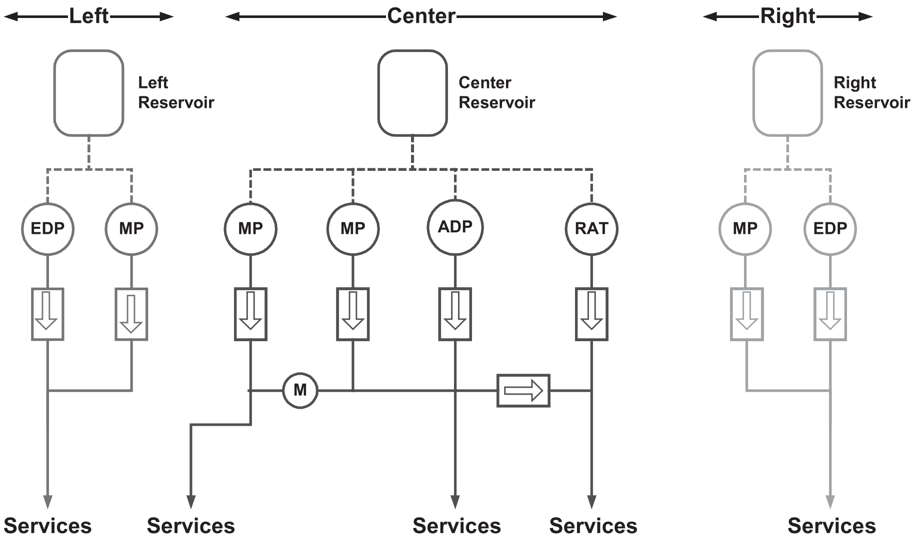


***Fig. 10.18** Simplified Boeing 767 hydraulic system*

Table 10.2 Boeing 767 simplified hydraulic schedule

System	Hydraulic power summary		Operating conditions
	Pump continuous	Pump demand	
Left or right	EDP		Basic system pressure
		ACMP	Supplements EDP
Centre	ACMP 1		Basic system pressure – maintains isolated system pressure
	ACMP 2		Basic system pressure – does not operate when one engine is out or left and right ACMPs are on
Centre		ADP	Supplements ACMPs 1 and 2
Centre (Emergency)		RAT	Operates when deployed

The RAT supplies emergency power in flight once the engine speed (N2) has fallen below 50 per cent on both engines and the airspeed is in excess of 80 knots. The RAT may only be restored to its stowed position on the ground.

While this description outlines the B767 system at a top level, the systems on the B747-400 and B777 also use a combination of engine-driven (EDP), air-driven (ADP), and electric motor driven pumps and a RAT, albeit in different architectures with a different pump configuration. The Boeing philosophy appears to favour fewer accumulators but uses more pumps with a more diverse selection of prime pump energy.

Boeing entrust the control and monitoring of the aircraft hydraulic systems to removable cards located in card files. Separate modules – known as HYDIM modules – are provided for each system, thereby providing segregation of control and monitoring functions.

Reference (**4**) very usefully summarizes the key hydraulic system characteristics of virtually all wide-body, narrow-body, and turboprop/commuter aircraft flying today.

Landing gear systems

The following systems are associated with the landing gear and usually utilize dedicated electronic means of control and monitoring:

- Landing gear extension and retraction.
- Undercarriage monitoring.
- Brake temperature and tyre pressure monitoring.
- Nosewheel steering.
- Main bogy steering.
- Brakes and antiskid.

Examples of some of these systems will be described.

Landing gear monitoring

Landing gear and door position need to be monitored during the retraction and extension sequence. This used to be achieved by using microswitches, but these devices were unreliable owing to their susceptibility to moisture ingress, the landing gear being a difficult environment in which to operate.

In recent years, proximity switches have been increasingly used as they depend upon the proximity of a target to a magnetic device. When the two portions, located on different parts of the landing, are in close proximity to one another, a change in magnetic coupling may be detected, signalling that the landing gear has reached a particular position. Proximity switches are used to detect a ‘Weight-On-Wheels’ (WOW) when the aircraft is sitting on the ground rather than flying. Many aircraft and avionics systems perform differently when on the ground rather than in the air, and WOW sensing is a critical function.

Figure 10.19 shows the functionality provided by the two Proximity Switch Electronics Units (PSEUs) used on the Boeing 777. As well as interfacing to some 30 landing gear elements, the PSEUs also interface to aircraft doors (50 sensors) and engine reverse thrust systems (14 sensors).

Brakes and antiskid

The principles of operation of a brake and antiskid system are shown in Fig. 10.20. This function is invariably controlled by electronic means today. The brake and antiskid unit feeds a demand to the brake control valve which modulates the hydraulic system pressure to apply the pressure to the aircraft brake units. As the pressure increases, the wheel may approach the point where it locks. Wheel-mounted sensors measure the wheel speed, and the brake control system is able to detect the point at which the wheel is about to lock. At this point the controller backs off the demand so that the wheel remains close to the point of locking but does not actually do so – this represents the

Fig. 10.19 Typical proximity switch sensing functions

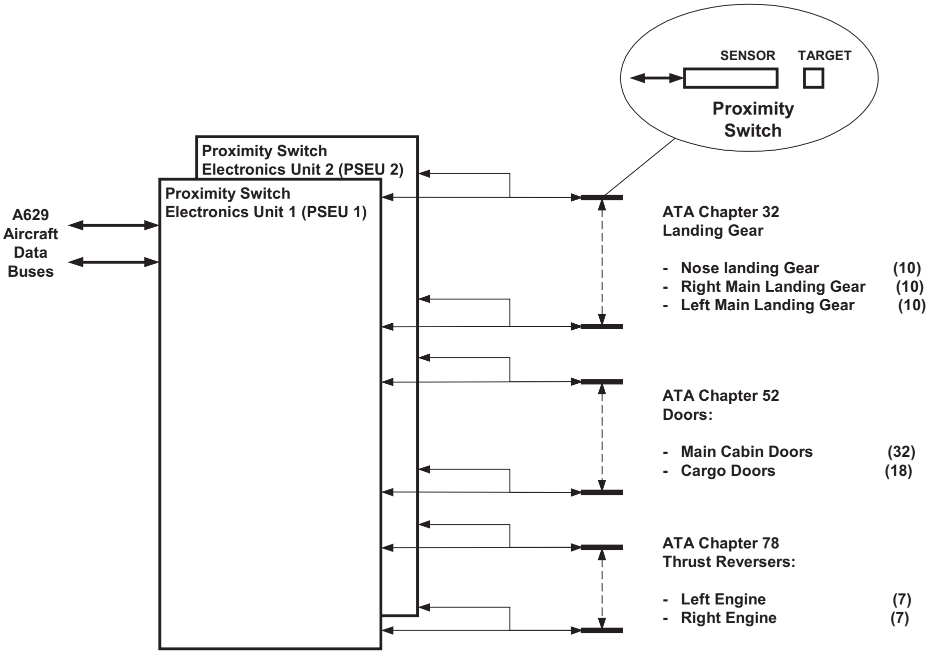
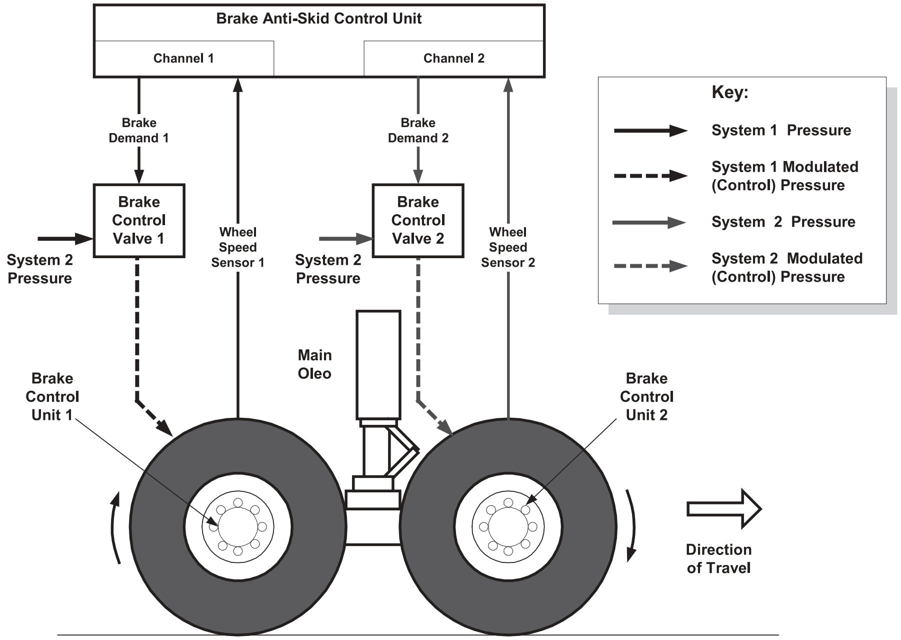


Fig. 10.20 Principles of braking and antiskid control



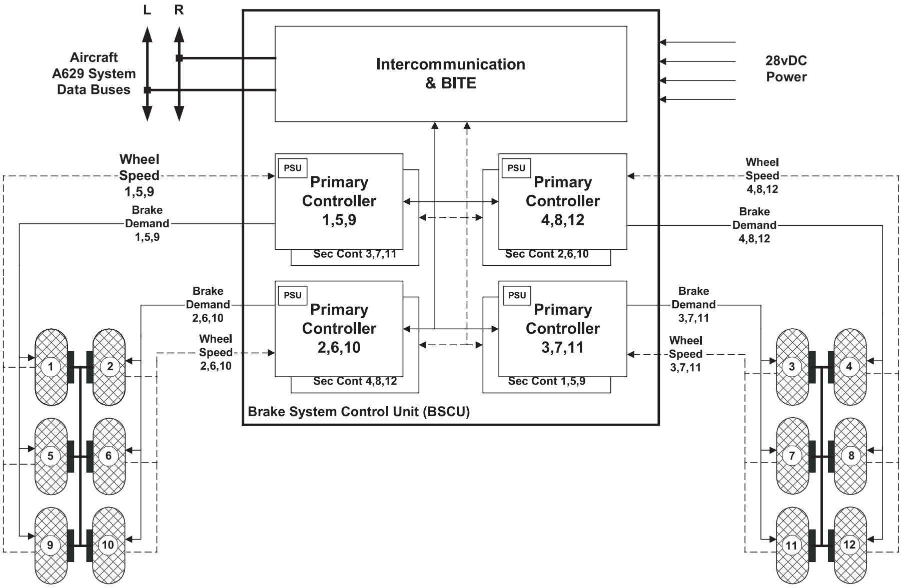


Fig. 10.21 *Simplified B777 braking configuration*

point of maximum braking efficiency and is identical to the operation of the ABS on cars. Figure 10.20 shows a dual-channel system, but in practice multiple-channel systems are commonly used on large aircraft with multiple bogies.

The system also allows the flight crew to preselect a deceleration to be applied during the landing run. In this case the system controls the aircraft to a constant deceleration, which can be a very useful feature when landing on wet or slippery runway surfaces.

The systems described thus far apply to most simple aircraft braking systems. However, large aircraft have multiwheel bogies and sometimes more than two main landing gears. The B747-400 has four main oleos, each with a total of four wheels each. The B777 has two main bogies with six wheels each. These systems tend to be more complex and utilize multi-lane dual-redundant control. The B777 main gear shown in Fig. 10.21 is an example.

For control purposes the wheels are grouped in four lines of three wheels, each corresponding to an independent control channel as shown in Fig. 10.21. Each of the lines of three wheels – 1,5,9; 2,6,10, and so on – is controlled by a dual-redundant controller located in the Brake System Control Unit (BSCU). Brake demands and wheel speed sensor readings are grouped by each channel and interfaced with the respective channel control. Control channels have individual power supplies to maintain channel segregation and integrity. The BSCU interfaces with the rest of the aircraft by means of left and right A629 aircraft systems data buses. This system is supplied by the Hydro-Aire division, part of Crane Aerospace, and is indicative of the sophistication that modern brake systems offer for larger aircraft.

Central maintenance systems

The need for maintenance data on-board the aircraft carrying complex integrated avionics systems of today is paramount. One of the keystones of on-board avionics systems is the Built-In Test (BIT) function that most systems embrace. BIT embraces a test capability within the equipment that enables the correct functioning of the equipment to be assured. BIT is performed continuously (CBIT) or can be initiated by the crew during preflight checks (IBIT). Early manifestations of BIT were unreliable and not always able to ascertain the correct functioning of the equipment, let alone make accurate diagnostic assessments of equipment failures. Many years of hard-learned lessons have resulted in aircraft and avionics manufacturers fielding much more capable and potent equipment in the last decade or so. While never perfect, the maintenance systems of today offer much more help than hindrance in the battle to keep the aircraft airborne and earning revenue.

The availability of improved BIT equipment (BITE) capabilities, together with the increased use of aircraft-level data buses such as ARINC 429 and ARINC 629, have enabled the various avionics equipment to share copious quantities of diagnostic data. The ready availability of improved systems-level multifunction displays such as EICAS and ECAM on most modern aircraft has resulted in much more information relating to system function and status being displayed at the system level on the flight deck displays. However, it is the Control and Display Unit (CDU) – normally associated with the Flight Management System (FMS) – that has made the greatest contribution to on-board maintenance test and diagnosis.

It is by its very nature – text display (normally in colour), hard alpha-numeric and menu-driven soft keys – that the CDU offers a multipurpose man–machine interface for the technician on the ground in the same way in which it provides similar operational functions for the flight crew while airborne. These features, coupled with the highly integrated nature of data transfer in the modern avionics system, enable the CDU to be used as a very powerful test and diagnostic tool.

Two distinct but separate implementations are examined, both of which offer considerable improvements above earlier-generation systems. Many modern regional, business jet, and large, medium, and small transport aircraft will have similar functionality embedded. What will be described is indicative of the industry norm rather than one or two specific examples. The examples to be described are:

- Airbus A330/340 Central Maintenance System (CMS).
- Boeing 777 Central Maintenance Computing System (CMCS).

Airbus A330/340 Central Maintenance System

The implementation of the A330/340 Central Maintenance System (CMS), also adopted for later models of the A320 family, is described herein. The Airbus aircraft predominantly use ARINC 429 data buses throughout, and this system is based upon the integration of data based upon this level data fusion. This contrasts with the Boeing ARINC 629 centralized approach which will be described later. Both represent extremely capable systems; the fact that the approach is different is of interest.

The key display elements as shown in Fig. 10.22 are:

- Display of aircraft system synoptics and status displays available to the flight crew on the ECAM displays.
- Use of the three CDUs as a man–machine interface for system test and diagnostic purposes.

The ECAM displays typically portray system synoptics relating to the following provided by the Display Management Computers (DMCs):

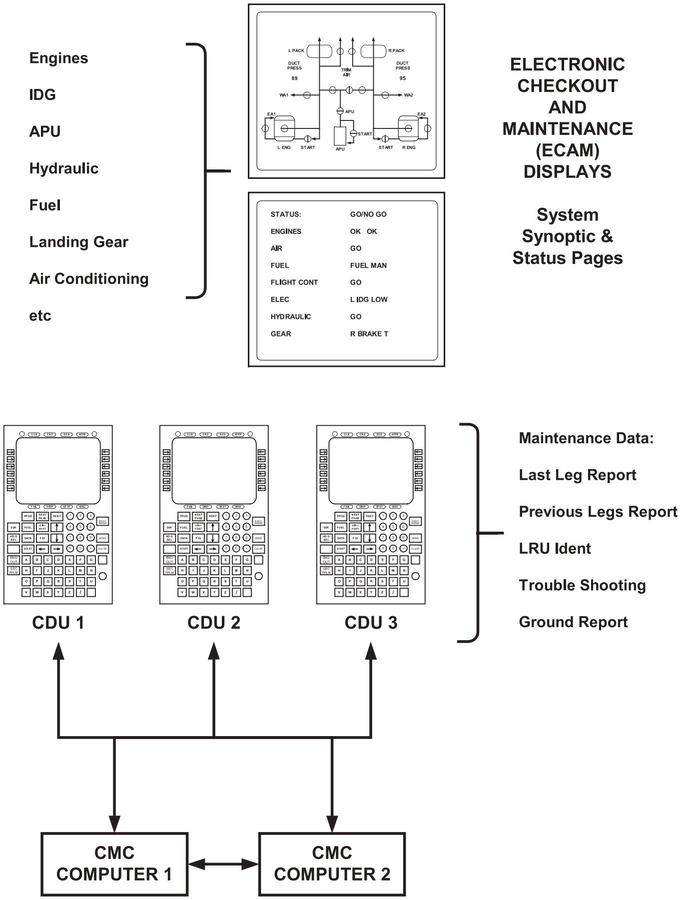


Fig. 10.22 *Airbus CMC computation and display*

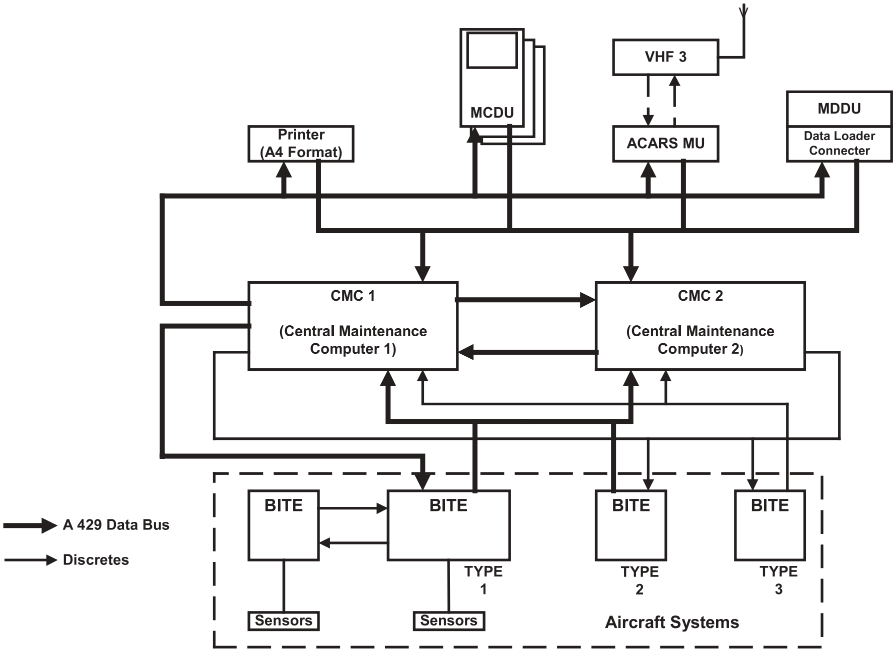
- Engines.
- Electrical system.
- APU.
- Hydraulic system.
- Fuel system.
- Air conditioning.
- Landing gear.

Similar data are displayed for many other systems.

The CDUs are provided with data in this instance from one of the two dual-redundant Central Maintenance Computers (CMCs), as shown in the top-level architecture in Fig. 10.23. These data are menu driven according to the maintenance page selected on the CDU and further down-selection using the soft keys at the side of the display. The CMCs provide data to the CDUs for display, to the flight deck printer, and to VHF system 3 for ACARS transmission as required. Reference (**5**) describes the operation of the system in significant detail.

In the Airbus approach, avionics systems are defined by three types:

Fig. 10.23 Airbus central maintenance computing architecture



- Type 1. These are characterized by ARINC 429 digital data bus inputs and outputs. A total of 34 basic and nine optional systems, comprising a total of 75 units, are in this category.
- Type 2. These systems comprise a discrete and an ARINC 429 data bus input from the CMC. There are a total of ten basic systems, comprising a total of 19 units.
- Type 3. These systems typically are characterized by discrete inputs and outputs. There are a total of four basic and one optional system, comprising a total of 8 units.

Therefore, the system as described can interface to a total of 102 line replaceable units (LRUs) if all options are fitted (91 basic fit LRUs).

The most powerful aspect of the system relates to the menu-driven function provided by the CDU(s) (see Fig. 10.24). On this display, several of the soft keys are highlighted and enable selection of sub-menus as stated on the CDU screen.

The Airbus system, in common with many other similar systems, utilizes the total capabilities of the CDU, albeit focused in the maintenance sense. In the example shown, various soft keys are highlighted, and the associated maintenance/diagnostic function is illustrated alongside the appropriate key. Typical systems allow the following capabilities at this level:

- Last leg reporting.
- Previous leg reporting.
- LRU identification.
- Troubleshooting options.

The precise options available will depend upon the implementations in a specific system. Suffice it to say that the menu-driven option allows the interrogation of system functions at an individual sensor level, complete with status and scaling (accuracy

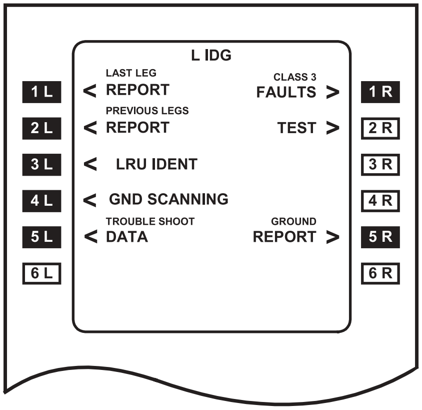


Fig. 10.24 *CDU maintenance menu driven display*

information). Modern systems are usually also capable of interrogating data down to the module/sensor level and may also include a shop-level (second/third level) fault history for every module within the system. In this way, modules with previous fault histories may be identified and correlated with the reported fault before removal from the aircraft (rogue modules with persistent and repeatable fault histories may also be identified and investigated).

The system outlined provides a multilevel diagnostic capability based upon an Air Transport Association ATA (functional) chapter basis. The system permits the download of diagnostic data via a 1.44 Mb diskette if required.

Boeing 777 Central Maintenance Computing System

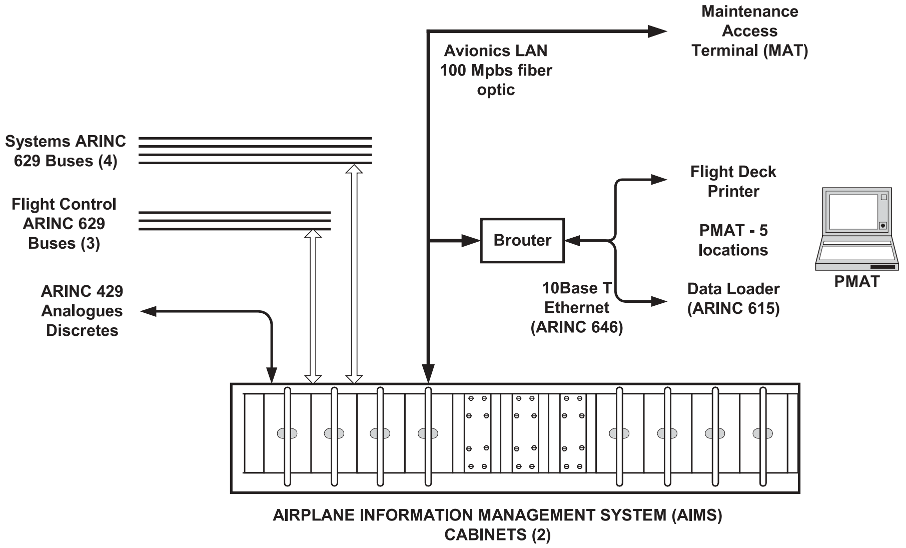
The Boeing Central Maintenance Computing System (CMCS) also uses an integrated approach to gathering maintenance data and displaying it to the technician. In the Boeing 777, the Airplane Information Management System (AIMS) is used to assemble the necessary data for the CMCS function, as depicted in Fig. 10.25. The AIMS comprises two integrated cabinets which interface to the aircraft systems as follows:

- To the aircraft systems and engines via four aircraft systems ARINC 629 data buses.
- To the flight control system (see Chapter 9) via three flight control ARINC 629 data buses.
- To other aircraft equipment by means of ARINC 429 data buses and analogue and discrete signals.

Either cabinet has the ability to pass data via a fibre-optic avionics Local Area Network (LAN) to the maintenance interface. This is a high data rate Fibre Distributed Data Interface (FDDI) bus operating at 100 Mb/s that connects directly to the Maintenance Access Terminal (MAT) and keyboard. Data are also passed via a router that performs a network interface function enabling several devices to be connected via 10BaseT, 10 Mb/s Ethernet (ARINC 645) as follows:

- Flight deck printer.
- Portable Maintenance Access Terminal (PMAT) at five receptacles around the aircraft.
- Data loader operating in accordance with ARINC 615.

Fig. 10.25 Boeing 777 Central Maintenance Computing System



The Boeing philosophy is therefore to use dedicated display equipment, whereas Airbus uses the FMS CDUs. However, Boeing uses the existing AIMS cabinets for the data gathering and processing function, whereas Airbus uses dedicated computing resources – the two CMCs.

The Boeing system has the following capabilities:

- LRU software loading.
- Input monitoring.
- Configuration reporting.
- Access to shop fault history data (contained within local non-volatile memory on every LRU/module).
- On-board engine balancing.
- PSEU and air/ground rigging.
- Reporting capabilities.

In-Flight Entertainment

In-Flight Entertainment (IFE)

The provision of IFE equipment is becoming a powerful marketing and customer satisfaction tool in the light of ever-increasing competition between airlines in the medium- and long-range air transport markets. The importance of IFE in the battle for customers is such that the equipment can in some cases be dispatch critical, for reasons of passenger appeal rather than flight safety.

A very simplified overview of an IFE system is shown in Fig. 10.26, depicting the main component elements of an early Video-On-Demand (VOD) system for a three-class seating arrangement. This simplified overview highlights some of the major areas with which the system has to interface:

- Cabin management head-end video equipment or peripherals.

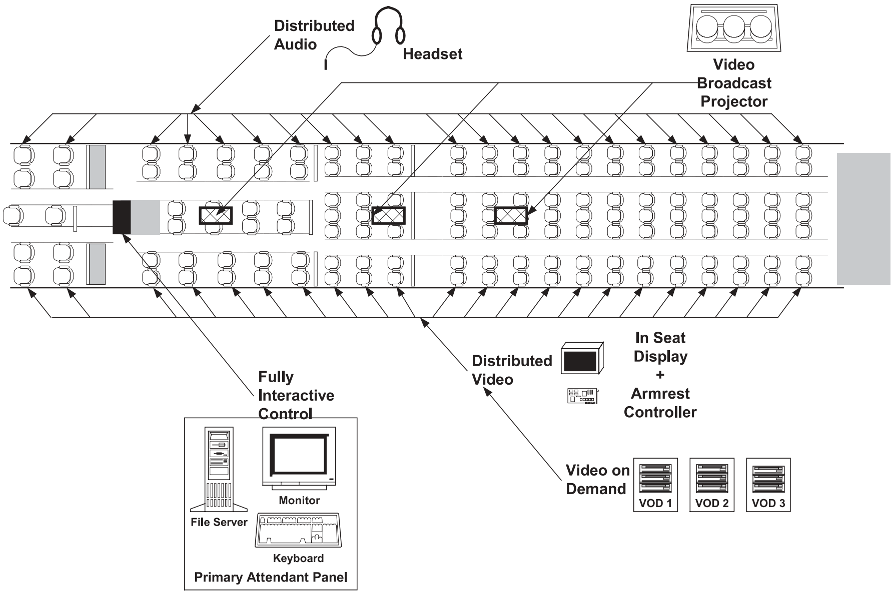


Fig. 10.26 *Simplified overview of an IFE system*

- Interface with the aircraft and cabin crew.
- Seat equipment and overhead video displays.
- Interconnecting databuses, wiring, and power supplies.

More recent, rapidly evolving functions are being developed. In their simplest forms these include:

- PC interfacing with the aircraft communications ‘spine’, conveying the ability to send faxes/e-mail and access the Internet.
- More advanced passenger game and shopping/entertainment options.
- Broadband communications via satellite en route.
- Direct TV broadcast.
- Wireless gateway communications with the aircraft on the gate.
- For the flight crew – the ability to interface with the airline’s own IT networks for transmitting receiving maintenance data, uplinking flight plan updates, etc.

The principles of these enhanced air to ground communications will be described later.

The IFE market is very dynamic, given that IFE and the enhanced capabilities that the system can offer are becoming major marketing tools, particularly for the first-tier airlines competing in the medium- and long-range markets. The ability to shop, play games and, in the case of the businessman, work and communicate with the office via phone, fax or e-mail is becoming of paramount importance when offering a competitive service. The importance of the system in terms of availability and customer satisfaction is such that the IFE system can be a dispatch-critical system.

The complexity of the hardware and equipment of the IFE system on large aircraft is enormous in terms of hardware, electrical power, and wiring. An IFE system can easily approach or exceed 200 MCU equivalent, and some of the more capable systems consume considerable quantities of aircraft electrical power – commonly estimated as up to 100 W per passenger seat for VOD. A 300-seat aircraft could easily consume 30 kVA of power within the IFE system. The cost of an IFE system can easily exceed

US$ 1 000 000 per system for a large aircraft.

The technology baseline of the IFE system shares many features with the domestic electronic hardware and PC/IT markets. Rapid system development is predominant owing to the market pressures, and standard avionics equipment technical disciplines such as rationalization and standardization, commonality, and configuration control are the victims in this rapidly evolving marketplace. ARINC 628 is a multipart specification that lays down standards for the implementation of the IFE system.

The generic technology implementations that have been mentioned in regard to IFE systems have included:

- Ethernet – 10BaseT (IEEE 802.3) standard 10 Mb/s transmission.
- Token ring (IEEE 802.5) using 16 Mb/s data rates.
- Sony – passport system using ‘FireWire’ (IEEE 1394) connection technology commonly used in commercial camcorder and PC digital video interface applications. This is a scaleable LAN technology that may use 100, 200, 300, or 400 Mb/s on a bus. Sony have also been looking at 150 Mb/s commercial ATM networks to provide the communications ‘spine’ for the IFE system.

These technologies are widely used in ground-based computer and PC applications and lend themselves to the use of standard commercial digital video and PC/computer industry standards.

IFE systems are highly competitive in nature, as has already been mentioned. There is a convergence of diverse technology applications:

- Satellite communications.
- Avionics technology in the satellite and aircraft systems interfaces.
- Commercial consumer/digital communications, TV, and video.
- Software protocols and programme content.
- Wireless Access Protocols (WAPs).

An example of a system being developed to integrate these features is Connexion-by-Boeing^{SM}. Connexion-by-Boeing^{SM} is a declared intention to provide an end-to-end service provider function for IFE and ground–air–ground broadband communications. Initially, this implementation is expected to accommodate bandwidths of 5 Mb/s receive and 1.5 Mb/s transmit. The system will comprise three basic levels:

- Airborne antennae, servers, routers, and the associated wiring.
- Ground infrastrusture: network control, satellite uplink/downlink, and business control centre.
- Space element: leased satellite transponders.

Boeing are teamed with a total of six specialised companies to provide this overall capability

Rockwell Collins I^{2}S

A good example of the operation of the extended air-to-ground communications feature is the Integrated Information System (I^{2}S) being developed by Rockwell Collins, though this is by no means the only implementation. It does serve well the purpose of outlining the method of operation. This system extends the range of data available to pilots, passengers, cabin staff, and maintenance specialists as part of an airline information network.

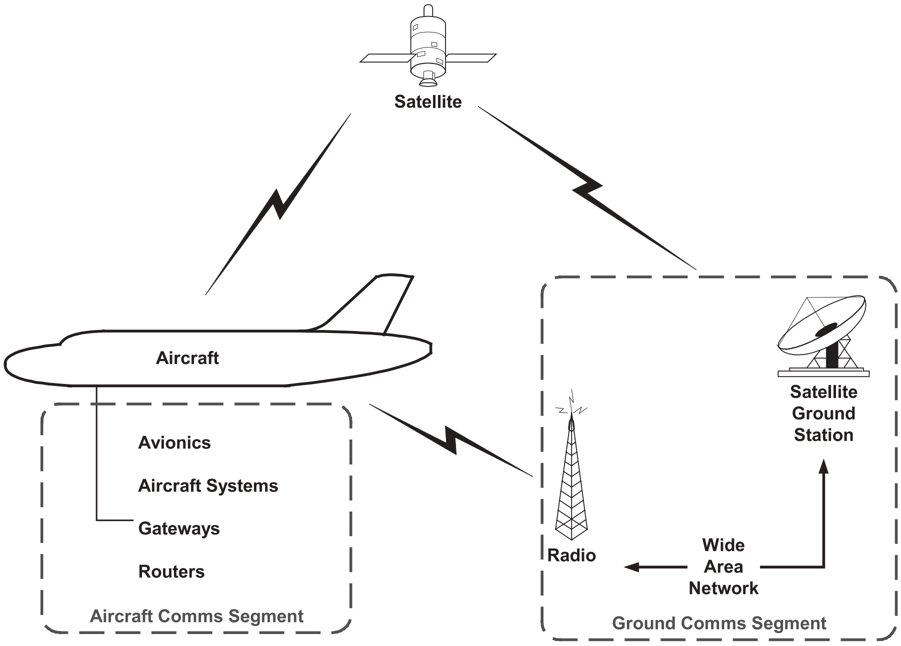


Fig. 10.27 Principle of air to ground communications exchange

Air-to-ground information exchange

The principle of an aircraft-to-ground information system is shown in Fig. 10.27. The principle involves the linking of an aircraft communications segment to a ground communications segment via three primary communications means:

- VHF voice/data link – this uses on-board VHF radios but is limited to line-of-sight communication owing to the transmission properties of VHF radio.
- HF/HF voice/data link – this uses on-board HF radios which may be limited by upper ionospheric characteristics, particularly during active sunspot periods.
- SATCOM – this uses on-board satellite communication equipment, where fitted, to communicate with the INMARSAT geostationary satellites. SATCOM coverage is reasonable over most parts of the globe but is limited to latitudes between 85° north and 55° south owing to the low grazing angle while flying in polar regions.

The operation and limitations of these methods of communication were described in Chapter 6.

Present air-to-ground data links have been limited to low-level communications by means of the Airborne Communications, Addressing, and Reporting System (ACARS) which has incorporated a limited dataset associated with airline operations. I^2S expands this dataset to a far greater range of data services.

Ground communications segment

The ground communications segment is shown in Fig. 10.28. The ground communications segment embraces a wide area network integrating the following services:

- SATCOM.
- Wireless gatelink – this gives the aircraft a local wireless link while in the proximity of the passenger terminal and allows a seamless transition on to the gate.
- Telephone.
- VHF voice/data.
- HF data.
- Various ground-based airline networks associated with:
  - dispatch: flight planning, weather, dispatch host;
  - airline maintenance network: maintenance network and maintenance host;
  - passenger services, ticketing, etc.;
  - value-added services, duty free, shopping, etc.

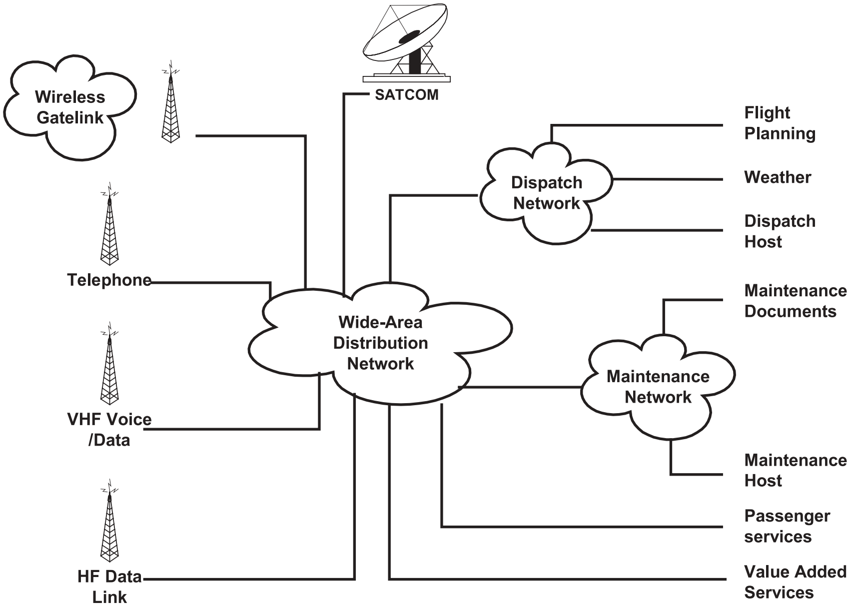


Fig. 10.28 *Ground communications segment*

Aircraft communications segment

The aircraft communications segment is shown in Fig. 10.29. The aircraft communications segment provides an interface with the aircraft communications suite to facilitate communications to and from the ground. The secure interface provides a firewall with the passenger interface:

- Passenger entertainment.
- E-mail.
- Internet.
- Fax, etc.

and the higher-integrity aircraft functions such as:

- FMS/MCDU.
- ACARS.
- Printer.
- Digital Flight Data Recording System (DFDRS).

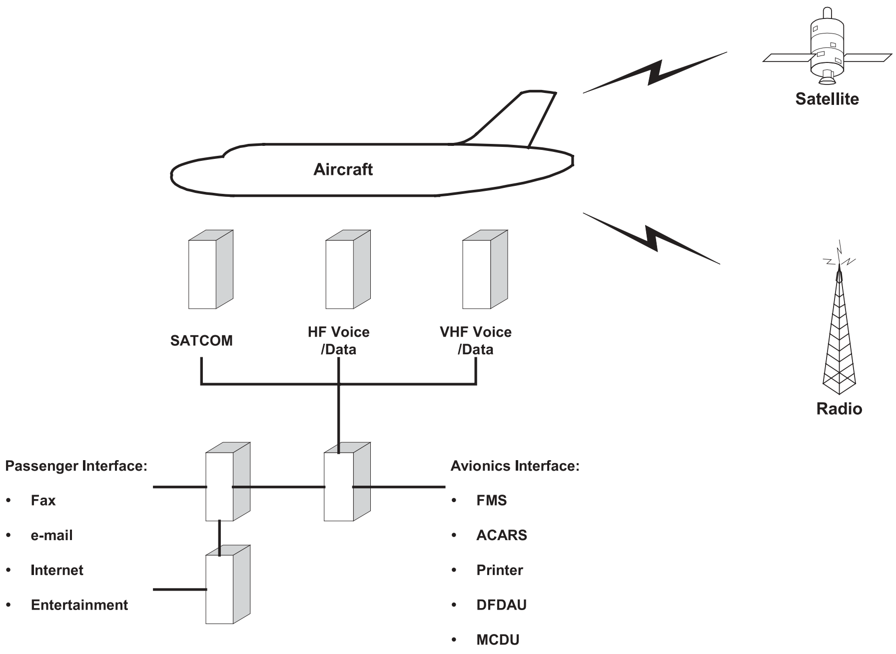


Fig. 10.29 *Aircraft communications segment*

For the flight crew, I^2S provides weather updates, an extensive electronic library, electronic charts and maps, and, ultimately, the ability to upload navigational databases. The cabin crew will be provided access to manifest data and duty-free shopping information. The passenger will be able to access the aircraft file server for audio/video, shopping, and games, and interface to e-mail and the Internet.

Present data rates are very low: typically around 2.5 kbit/s for voice, though some systems offer ~10 kbit/s at higher cost. Developments are in hand to expand this data rate to 64 kbit/s, which will permit air–ground/ground–air transmission of e-mail. Passengers will also be able to access an Internet cache stored on-board the aircraft, but will not be able to access the Internet in real time.

Rockwell Collins I^2S implementation

The Rockwell Collins I^2S comprises the following units:

- A Secure Interface Unit (SIU) to provide the firewall function between passenger services and aircraft avionics/systems. This is a 4MCU unit.
- A File Server Unit (FSU) to host the applications run on-board the aircraft. This is a 4MCU unit.
- An antenna and receiver for the wireless gatelink; one antenna is envisaged for the A320, but a larger aircraft such as a B747 might require three links owing to the increased amount of data.

References

(1) Advisory Circular AC 120-42A (1988) Extended Range Operation with Two-Engine Airplanes.

(2) Parker Aerospace Marketing Publication GPDS9709-FM.
(3) *Airbus FAST Magazine*, 26, 2000.
(4) SAE Aerospace Information Report (AIR) 5005 (2000) Aerospace – Commercial Aircraft Hydraulic Systems.
(5) *Airbus FAST Magazine*, 16, 1994.

CHAPTER 11

Systems Integration

The descriptions and functions of avionics systems so far in this book will have provided the reader with an insight into the functions and capabilities of a wide range of avionics systems. What is also clear is the fact that many of these systems and subsystems need to be integrated to perform the top-level aircraft avionics functions such as navigation, autopilot/flight direction, and so on. This process has become steadily more necessary as functional and operational requirements have increased.

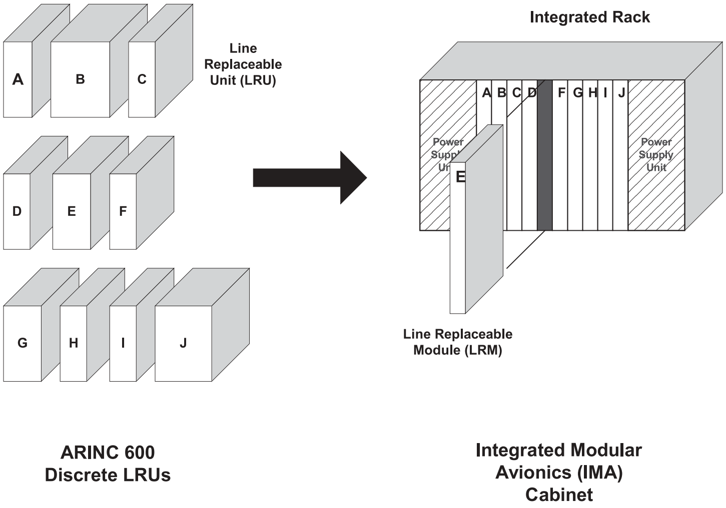
Many of the examples of systems integration described in this book relate to avionics solutions using ARINC 600 specification Line Replaceable Units (LRUs) interconnected using digital data buses: usually A429 for civil aircraft, but also A629 in the case of the Boeing 777. Other digital data buses may also be used on civil aircraft. Where military equipment is fitted to a basic civil aircraft platform to fulfil a military role, as described in Chapter 13, MIL-STD-1553B or other military data buses may be used to perform integration.

Further savings can be achieved by the physical integration of functional implementations into integrated racks or cabinets, as shown in Fig. 11.1. In a typical example, 20–25 different LRU functions may be located into two or more cabinets, with significant savings in volume weight and reliability. The net result is a system that is integrated in a physical and data communications network. Further functional integration is achieved by the use of integrated systems design and open architectures.

This chapter will examine some of the issues associated with the introduction of Integrated Modular Avionics (IMA) architectures in civil aircraft. There are a number of implementations already in being across a range of aircraft, and the practice is becoming adopted in a more widespread fashion.

One of the most obvious benefits of adopting the IMA approach is the possibility of sharing two or more Power Supply Units (PSUs) in the cabinet, each of which is capable of supplying the suite of modules with the necessary regulated DC power supplies.

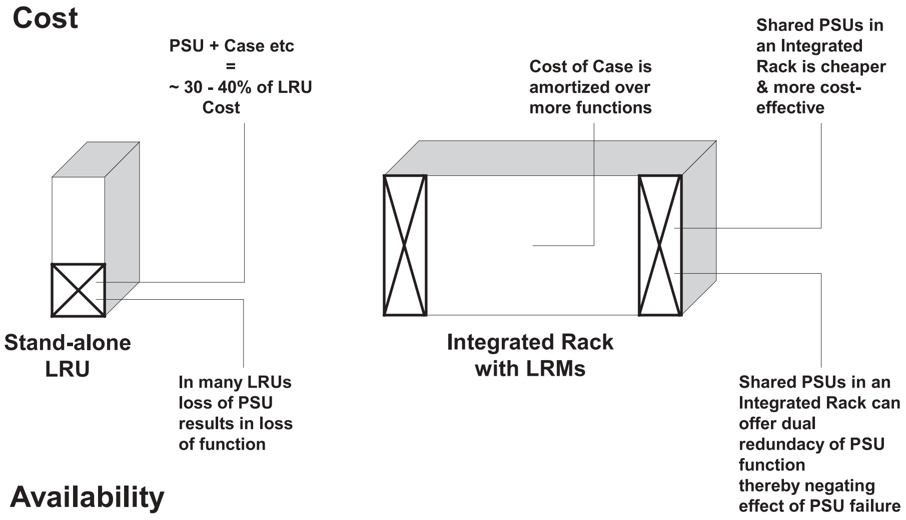
Fig. 11.1 Concept of using integrating cabinets



The effects of sharing PSUs across a number of electronic modules are highlighted in Fig. 11.2. In a stand-alone LRU, the cost of providing PSU, case EMI filters, and connectors may account for between 30 and 40 per cent of the total cost of the unit. Furthermore, in most LRUs the PSU is simplex, and if it fails the total function of the unit is lost. In an integrated rack, two or more PSUs may power the entire complement of electronic modules, and therefore economies of scale are possible. In this situation, when a PSU fails, the redundancy provided by the other unit(s) enables the rack to continue operating and full system functionality is retained. The move to the integrated rack concept thus offers significant savings in terms of cost and availability for this reason alone.

Further advantages are realized by the technology advances brought by later technology when applied within the context of the integrated rack implementation. Therefore, while some of the benefits are brought about by the integration concept,

Fig. 11.2 Benefits of using shared PSUs



others are due to a parallel improvement in microcircuit electronic technology and consequent reduction in the size of the electronic real estate. This can be seen by referring to the broad comparison between a typical ARINC 600 LRU produced in the mid-1980s and an integrated rack module manufactured in the mid-1990s in Fig. 11.3.

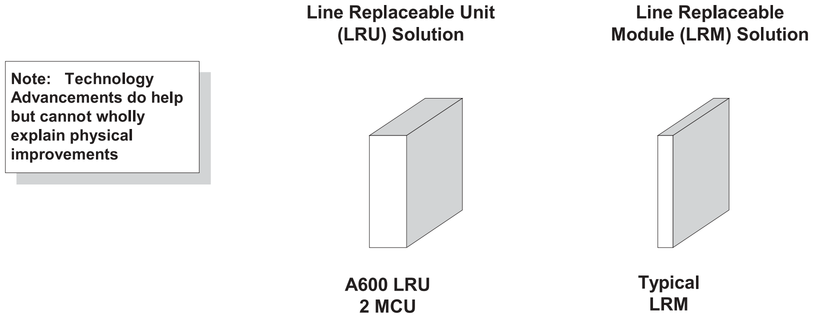
Advantages

A comparison of key unit characteristics in Fig. 11.3 highlights the following benefits for the later LRM implementation:

- Volume reduced by around 50 per cent.
- Weight reduced to approximately 30 per cent.
- Power dissipation reduced to about 16 per cent.
- Reliability improved by a factor of greater than 20 – this a significant increase that is partly due to micro-miniaturization of ICs and partly due to improved packaging.

These represent significant savings when applied across the total complement of LRMs that comprise the total aircraft system.

Other factors weigh in favour of IMA, which offers a unified approach to system design. Common processors may be used, whereas in an integrated but federated A600 solution a variety of different processors and accompanying software languages might be employed. Therefore, the IMA approach offers commonality of design and economies of scale that might be difficult to match using a more conventional solution. This has a beneficial impact on supportability, since fewer spares are required at maintenance centres and spares planning is eased.



	A600 LRU 2 MCU	Typical LRM
Volume:	**210 cu in** **[2.25 x 7.5 x 12.5]**	**~100 cu in** **[~ 10 x 12 x 0.85]**
Weight:	**6 - 8lbs**	**2.2lbs**
Power Dissipation:	**20 - 40W**	**~ 5W**
Reliability:	**~ 10,000hrs**	**> 200,000 hours**

Fig. 11.3 *Comparison of ARINC 600 LRU with an LRM*

Disadvantages

These advantages must be weighed against the disadvantages of using an integrated, common approach:

- The implementation might possibly be more expensive – in some areas the adoption of a common solution may lead to overkill.

- The solution may possibly be more risky as new designs and packaging are required.
- The IMA concept may pose proprietary problems by having different vendors working more closely together and each carefully guarding their intellectual property rights – to the overall detriment to the system and aircraft implementation.
- Segregation considerations (more eggs in one basket).
- The use of an 'open' or 'closed' architecture. The characteristics of an 'open' architecture are described in the next section.
- What standards will apply? A429 and A600 specify the data bus and LRU format in standard terms, but what standards should be used in the IMA/LRM solution for backplane interconnections, module intercommunication, and so forth?
- The implementation raises the spectre of who takes responsibility in the end.
- In some conditions, depending upon the input/output signal mix, there may be concerns regarding EMI compatibility and the prospect that certain modules in the cabinet may pose EMI hazards to other modules in the vicinity.
- Owing to some of the difficulties outlined, the IMA solution may possibly be more difficult to certify.

Clearly, there are some knotty questions to be answered. This is a major debate, and some of the early implementations of the concept that exist today will be described in this chapter.

In the view of many Original Equipment Manufacturers (OEMs), the advantages outweigh the disadvantages, which is why IMA is being so readily embraced for integrated avionics systems. In some cases, the integration may only be applied to a system or subsystems with the OEMs or subsystem integrators taking the responsibility of system integration. In others, particularly in the business jet and regional jet community, a more total system integration responsibility may be taken on by a major avionics systems company integrating the entire avionics systems functions: flight control and guidance, navigation, display, engine and utilities control, radio communications, and so on. Therefore, the avionics systems integrator takes on the system integration risk with the prize of having a much greater hardware and therefore value content to offset the integration development cost and risk. Examples of these contrasting approaches will be examined later in the chapter.

Open architecture issues

The use of open, user-friendly architecture is a common discussion theme in many of the Information Technology (IT) architectures available today, and similar arguments apply to the adoption of open architectures for IMA implementations. An open system may be defined as (**1**)

> A system composed of components with well-defined interfaces between components conforming to standard interface specifications.

A major prerequisite to the definition of an open system is the use of standard interfaces. These interfaces may address a number of characteristics:

- Standard module mechanical interface: dimensions, securing arrangements, weight.
- Standardized electrical characteristics: power requirements, types, power consumption.
- Standardized serial data bus interfaces.

- Standardized parallel/backplane bus characteristics.
- Analogue and discrete signal defined interfaces.

The use of all of these standardized elements infers the use of a modular system where individual modules may be updated when technology improvements permit. This enables the benefits of technology improvements to be adopted while ensuring that the architecture is still capable of functioning according to the original intent. Reference (**1**) is a guidance document relating to the use of open systems for avionics.

The use of Commercial Off-The Shelf (COTS) technology is complementary to the adoption of an open architecture: it does not supplant the need. Adoption of COTS is key in both military and civil avionics as the life cycle of commercial PC and IT components is a fraction of that of an avionics system. Moreover, the proportion of microelectronic components that the aerospace/avionics industry procures is an increasingly small fraction of the total procured by all industries.

Component obsolescence

The relatively small and increasingly diminishing proportion of electronic components used by the aerospace avionics industry is illustrated in Fig. 11.4. This diagram shows

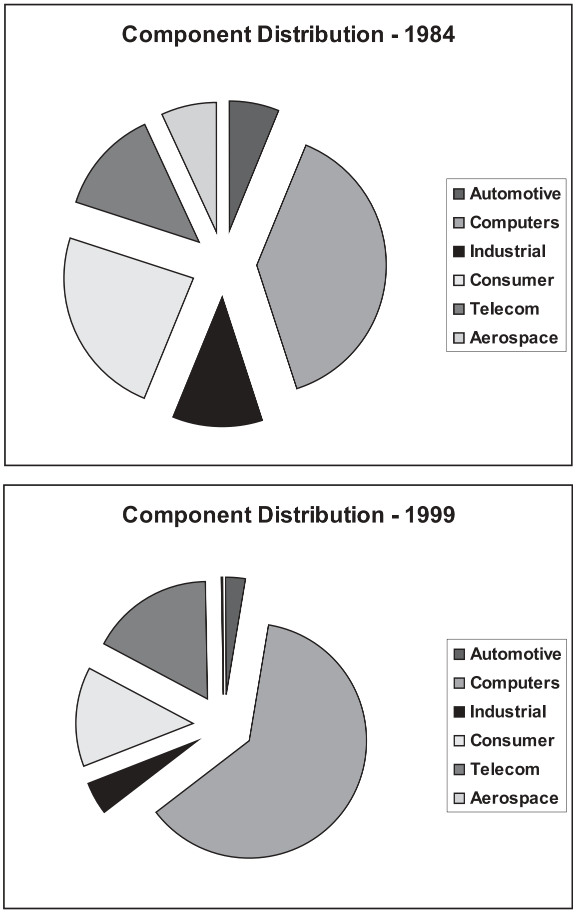


Fig. 11.4 *Comparison of electronic component use by industry*

pie chart presentations for the proportion of electronic components procured by various industries in 1984 and in 1999. The aerospace industry procured 7 per cent of all components in 1984 but only 0.7 per cent in 1999. This contrasts with the computer industry 39 per cent (1984) and 62 per cent (1999) and the telecommunications industry 13 per cent (1984) and 17 per cent (1999). This brings other problems such as component obsolescence to the fore. The gestation period and life cycle of components and technologies used in the computer industry is a fraction of that in the aerospace industry where the total life cycle of a system may extend to 40 years. This problem is particularly acute in the field of military avionics where the development time-scale for a fighter aircraft – from launch to entry into service – can be 10 or 12 years. While civil aircraft programmes have a quicker development time-scale, the problem may still be significant and requires a strategy to manage component obsolescence. Reference (**2**) outlines an approach being followed by Airbus to address component obsolescence.

Definition of IMA cabinets – first-, second- and third-generation implementations

As has already been mentioned, a number of IMA style implementations have been embodied in civil aircraft over the past decade, though there is a wide variation in the approach taken. Most of the present implementations have not embraced a totally open architecture; in some cases proprietary or non-standard backplanes have been adopted. Also, although COTS has been applied in some implementations, its adoption and use is not widespread.

In order to frame the IMA implementations developed so far in some sort of context, it is useful to address them in terms of first-, second-, and third-generation systems. In this contextual framework the first generations represent an earlier and more conservative approach to IMA integration, whereas the third generation addresses a more aggressive approach to solving the integration issues. While in no way being judgemental in comparing these differing and equally valid approaches, it is useful to compare and contrast each of the implementations at a top level to understand the direction in which the technology is evolving (see Fig. 11.5). Each generation may be broadly characterized by a summary of the salient technical details together with typical applications.

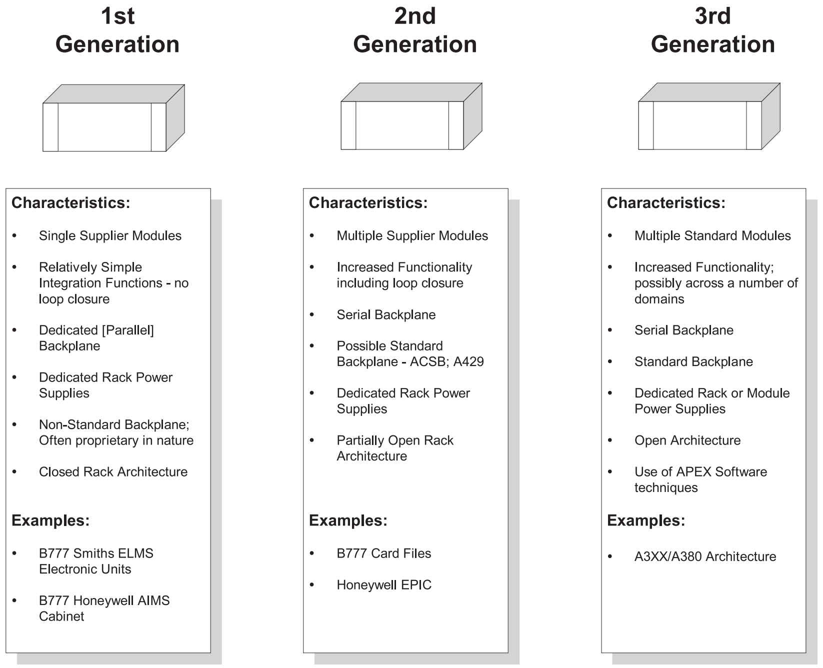
First-generation IMA

Characteristics

- Single supplier of modules. Functional software provided by the rack integrator.
- Relatively simple implementation – few or no examples of loop closure.
- Dedicated (parallel) backplane. Non-standard or single-use backplane or proprietary in nature.
- Dedicated/multiple-rack PSUs.
- Closed rack/cabinet architecture.

Application

- Boeing 777 Smiths Electrical Load Management System (ELMS) Electronic Units (EUs) – first revenue service June 1995.
- Boeing 777 Honeywell Airplane Information Management System (AIMS) – first revenue service June 1995.



1st Generation

Characteristics:

- Single Supplier Modules
- Relatively Simple Integration Functions - no loop closure
- Dedicated [Parallel] Backplane
- Dedicated Rack Power Supplies
- Non-Standard Backplane; Often proprietary in nature
- Closed Rack Architecture

Examples:

- B777 Smiths ELMS Electronic Units
- B777 Honeywell AIMS Cabinet

2nd Generation

Characteristics:

- Multiple Supplier Modules
- Increased Functionality including loop closure
- Serial Backplane
- Possible Standard Backplane - ACSB; A429
- Dedicated Rack Power Supplies
- Partially Open Rack Architecture

Examples:

- B777 Card Files
- Honeywell EPIC

3rd Generation

Characteristics:

- Multiple Standard Modules
- Increased Functionality; possibly across a number of domains
- Serial Backplane
- Standard Backplane
- Dedicated Rack or Module Power Supplies
- Open Architecture
- Use of APEX Software techniques

Examples:

- A3XX/A380 Architecture

***Fig. 11.5** Definition of IMA generations – first, second, third*

Second-generation IMA

Characteristics

- Multiple suppliers of modules. Functional software provided by the rack integrator and specialist module supplier – e.g. fuel system, proximity switch integration.
- Increased functionality, including more examples of loop closure.
- Serial backplane. Possibly a standard defined backplane: Avionics Standard Communications Bus (ASCB) or A429. However, the backplane implementation may be proprietary in nature which detracts from an open architecture.
- Dedicated/multiple-rack PSUs.
- Partially open rack/cabinet architecture.

Application

- Boeing 777 card files – first revenue service June 1995.
- Honeywell EPIC – Raytheon Horizon: first flight due 2002.

Third-generation IMA

Characteristics

- Multiple standard modules.
- Increased functionality across a number of domains.
- Serial backplane. Standard backplane using COTS components.
- Aircraftwide COTS interfacing data bus standard.
- Dedicated/multiple-rack PSUs.

- Open architecture.
- Use of Application Executive (APEX) software integration techniques.

Application

- Airbus A3XX/A380 integrated IMA/Avionics Fast Switched Ethernet (AFDX) aircraftwide architecture: first flight due 2005.

IMA Examples

To broaden the understanding of the IMA issues and address the architectures and respective functionality adopted so far, it is worthy reviewing several civil transport IMA examples. These are:

- Boeing 777 ELMS EUs.
- Boeing 777 AIMS.
- Boeing 777 card files.
- Honeywell EPIC.
- A380 IMA.

In each case the aircraft-level and backplane data buses are identified, together with the nature of the functions being integrated. Table 11.1 gives a useful summary.

Table 11.1 Top-level comparison of IMA implementations

Application	Generation data buses [aircraft/backplane]	Function	Characteristics and application
Smiths Boeing 777 ELMS	First [A629/parallel backplane]	Aircraft electrical and utilities	Load management, electrical power distribution load shed
Boeing 777 card files	Second [A629/A429 serial]	Aircraft utilities	Fire detection, hydraulics control, ECS functions, WOW interfacing
Honeywell Boeing 777 AIMS	First [A629/SAFEbusTM]	Avionics functions	Avionics suite including displays, flight management, communications
Honeywell EPIC	Second [ASCB/proprietary]	Avionics + some utilities (fuel, landing gear, etc.)	Integrated avionics suite including some utilities
Airbus AFDX network – furnished by Thales/Diehl	Third [AFDX/AFDX]	Aircraft-level interconnecting several aircraft level functional domains	Integrated system interconnecting cockpit, cabin, energy management, utilities, flight and engine control and domains

Boeing 777 – ELMS

The Boeing 777 ELMS architecture is portrayed in Fig. 11.6. The electrical power distribution power aspects have already been described in Chapter 4. It is the electronic architecture that is of interest in this chapter. Three of the power management panels host dual-redundant EUs, each of which are connected to the left and right A629 aircraft system data buses. Therefore, all three EUs are connected in a dual-redundant manner to all of the other aircraft systems controllers.

The EUs themselves are modular in construction and comprise a large number of common modules, all of which are line replaceable. Of a possible total of 39 electronic modules across the aircraft set (all three EUs), commonality is achieved as described in Table 11.2.

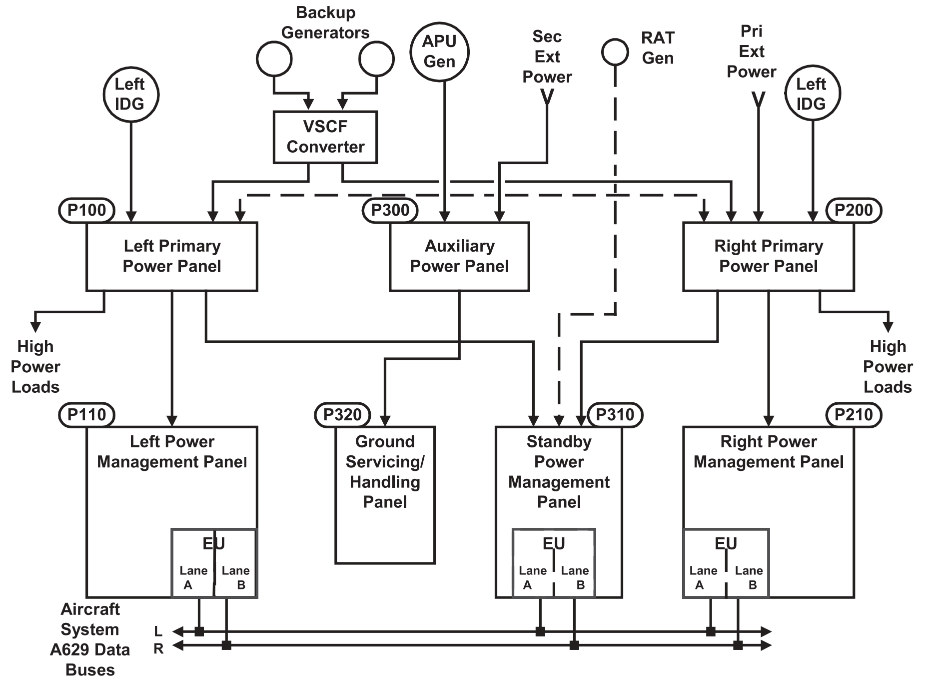


Fig. 11.6 *Boeing 777 ELMS EU architecture (Smiths Aerospace)*

Table 11.2 Boeing 777 ELMS EU commonality

Electronic module type	Quantity (per shipset)
CPU module	6
ARINC 629 module	6
Input/output type 1 module	18
Input/output type 2 module	6
Special function module	1
DC Subsystem module	1
PSU module	6
Total	**38**

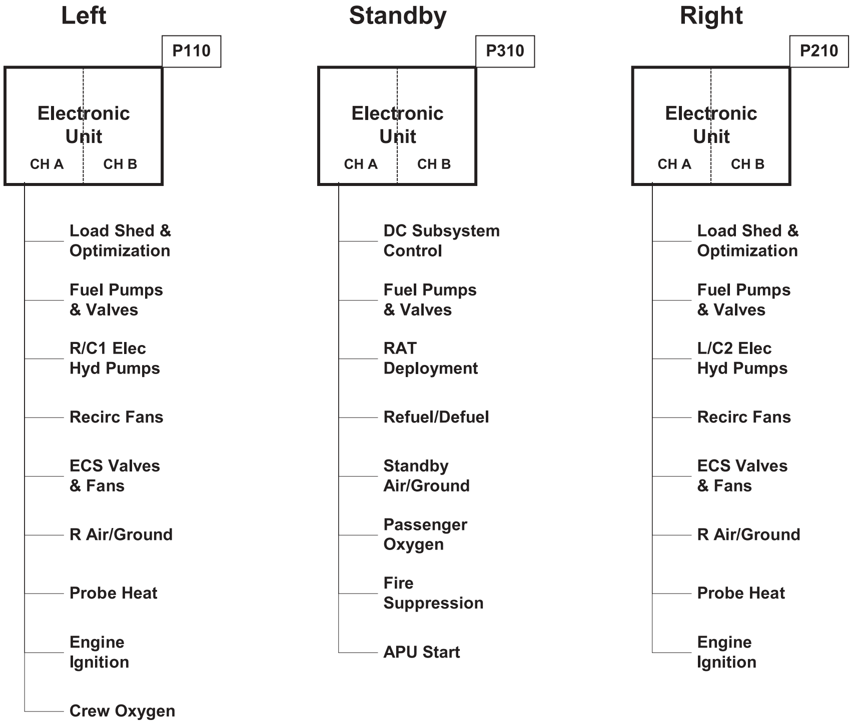
These modules were initially provisioned to provide intelligence for the electrical power distribution functions – both primary and secondary power distribution and protection. However, it was soon realized that the proximity of intelligence to the power distribution, coupled with the A629 data bus interfaces to other aircraft systems controllers, enabled additional functions to be embedded. A list of the major ELMS functions are tabulated in Fig. 11.7.

Some of these functions are flight critical in nature, e.g. as the probe heating and Ram Air Turbine (RAT) deployment capabilities. Other functions may be essential or even non-essential in nature. Therefore, by judiciously partitioning these functions between EUs and by functional module within the EU, any combined criticality effects were minimized.

Another advantage that the aggressive use of common electronic modules brought was an accelerated module experience database when the system entered airline service. United Airlines, the launch customer for the Boeing 777, achieved utilization rates of 4000 flying hours per annum on their Boeing 777 fleet. Because of the high levels of modularity within the system, individual module experience rates therefore increased by the appropriate multiple: A629 and CPU modules accrued service history at 24 000 h per annum per aircraft, whereas the popular I/O module type 1 gained service history at a rate of 72 000 h per annum per aircraft. This rapidly accruing service history led to module maturity being attained at a very early stage in the service history. The physical characteristics of the ELMS LRMs are shown in Fig. 11.8.

One other point worthy of note is that, owing due to the relatively low processing and high input/output nature of the system, power densities for the ELMS modules have been retained at low levels. Power dissipation per module is ~5 W, which means the

Fig. 11.7 Boeing 777 ELMS functions (Smiths Aerospace)



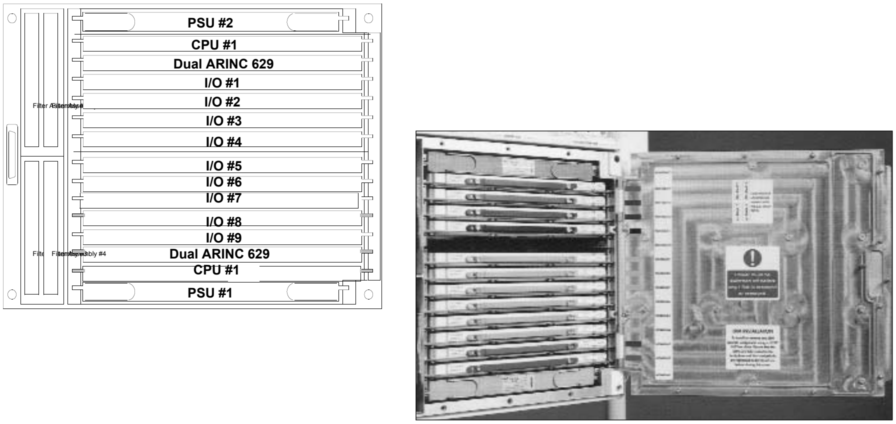


Fig. 11.8 *Boeing 777 ELMS EU and LRMs (Smiths Group)*

EUs do not require cooling air. This is a significant advantage compared with the higher power densities attained by the integration of avionics system functions.

Boeing 777– AIMS

By contrast with the ELMS, which is confined to the control of the aircraft electrical power distribution and a number of aircraft utility system functions, the AIMS represents the very heart of the aircraft avionics systems, as illustrated in Fig. 11.9. It performs the total integration of the following functions:

- Flight management – utilizing the experience gained on Boeing 757, 767, and 747-400 aircraft.
- Displays.
- Navigation.
- Central maintenance.
- Flight deck communications.
- Thrust management.
- Digital flight data.
- Engine data interface.
- Data conversion gateway.

The AIMS comprises two cabinets which perform all of these functions, interfacing with the aircraft by means of ARINC 629 aircraft system buses (4), ARINC 629 flight control buses (3), ARINC 429 data buses, and discrete and analogue I/O.

Information is displayed on the six primary flight deck Active Matrix Liquid Crystal Displays (AMLCDs) and three colour Control and Display Units (CDUs). A description of the Boeing 777 display philosophy is included in Chapter 7. Each crew member is provided with a cursor control unit to facilitate the flight crew–system interface.

The basic AIMS cabinet architecture is shown in Fig. 11.10. The architecture comprises Core Processing Modules (CPMs) and I/O modules (IOMs), which are line replaceable items. Functional partitions are divided within the CPMs according to the tasks indicated in the figure. The use of multiple functions hosted within common computing resources requires a robust software partitioning environment. The power dissipated by the cabinet is sufficiently high for forced air cooling to be required.

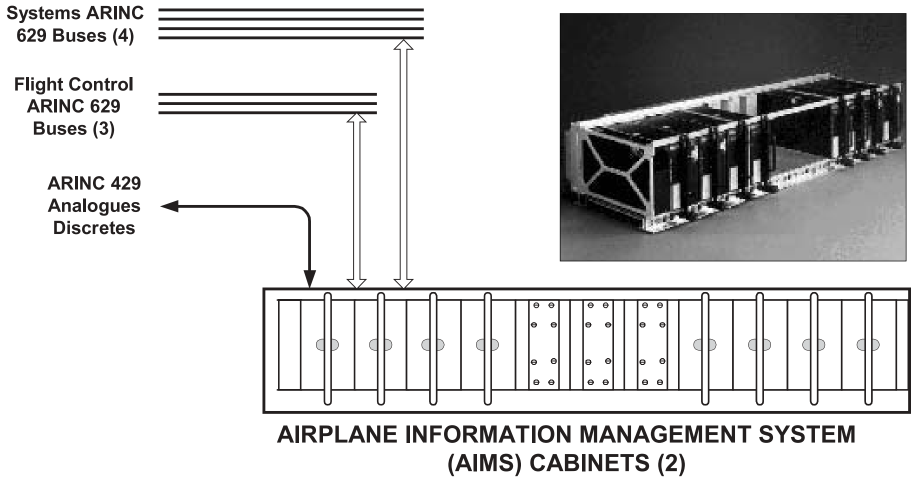


Fig. 11.9 AIMS cabinet (Honeywell)

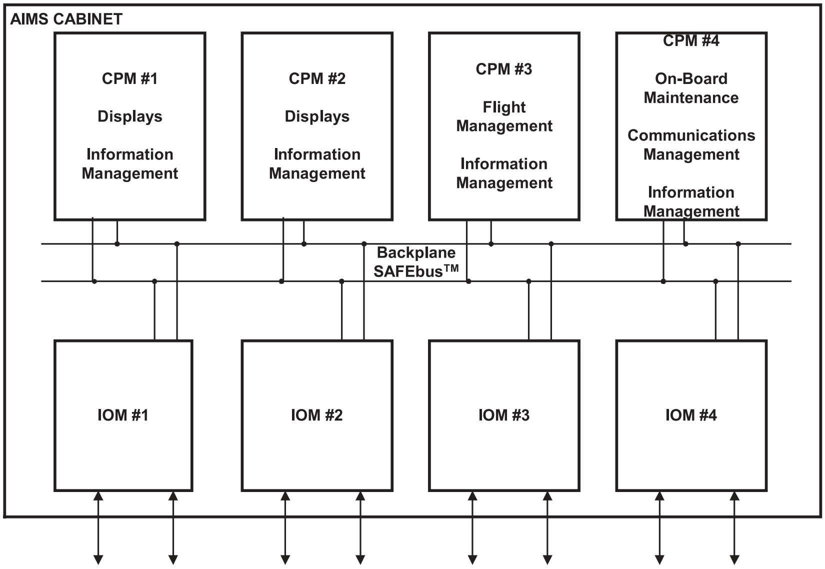


Fig. 11.10 Boeing 777 AIMS architecture

Modules communicate with one another via the SAFEbus® backplane which operates with the processors in a lock-step hardware architecture. BITE requirements are very demanding to assure the correct working of this architecture. It is claimed that the probability of an undetected hardware fault in the Honeywell proprietary CPM lock-step architecture is $<1 \times 10^{-9}$ per hour. The probability of generic hardware faults or faults that occur in both lanes on the same clock cycle are reported to have a probability of 1×10^{-12} per hour. References (**3**) and (**4**) give a more complete description of AIMS operation.

The AIMS is entirely manufactured by Honeywell using their proprietary backplane and as such is a closed architecture.

Boeing 777 – card files

The Boeing Commercial Aircraft Group (BCAG) took responsibility for the card file architecture used on the Boeing 777. The Boeing philosophy has been to use line replaceable cards for low-level aircraft utilities control functions on all their aircraft, but the 777 implementation of the P84 and P85 card files is more advanced.

Figure 11.11 shows the simplified interrelationship of the P84/P85 right and left card files with the AIMS cabinets, dedicated systems controllers, and the overhead panel interface. All are interconnected via the left and right ARINC 629 aircraft system data buses. The typical functions hosted within card files are illustrated in Fig. 11.12.

The main systems served by the P84 card file and functionally partitioned by hardware are:

- Hydraulic control and indication.
- ECS.
- Duct overheat detection.
- Fire detection.
- Weight-on-wheels.

The P85 card file serves a similar range of systems.

It can be seen that the card file architecture is that of a typical integrated rack or cabinet with dual PSUs, dual ARINC 629 gateways, and a serial backplane which interconnects the modules; in this case ARINC 429 data buses are used. Some of the specialized cards are outsourced by Boeing to the appropriate specialist suppliers who also supply the relevant sensors. Therefore, there are multiple suppliers for the cards in the card file and the architecture is open using standard avionic buses – ARINC 629/ARINC 429. Functional partitioning using hardware has advantages: as functions are segregated, they cannot interfere with each other, and software partitioning does not have to be considered. A further advantage of hardware partitioning is that fault diagnosis and rectification are eased.

The power dissipation of the cards that populate the P84 and P85 card files are such that convention cooling is sufficient.

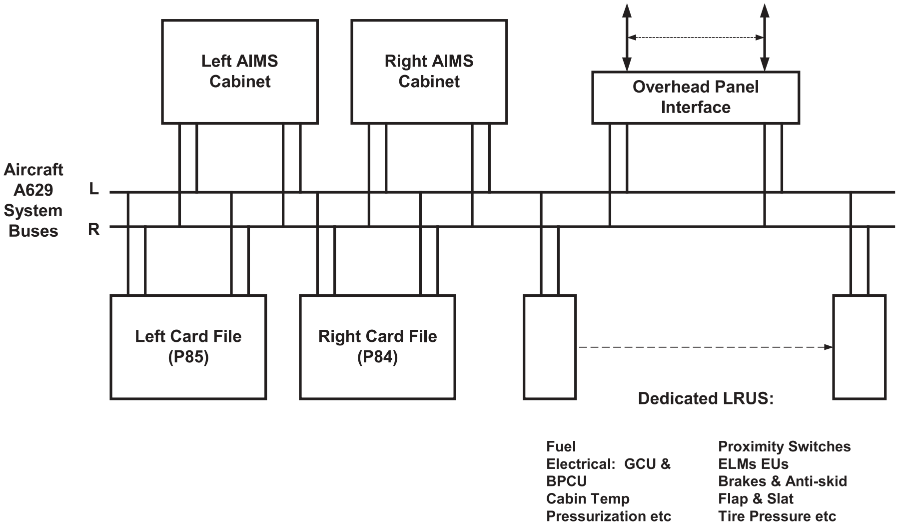
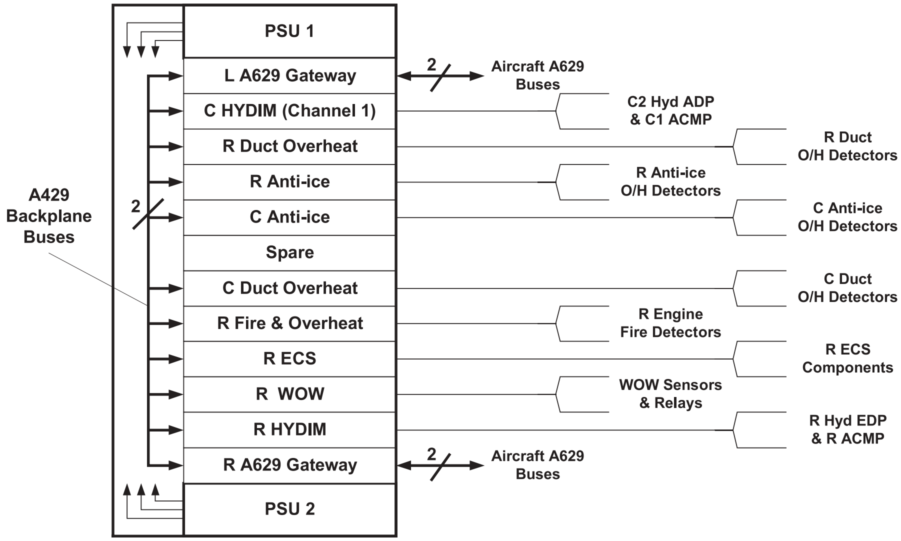


Fig. 11.11 *Interface of card files with aircraft system controllers – simplified*

Fig. 11.12 Boeing 777 P84 card file – simplified architecture



Honeywell EPIC

The Honeywell EPIC system is typical of the single-supplier, totally integrated system used on-board business jets and regional airliners. Rockwell Collins provide a similar system with their Pro-Line range of systems. The Honeywell EPIC system used as the example has been adopted for the Raytheon Horizon and Embraer 170/190 programmes.

In a way, the levels of integration used on-board business jets in particular represent the highest levels of integration of any civil avionics systems. This is because the avionics systems requirements are every bit as demanding for the top-level systems – Global Express, Gulfstream V, and Falcon 800 – as they are for aircraft such as Boeing 777 and A330/340. Many of these business jets have ranges in excess of 4000 miles, and, in the case of Global Express and Gulfstream GV, in excess of 6000 miles. Therefore, the levels of performance, accuracy, and integrity are equivalent to the larger long-range aircraft.

These aircraft are operated in a very competitive and dynamic marketplace. The introduction of fractional ownership means that these aircraft are heavily utilized and are no longer 'corporate toys' as perhaps earlier generations of business jets were. The high levels of utilization, coupled with demanding technical and systems availability requirements, mean that some of these systems are at the leading edge of modern integrated avionics systems thinking.

Two other factors propel the business jet community to the top of the systems integration pyramid. The first is the fact that the time to certify or recertify a new system or systems is shorter as:

- The basic airframe, though sophisticated, is simpler to certify than a commercial transport aircraft.
- The business drivers are very acute, which accentuates the risk/return benefits.

The Honeywell EPIC therefore demonstrates higher levels of integration than the Boeing 777 or Airbus A330/340 systems. The second major factor is that the shorter gestation period/development time-scales are extremely valuable in terms of being able to infuse COTS technology. The Honeywell EPIC is based upon an adaptation of commercial Ethernet 10BaseT technology, which was originally developed to serve the academic community. Honeywell have of course developed, adapted, and ruggedized the technology to the point where it can be used for flight-critical applications

The baseline EPIC architecture is portrayed in Fig. 11.13. The Avionics Systems Communication Bus (ASCB)-D variant is an adaptation of COTS Ethernet. Key features are:

- Five 8 × 10 in flat-panel AMLCD displays.
- Dual-redundant 10BaseT data buses.
- Bus architecture ruggedized to suit flight-critical functions.
- 10 Mb/s data rate.
- Up to 48 terminals (stations).
- Use of Cursor Control Devices (CCDs).
- Hosting of functional LRM modules in two or four Modular Avionics Units (MAUs).
- Software integration using a virtual backplane network, making the system highly adaptive and flexible and allowing customer options and preferences to be easily accommodated.
- Digital Engine Operating System (DEOS) providing an integral environment for differing levels of criticality: non-essential, essential, and flight critical. This enables the development engineer to use standard tools while still meeting FAA certification requirements.
- Integrated Sensor Suite (ISS) incorporating the following avionics sensor functions: ADC, GPS, IRS, AHRS.
- Integrated radio and audio system embracing the following communications and navaid functions: VOR, ADF, DME, ILS, VHF communications, ATC Mode S, and others.

For the flight crew, the interface with the EPIC system has been refined to provide the following interface capabilities:

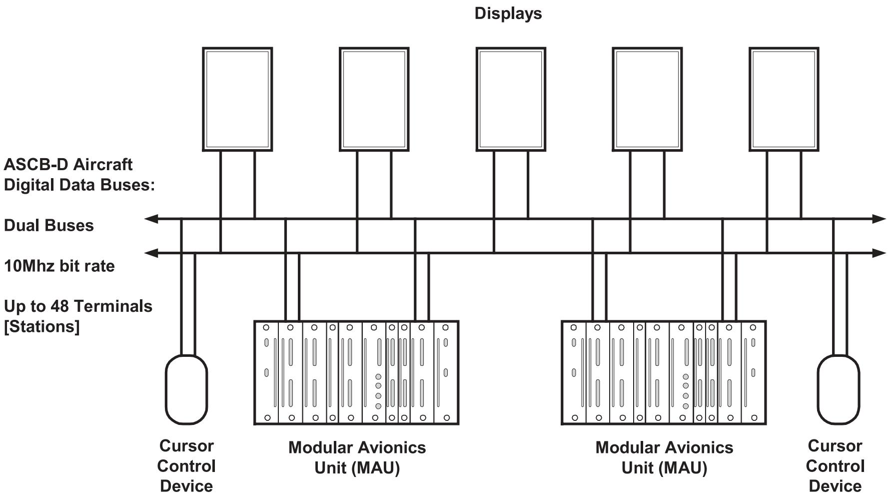


Fig. 11.13 *Honeywell EPIC architecture*

- AMLCD displays.
- EICAS.
- CCDs.
- Voice Command System (VCS).
- Head-Up Display (HUD).

From the functional viewpoint the system integrates many of the usual avionics functions such as fuel quantity gauging, weight-on-wheels and proximity switch sensing, hydraulics, air conditioning and pressurization, APU control, nosewheel steering, landing gear actuation, and so on. Therefore, many of the aircraft utilities functions are integrated alongside the classical avionics functions. Some of the heavier power applications are integrated into Power Distribution Assemblies (PDAs) which are supplied by the electrical power generation system integrator (Hamilton-Sundstrand).

A useful overview and description as to how the fuel quantity gauging system is integrated into the EPIC/MAUs architecture may be found in Reference (**5**). This presentation outlines how the Raytheon Horizon fuel system is integrated into the MAUs. A typical Honeywell EPIC MAU is shown in Fig. 11.14.

Therefore, in contrast to the AIMS implementation, the EPIC system uses COTS and an open architecture. Module power dissipation is such that forced air cooling is required to maintain the modules at a reasonable operating temperature.

Fig. 11.14 *Honeywell EPIC Modular Avionics Unit (MAU) (Honeywell)*

A380 IMA

The IMA implementations described so far generally, with the possible exception of EPIC as applied to business jets, only accomplish integration of part of the overall aircraft avionics fit. To achieve the full benefits of IMA it is necessary to craft a solution that may be applied aircraft wide. This raises other difficulties as, the broader the application, the more complex the implementation becomes in terms of diverging processor, I/O, and integrity drivers.

The benefits of applying IMA concepts aircraft wide may be summarized as follows:

- Common set of core hardware modules used across all functional domains.
- Standardized processing modules.

- Common use of software tools, standards, and languages.
- Dispenses with a multitude of specialized and dedicated LRUs.
- Ability to accommodate specialist interfaces: fuel, landing gear, etc.
- Realization of the benefits of scale across the entire aircraft.
- Improved logistics for the OEM and the airline.
- Provision of a scalable architecture with scope for application across the entire range of aircraft models in future implementations.

The Airbus IMA implementation seeks to realize these benefits by adopting an aircraftwide IMA architecture based upon a COTS adaptation of Fast Switched Ethernet (FDX) technology. The data bus selection called Avionics FDX (AFDX) is an adaptation of fast switched Ethernet technology using 100BaseT or 100 Mb/s twin wire technology.

The A380 implementation envisages a multicabinet architecture, which serves the following functional domains:

1. Cockpit domain. Encompassing the avionics functions, the cockpit domain is involved with interfacing with the primary displays. The cockpit domain comprises four cabinets.
2. Cabin domain. The cabin domain is associated with all those functions associated with the passenger cabin – environmental, water and waste, cabin information distribution system. The cabin domain functions are hosted in two cabinets.
3. Energy domain. Electrical power generation and hydraulic and bleed air control functions are contained within the energy domain, which consists of two cabinets.
4. Utility domain. Utility management functions associated with the landing gear fuel system and main gear and nosewheel steering are encompassed within the utility domain, housed in two cabinets.

The flight control and engine control functions are not contained within the overall IMA/AFDX switching complex but are functional domains within their own right that interface with the rest of the aircraft systems by more conventional means.

Functional domains are interconnected by means of a dual-redundant AFDX switching network using COTS-derived AFDX switches. The overall architecture implementation is more reminiscent of a typical telecommunications switching network rather than an aircraft avionics data bus implementation with which avionics engineers are familiar. Figure 11.15 portrays a top-level overview of the A380 AFDX system, showing the layout of the key functional domains and their interfaces with the AFDX switches. The very high bandwidth AFDX switching network effectively allows message packets to be dispatched around the network, with data transmissions interleaved rather being passed directly in discrete messages as in conventional avionics data buses.

Key elements of these functional domains are:

- Common Processor Modules (CPMs). All the CPMs are sourced from a single supplier – for the A380 this will be a consortium comprising Thales and Diehl.
- Standard I/O Modules (SIOMs).
- Specialized I/O modules for systems such as fuel, landing gear, etc., procured from the specialist suppliers who also supply the appropriate sensors.
- Software integration is accomplished by means of an Application Executive

Fig. 11.15 A380 AFDX switching network

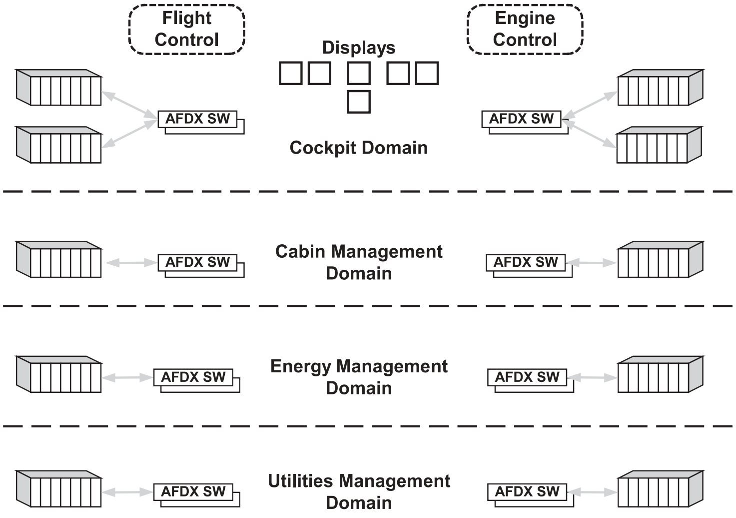
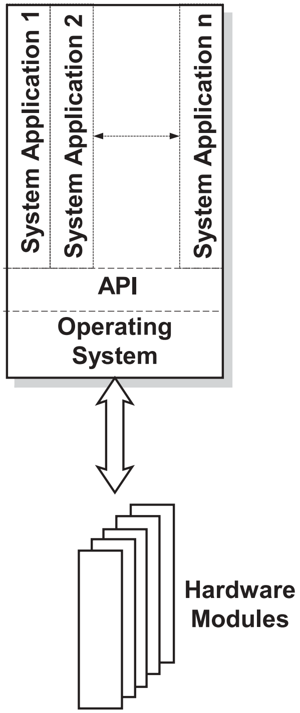


Fig. 11.16 Application executive



(APEX) where differing software functions hosted by the CPM are robustly partitioned from one another. Therefore, all the CPMs are multifunctional, hosting the application software for many aircraft system functions (see Fig. 11.16).

- Single supplier for AFDX switches – for the A380 this will be Rockwell Collins.

This method of implementation significantly alters the responsibilities for systems integration. Systems suppliers will encode application software, which will be hosted within the Thales/Diehl CPM. Significant systems integration responsibilities will fall upon the shoulders of the OEM companies who take on the task of integrating the aircraft functional domains.

References

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CHAPTER 12

Future Air Navigation System

The rapidly increasing air traffic density is leading to a pressing need to improve the Air Transport Management (ATM) system by all available means and move on from the techniques and technologies that have served the industry for the last 40 years. This evolution will embrace the use of new technologies mixed with existing capabilities to offer improved air traffic management.

The aims of ATM may be summarized as follows:

- To maintain or increase levels of safety.
- To allow dynamic accommodation of user-preferred three- and four-dimensional flight trajectories.
- To improve the provision of information to users in terms of weather, traffic situation, and services.
- To increase user involvement in ATM decision-making, including air–ground computer dialogues.
- To organize airspace in accordance with ATM procedures.
- To increase system capacity to meet traffic demand.
- To accommodate a full range of aircraft types and capabilities.
- To improve navigation and landing capabilities to support advanced approach and departure procedures.
- To create, to the maximum extent possible, a seamless continuum of airspace where boundaries are transparent to the user.

To this end, the air traffic control authorities, airline industry, regulatory authorities, and airframe and equipment manufacturers are working to create the Future Air Navigation System (FANS) to develop the necessary equipment and procedures.

The areas where improvements may be made relate to Communications, Navigation, and Surveillance, commonly referred to as CNS. The key attributes of these improvements may be briefly summarized as:

1. Communication. The use of data links to increase data flow and permit the delivery of complex air traffic control clearances.
2. Navigation. The use of GPS in conjunction with other navigational means to improve accuracy and allow closer spacing of aircraft.
3. Surveillance. The use of datalinks to signal aircraft position and intent to the ground and other users.

These headings form a useful framework to examine the CNS/ATM improvements already made and those planned for the future.

Communications

The main elements of improvement in communications are:

- Air–ground VHF data link for domestic communications.
- Air–ground SATCOM communications for oceanic communications.
- High-Frequency Data Link (HFDL).
- 8.33 kHz VHF voice communications.

Air–ground VHF data link

The emergence of data links as means of communications versus conventional voice communications is shown in Fig. 12.1. Voice links have been used in the past for communications between the air traffic control system or ATM and the Airline Operational Centre (AOC). The use of data links, controlled and monitored by the FMS or other suitable method on-board the aircraft, may communicate with the AOC and ATM system via data links. These data links may be implemented using one or more of the following:

- VHF communications.
- Mode S transponder.
- Satellite links.

Data link communications are being designed to provide more efficient communications for ATC and Flight Information Services (FIS). Although these systems essentially replace voice communication, there will be a provision for voice back-up in the medium term.

Flight plan data, including aircraft position and intent in the form of future waypoints, arrival times, selected procedures, aircraft trajectory, destination airport, and alternatives, will all be transferred to the ground systems for air traffic management. The data sent to the ground ATM system will aid the process of predicting a positional vector for each aircraft at a specific time. This information will aid the task of the ground controllers for validating or reclearance of an aircraft's flight plan. Furthermore, use of the Required Time of Arrival (RTA) feature will enable the air traffic controllers to reschedule aircraft profiles in order that conflicts do not arise.

For ATC flight service and surveillance, VHF Data Link (VDL) communications will be increasingly used for domestic communications. VHF communications are line-of-sight limited, as has already been explained. A number of options exist:

- VDL mode 1. Compatible with existing ACARS transmitting at 2.4 kb/s. This mode suffers from the disadvantage that it is character oriented.

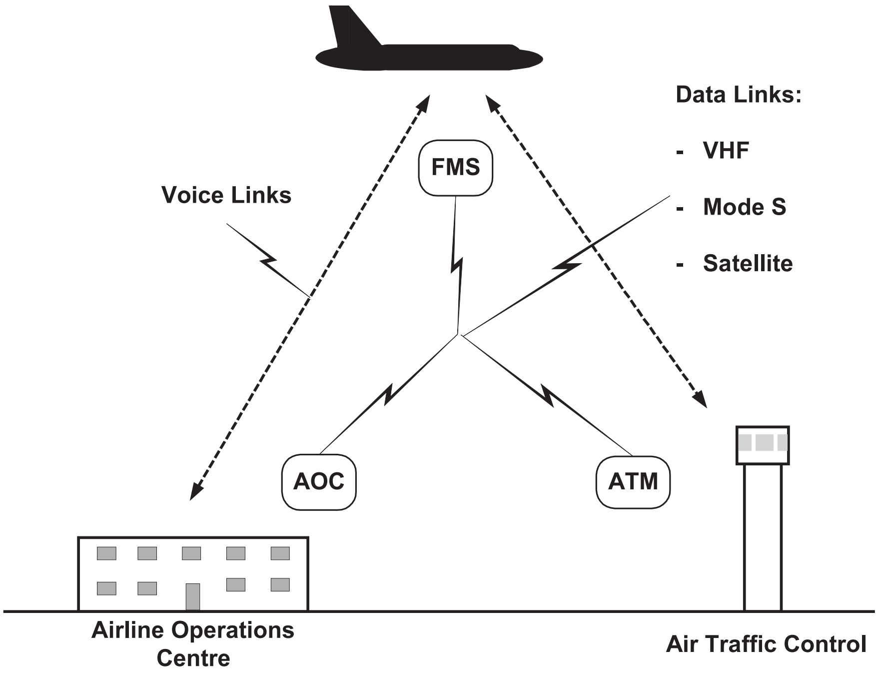


Fig. 12.1 *The emergence of data links*

- VDL mode 2. Data transmitted at 31.5 kb/s. As well as having a higher bandwidth, this protocol is bit rather than character oriented, making it 50–70 per cent more efficient than the ACARS protocol. VDL mode 2 is able to support Controller-to-Pilot Data-Link Communications (CPDLC).
- VDL mode 3. Simultaneous data and analogue voice communications using Time Division Multiple-Access (TDMA) techniques.
- VDL mode 4. Used with the 1090 MHz signal of ATC mode S.

It is expected that the introduction of data-link technology will benefit all users owing to the more efficient and less ambiguous nature of the messages passed. Significant improvements in dispatch delays and fuel savings are expected as these technologies reach maturity.

Air–ground SATCOM communications

SATCOM is a well-proven data link that, as has already been explained, is limited at very high latitudes in excess of about 82º. The SATCOM system is supported by the INMARSAT constellation already described in Chapter 6.

HF data link

Modern technology enables HF data-link transmissions to be more robust than HF voice and therefore less susceptible to the effects of the sunspot cycle. HF data link provides primary coverage out to 2700 nautical miles and secondary coverage beyond that should propagation conditions be favourable. There is extensive cover by ground stations located in the northern hemisphere such that HF data link is a viable alternative to SATCOM for north polar transitions (see Chapter 6).

8.33 kHz VHF voice communications

Conventional VHF voice channels are spaced at intervals of 25 kHz throughout the spectrum. A denser communications environment has resulted in the introduction of digital radios that permit spacing at 8.33 kHz, allowing three channels to be fitted in the spectrum where only one could be used previously. With effect from the 7 October 1999, these radios have already been mandated in Europe for operation above 20 000 ft and mandatory use will follow in the United States within a number of years; one of the difficulties in predicting the time-scale is the vast number of radios that have to be replaced/retrofitted.

Navigation

A number of navigational improvements are envisaged:

- Introduction of Required Navigation Performance (RNP) and Actual Navigation Performance (ANP) criteria. This defines absolute navigational performance requirements for various flight conditions and compares these with the actual performance the aircraft system is capable of providing.
- Reduced Vertical Separation Minima (RVSM).
- Differential GPS (DGPS) enhancements:
  - WAAS,
  - LAAS.
- Protected Instrument Landing System (ILS).
- Introduction of Microwave Landing System (MLS)
- Polar routes.

Classical method for defining navigation performance

Area navigation (RNAV)

RNAV navigation systems allow the aircraft to operate within any desired course within the coverage of station-referenced signals (VOR, DME) or within the limits of a self-contained system capability (INS, GPS) or a combination of these. RNAV systems have a horizontal two-dimensional capability using one or more of the on-board navigational sensors to determine a flight path determined by navigation aids or waypoints referenced to latitude and longitude. In addition, the RNAV system provides guidance cues or tracking of the flight path. Many modern RNAV systems include a three-dimensional capability to define a vertical flight path based upon altimetry, and some include a full aircraft and engine performance model.

The benefits of RNAV may be shown by a simple example. A typical route length from Boston to Miami using conventional airway routing would typically be around 1198 nautical miles. The same route flown using RNAV could be in the region of 1113 nautical miles, a saving in distance of 85 nautical miles or about 7 per cent with a corresponding saving in time and fuel.

The performance of pre-RNAV systems has historically been defined according to the criteria in Fig. 12.2:

- Along-track error.
- Across-track error.
- Flight Technical Error (FTE)

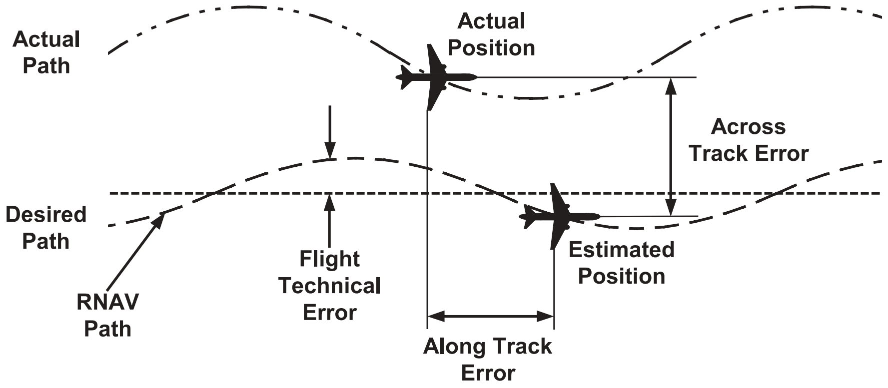


Fig. 12.2 *Historic RNAV error composition*

The total navigation error is the Root Sum Squared (RSS) of these elements for a given navigation means or phase of flight.

The availability of the navigation capability is defined at 99.999 per cent and the integrity requirement for misleading navigation information is set at 99.9999 per cent. Navigation accuracies and FTE are typically demonstrated within the values given in Table 12.1 for differing phases of flight

Table 12.1 Demonstrated accuracy and FTE per phase of flight

Airspace/Operation	Accuracy (95%) (nautical miles)	FTE (nautical miles)
Oceanic/en route remote	± 3.8	± 1.0
Domestic en route	± 2.8	± 1.0
Terminal	± 1.7	± 0.5
Approach – VOR/DME	± 0.5	± 0.125
Approach – multisensor	± 0.3	± 0.125

Vertical navigation (VNAV)

A VNAV capability further enhances flight operations by specifying a vertical flight path to be flown coincident with the lateral navigation guidance, providing a three-dimensional path for the aircraft to follow. As for the lateral guidance case, path deviation and guidance for tracking the path are provided. Key elements in VNAV are:

- Computed vertical flight paths as the basis for aircraft guidance.
- Assured repeatability of performance.
- Tailoring the flight path to aircraft/engine performance – full-performance VNAV.
- Providing situational awareness cues to the flight crew.

VNAV capabilities have been defined by the error components defined in Fig. 12.3. and listed below:

- Flight path definition error.
- Altimetry error (99.7 per cent).
- FTE.

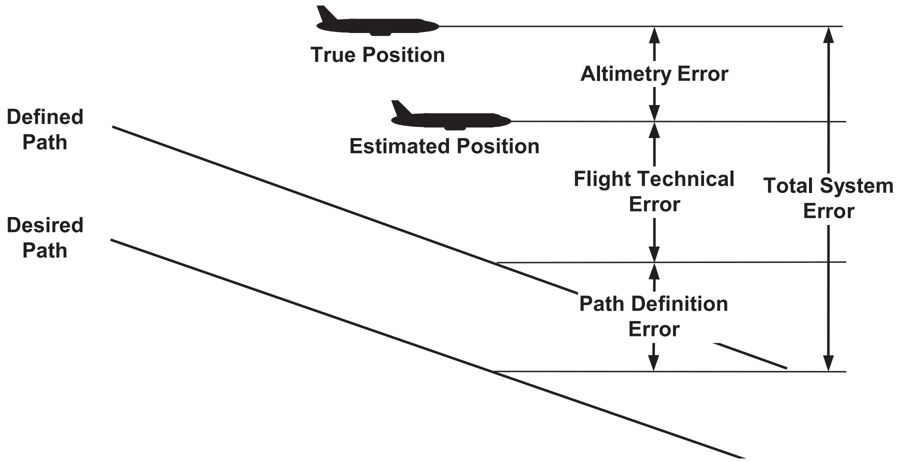


Fig. 12.3 *Historic VNAV error composition*

The total demonstrated VNAV system error is the RSS of these values, depending upon the altitude and flight trajectory conditions given in Table 12.2.

Table 12.2 Demonstrated VNAV performance and FTE by altitude and flight condition

Airspace/Operation	Level 99.7%	Level/intercept FTE	Climb/descent 99.7%	Climb/descent FTE
At or below 5000 ft	±50 ft	±150 ft	±100 ft	±200 ft
5 000 to 10 000 ft	±50 ft	±240 ft	±150 ft	±300 ft
Above 10 000 ft	±50 ft	±240 ft	±220 ft	±300 ft

Note: This performance is consistent with the 1000 ft RVSM criteria for FL290 and above described later.

Each type of navigation aid has its own error characteristic: those for VOR/DME and DME/DME will be different as indicated below.

VOR/DME error

VOR/DME characteristics will vary according to:

- Distance from station, for a specified angular error the lateral error will vary with distance.
- Altitude will affect coverage and slant range.
- Station to station distance will affect the error; the further the stations are apart, the greater the error at the mid-point.
- Course error.

The maximum angular error defined by reference (**1**) is ±4.5° (see Fig. 12.4).

DME/DME error

DME/DME error characteristics will vary according to:

- Station site.
- Slant range.
- Number of stations – choice of different stations upon which to take a fix.
- Relative geometry between stations, as shown in Fig. 12.5.

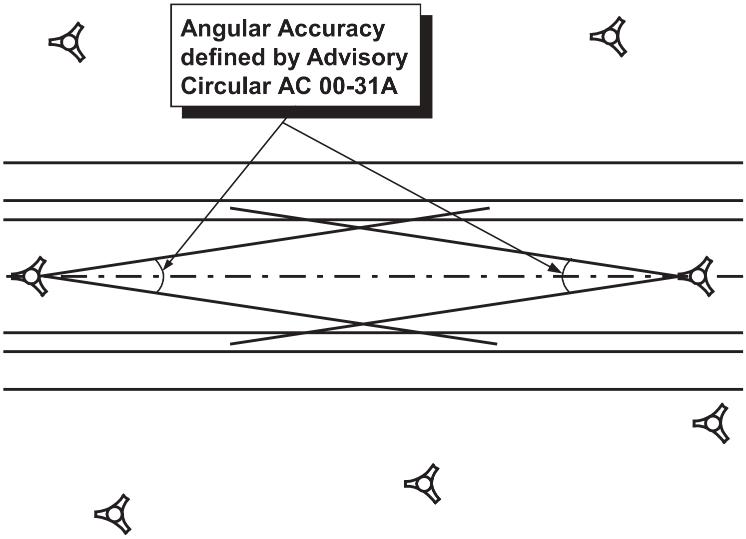


Fig. 12.4 *VOR/DME error characteristic*

The maximum range error defined by reference (**1**) is ±0.5 nautical miles or 3 per cent, whichever is the greater.

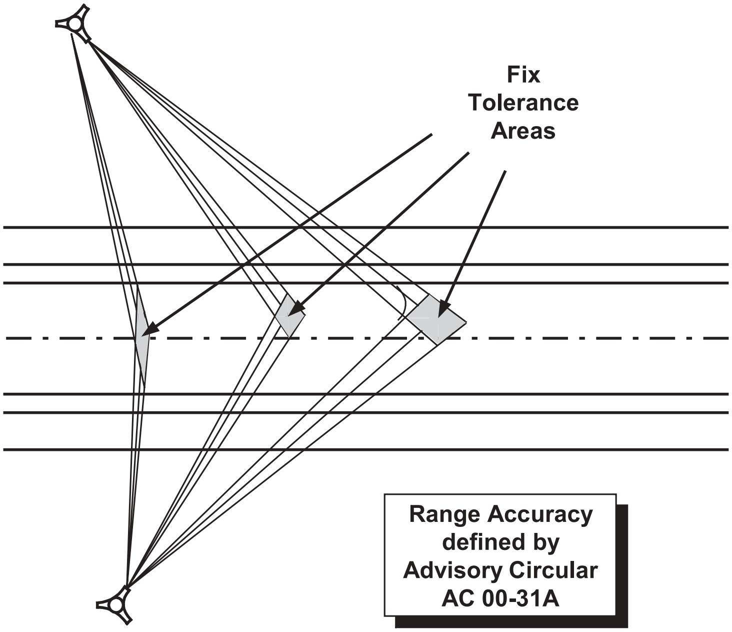


Fig. 12.5 *DME fix tolerance areas*

The overall effect of these and other factors leads to inefficient airspace design and procedures in the form of large airspace buffers and non-optimum separation minima for aircraft. The effect of different navigation infrastructures with different performance has a major impact on route widths, airspace use, and buffers. Fix tolerance areas are determined on the basis of the uncertainty resulting from bearing and range variations. This is compounded by the fact that historical airspace design and usage accounts for the lowest common denominator – the aircraft with the lowest performance navigation system. Although aircraft with more capable navigation systems may be able to navigate more accurately, they are unable to capitalize upon this fact as the airspace has always to allow for the worst case. As air traffic densities continued to increase, it became apparent that this situation could not continue.

RNP RNAV

Actual Navigation Performance

The Actual Navigation Performance (ANP) of the aircraft navigation system is represented by a circle defining the accuracy of the aircraft navigation system for 95 per cent of the time. The value of ANP is derived by taking the value of all of the navigation sensors and statistically weighing them against the other sensors. After a period of time, a degree of confidence is established as to which are the most accurate sensors, and therefore the ANP value is established. The 95 per cent probability circle is compared with RNP to decide whether the navigation system performance is good enough for the route segment being flown. The ANP and RNP values are displayed on the FMS CDU such that the flight crew can readily check on the navigation system status. Should the ANP exceed the RNP value for a given route sector for any reason, e.g. because of failure of a critical navigation sensor, the crew are alerted to the fact that the system is not maintaining the accuracy necessary. This will result in the aircraft reverting to some lower-capability navigational means, and in an approach guidance mode it may necessitate the crew executing a go-around and reinitiating the approach using a less accurate guidance means.

Required Navigation Performance

The RNP defines the lateral track limits within which the ANP circle should be constrained for various phases of flight. The general requirements are:

- For oceanic crossings the RNP is ±12 nautical miles, also referred to as RNP-12.
- For en route navigation the RNP is ±2 nautical miles (RNP-2).
- For terminal operations the RNP is ±1 nautical miles (RNP-1).
- For approach operations the RNP is ±0.3 (RNP-0.3).

Other specific RNP requirements may apply in certain geographical areas, e.g. RNP-4 and RNP-10 (see Fig. 12.6).

It is clear that this represents a more definitive way of specifying aircraft navigational performance, versus the type of leg being flown, than has previously been the case. Other more specific criteria exist: RNP-5 (also known as BRNAV or basic area navigation) has already been introduced in parts of the European airspace with the prospect that RNP-1 (also known as PRNAV or precision navigation) will be introduced in a few years. There are precision approaches in being – notably those in Juneau, Alaska – where RNP-0.15 is required for new precision approaches developed for mountainous terrain. The characteristics of this approach will be described later.

The effect of using RNP to specify the route structure versus the historical approach is shown graphically in Fig. 12.7. The pre-RNP approach is shown on the left of the diagram. The route width ranges from 60 up to 100 nautical miles. This enormous buffer is required to mitigate against the combined effect of navigation errors, navigation performance, route, traffic density, surveillance, communications, and ATC.

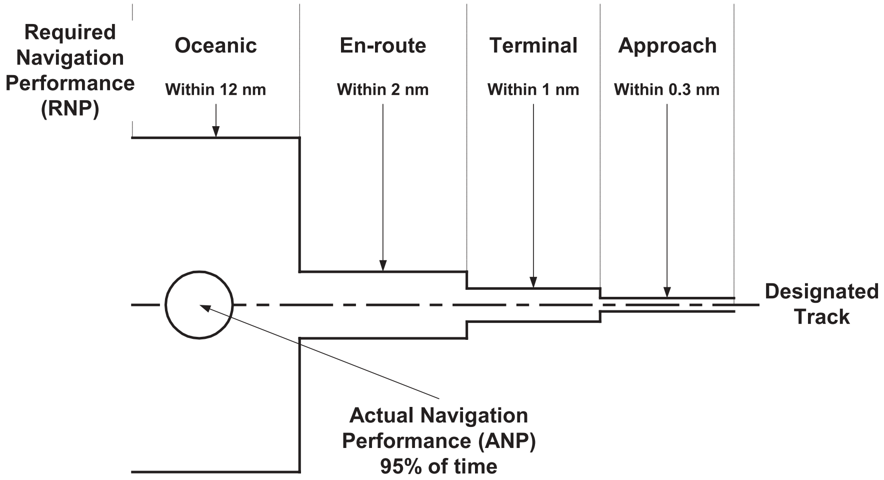


Fig. 12.6 *ANP versus RNP requirements*

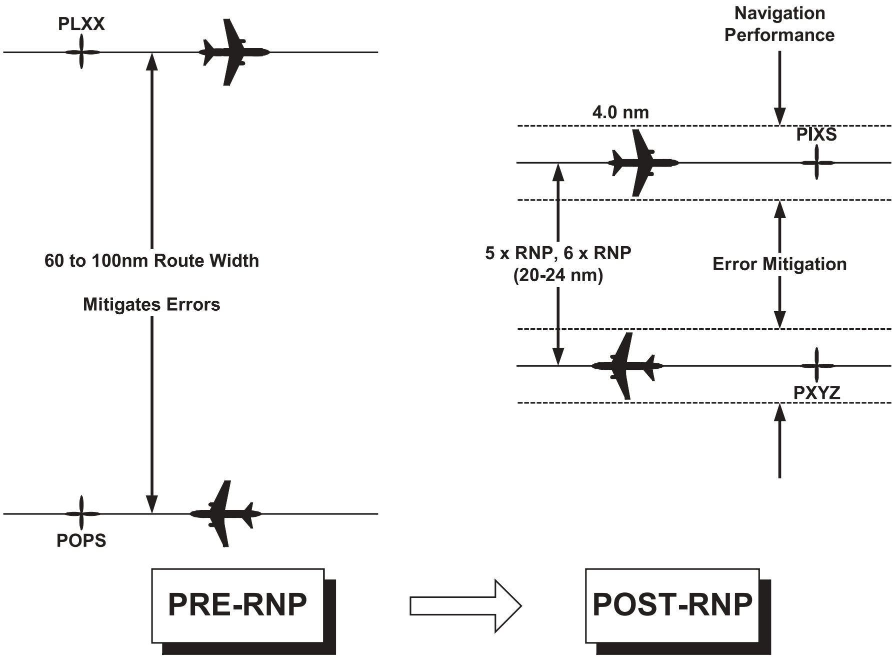


Fig. 12.7 *Comparison of historical and RNP definitions of route structure*

This approach considers the worst combination of factors that may apply. The RNP approach, together with other FANS improvements in surveillance and communications, leads to the situation shown on the right.

The post-RNP philosophy defines the route separation and error mitigation buffer in terms of multiples of the RNP performance. For the 4.0 nautical miles route (RNP-4) example shown, the route width is ~20–24 nautical miles depending upon the RNP multiple used. This compares most favourably with the pre-RNP route structure.

Similar spacing considerations apply along the route. As the navigational performance also affects along-track accuracy, similar trackwise improvements in separation may be achieved.

RNAV standards within Europe

Two RNAV standards are being developed in Europe, these are:

- Basic RNAV (BRNAV). BRNAV was introduced in 1988 and is equivalent to RNP-5 for RNAV operations. Navigation may be accomplished by using the following means:
  - DME/DME,
  - VOR/DME with a 62 nautical miles VOR range limit,
  - INS with radio updating or limited to 2 h since last on-ground position update,
  - LORAN C with limitations,
  - GPS with limitations.

  Until 2005, primary sources of navigation will be DME/DME, VOR/DME, and GPS. Reference (**3**), Advisory Circular AC 90-96, Approval of US Operators and Aircraft to Operate under Instrument Flight Rules (IFR) in European Airspace designated for Basic Area Navigation (BRNAV), 20 March 1998, approves the operation of US aircraft in European airspace under the application of existing advisory circulars.
- Precision RNAV (PRNAV). PRNAV is intended to be introduced at some time in the future but not before 2005. PRNAV will invoke the use of navigation under RNP-1 accuracy requirements or better.

The effect of using RNP techniques allied to the navigational capabilities of a modern aircraft needs to be examined in the context of the key areas of error defined above, namely:

- Path definition error – the difference between the desired flight path and actual flight path.
- Path steering error – the ability of the pilot and/or autopilot system to conform to the defined flight path.
- Position estimation error – the ability of the navigation system to estimate position and the level of integrity that may be assured in that process.

Path definition error/RNP fixes

The process of introducing higher-precision navigation criteria has led to a critical review of the nature of fixes presently allowed and their viability within the RNP framework. Certain types of fix are rigidly defined and relate to the position of a beacon or as defined by inertial coordinates – today usually GPS derived. Examples are given of these fixes (illustrated in Fig. 12.8). These show that they are absolute in their characterization:

- Initial Fix (IF)
- Track to a Fix (TF)
- Constant Radius to a Fix (RF)
- Holding to Fix (HX)

Each of these fixes is unambiguously defined to a particular point in the route structure; there are no conditional constraints that apply. Fixes with this type of characterization are ideal for specifying a robust route structure.

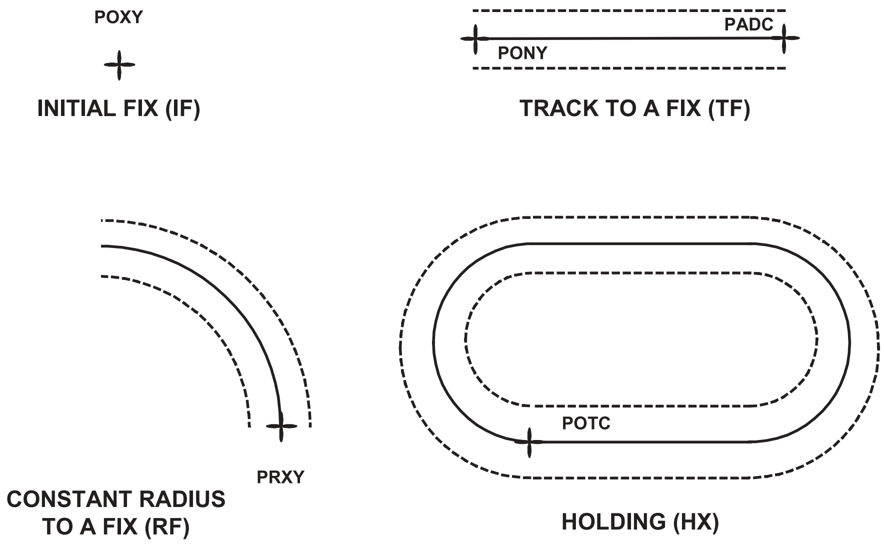


Fig. 12.8 *Preferred fixes for RNP*

By contrast, fixes that are conditional or depend to some degree upon a time conditional or dynamic constraint are less favoured in the proposed route structure. The examples shown in Fig. 12.9 all have conditional constraints. The Direct to a Fix (DF) example has a variable starting point that will affect the aircraft dynamics when arriving at the (DF) fix point. Similarly, Fix to an altitude (FA) will depend upon an individual aircraft performance or ability to climb to an assigned altitude, and other key navigation parameters may be suppressed. In essence, although the start point is defined, the end point is indefinite. Finally, the example given for Course to a Fix (CF) only relates to the course that the aircraft is flying when the designated fix point is reached. The future course of the aircraft beyond this point will depend upon errors accumulated in the previous leg on account of accuracy of the previous fix, wind variation, and other sources. The course to be followed after this fix will not necessarily agree with the aircraft's defined route/flight plan.

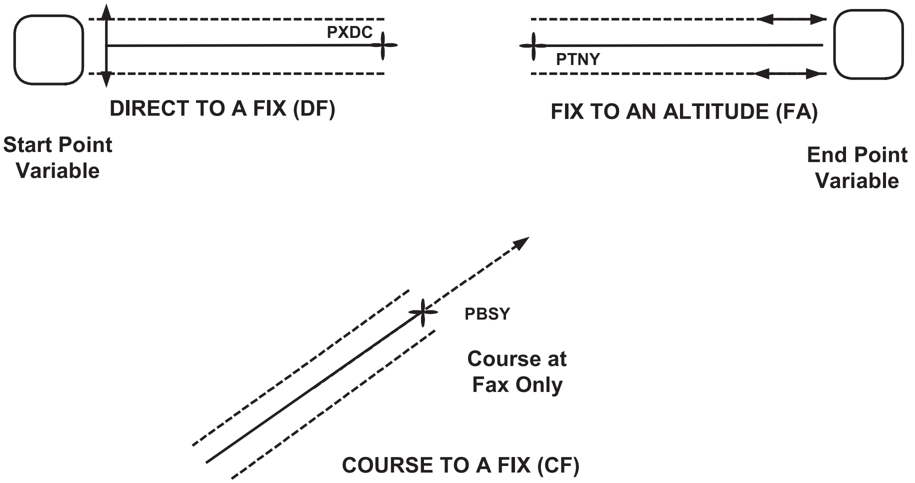
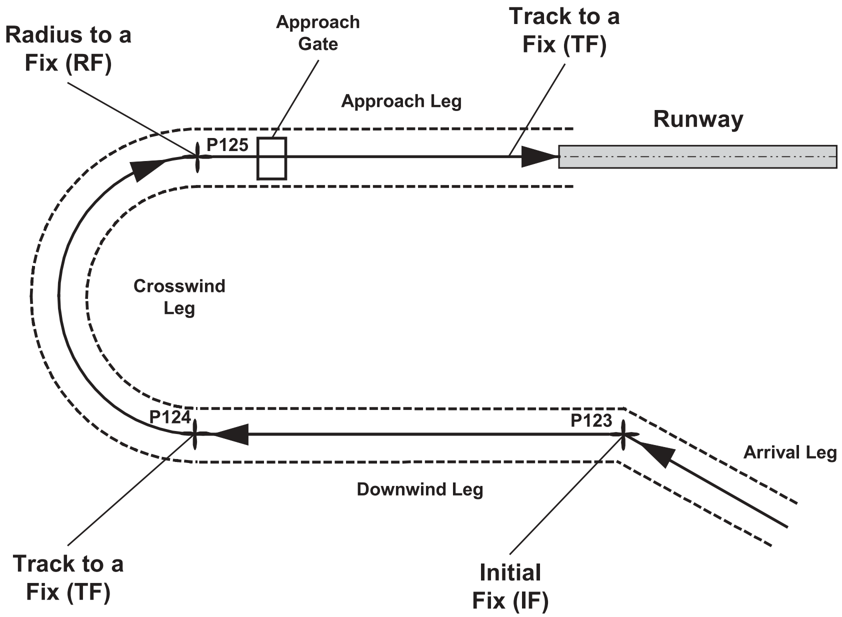


Fig. 12.9 *Fixes which are discouraged for RNP*

Fig. 12.10 Route structures developed for RNP



It follows that, to define and maintain a robust and repeatable route structure, the emphasis needs to be placed upon those navigation legs that are most rugged in their ability to minimize flight path error. The combination of such flight legs may lead to a structure similar to that shown in Fig. 12.10. In this example a series of robust legs is combined to define a highly repeatable approach comprising arrival, downwind, crosswind, and approach legs. The repeatability of this philosophy has been amply demonstrated in a number of terminal approach examples, including Frankfurt in Germany, Schipol in the Netherlands, and San Francisco and Boston in the United States, to name a few. Not only can these approaches be flown with greater navigational accuracy, but the precise alignment of arrival and departure routes can be aligned greatly to improve noise abatement and the overall environmental impact upon communities in the airport locality.

Path steering error

The assumed values for FTE are given in Table 12.3. Studies on recent aircraft suggest that actual values achieved in modern aircraft using digital autopilots are in fact much better than the assumed values (see Table 12.4).

Position estimation error

As well as the data given in the tables above, reference (**2**) gives a comprehensive overview of the considerations to be addressed when considering RNP/RNAV, and of future likely developments. The specified increases in traffic density described so far only affect the lateral spacing of aircraft; vertical spacing is another consideration. The desire also to reduce the vertical separation between aircraft is addressed in another aspect of FANS – Reduced Vertical Separation Minima (RVSM).

Table 12.3 Assumed values for path steering FTE (5 per cent probability) (2)

Flight Phase	Manual (nautical miles)	Flight Director (nautical miles)	Autopilot (nautical miles)
Oceanic	2.0	0.5	0.25
En-route	1.0	0.5	0.25
Terminal	1.0	0.5	0.25
Approach	0.50	0.25	0.125

Table 12.4 Actual values for path steering FTE (Boeing study (2))

	Manual flight with map display (nautical miles)	LNAV with flight director coupled (nautical miles)	LNAV with autopilot coupled (nautical miles)
En route	0.502–0.918	0.111–0.232	0.055–0.109
Terminal	0.208–0.402	0.073–0.206	0.068–0.088

RVSM

One of the other ways of increasing traffic density is the introduction of the RVSM criteria. For many years, aircraft have operated with a 2000 ft vertical separation at flight levels between FL290 and FL410. As traffic density has increased, this has proved to be a disadvantage for the busiest sections of airspace. Examination of the basic accuracy of altimetry indicated that there were no inherent technical reasons why this separation should not be reduced. Accordingly, RVSM was introduced to increase the available number of flight levels in this band and effectively permit greater traffic density. The principle is to introduce additional usable flight levels such that the flight level separation is 1000 ft throughout the band, as shown in Fig. 12.11.

Originally, a trial was mounted in 1997 to test the viability of the concept on specific flight levels – FL340 and FL360 as shown in the Fig. 12.11. RVSM is now implemented throughout most of Europe from FL290 to FL410, introducing six new flight levels compared with before. All the specified flight levels will be implemented on the North Atlantic by early 2002. Other regions in the globe will have RVSM selectively implemented to increase air traffic density according to Fig. 12.12. and Table 12.5.

RVSM implementation

At the time of writing, the time-scale for the worldwide implementation of RVSM is shown in Fig. 12.12 and in the accompanying Table 12.5 (**3**). RVSM operation requires the aircraft to possess two independent means of measuring altitude and an autopilot with an accurate height hold capability. The operators of RVSM-equipped aircraft are not taken on trust: independent height monitoring stations survey aircraft passing overhead, measuring actual height compared with flight plan details and the individual aircraft fin number aircraft and route plan. RVSM implementation therefore embraces a watchdog function that ensures that all users are conforming to the RVSM accuracy and performance provisions.

Fig. 12.11 *RVSM – insertion of new flight levels*

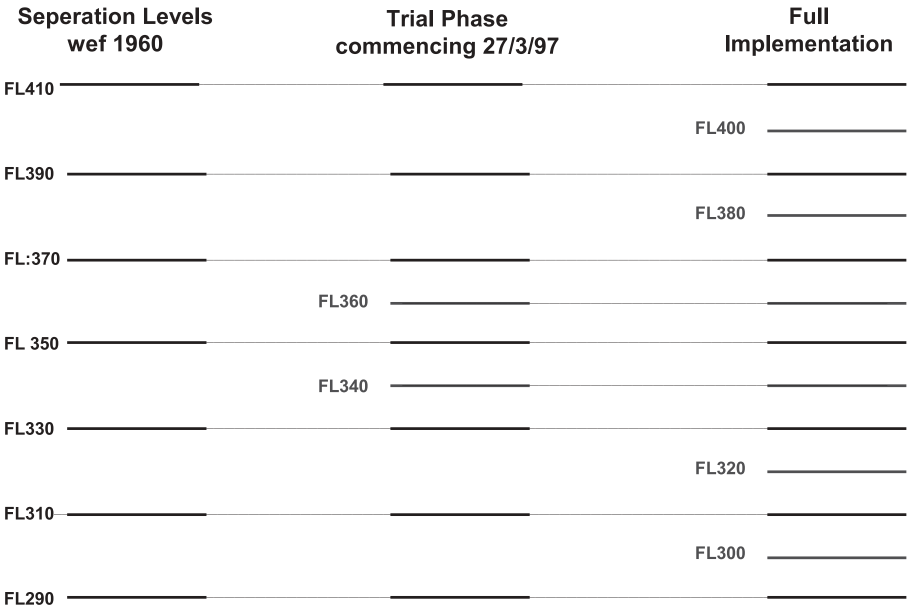


Fig. 12.12 *RVSM implementation – world wide*

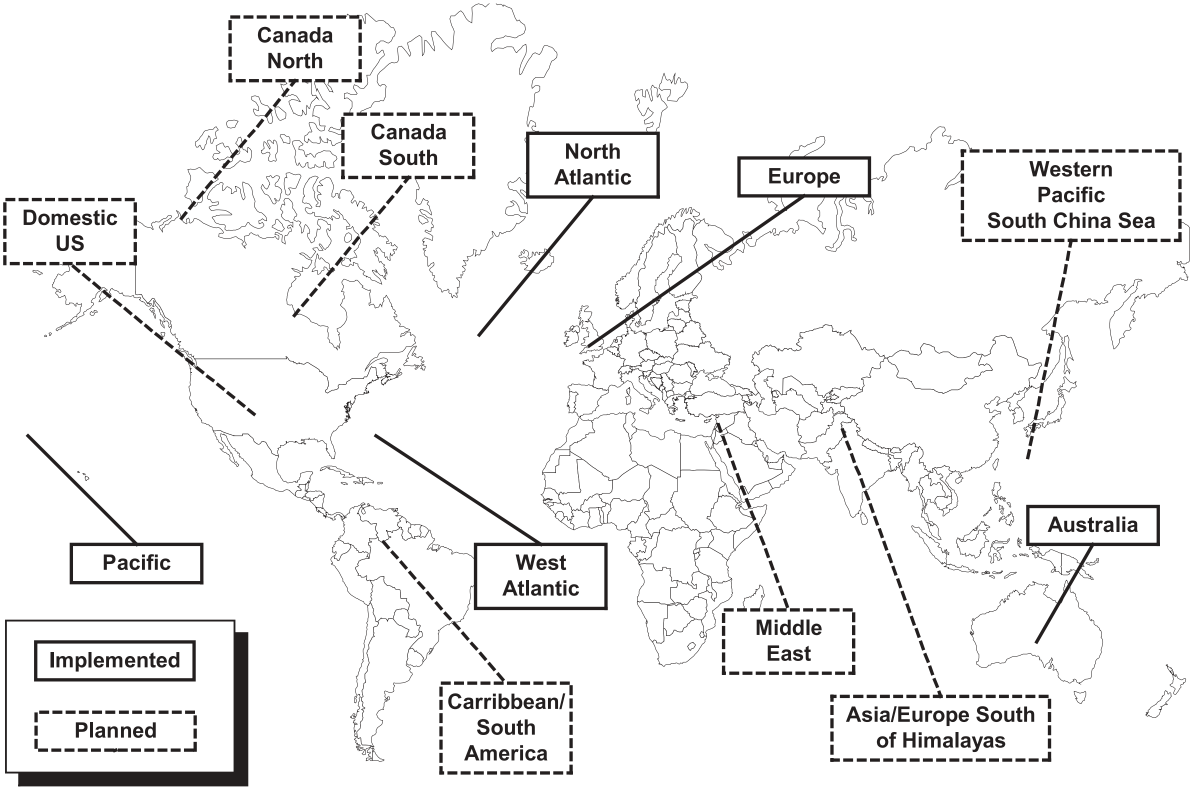


Table 12.5 RVSM – worldwide status

RVSM status - Americas and Europe		
North Atlantic	March 1997	FL330-370
	October 1998	FL310-390
	24 January 2002	FL290-410
West Atlantic route system (WATRS)	1 November 2001	FL310-390
	24 January 2002	FL290-410
Europe tactical (UK, Ireland, Germany, Austria)	April 2001	FL290-410
Europe-wide	24 January 2002	FL290-410
South Atlantic	24 January 2002	FL290-410
Canada North domestic	April 2002	FL290-410
Canada South domestic	Coordinate with US domestic	
Domestic US – phase 1 (1)	1 December 2004	FL350-390
Domestic US – phase 2 (1)	Late 2005-2006	FL 290-390 (FL410)
Note: (1) DRVSM plan to be finalised not later than January 2002 based on ATC simulation results and user inputs		
Caribbean/South America	RVSM Group Established	
RVSM status – Asia/Pacific		
Pacific	February 2000	FL290-390
	Tactical Use	FL400-410
Australia	November 2001	FL290-410
Western Pacific/South China Sea	21 February, 2002	Consult Publications
Middle East	November 2003	TBD
Asia-Europe/South of Himalayas	November 2003	TBD

The stated benefits of RVSM in the North Atlantic (NAT) RVSM area are:

- Improved operations/efficiency.
- Greater availability of more fuel-efficient altitudes.
- Greater availability of the most fuel-efficient tracks or routes.
- Increased probability that an operator will be cleared onto the desired track or altitude.
- Enhanced controller flexibility to manage traffic through an increased number of available altitudes.

Differential GPS enhancements

DGPS enhancements are being developed for en route and precision landings in the United States. The GPS enhancements WAAS and LAAS have already been described in Chapter 8 – Navigation, and their introduction should lead to the following accuracies being achieved as a matter of course:

- WAAS is anticipated to yield an accuracy of ~7 m, which will be sufficient for Cat I approaches.
- LAAS is expected to provide enhanced accuracies of ~1 m, which will be sufficient for precision approaches Cat II and Cat III.

The introduction of DGPS technology is also envisaged for Europe and the Far East. In Europe there are two programmes in the planning stage that will enhance satellite navigation. The European Space Agency (ESA), the European Commission (EC), and the European Organization for the Safety of Air Navigation (Eurocontrol) are working together on the development of a Global Positioning and Navigation Satellite System (GNSS) plan. The GNSS programme is being carried out in two phases:

- GNSS-1. This involves the development of the European Geostationary Navigation Overlay System (EGNOS) which will augment the existing US GPS and Russian GLONASS systems.
- GNSS-2. This involves the development of a second-generation satellite navigation system, including the deployment of Europe's own satellite system – Galileo. At the time of writing, the EU nations had agreed to secure funding for the deployment of the Galileo system

Within GNSS-l , Europe is contributing EGNOS which is aimed at augmenting the performance of GPS and GLONASS in terms of precision and data integrity. EGNOS is now becoming a reality as a test bed – a simplified version of the fully fledged system has recently been readied by Alcatel Space, the prime contractor leading the international industrial team that is developing the system.

The system is based on the use of ground infrastructure and three geostationary satellites. These are INMARSAT-3 Indian Ocean region (IOR) and Atlantic Ocean region east (AOR-E) and, soon, ESA's Artemis telecommunications satellite which is due to be launched into geo stationary orbit above Africa in 2002. These spacecraft are equipped with dedicated navigation transponders to augment the positioning services currently offered by the GPS and GLONASS constellations. The EGNOS ground infrastructure will be deployed over more than 40 sites, mostly in Europe. The ground infrastructure for the preoperational version has already been deployed at many sites around Europe: France, Iceland, Italy, the Netherlands, Norway, Spain, Turkey, and the United Kingdom, and at two sites outside Europe: Kourou (French Guyana) and Hartebeeshoek (South Africa).

The EGNOS system will be qualified at the end of 2003 and will provide in Europe an operational satellite navigation service that will later be improved with the operational introduction of the Galileo satellite system – scheduled for 2008. The European EGNOS, the US WAAS, and the Japanese Multifunction Satellite Augmentation System (MSAS) will be fully interoperable

Following the commissioning of the EGNOS test bed system, an EGNOS-like signal has since mid-February been transmitted from space, providing users with a GPS augmentation signal and enabling them to compute their positions to an accuracy of a few metres. The EGNOS test bed signal is currently available in the coverage area of the AOR-E satellite and as of May 2000 became available in the coverage area of the IOR spacecraft.

This pre-operational version of EGNOS will allow Europe to support demonstration of the operational benefits of GNSS to user communities. The test bed will be used for all modes of transport (air, land, and maritime) that require positioning services to accuracies of a few metres, and more particularly safety-critical services. For aviation users, for instance, EGNOS will provide for en route navigation as well as non-precision approach and precision approach phases of flight.

The European Commission is more specifically responsible for the promotion of applications and user equipment, an activity that relies largely on the use of the EGNOS system test bed. Financial support is also provided by EC for the EGNOS project, in particular for the lease of INMARSAT-3 transponders and implementation of the Artemis navigation payload.

In Japan, the MSAS is intended to provide a differential satellite-based system for a wide range of users including the aviation community. Unfortunately, the first satellite was lost during launch in 1999, and no further progress has been made.

Protected ILS

Within Europe, some ILS installations suffer interference from high-power FM local radio stations. Modifications have been mandated that introduce receiver changes to protect the ILS systems from this interference.

Introduction of the Microwave Landing System (MLS)

The MLS is an approach aid that was conceived to redress some of the shortcomings of ILS. The specification of a time-reference scanning beam MLS was developed through the late 1970s/early 1980s, and a transition to MLS was envisaged to begin in 1998. However, with the emergence of satellite systems such as GPS there was also a realization that both ILS and MLS could be rendered obsolete when such systems reach maturity. In the event, the US civil community is embarking upon higher-accuracy developments of the basic GPS system: the WAAS and LAAS implementations described earlier. In Europe, the introduction of the similar EGNOS system may also render the need for dedicated approach aids unnecessary. For the moment, the United Kingdom, the Netherlands, and Denmark have embarked upon a modest programme of MLS installations at major airports.

Polar routes

The increased range of modern aircraft, together with the improvement in navigation systems, has led in recent years to the exploitation of the north polar route. Certain airlines are providing direct flights from New York (JFK), Chigaco, and Los Angeles to Far Eastern destinations using direct polar routes that realize considerable fuel economies. There are areas where significant caution needs to be applied on these routes as some of the diversion airfields are not well equipped and can suffer from severe weather conditions. The performance of certain parts of the aircraft avionics equipment can suffer at high latitudes, and communications coverage in particular can be affected. Nevertheless, a great deal of attention is being given to utilizing the modern advances to exploit these new routes [see reference (**4**)]. A supplementary publication outlines some of the navigation difficulties experienced at extreme latitudes, especially when it is anticipated that the aircraft will transit at or near the north pole, creating specific difficulties for the navigation system (e.g. an abrupt singularity where north turns to south while flying over the pole).

Surveillance

Surveillance enhancements include the following:

- TCAS II.
- ATC mode S.
- Automatic Dependent Surveillance A (ADS-A).
- Automatic Dependent Surveillance B (ADS-B).

The operation of TCAS and ATC mode S has been described in Chapter 6, but their use in a FANS context will be examined here.

TCAS

The operation of TCAS has already been described in Chapter 6. When operating together with a mode S transponder and a stand-alone display or EFIS presentation, TCAS is able to monitor other aircraft in the vicinity by means of airborne interrogation and assessment of collision risk. TCAS II provides vertical avoidance manoeuvre advice by the use of RAs. TCAS II will soon be made mandatory for civil airliners – aircraft with a weight exceeding 15 000 kg or 30 or more seats – operating in Europe. This will be extended to aircraft exceeding 5700 kg or more than 10 seats, probably by 2005.

ATC mode S

The use of the ATC mode S transponder in providing digital air–ground and air–air data links (VDL mode 4) has already been described in Chapter 6. This provides a basic ADS-B capability.

Mode S also has the capability of providing a range of data formats – from level 1 to level 4. These are categorized as follows:

- Level 1. This is defined as the minimum capability mode S transponder. It has the capability of reply to mode S interrogations but has no data link capability. All the messages provided by level 1 are short (56 bit) messages.
- Level 2. These transponders support all the features of the level 1 transponder with the addition of standard-length, data-link word formats. This can entail the use of longer messages (112 bit). Some of the messages are used for TCAS air–air communication while others are utilized for air–ground and ground–air communication as part of the enhanced surveillance Data Access Protocol System (DAPS) requirements.
- Level 3. The level 3 transponders embrace the same functionality as level 2 with the additional ability to receive Extended-Length Messages (ELMs) which comprise 16 segments of information, each containing a 112 bit message
- Level 4. Level 4 has the full functionality of level 3, with the capability of transmitting ELM messages of up to 16 segments of 112 bit word messages.

Originally it was envisaged that ATC mode S would be the primary contender to provide the CNS/ATM functionality by providing large block transfers of information. More recently it has been realized that VDL mode 4 might better serve this need, and levels 3 and 4 are no longer required.

Automatic Dependent Surveillance – address mode (ADS-A)

ADS-A will be used to transmit aircraft four-dimensional position and flight plan intent based upon GPS position during oceanic crossings. The communications media will be SATCOM or HF Data Link (HFDL). ADS-A requires the aircraft to be fitted with an FMS and CDU and with some means of displaying message alerts and annunciation.

Automatic Dependent Surveillance – broadcast mode (ADS-B)

ADS-B will be used to transmit four-dimensional position and flight plan intent based upon GPS position using line-of-sight VHF communications. Either mode S or digital VHF radio will be used to transmit the data. ABS-B requires a cockpit display of traffic information.

An extended demonstration project in the Louisville area conducted by the FAA in conjunction with United Parcels Services (UPS) and others has helped to demonstrate the ADS-B concept and has paved the way for UPS aviation technologies to secure certification for new displays that can provide improved runway, aircraft, and weather information displays facilitating the movement of aircraft in the terminal area. UPS have predicted that this technology will yield a 20 per cent capacity increase at Louisville, as reducing the arrival spacing of aircraft by 20–30 s can reduce the nightly sorting operation by around 30 min. Time savings of this proportion are critical in maintaining delivery of parcels on time.

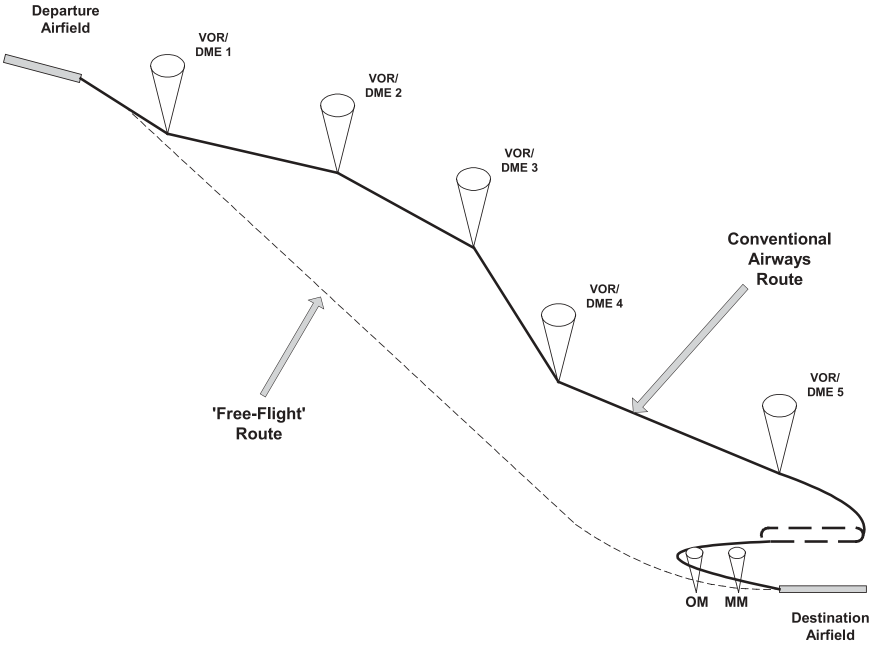
Direct routing

The CNS triad offers a suite of methods for addressing the FANS objectives. The ultimate aim of FANS is to provide a 'free flight' capability as shown in Fig. 12.13.

Under prevailing navigational means, an aircraft flying from a departure airfield to a destination airfield would need to file a flight plan that entails flying down prescribed airways flying from VOR/DME beacon to beacon. Upon arrival in the vicinity of the destination airfield, the aircraft may have to enter a holding pattern while waiting for clearance to land. The approach may necessitate an ILS approach that would involve overflight of the outer marker some 5–7 nautical miles on the extended runway centreline. Many inefficiencies exist in this type of routing which is typical of many flights carried out over land today.

Free flight on the other hand would enable a more direct routing from the departure to the destination airfield, clearly yielding savings in time and fuel, as indeed RNAV allows today. Free flight is rather more than RNAV: it will encompass a wider range of ADS capabilities such that data links may be used to modify ATC flight plans and clearances in a dynamic fashion. Furthermore, upon arrival at the destination airfield, the aircraft could execute a more direct and efficient approach and landing; most probably enabled by phasing its approach with that of other aircraft by means of four-dimensional, required time of arrival navigation techniques. An FMS embodying a full-performance aircraft model could make further savings by optimizing the use of fuel during flight such that the aircraft could meet the necessary four-dimensional waypoints at the appropriate time with a minimum fuel burn.

Fig. 12.13 *Principle of 'free flight'*



The universal application of the principles of free flight is some way off. Free flight may already be used on specific flight segments where air traffic density is relatively low, but the universal application depends upon the proven maturity of the technologies described. There are complex political and financial factors that compound the technical issues. Free flight in Europe will probably not be fully available until the partitioning of the airspace is simplified and rationalized compared with the complex structure that exists today. The cost of new equipment, both air and ground based, is not trivial. Airlines will wish to be assured that the investment in new equipment will provide an adequate return as new FANS capabilities are introduced. Nevertheless, in the medium and long term, the pressures of air traffic density and fuel economy will ensure that many FANS features will attain maturity.

Need for flight management system (FMS)

The need to be able to manage all of these techniques in a timely manner is crucial to the success of FANS. The need for an FMS with the capabilities outlined in Chapter 8 will continue to be a key factor in integrating these features in a robust manner. The ability to provide software upgrades to the FMS will enable present route structures and communications to be improved while new ones are added. It is vital, therefore, that the FMS contains sufficient spare memory and computation capacity to allow for future growth in these areas.

Boeing FANS 1 Implementation

The Boeing approach to providing a FANS capability has been to provide software upgrades for older models that impart additional navigation and communications functionality to existing avionics hardware. Newer models such as the Boeing 777 and Boeing 767-400 will have these functions embedded. This approach may be summarized by reference to Fig. 12.14 which shows a typical Boeing display/FMS interface.

The major features are:

- Conventional navigation information displayed on the captain and first officer's Navigation Displays (NDs).
- On the Boeing 747-400, when the FANS 1 package is loaded, the captain's CDU acts as a host to display additional ATC communications functions as well as the existing navigation functions.
- On the Boeing 737 aircraft with the appropriate software upgrade loaded, additional guidance information is shown on the Primary Flight Display (PFD), as described in reference (**9**). The PFD is enhanced by the display of lateral and vertical deviation scales along the bottom and right-hand edge of the PFD. By flying the aircraft to maintain zero lateral and vertical deviation (the same as for an ILS approach), deviations from the desired flight path are minimized. This enhancement enables the same flight technical error deviation to be achieved in Flight Director (FD) as in autopilot mode.

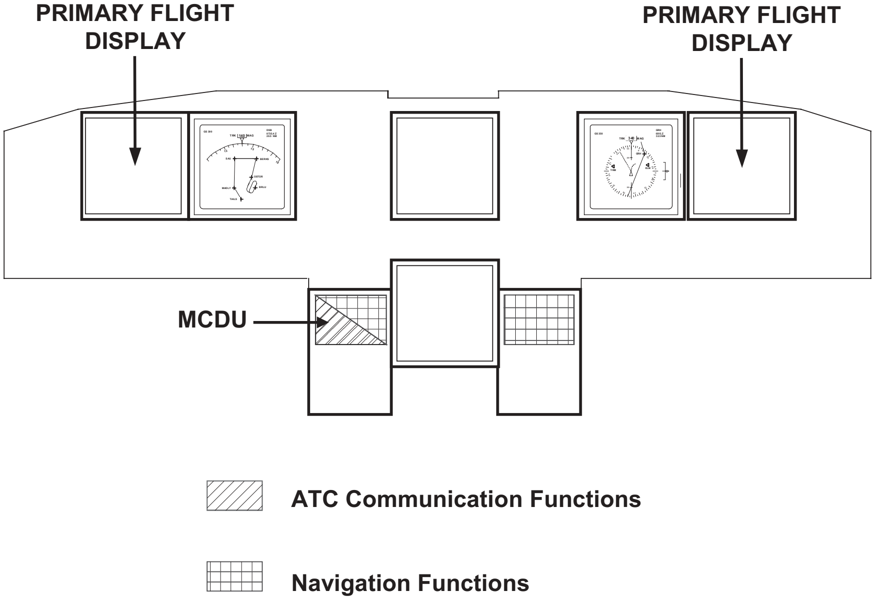


Fig. 12.14 *Boeing FANS 1 – display of information*

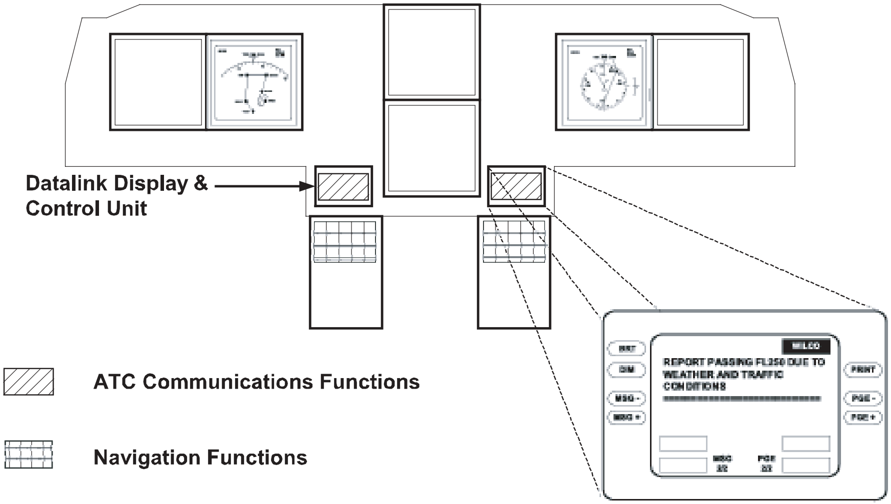
The stated performance of the Boeing aircraft in RNP navigation modes is quoted in references (**5**–**9**).

Airbus FANS-A Implementation

Airbus has taken a slightly different approach to implementing its FANS upgrade by adding additional equipment rather than by adding software functionality to existing hardware. The units added include:

- An Air Traffic Services Unit (ATSU) incorporating an Airborne Communications Addressing and Reporting System (ACARS) and airline operation communication (AOC) functions.
- Replacement of existing radios with VHF Digital Radios (VDRs).
- Introduction of Dual Data-link Control and Display Units (DCDUs) located on the centre console (see Fig. 12.15).
- An upgraded FMS module hosted in the Flight Management and Guidance Envelope Computer (FMGEC).

Initially, this was fitted to the A330/340 family with a view to moving towards a FANS-B version implementing Aeronautical Telecommunications Network (ATN) functionality.



REPORT PASSING FL250 DUE TO WEATHER AND TRAFFIC CONDITIONS

Fig. 12.15 *Airbus FANS A – display of information*

High-precision approaches

The judicious application of the FANS techniques described in this chapter have allowed major advances to be made and approaches to be designed that would not previously have been possible. One such example is the new approach designed for RW26 at Juneau, Alaska. The airport at Juneau is surrounded by high mountains, and the approach to RW26 requires flying north-west up a glacial valley embracing the Gastineau Channel. Figure 12.16 presents a view of Juneau airport and the Gastineau channel viewed from the north-west, showing the rugged nature of the terrain.

Figure 12.17 depicts the horizontal profile of the approach to RW26. Before the design and implementation of the RNP approach, this access to Juneau was impossible in all but the best weather conditions. The procedures have been designed to accommodate RNP_0.15, RNP_0.2, and RNP_0.3 approaches which allow Decision Heights (DHs) of 337, 437, and 1810 ft respectively. The approaches were designed by the Alaska Airlines in conjunction with Boeing and the FMS manufacturer, Smiths Aerospace. A full description of the considerations involved in developing the approach is given in reference (**10**).

Fig. 12.16 *Juneau, Alaska*

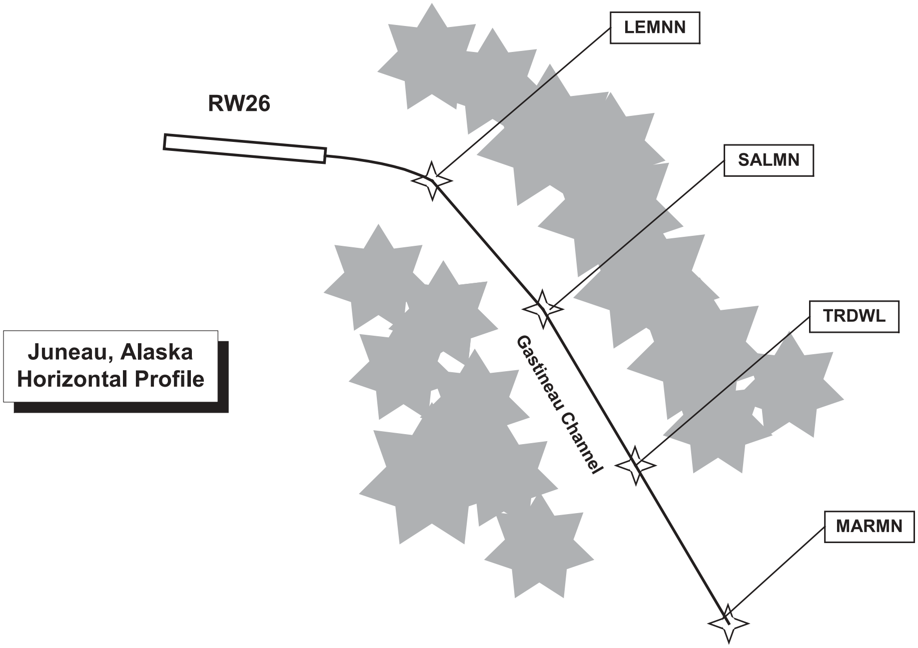


Fig. 12.17 *Juneau, Alaska – horizontal profile*

Using the new approach allowed shorter arrival and departure procedures when flying to Juneau from Seattle – about 5 min inbound and 6 min outbound. In terms of annual savings it has been estimated that about 245 flight hours, or 740 000 lb of fuel, could be saved assuming present flight schedules.

The use of RNP procedures also allows a more stable approach path (see Fig. 12.18). Usually, a conventional approach would involve a series of short descents followed by level flight as the aircraft ‘steps down’ the approach path. Using improved guidance, the aircraft is able to follow a gradual approach path starting at the outer waypoint (MARMN), some 15 miles out, and continue this descent down to the DH. This reduces crew workload and allows the flight crew to spend more time monitoring the status of the approach rather than continually changing power and trim settings. This is a further benefit of an RNP-based approach to add to the other benefits.

The aircraft used to fly this approach is a Boeing 737. The minimum navigation equipment fit required to execute this approach is listed below:

- Dual FMS.
- Dual IRS.
- Dual GPS.
- Dual EFIS.

Experience showed that the weather radar provided terrain information that was most useful to monitor the status of the approach, and a serviceable weather radar was added to the minimum equipment fit.

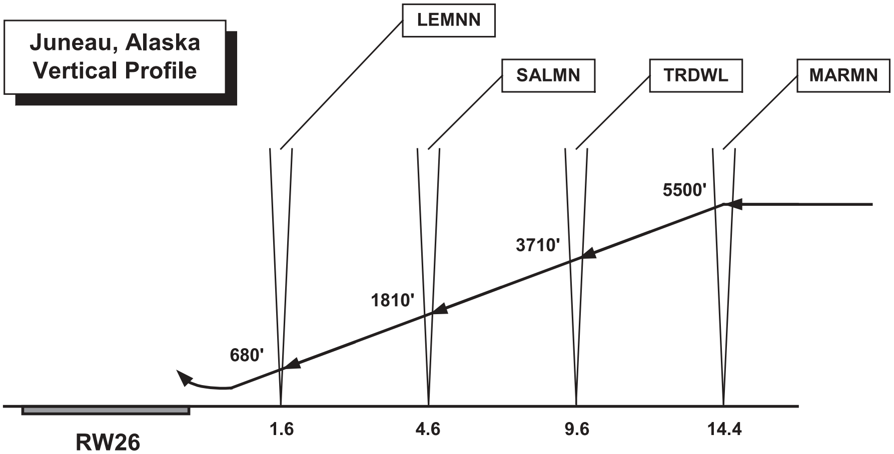


Fig. 12.18 *Juneau, Alaska – vertical profile*

References

(1) Advisory Circular AC 00-31A (1982) National Aviation Standard for the Very High Frequency Omnidirectional Radio Range(VOR(/Distance Measuring Equipment (DME)/Tactical Air Navigation Systems.

(2) **Nakamura, D.** (2000) Boeing FMS RNAV Workshop – General Information on the Functional and Technical Aspects of Required Navigation Performance (RNP) Area Navigation (RNAV) and Applications

(3) Federal Aviation Authority (FAA), RVSM Website: www.faa.gov.ats/ato/rvsm1.htm.

(4) Polar routes (2001) Boeing Aero 16.

(5) RNP Capability of FMS Equipped B737 Generation 2,-737-300/400/500, 737-600/700/800, (1997), D6-39067-2.

(6) RNP Capability of FANS 1 FMCS Equipped 747-400, Generation 3, -747-400, Rev B, (2000), D926U0280.

(7) RNP Capability of FANS 1 FMCS Equipped 757/767, Generation 1, -757/767 Rev B, (2000), D926T0120.

(8) RNP Capability of FANS 1 Equipped 777, -777 Rev B, (1999) D234W018-7.

(9) Lateral and vertical navigation (2001) Boeing Aero 16.

(10) **Kramer, M.** (1998) RNP RNAV Terminal areas – procedures design and approval for Juneau, Alaska, *FITEC '98.*

CHAPTER 12

Future Air Navigation System

The rapidly increasing air traffic density is leading to a pressing need to improve the Air Transport Management (ATM) system by all available means and move on from the techniques and technologies that have served the industry for the last 40 years. This evolution will embrace the use of new technologies mixed with existing capabilities to offer improved air traffic management.

The aims of ATM may be summarized as follows:

- To maintain or increase levels of safety.
- To allow dynamic accommodation of user-preferred three- and four-dimensional flight trajectories.
- To improve the provision of information to users in terms of weather, traffic situation, and services.
- To increase user involvement in ATM decision-making, including air–ground computer dialogues.
- To organize airspace in accordance with ATM procedures.
- To increase system capacity to meet traffic demand.
- To accommodate a full range of aircraft types and capabilities.
- To improve navigation and landing capabilities to support advanced approach and departure procedures.
- To create, to the maximum extent possible, a seamless continuum of airspace where boundaries are transparent to the user.

To this end, the air traffic control authorities, airline industry, regulatory authorities, and airframe and equipment manufacturers are working to create the Future Air Navigation System (FANS) to develop the necessary equipment and procedures.

The areas where improvements may be made relate to Communications, Navigation, and Surveillance, commonly referred to as CNS. The key attributes of these improvements may be briefly summarized as:

1. Communication. The use of data links to increase data flow and permit the delivery of complex air traffic control clearances.
2. Navigation. The use of GPS in conjunction with other navigational means to improve accuracy and allow closer spacing of aircraft.
3. Surveillance. The use of datalinks to signal aircraft position and intent to the ground and other users.

These headings form a useful framework to examine the CNS/ATM improvements already made and those planned for the future.

Communications

The main elements of improvement in communications are:

- Air–ground VHF data link for domestic communications.
- Air–ground SATCOM communications for oceanic communications.
- High-Frequency Data Link (HFDL).
- 8.33 kHz VHF voice communications.

Air–ground VHF data link

The emergence of data links as means of communications versus conventional voice communications is shown in Fig. 12.1. Voice links have been used in the past for communications between the air traffic control system or ATM and the Airline Operational Centre (AOC). The use of data links, controlled and monitored by the FMS or other suitable method on-board the aircraft, may communicate with the AOC and ATM system via data links. These data links may be implemented using one or more of the following:

- VHF communications.
- Mode S transponder.
- Satellite links.

Data link communications are being designed to provide more efficient communications for ATC and Flight Information Services (FIS). Although these systems essentially replace voice communication, there will be a provision for voice back-up in the medium term.

Flight plan data, including aircraft position and intent in the form of future waypoints, arrival times, selected procedures, aircraft trajectory, destination airport, and alternatives, will all be transferred to the ground systems for air traffic management. The data sent to the ground ATM system will aid the process of predicting a positional vector for each aircraft at a specific time. This information will aid the task of the ground controllers for validating or reclearance of an aircraft's flight plan. Furthermore, use of the Required Time of Arrival (RTA) feature will enable the air traffic controllers to reschedule aircraft profiles in order that conflicts do not arise.

For ATC flight service and surveillance, VHF Data Link (VDL) communications will be increasingly used for domestic communications. VHF communications are line-of-sight limited, as has already been explained. A number of options exist:

- VDL mode 1. Compatible with existing ACARS transmitting at 2.4 kb/s. This mode suffers from the disadvantage that it is character oriented.

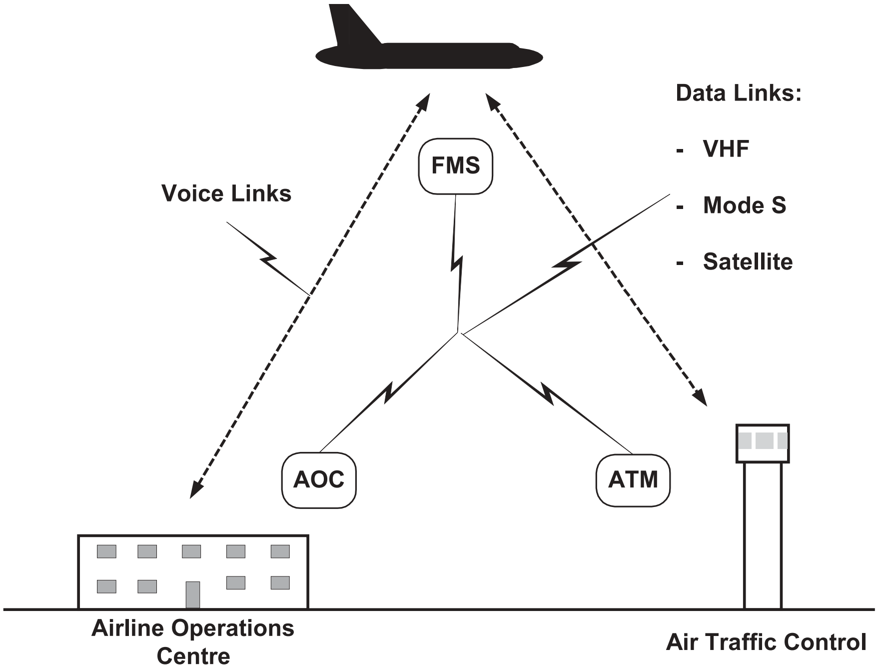


Fig. 12.1 *The emergence of data links*

- VDL mode 2. Data transmitted at 31.5 kb/s. As well as having a higher bandwidth, this protocol is bit rather than character oriented, making it 50–70 per cent more efficient than the ACARS protocol. VDL mode 2 is able to support Controller-to-Pilot Data-Link Communications (CPDLC).
- VDL mode 3. Simultaneous data and analogue voice communications using Time Division Multiple-Access (TDMA) techniques.
- VDL mode 4. Used with the 1090 MHz signal of ATC mode S.

It is expected that the introduction of data-link technology will benefit all users owing to the more efficient and less ambiguous nature of the messages passed. Significant improvements in dispatch delays and fuel savings are expected as these technologies reach maturity.

Air–ground SATCOM communications

SATCOM is a well-proven data link that, as has already been explained, is limited at very high latitudes in excess of about 82º. The SATCOM system is supported by the INMARSAT constellation already described in Chapter 6.

HF data link

Modern technology enables HF data-link transmissions to be more robust than HF voice and therefore less susceptible to the effects of the sunspot cycle. HF data link provides primary coverage out to 2700 nautical miles and secondary coverage beyond that should propagation conditions be favourable. There is extensive cover by ground stations located in the northern hemisphere such that HF data link is a viable alternative to SATCOM for north polar transitions (see Chapter 6).

8.33 kHz VHF voice communications

Conventional VHF voice channels are spaced at intervals of 25 kHz throughout the spectrum. A denser communications environment has resulted in the introduction of digital radios that permit spacing at 8.33 kHz, allowing three channels to be fitted in the spectrum where only one could be used previously. With effect from the 7 October 1999, these radios have already been mandated in Europe for operation above 20 000 ft and mandatory use will follow in the United States within a number of years; one of the difficulties in predicting the time-scale is the vast number of radios that have to be replaced/retrofitted.

Navigation

A number of navigational improvements are envisaged:

- Introduction of Required Navigation Performance (RNP) and Actual Navigation Performance (ANP) criteria. This defines absolute navigational performance requirements for various flight conditions and compares these with the actual performance the aircraft system is capable of providing.
- Reduced Vertical Separation Minima (RVSM).
- Differential GPS (DGPS) enhancements:
  - WAAS,
  - LAAS.
- Protected Instrument Landing System (ILS).
- Introduction of Microwave Landing System (MLS)
- Polar routes.

Classical method for defining navigation performance

Area navigation (RNAV)

RNAV navigation systems allow the aircraft to operate within any desired course within the coverage of station-referenced signals (VOR, DME) or within the limits of a self-contained system capability (INS, GPS) or a combination of these. RNAV systems have a horizontal two-dimensional capability using one or more of the on-board navigational sensors to determine a flight path determined by navigation aids or waypoints referenced to latitude and longitude. In addition, the RNAV system provides guidance cues or tracking of the flight path. Many modern RNAV systems include a three-dimensional capability to define a vertical flight path based upon altimetry, and some include a full aircraft and engine performance model.

The benefits of RNAV may be shown by a simple example. A typical route length from Boston to Miami using conventional airway routing would typically be around 1198 nautical miles. The same route flown using RNAV could be in the region of 1113 nautical miles, a saving in distance of 85 nautical miles or about 7 per cent with a corresponding saving in time and fuel.

The performance of pre-RNAV systems has historically been defined according to the criteria in Fig. 12.2:

- Along-track error.
- Across-track error.
- Flight Technical Error (FTE)

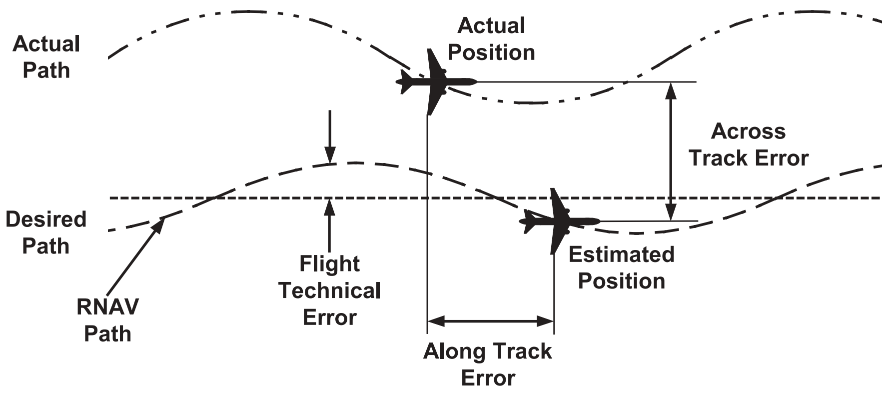


Fig. 12.2 *Historic RNAV error composition*

The total navigation error is the Root Sum Squared (RSS) of these elements for a given navigation means or phase of flight.

The availability of the navigation capability is defined at 99.999 per cent and the integrity requirement for misleading navigation information is set at 99.9999 per cent. Navigation accuracies and FTE are typically demonstrated within the values given in Table 12.1 for differing phases of flight

Table 12.1 Demonstrated accuracy and FTE per phase of flight

Airspace/Operation	Accuracy (95%) (nautical miles)	FTE (nautical miles)
Oceanic/en route remote	± 3.8	± 1.0
Domestic en route	± 2.8	± 1.0
Terminal	± 1.7	± 0.5
Approach – VOR/DME	± 0.5	± 0.125
Approach – multisensor	± 0.3	± 0.125

Vertical navigation (VNAV)

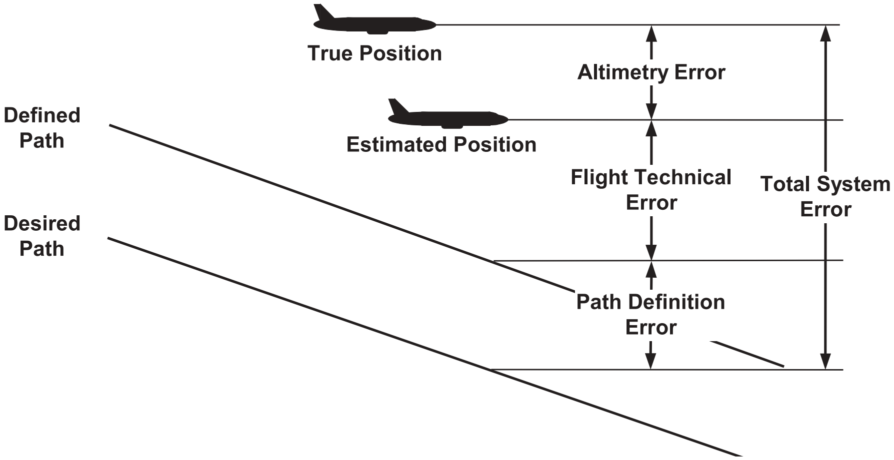
A VNAV capability further enhances flight operations by specifying a vertical flight path to be flown coincident with the lateral navigation guidance, providing a three-dimensional path for the aircraft to follow. As for the lateral guidance case, path deviation and guidance for tracking the path are provided. Key elements in VNAV are:

- Computed vertical flight paths as the basis for aircraft guidance.
- Assured repeatability of performance.
- Tailoring the flight path to aircraft/engine performance – full-performance VNAV.
- Providing situational awareness cues to the flight crew.

VNAV capabilities have been defined by the error components defined in Fig. 12.3. and listed below:

- Flight path definition error.
- Altimetry error (99.7 per cent).
- FTE.

Fig. 12.3 *Historic VNAV error composition*



The total demonstrated VNAV system error is the RSS of these values, depending upon the altitude and flight trajectory conditions given in Table 12.2.

Table 12.2 Demonstrated VNAV performance and FTE by altitude and flight condition

Airspace/Operation	Level 99.7%	Level/intercept FTE	Climb/descent 99.7%	Climb/descent FTE
At or below 5000 ft	±50 ft	±150 ft	±100 ft	±200 ft
5 000 to 10 000 ft	±50 ft	±240 ft	±150 ft	±300 ft
Above 10 000 ft	±50 ft	±240 ft	±220 ft	±300 ft

Note: This performance is consistent with the 1000 ft RVSM criteria for FL290 and above described later.

Each type of navigation aid has its own error characteristic: those for VOR/DME and DME/DME will be different as indicated below.

VOR/DME error

VOR/DME characteristics will vary according to:

- Distance from station, for a specified angular error the lateral error will vary with distance.
- Altitude will affect coverage and slant range.
- Station to station distance will affect the error; the further the stations are apart, the greater the error at the mid-point.
- Course error.

The maximum angular error defined by reference (**1**) is ±4.5° (see Fig. 12.4).

DME/DME error

DME/DME error characteristics will vary according to:

- Station site.
- Slant range.
- Number of stations – choice of different stations upon which to take a fix.
- Relative geometry between stations, as shown in Fig. 12.5.

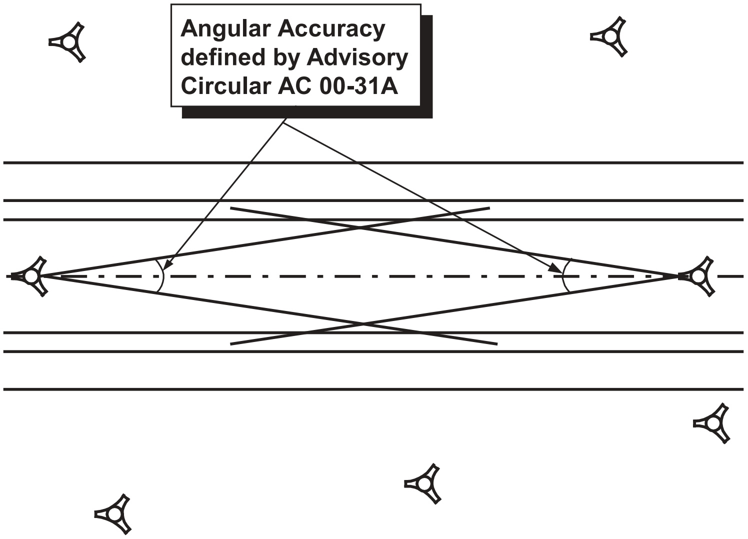


Fig. 12.4 *VOR/DME error characteristic*

The maximum range error defined by reference (**1**) is ±0.5 nautical miles or 3 per cent, whichever is the greater.

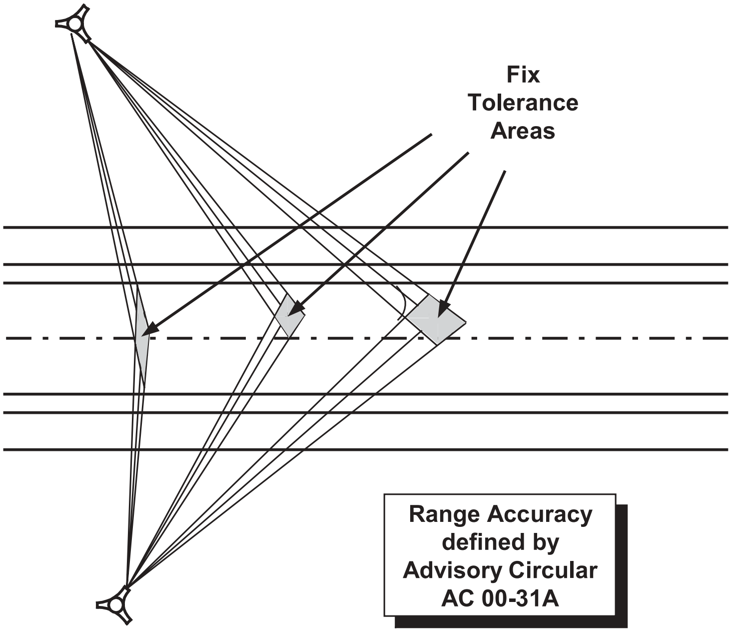


Fig. 12.5 *DME fix tolerance areas*

The overall effect of these and other factors leads to inefficient airspace design and procedures in the form of large airspace buffers and non-optimum separation minima for aircraft. The effect of different navigation infrastructures with different performance has a major impact on route widths, airspace use, and buffers. Fix tolerance areas are determined on the basis of the uncertainty resulting from bearing and range variations. This is compounded by the fact that historical airspace design and usage accounts for the lowest common denominator – the aircraft with the lowest performance navigation system. Although aircraft with more capable navigation systems may be able to navigate more accurately, they are unable to capitalize upon this fact as the airspace has always to allow for the worst case. As air traffic densities continued to increase, it became apparent that this situation could not continue.

RNP RNAV

Actual Navigation Performance

The Actual Navigation Performance (ANP) of the aircraft navigation system is represented by a circle defining the accuracy of the aircraft navigation system for 95 per cent of the time. The value of ANP is derived by taking the value of all of the navigation sensors and statistically weighing them against the other sensors. After a period of time, a degree of confidence is established as to which are the most accurate sensors, and therefore the ANP value is established. The 95 per cent probability circle is compared with RNP to decide whether the navigation system performance is good enough for the route segment being flown. The ANP and RNP values are displayed on the FMS CDU such that the flight crew can readily check on the navigation system status. Should the ANP exceed the RNP value for a given route sector for any reason, e.g. because of failure of a critical navigation sensor, the crew are alerted to the fact that the system is not maintaining the accuracy necessary. This will result in the aircraft reverting to some lower-capability navigational means, and in an approach guidance mode it may necessitate the crew executing a go-around and reinitiating the approach using a less accurate guidance means.

Required Navigation Performance

The RNP defines the lateral track limits within which the ANP circle should be constrained for various phases of flight. The general requirements are:

- For oceanic crossings the RNP is ±12 nautical miles, also referred to as RNP-12.
- For en route navigation the RNP is ±2 nautical miles (RNP-2).
- For terminal operations the RNP is ±1 nautical miles (RNP-1).
- For approach operations the RNP is ±0.3 (RNP-0.3).

Other specific RNP requirements may apply in certain geographical areas, e.g. RNP-4 and RNP-10 (see Fig. 12.6).

It is clear that this represents a more definitive way of specifying aircraft navigational performance, versus the type of leg being flown, than has previously been the case. Other more specific criteria exist: RNP-5 (also known as BRNAV or basic area navigation) has already been introduced in parts of the European airspace with the prospect that RNP-1 (also known as PRNAV or precision navigation) will be introduced in a few years. There are precision approaches in being – notably those in Juneau, Alaska – where RNP-0.15 is required for new precision approaches developed for mountainous terrain. The characteristics of this approach will be described later.

The effect of using RNP to specify the route structure versus the historical approach is shown graphically in Fig. 12.7. The pre-RNP approach is shown on the left of the diagram. The route width ranges from 60 up to 100 nautical miles. This enormous buffer is required to mitigate against the combined effect of navigation errors, navigation performance, route, traffic density, surveillance, communications, and ATC.

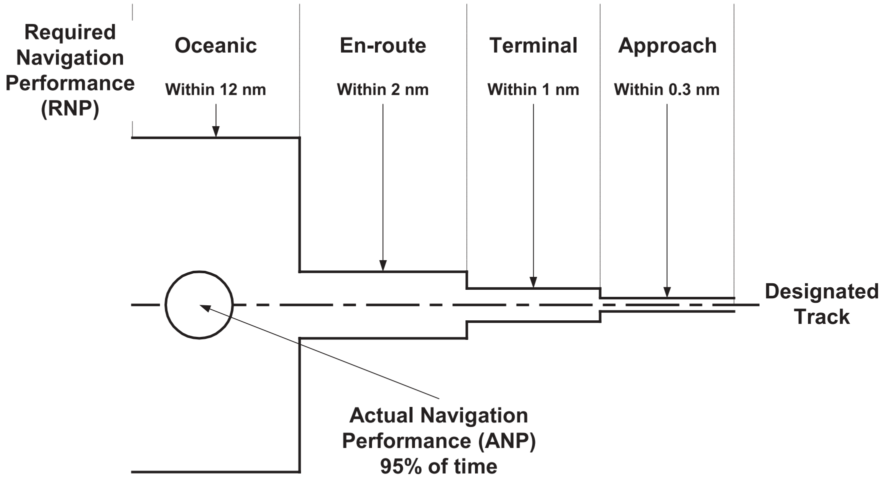


Fig. 12.6 *ANP versus RNP requirements*

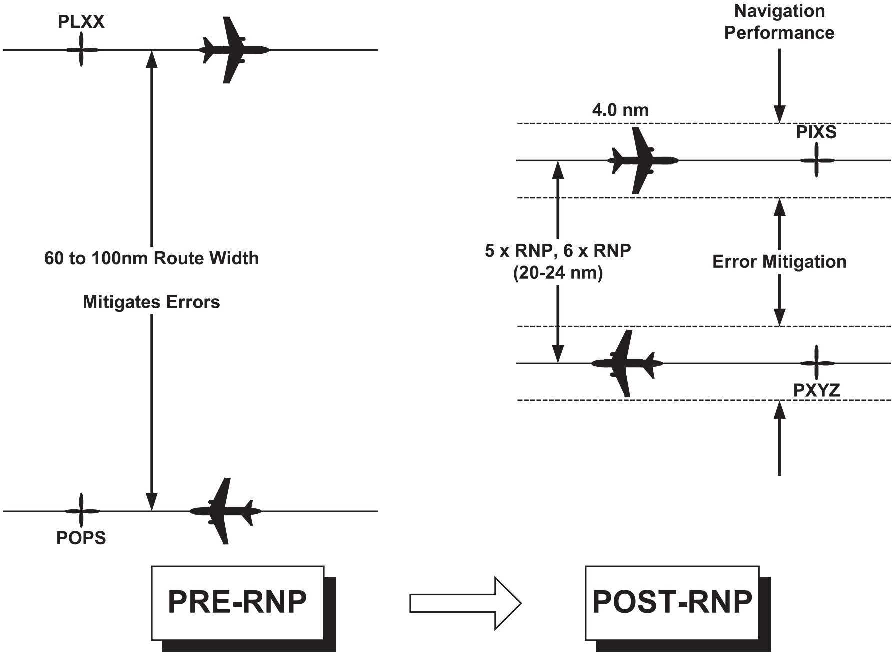


Fig. 12.7 *Comparison of historical and RNP definitions of route structure*

This approach considers the worst combination of factors that may apply. The RNP approach, together with other FANS improvements in surveillance and communications, leads to the situation shown on the right.

The post-RNP philosophy defines the route separation and error mitigation buffer in terms of multiples of the RNP performance. For the 4.0 nautical miles route (RNP-4) example shown, the route width is ~20–24 nautical miles depending upon the RNP multiple used. This compares most favourably with the pre-RNP route structure.

Similar spacing considerations apply along the route. As the navigational performance also affects along-track accuracy, similar trackwise improvements in separation may be achieved.

RNAV standards within Europe

Two RNAV standards are being developed in Europe, these are:

- Basic RNAV (BRNAV). BRNAV was introduced in 1988 and is equivalent to RNP-5 for RNAV operations. Navigation may be accomplished by using the following means:
  - DME/DME,
  - VOR/DME with a 62 nautical miles VOR range limit,
  - INS with radio updating or limited to 2 h since last on-ground position update,
  - LORAN C with limitations,
  - GPS with limitations.

  Until 2005, primary sources of navigation will be DME/DME, VOR/DME, and GPS. Reference (**3**), Advisory Circular AC 90-96, Approval of US Operators and Aircraft to Operate under Instrument Flight Rules (IFR) in European Airspace designated for Basic Area Navigation (BRNAV), 20 March 1998, approves the operation of US aircraft in European airspace under the application of existing advisory circulars.
- Precision RNAV (PRNAV). PRNAV is intended to be introduced at some time in the future but not before 2005. PRNAV will invoke the use of navigation under RNP-1 accuracy requirements or better.

The effect of using RNP techniques allied to the navigational capabilities of a modern aircraft needs to be examined in the context of the key areas of error defined above, namely:

- Path definition error – the difference between the desired flight path and actual flight path.
- Path steering error – the ability of the pilot and/or autopilot system to conform to the defined flight path.
- Position estimation error – the ability of the navigation system to estimate position and the level of integrity that may be assured in that process.

Path definition error/RNP fixes

The process of introducing higher-precision navigation criteria has led to a critical review of the nature of fixes presently allowed and their viability within the RNP framework. Certain types of fix are rigidly defined and relate to the position of a beacon or as defined by inertial coordinates – today usually GPS derived. Examples are given of these fixes (illustrated in Fig. 12.8). These show that they are absolute in their characterization:

- Initial Fix (IF)
- Track to a Fix (TF)
- Constant Radius to a Fix (RF)
- Holding to Fix (HX)

Each of these fixes is unambiguously defined to a particular point in the route structure; there are no conditional constraints that apply. Fixes with this type of characterization are ideal for specifying a robust route structure.

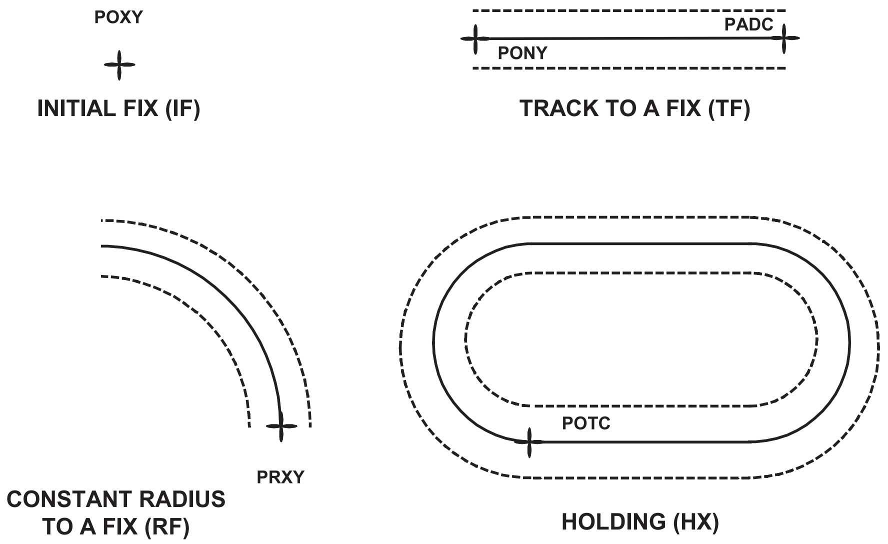


Fig. 12.8 *Preferred fixes for RNP*

By contrast, fixes that are conditional or depend to some degree upon a time conditional or dynamic constraint are less favoured in the proposed route structure. The examples shown in Fig. 12.9 all have conditional constraints. The Direct to a Fix (DF) example has a variable starting point that will affect the aircraft dynamics when arriving at the (DF) fix point. Similarly, Fix to an altitude (FA) will depend upon an individual aircraft performance or ability to climb to an assigned altitude, and other key navigation parameters may be suppressed. In essence, although the start point is defined, the end point is indefinite. Finally, the example given for Course to a Fix (CF) only relates to the course that the aircraft is flying when the designated fix point is reached. The future course of the aircraft beyond this point will depend upon errors accumulated in the previous leg on account of accuracy of the previous fix, wind variation, and other sources. The course to be followed after this fix will not necessarily agree with the aircraft's defined route/flight plan.

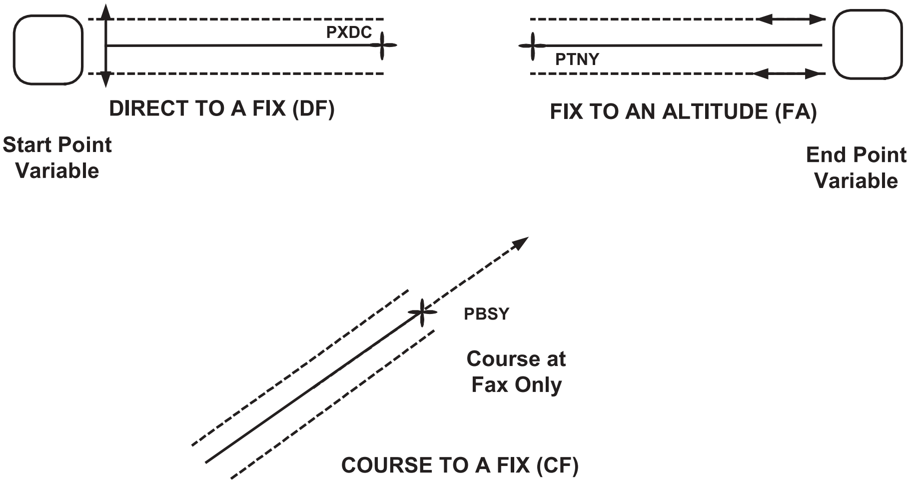
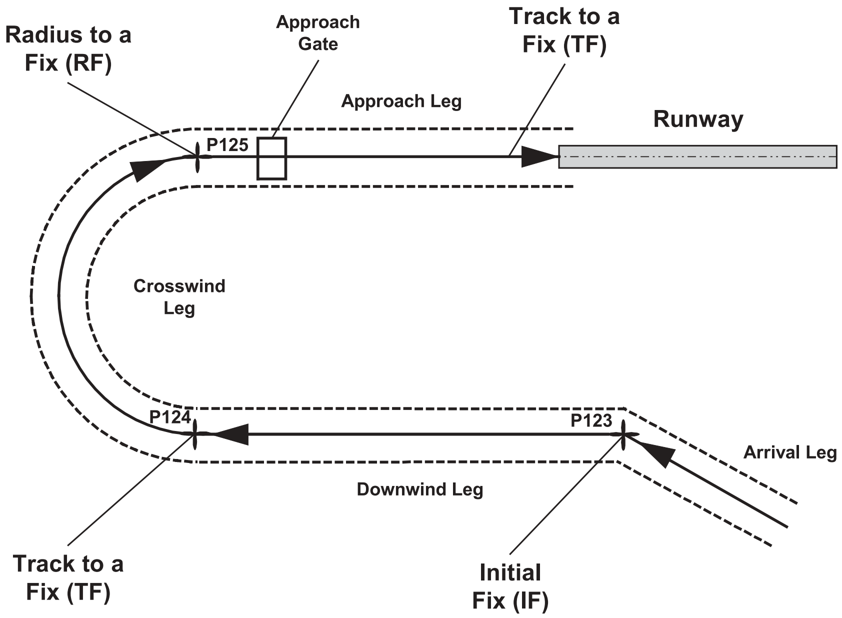


Fig. 12.9 *Fixes which are discouraged for RNP*

Fig. 12.10 Route structures developed for RNP



It follows that, to define and maintain a robust and repeatable route structure, the emphasis needs to be placed upon those navigation legs that are most rugged in their ability to minimize flight path error. The combination of such flight legs may lead to a structure similar to that shown in Fig. 12.10. In this example a series of robust legs is combined to define a highly repeatable approach comprising arrival, downwind, crosswind, and approach legs. The repeatability of this philosophy has been amply demonstrated in a number of terminal approach examples, including Frankfurt in Germany, Schipol in the Netherlands, and San Francisco and Boston in the United States, to name a few. Not only can these approaches be flown with greater navigational accuracy, but the precise alignment of arrival and departure routes can be aligned greatly to improve noise abatement and the overall environmental impact upon communities in the airport locality.

Path steering error

The assumed values for FTE are given in Table 12.3. Studies on recent aircraft suggest that actual values achieved in modern aircraft using digital autopilots are in fact much better than the assumed values (see Table 12.4).

Position estimation error

As well as the data given in the tables above, reference (**2**) gives a comprehensive overview of the considerations to be addressed when considering RNP/RNAV, and of future likely developments. The specified increases in traffic density described so far only affect the lateral spacing of aircraft; vertical spacing is another consideration. The desire also to reduce the vertical separation between aircraft is addressed in another aspect of FANS – Reduced Vertical Separation Minima (RVSM).

Table 12.3 Assumed values for path steering FTE (5 per cent probability) (2)

Flight Phase	Manual (nautical miles)	Flight Director (nautical miles)	Autopilot (nautical miles)
Oceanic	2.0	0.5	0.25
En-route	1.0	0.5	0.25
Terminal	1.0	0.5	0.25
Approach	0.50	0.25	0.125

Table 12.4 Actual values for path steering FTE (Boeing study (2))

	Manual flight with map display (nautical miles)	LNAV with flight director coupled (nautical miles)	LNAV with autopilot coupled (nautical miles)
En route	0.502–0.918	0.111–0.232	0.055–0.109
Terminal	0.208–0.402	0.073–0.206	0.068–0.088

RVSM

One of the other ways of increasing traffic density is the introduction of the RVSM criteria. For many years, aircraft have operated with a 2000 ft vertical separation at flight levels between FL290 and FL410. As traffic density has increased, this has proved to be a disadvantage for the busiest sections of airspace. Examination of the basic accuracy of altimetry indicated that there were no inherent technical reasons why this separation should not be reduced. Accordingly, RVSM was introduced to increase the available number of flight levels in this band and effectively permit greater traffic density. The principle is to introduce additional usable flight levels such that the flight level separation is 1000 ft throughout the band, as shown in Fig. 12.11.

Originally, a trial was mounted in 1997 to test the viability of the concept on specific flight levels – FL340 and FL360 as shown in the Fig. 12.11. RVSM is now implemented throughout most of Europe from FL290 to FL410, introducing six new flight levels compared with before. All the specified flight levels will be implemented on the North Atlantic by early 2002. Other regions in the globe will have RVSM selectively implemented to increase air traffic density according to Fig. 12.12. and Table 12.5.

RVSM implementation

At the time of writing, the time-scale for the worldwide implementation of RVSM is shown in Fig. 12.12 and in the accompanying Table 12.5 (**3**). RVSM operation requires the aircraft to possess two independent means of measuring altitude and an autopilot with an accurate height hold capability. The operators of RVSM-equipped aircraft are not taken on trust: independent height monitoring stations survey aircraft passing overhead, measuring actual height compared with flight plan details and the individual aircraft fin number aircraft and route plan. RVSM implementation therefore embraces a watchdog function that ensures that all users are conforming to the RVSM accuracy and performance provisions.

Fig. 12.11 *RVSM – insertion of new flight levels*

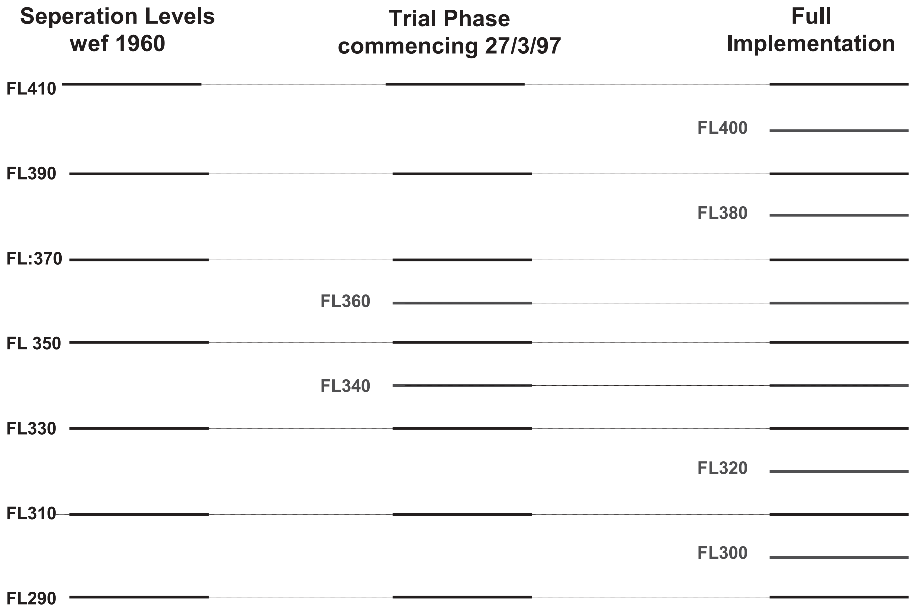


Fig. 12.12 *RVSM implementation – world wide*

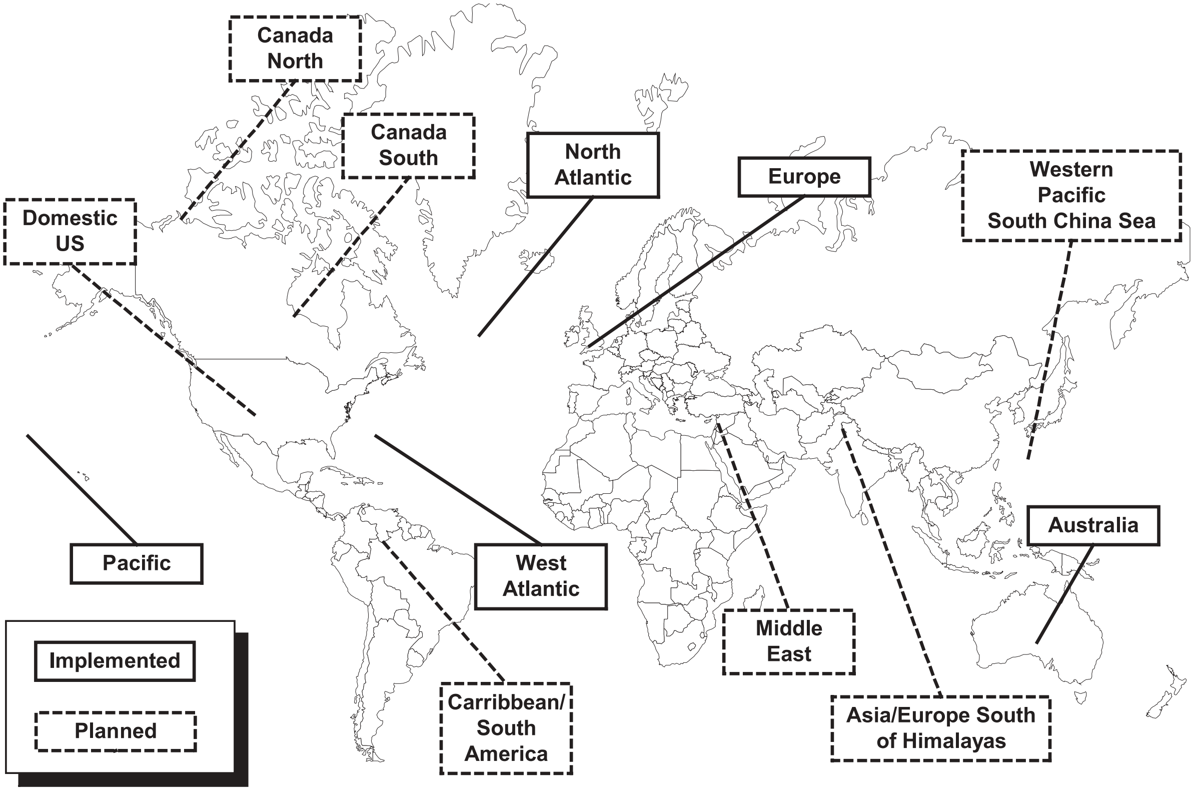


Table 12.5 RVSM – worldwide status

	RVSM status - Americas and Europe	
North Atlantic	March 1997	FL330-370
	October 1998	FL310-390
	24 January 2002	FL290-410
West Atlantic route system (WATRS)	1 November 2001	FL310-390
	24 January 2002	FL290-410
Europe tactical (UK, Ireland, Germany, Austria)	April 2001	FL290-410
Europe-wide	24 January 2002	FL290-410
South Atlantic	24 January 2002	FL290-410
Canada North domestic	April 2002	FL290-410
Canada South domestic	Coordinate with US domestic	
Domestic US – phase 1 (1)	1 December 2004	FL350-390
Domestic US – phase 2 (1)	Late 2005-2006	FL 290-390 (FL410)
Note: (1) DRVSM plan to be finalised not later than January 2002 based on ATC simulation results and user inputs		
Caribbean/South America	RVSM Group Established	
	RVSM status – Asia/Pacific	
Pacific	February 2000	FL290-390
	Tactical Use	FL400-410
Australia	November 2001	FL290-410
Western Pacific/South China Sea	21 February, 2002	Consult Publications
Middle East	November 2003	TBD
Asia-Europe/South of Himalayas	November 2003	TBD

The stated benefits of RVSM in the North Atlantic (NAT) RVSM area are:

- Improved operations/efficiency.
- Greater availability of more fuel-efficient altitudes.
- Greater availability of the most fuel-efficient tracks or routes.
- Increased probability that an operator will be cleared onto the desired track or altitude.
- Enhanced controller flexibility to manage traffic through an increased number of available altitudes.

Differential GPS enhancements

DGPS enhancements are being developed for en route and precision landings in the United States. The GPS enhancements WAAS and LAAS have already been described in Chapter 8 – Navigation, and their introduction should lead to the following accuracies being achieved as a matter of course:

- WAAS is anticipated to yield an accuracy of ~7 m, which will be sufficient for Cat I approaches.
- LAAS is expected to provide enhanced accuracies of ~1 m, which will be sufficient for precision approaches Cat II and Cat III.

The introduction of DGPS technology is also envisaged for Europe and the Far East. In Europe there are two programmes in the planning stage that will enhance satellite navigation. The European Space Agency (ESA), the European Commission (EC), and the European Organization for the Safety of Air Navigation (Eurocontrol) are working together on the development of a Global Positioning and Navigation Satellite System (GNSS) plan. The GNSS programme is being carried out in two phases:

- GNSS-1. This involves the development of the European Geostationary Navigation Overlay System (EGNOS) which will augment the existing US GPS and Russian GLONASS systems.
- GNSS-2. This involves the development of a second-generation satellite navigation system, including the deployment of Europe's own satellite system – Galileo. At the time of writing, the EU nations had agreed to secure funding for the deployment of the Galileo system

Within GNSS-l , Europe is contributing EGNOS which is aimed at augmenting the performance of GPS and GLONASS in terms of precision and data integrity. EGNOS is now becoming a reality as a test bed – a simplified version of the fully fledged system has recently been readied by Alcatel Space, the prime contractor leading the international industrial team that is developing the system.

The system is based on the use of ground infrastructure and three geostationary satellites. These are INMARSAT-3 Indian Ocean region (IOR) and Atlantic Ocean region east (AOR-E) and, soon, ESA's Artemis telecommunications satellite which is due to be launched into geo stationary orbit above Africa in 2002. These spacecraft are equipped with dedicated navigation transponders to augment the positioning services currently offered by the GPS and GLONASS constellations. The EGNOS ground infrastructure will be deployed over more than 40 sites, mostly in Europe. The ground infrastructure for the preoperational version has already been deployed at many sites around Europe: France, Iceland, Italy, the Netherlands, Norway, Spain, Turkey, and the United Kingdom, and at two sites outside Europe: Kourou (French Guyana) and Hartebeeshoek (South Africa).

The EGNOS system will be qualified at the end of 2003 and will provide in Europe an operational satellite navigation service that will later be improved with the operational introduction of the Galileo satellite system – scheduled for 2008. The European EGNOS, the US WAAS, and the Japanese Multifunction Satellite Augmentation System (MSAS) will be fully interoperable

Following the commissioning of the EGNOS test bed system, an EGNOS-like signal has since mid-February been transmitted from space, providing users with a GPS augmentation signal and enabling them to compute their positions to an accuracy of a few metres. The EGNOS test bed signal is currently available in the coverage area of the AOR-E satellite and as of May 2000 became available in the coverage area of the IOR spacecraft.

This pre-operational version of EGNOS will allow Europe to support demonstration of the operational benefits of GNSS to user communities. The test bed will be used for all modes of transport (air, land, and maritime) that require positioning services to accuracies of a few metres, and more particularly safety-critical services. For aviation users, for instance, EGNOS will provide for en route navigation as well as non-precision approach and precision approach phases of flight.

The European Commission is more specifically responsible for the promotion of applications and user equipment, an activity that relies largely on the use of the EGNOS system test bed. Financial support is also provided by EC for the EGNOS project, in particular for the lease of INMARSAT-3 transponders and implementation of the Artemis navigation payload.

In Japan, the MSAS is intended to provide a differential satellite-based system for a wide range of users including the aviation community. Unfortunately, the first satellite was lost during launch in 1999, and no further progress has been made.

Protected ILS

Within Europe, some ILS installations suffer interference from high-power FM local radio stations. Modifications have been mandated that introduce receiver changes to protect the ILS systems from this interference.

Introduction of the Microwave Landing System (MLS)

The MLS is an approach aid that was conceived to redress some of the shortcomings of ILS. The specification of a time-reference scanning beam MLS was developed through the late 1970s/early 1980s, and a transition to MLS was envisaged to begin in 1998. However, with the emergence of satellite systems such as GPS there was also a realization that both ILS and MLS could be rendered obsolete when such systems reach maturity. In the event, the US civil community is embarking upon higher-accuracy developments of the basic GPS system: the WAAS and LAAS implementations described earlier. In Europe, the introduction of the similar EGNOS system may also render the need for dedicated approach aids unnecessary. For the moment, the United Kingdom, the Netherlands, and Denmark have embarked upon a modest programme of MLS installations at major airports.

Polar routes

The increased range of modern aircraft, together with the improvement in navigation systems, has led in recent years to the exploitation of the north polar route. Certain airlines are providing direct flights from New York (JFK), Chigaco, and Los Angeles to Far Eastern destinations using direct polar routes that realize considerable fuel economies. There are areas where significant caution needs to be applied on these routes as some of the diversion airfields are not well equipped and can suffer from severe weather conditions. The performance of certain parts of the aircraft avionics equipment can suffer at high latitudes, and communications coverage in particular can be affected. Nevertheless, a great deal of attention is being given to utilizing the modern advances to exploit these new routes [see reference (**4**)]. A supplementary publication outlines some of the navigation difficulties experienced at extreme latitudes, especially when it is anticipated that the aircraft will transit at or near the north pole, creating specific difficulties for the navigation system (e.g. an abrupt singularity where north turns to south while flying over the pole).

Surveillance

Surveillance enhancements include the following:

- TCAS II.
- ATC mode S.
- Automatic Dependent Surveillance A (ADS-A).
- Automatic Dependent Surveillance B (ADS-B).

The operation of TCAS and ATC mode S has been described in Chapter 6, but their use in a FANS context will be examined here.

TCAS

The operation of TCAS has already been described in Chapter 6. When operating together with a mode S transponder and a stand-alone display or EFIS presentation, TCAS is able to monitor other aircraft in the vicinity by means of airborne interrogation and assessment of collision risk. TCAS II provides vertical avoidance manoeuvre advice by the use of RAs. TCAS II will soon be made mandatory for civil airliners – aircraft with a weight exceeding 15 000 kg or 30 or more seats – operating in Europe. This will be extended to aircraft exceeding 5700 kg or more than 10 seats, probably by 2005.

ATC mode S

The use of the ATC mode S transponder in providing digital air–ground and air–air data links (VDL mode 4) has already been described in Chapter 6. This provides a basic ADS-B capability.

Mode S also has the capability of providing a range of data formats – from level 1 to level 4. These are categorized as follows:

- Level 1. This is defined as the minimum capability mode S transponder. It has the capability of reply to mode S interrogations but has no data link capability. All the messages provided by level 1 are short (56 bit) messages.
- Level 2. These transponders support all the features of the level 1 transponder with the addition of standard-length, data-link word formats. This can entail the use of longer messages (112 bit). Some of the messages are used for TCAS air–air communication while others are utilized for air–ground and ground–air communication as part of the enhanced surveillance Data Access Protocol System (DAPS) requirements.
- Level 3. The level 3 transponders embrace the same functionality as level 2 with the additional ability to receive Extended-Length Messages (ELMs) which comprise 16 segments of information, each containing a 112 bit message
- Level 4. Level 4 has the full functionality of level 3, with the capability of transmitting ELM messages of up to 16 segments of 112 bit word messages.

Originally it was envisaged that ATC mode S would be the primary contender to provide the CNS/ATM functionality by providing large block transfers of information. More recently it has been realized that VDL mode 4 might better serve this need, and levels 3 and 4 are no longer required.

Automatic Dependent Surveillance – address mode (ADS-A)

ADS-A will be used to transmit aircraft four-dimensional position and flight plan intent based upon GPS position during oceanic crossings. The communications media will be SATCOM or HF Data Link (HFDL). ADS-A requires the aircraft to be fitted with an FMS and CDU and with some means of displaying message alerts and annunciation.

Automatic Dependent Surveillance – broadcast mode (ADS-B)

ADS-B will be used to transmit four-dimensional position and flight plan intent based upon GPS position using line-of-sight VHF communications. Either mode S or digital VHF radio will be used to transmit the data. ABS-B requires a cockpit display of traffic information.

An extended demonstration project in the Louisville area conducted by the FAA in conjunction with United Parcels Services (UPS) and others has helped to demonstrate the ADS-B concept and has paved the way for UPS aviation technologies to secure certification for new displays that can provide improved runway, aircraft, and weather information displays facilitating the movement of aircraft in the terminal area. UPS have predicted that this technology will yield a 20 per cent capacity increase at Louisville, as reducing the arrival spacing of aircraft by 20–30 s can reduce the nightly sorting operation by around 30 min. Time savings of this proportion are critical in maintaining delivery of parcels on time.

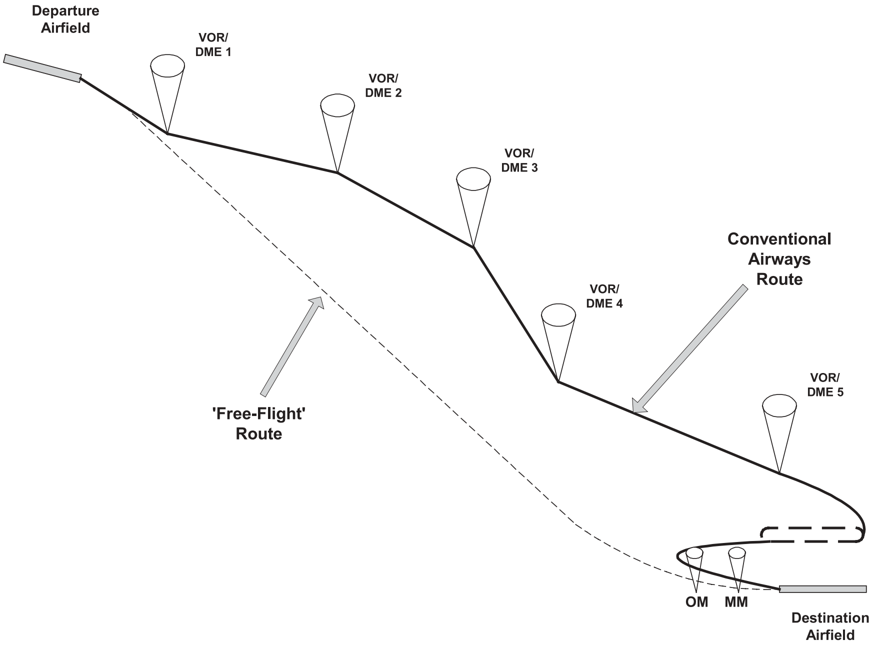
Direct routing

The CNS triad offers a suite of methods for addressing the FANS objectives. The ultimate aim of FANS is to provide a ‘free flight’ capability as shown in Fig. 12.13.

Under prevailing navigational means, an aircraft flying from a departure airfield to a destination airfield would need to file a flight plan that entails flying down prescribed airways flying from VOR/DME beacon to beacon. Upon arrival in the vicinity of the destination airfield, the aircraft may have to enter a holding pattern while waiting for clearance to land. The approach may necessitate an ILS approach that would involve overflight of the outer marker some 5–7 nautical miles on the extended runway centreline. Many inefficiencies exist in this type of routing which is typical of many flights carried out over land today.

Free flight on the other hand would enable a more direct routing from the departure to the destination airfield, clearly yielding savings in time and fuel, as indeed RNAV allows today. Free flight is rather more than RNAV: it will encompass a wider range of ADS capabilities such that data links may be used to modify ATC flight plans and clearances in a dynamic fashion. Furthermore, upon arrival at the destination airfield, the aircraft could execute a more direct and efficient approach and landing; most probably enabled by phasing its approach with that of other aircraft by means of four-dimensional, required time of arrival navigation techniques. An FMS embodying a full-performance aircraft model could make further savings by optimizing the use of fuel during flight such that the aircraft could meet the necessary four-dimensional waypoints at the appropriate time with a minimum fuel burn.

Fig. 12.13 *Principle of 'free flight'*



The universal application of the principles of free flight is some way off. Free flight may already be used on specific flight segments where air traffic density is relatively low, but the universal application depends upon the proven maturity of the technologies described. There are complex political and financial factors that compound the technical issues. Free flight in Europe will probably not be fully available until the partitioning of the airspace is simplified and rationalized compared with the complex structure that exists today. The cost of new equipment, both air and ground based, is not trivial. Airlines will wish to be assured that the investment in new equipment will provide an adequate return as new FANS capabilities are introduced. Nevertheless, in the medium and long term, the pressures of air traffic density and fuel economy will ensure that many FANS features will attain maturity.

Need for flight management system (FMS)

The need to be able to manage all of these techniques in a timely manner is crucial to the success of FANS. The need for an FMS with the capabilities outlined in Chapter 8 will continue to be a key factor in integrating these features in a robust manner. The ability to provide software upgrades to the FMS will enable present route structures and communications to be improved while new ones are added. It is vital, therefore, that the FMS contains sufficient spare memory and computation capacity to allow for future growth in these areas.

Boeing FANS 1 Implementation

The Boeing approach to providing a FANS capability has been to provide software upgrades for older models that impart additional navigation and communications functionality to existing avionics hardware. Newer models such as the Boeing 777 and Boeing 767-400 will have these functions embedded. This approach may be summarized by reference to Fig. 12.14 which shows a typical Boeing display/FMS interface.

The major features are:

- Conventional navigation information displayed on the captain and first officer's Navigation Displays (NDs).
- On the Boeing 747-400, when the FANS 1 package is loaded, the captain's CDU acts as a host to display additional ATC communications functions as well as the existing navigation functions.
- On the Boeing 737 aircraft with the appropriate software upgrade loaded, additional guidance information is shown on the Primary Flight Display (PFD), as described in reference (**9**). The PFD is enhanced by the display of lateral and vertical deviation scales along the bottom and right-hand edge of the PFD. By flying the aircraft to maintain zero lateral and vertical deviation (the same as for an ILS approach), deviations from the desired flight path are minimized. This enhancement enables the same flight technical error deviation to be achieved in Flight Director (FD) as in autopilot mode.

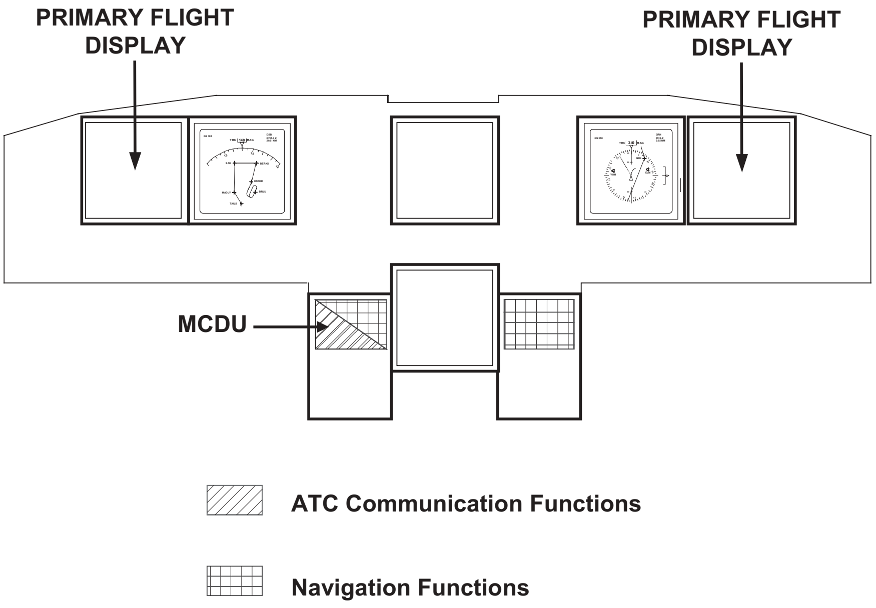


Fig. 12.14 *Boeing FANS 1 – display of information*

The stated performance of the Boeing aircraft in RNP navigation modes is quoted in references (**5**–**9**).

Airbus FANS-A Implementation

Airbus has taken a slightly different approach to implementing its FANS upgrade by adding additional equipment rather than by adding software functionality to existing hardware. The units added include:

- An Air Traffic Services Unit (ATSU) incorporating an Airborne Communications Addressing and Reporting System (ACARS) and airline operation communication (AOC) functions.
- Replacement of existing radios with VHF Digital Radios (VDRs).
- Introduction of Dual Data-link Control and Display Units (DCDUs) located on the centre console (see Fig. 12.15).
- An upgraded FMS module hosted in the Flight Management and Guidance Envelope Computer (FMGEC).

Initially, this was fitted to the A330/340 family with a view to moving towards a FANS-B version implementing Aeronautical Telecommunications Network (ATN) functionality.

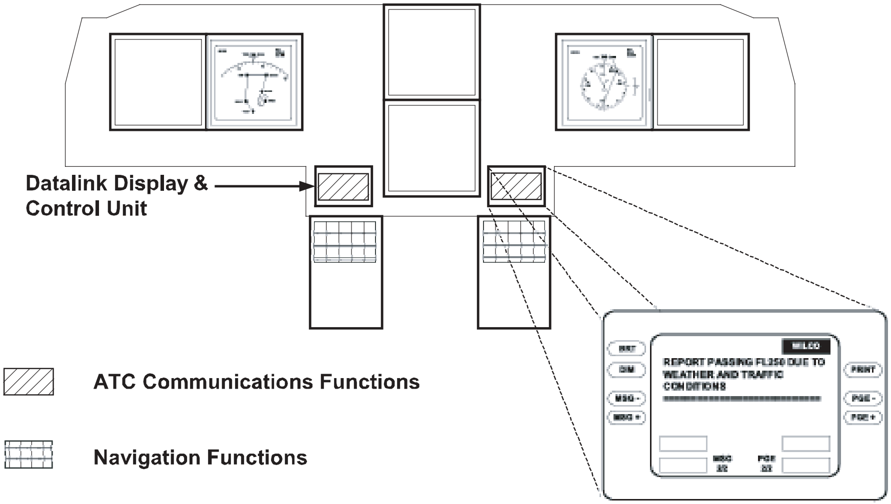


Fig. 12.15 *Airbus FANS A – display of information*

High-precision approaches

The judicious application of the FANS techniques described in this chapter have allowed major advances to be made and approaches to be designed that would not previously have been possible. One such example is the new approach designed for RW26 at Juneau, Alaska. The airport at Juneau is surrounded by high mountains, and the approach to RW26 requires flying north-west up a glacial valley embracing the Gastineau Channel. Figure 12.16 presents a view of Juneau airport and the Gastineau channel viewed from the north-west, showing the rugged nature of the terrain.

Figure 12.17 depicts the horizontal profile of the approach to RW26. Before the design and implementation of the RNP approach, this access to Juneau was impossible in all but the best weather conditions. The procedures have been designed to accommodate RNP_0.15, RNP_0.2, and RNP_0.3 approaches which allow Decision Heights (DHs) of 337, 437, and 1810 ft respectively. The approaches were designed by the Alaska Airlines in conjunction with Boeing and the FMS manufacturer, Smiths Aerospace. A full description of the considerations involved in developing the approach is given in reference (**10**).

Fig. 12.16 *Juneau, Alaska*

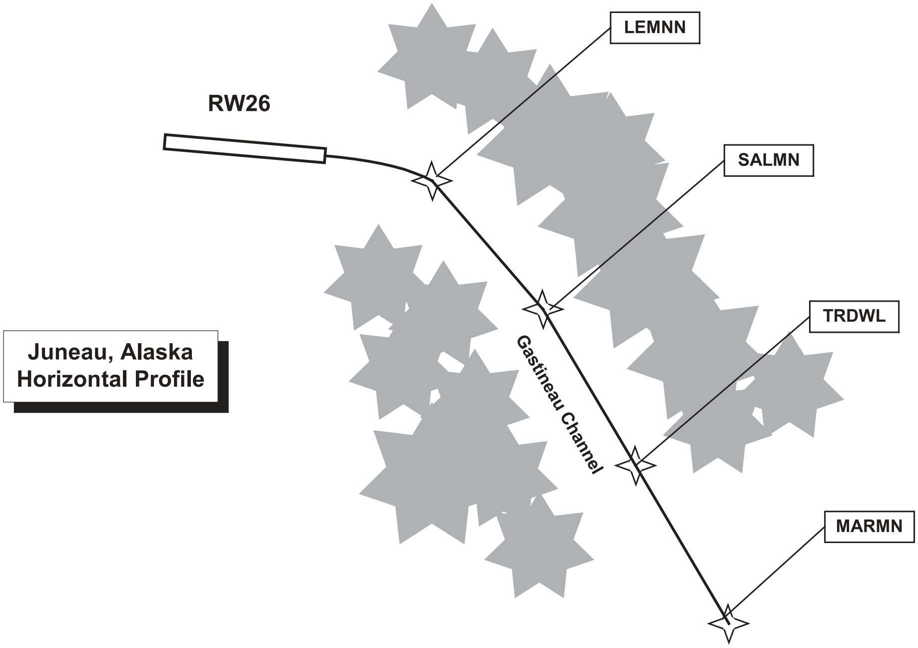


Fig. 12.17 *Juneau, Alaska – horizontal profile*

Using the new approach allowed shorter arrival and departure procedures when flying to Juneau from Seattle – about 5 min inbound and 6 min outbound. In terms of annual savings it has been estimated that about 245 flight hours, or 740 000 lb of fuel, could be saved assuming present flight schedules.

The use of RNP procedures also allows a more stable approach path (see Fig. 12.18). Usually, a conventional approach would involve a series of short descents followed by level flight as the aircraft 'steps down' the approach path. Using improved guidance, the aircraft is able to follow a gradual approach path starting at the outer waypoint (MARMN), some 15 miles out, and continue this descent down to the DH. This reduces crew workload and allows the flight crew to spend more time monitoring the status of the approach rather than continually changing power and trim settings. This is a further benefit of an RNP-based approach to add to the other benefits.

The aircraft used to fly this approach is a Boeing 737. The minimum navigation equipment fit required to execute this approach is listed below:

- Dual FMS.
- Dual IRS.
- Dual GPS.
- Dual EFIS.

Experience showed that the weather radar provided terrain information that was most useful to monitor the status of the approach, and a serviceable weather radar was added to the minimum equipment fit.

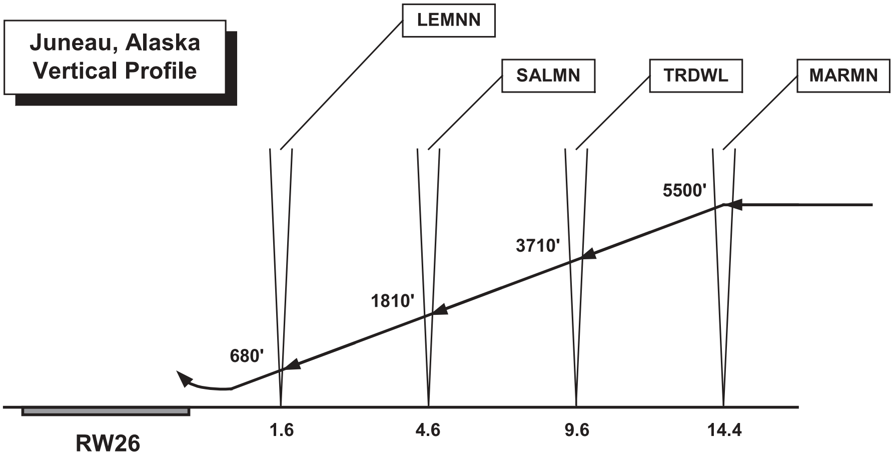


Fig. 12.18 *Juneau, Alaska – vertical profile*

References

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(2) **Nakamura, D.** (2000) Boeing FMS RNAV Workshop – General Information on the Functional and Technical Aspects of Required Navigation Performance (RNP) Area Navigation (RNAV) and Applications

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(6) RNP Capability of FANS 1 FMCS Equipped 747-400, Generation 3, -747-400, Rev B, (2000), D926U0280.

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(8) RNP Capability of FANS 1 Equipped 777, -777 Rev B, (1999) D234W018-7.

(9) Lateral and vertical navigation (2001) Boeing Aero 16.

(10) **Kramer, M.** (1998) RNP RNAV Terminal areas – procedures design and approval for Juneau, Alaska, *FITEC '98.*

CHAPTER 13

Military Aircraft Adaptation

There are two scenarios in which civil avionics are being used in military applications. The first is in the conversion of commercial aircraft to military roles, and the second is the updating of military types to make use of modern avionics where it is economically viable to use Commercial Off-The-Shelf (COTS) systems.

There are applications in the military field for which the civil aircraft platform together with its avionic systems is well suited. It may often be economically viable to convert an existing civil type rather than develop a new military project. Much of the development costs of the structure and basic avionics will have been recouped from airline sales for a new platform. Alternatively, a used aircraft bought from an airline may also be an economic solution. In either case, the basic avionics fitted will have been well tried and tested, and will be ideally suited for use in controlled airspace.

There have been numerous examples of successful conversions serving the armed forces around the world. The successful Nimrod maritime patrol aircraft was based on the De Havilland Comet airframe and has seen service with the Royal Air Force (RAF) as the MR1 and MR2 for over 30 years. The type is currently being extensively refurbished by BAE SYSTEMS with the designation of MRA4. The refurbishment includes a new wing, new engines, and new avionics and mission systems being fitted to the existing MR2 fuselage. The Comet airframe was also the platform for the Nimrod R Mk1 signals intelligence aircraft in service with the RAF as well as the ill-fated airborne early warning AEW Mk3. Similarly, the Lockheed P-3 Orion maritime patrol aircraft was based on the Lockheed Electra and has seen many years of service with the US Navy and many other operators worldwide. The HS 748 passenger aircraft was successfully converted to the Andover troop and cargo transporter for use by the RAF, whereas the VC10 and Tristar have been used for personnel transport only. Tanker aircraft have been developed from the Boeing 707 (KC-135), the VC10, and the Lockheed L1011 Tristar, and proposals are in place for use of the Boeing 767 and

Airbus A330 in this role. Commercial types are also used for carrying passengers, troops, and VIPs, such as the 125 Dominie, VC10, Tristar, and 707.

There are also instances where civilian agencies have made use of aircraft for surveillance. These include Police, Customs and Excise, Fisheries Protection and drug enforcement agencies. These applications are often civilian helicopters or small commercial aircraft.

Typical applications for re-use of commercial aircraft include:

- Personnel, matériel, and vehicle transport.
- Air-to-air refuelling tanker.
- Maritime patrol.
- Airborne early warning.
- Ground surveillance.
- Electronic warfare.
- Flying classroom.
- Range target/security aircraft.

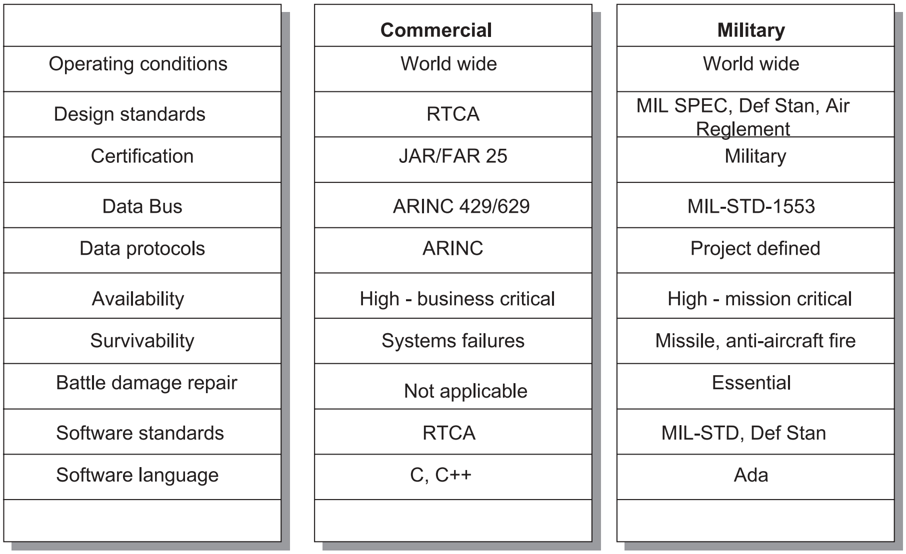
For these roles, a large slow-moving platform capable of transit over long distances and loitering for long periods of time, together with internal space for a mission crew and their workstations, is ideal. Many commercial aircraft are capable of conversion to these roles, retaining the basic structure, avionics, and flight deck with the installation of additional avionics, or mission systems to tailor the aircraft to a specific role. Retaining the basic avionic systems architecture makes good sense, since military aircraft make extensive use of civilian controlled airspace during peacetime, and will often transit to theatres of operation, for training and defence purposes, using commercial routes.

The conversion to a military role, especially if the carriage of weapons is included, requires a different approach to safety and qualification, challenging the aircraft design teams to make the best use of civil and military certification rules. This often poses interesting problems in the mixing of design standards and processes.

The basic avionic systems are complemented by a set of sensors and systems to perform specific surveillance tasks. This is a situation in which the basic navigation and communication systems become part of the role-specific systems, and in which there are particular issues of accuracy, integration, and security. These are especially important in instances where a commercial aircraft platform forms the basis of the military vehicle. In such instances there may be conflicts between the characteristics of the embedded systems on the commercial vehicle and the requirements of the military vehicle. These issues may affect the approach to design and certification of the resultant aircraft. Some of the issues can be seen in Fig. 13.1.

Avionic and mission system interface

The basic avionics provide information to the flight deck crew for safe flying, handling, and monitoring the behaviour of the aircraft and its systems, as described in the previous chapters in this book. In addition to this, the mission crew require assistance in performing the military task. For this they will have their own sensors and systems which will be supplemented by information from the basic avionics suite. There are a number of sources of information required by the crew to build up a picture of the tactical scenario, as illustrated in Fig. 13.2.



	Commercial	Military
Operating conditions	World wide	World wide
Design standards	RTCA	MIL SPEC, Def Stan, Air Reglement
Certification	JAR/FAR 25	Military
Data Bus	ARINC 429/629	MIL-STD-1553
Data protocols	ARINC	Project defined
Availability	High - business critical	High - mission critical
Survivability	Systems failures	Missile, anti-aircraft fire
Battle damage repair	Not applicable	Essential
Software standards	RTCA	MIL-STD, Def Stan
Software language	C, C++	Ada

Fig. 13.1 *Comparison of commercial and military certification issues*

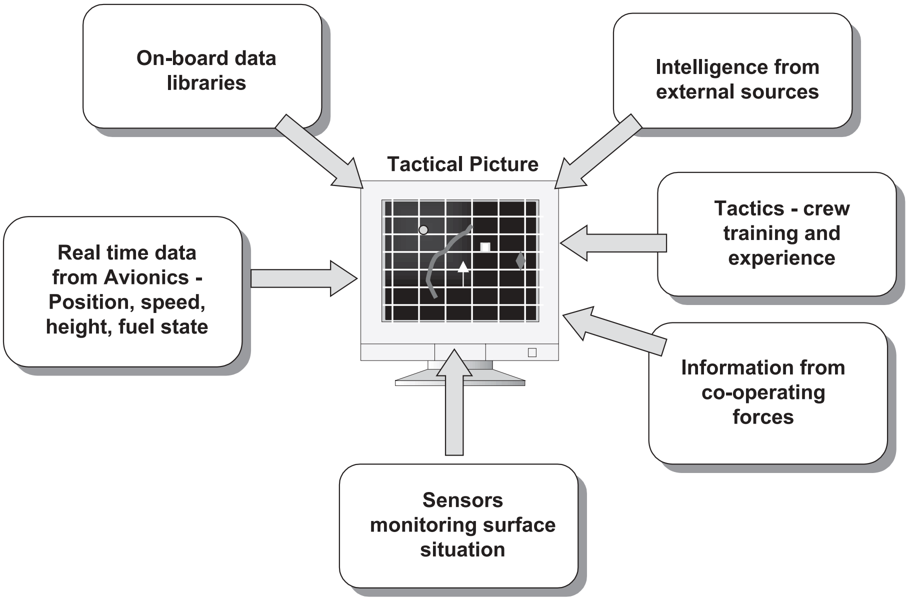


Fig. 13.2 *Contributors to the tactical practice*

The data that the crew have mentally to assimilate include:

1. Real-time data from avionics. Typical information such as position, speed, height, attitude, and heading is provided from the navigation and air data sensors to aid tactical navigation. Information on the quantity of fuel remaining is essential to allow the mission crew to determine how long they can safely remain on task. The flight deck crew will monitor all these parameters as an independent check that the aircraft is being handled safely.

2. Sensor information. Information from the mission sensors will be prepared by sensor operators and provided to form a single tactical picture from which information can be gathered to conduct the mission.
3. On-board data libraries. Data are stored that enable the sensor operators to estimate what type of target they have detected. This information may be historical and may be supplemented by data obtained on each mission. Experienced operators are able to identify a target type, and sometimes even an individual ship or submarine, by its combined sensor signature.
4. Tactics. All operating crews are taught the appropriate tactics to detect and track their targets. These tactics are built up from many years of operational experience and are often highly classified, since their disclosure would enable an enemy to evolve tactics to avoid detection.
5. Information from cooperating forces. Most surveillance aircraft operate in collaboration with other forces and platforms such as fighter aircraft, helicopters, and ground forces operated by allies. All intelligence gained is made available by secure communications to add to the tactical picture.
6. Intelligence from external sources. Headquarters teams may have access to other information in large databases or gleaned from the emerging scenario, especially if information is being provided by many sources. This is either sent to the aircraft by secure communications for inclusion in their own database, or decisions made at headquarters after analysis of the tactics may be sent, again by secure communications.

The requirement to provide a comprehensive system for capture, analysis, and presentation of this information must be clearly understood so that the system architecture is fit for purpose. The architecture will then be designed to ensure that the basic avionics is able to provide the appropriate data with the appropriate accuracy and data rate, and with the required level of integrity. The mission system will be designed to meet its own internal requirements for sensors, processing, and data display. The interfaces between the systems must then be designed to respect the need for information by the mission system, and to preserve the integrity of the basic avionics so that aircraft safety is not impaired. Figure 13.3 shows some examples of interfaces that may be required between the avionics and the mission system.

Figure 13.4 shows an example architecture that separates the commercial and military data bus structures. One mechanism for doing this is to retain much of the basic avionics architecture, including the ARINC bus structure and data protocols. The mission computer can be used to accept the ARINC interfaces, convert the data formats for its own use, and provide the interface and data formatting and the bus control function for the MIL-STD-1553 data bus. Any military avionics that need to be incorporated in the flight deck can be included in the MIL-STD-1553 architecture by extending the bus or by providing a long stub connection. More information on ARINC and MIL-STD data buses can be found in Chapter 2.

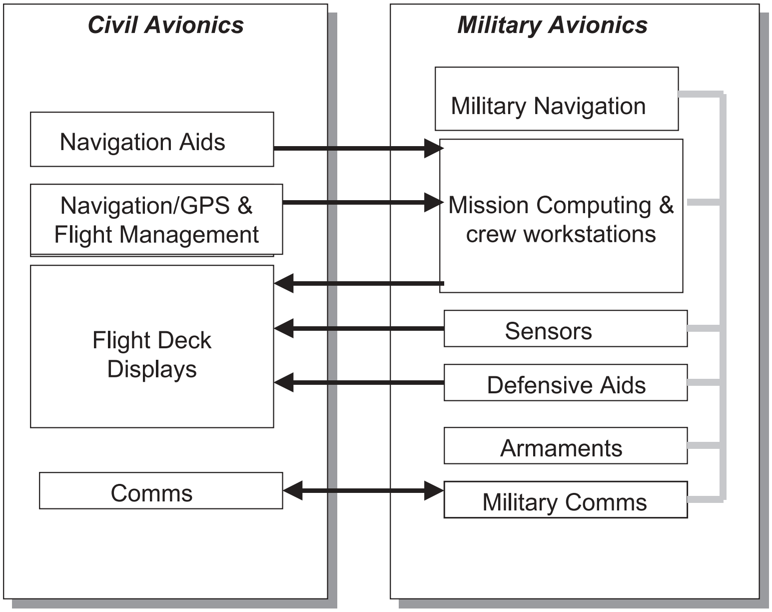


Fig. 13.3 *Generic interfaces between avionics and mission systems*

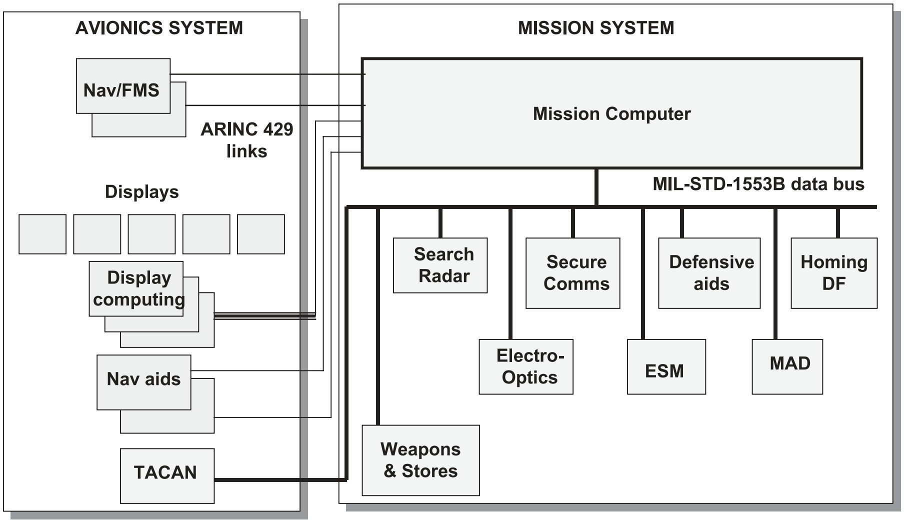


Fig. 13.4 *A typical architecture*

Navigation and flight management

Unlike a commercial aircraft, which routinely flies on well-planned and preordained commercial routes programmed into its Flight Management System (FMS), the military aircraft may only use such routes during training or transit to an operational area. Once on task, the aircraft may be following a target, such as tracking a marine vessel, surface vehicles, or a submarine. In these circumstances its route becomes unplanned or determined by the target it is following, and an accurate knowledge of position is vital for a number of reasons:

- The long-range maritime patrol aircraft may spend many hours on patrol over open ocean, up to 1000 miles from base. It is important that its position is known for coordination during rescue operations.
- To understand the position of the aircraft relative to its last known planned position, i.e. when it entered the operational area, and its location relative to other forces, both friendly and hostile.
- To ensure that the aircraft does not enter hostile aircraft or missile engagement zones.
- To report the position of the target to other forces in order to assemble an accurate surface picture.
- To hand over the target to another aircraft if required.
- To attack the target if required and to provide accurate position information to the weapon(s).
- To ensure that the return route home is known and to determine when to break off an engagement and return safely to base with adequate fuel reserves to cover a diversion at home base. This also requires an accurate knowledge of fuel quantity.
- Commercial Global Positioning System (GPS) based navigation systems (see Chapter 8) are sufficiently accurate for most purposes. A military code enables access to even more accurate navigation data, and an active GPS antenna ensures that access to the right number of the most suitable satellites is guaranteed in most areas of the world.

The FMS will be used to store planned routes to and from operational areas, often using commercial air lanes and routes for fuel economy reasons in peacetime. In such cases the use of navigation aids is mandatory. It may also be used in conjunction with the mission computing system to allow the aircraft to fly automatically on stored search patterns. This enables the aircraft to fly for long periods of time in a systematic search of the land or sea surface. The FMS also eases crew workload during normal operations, making it possible to reduce the flight deck crew numbers, with consequent savings in operating costs. For long-duration missions, in which there may be long periods of flying at low level in adverse weather conditions, this reduction in workload is valuable in reducing crew fatigue and contributing to the safety of the mission. This is especially the case when flying repetitive search or surveillance patterns at low level for long periods of time. These patterns can be input into the FMS as routes and called up by the crew to start and finish at defined geographical points. The FMS and autopilot are not normally expected to perform this sort of task, and additional safety analysis will be required to ensure that the systems can be certificated to operate at low level without endangering the aircraft.

Navigation aids

Modern navigation aids complement the role of many military aircraft. Apart from their use in normal airways flying and in using commercial airports, the systems may be used as follows:

- The radar altimeter (or rad alt, also known as radio altimeter) may be used to provide warning of deviation from a set height or to assist in the maintenance of height at very low levels (see Chapter 5). The radar altimeter measures absolute height above land or sea. It operates by sending either continuous or pulsed radio signals from a transmitter, and receives signals bounced back from the surface. The time taken for the signal to travel to the surface and back is converted to an absolute altitude that can be used by other systems, as well as provided as an indication on the flight deck. Aircraft flying low for long periods of time will use two radar altimeters, enabling the mission to be completed even if one altimeter fails. By positioning the transmitting and receiving antennae for each system on either side of the aircraft ventral surface, the radar altimeters continue to measure true height above the ground even when the aircraft is pulling a tight banking turn, or when the aircraft is flying in a steep-sided valley. This is especially used by maritime patrol aircraft where a height above the sea surface as little as 200 ft must be maintained even when operating in very tight turns (up to 2g). For such applications, a dual-radar altimeter is used of a type that has a low probability of intercept type signal using frequency hopping or pulse modulation to avoid detection by enemy assets.

- Landing aids may be employed using a MultiMode Receiver (MMR) to allow the use of a microwave landing system or differential GPS in addition to the usual Instrument Landing System (ILS) (see Chapter 6). This allows the aircraft to operate in remote areas where conventional aids are not available or may have been destroyed. Mobile MLS or differential GPS can be set up by occupying ground troops to provide a service to friendly aircraft.
- Ground Proximity Warning Systems (GPWSs) and Enhanced GPWSs or Terrain Avoidance Warning Systems (TAWSs) (see Chapter 8) will be used over foreign terrain and may be connected to a military digital map to improve accuracy and confidence in the system. Such a system is known as terrain profile mapping (Terprom). Terprom correlates stored terrain data against inputs from the aircraft's navigation and radio altimeters to achieve a highly accurate, drift-free navigation solution. This is used to provide predictive terrain awareness with no detectable forward emissions such as would be produced by terrain-following radar. This is especially useful in transport aircraft making covert approaches to dropping zones while using the terrain as defensive cover and wishing to avoid detection by hostile electronic support measures systems. The system can be updated with military intelligence to designate hostile areas and newly erected obstacles.
- A terminal area Traffic Collision Avoidance System (TCAS) is used not only in crowded airspace in the vicinity of airfields but also in training where there may be a concentration of aircraft in a small space, for example a designated low-flying area (see Chapter 6).
- TACAN is a chain of military navigation beacons used to obtain range and bearing. It can also be used to provide homing to a tanker aircraft (see Chapter 6).

Flight deck displays

A problem with early aircraft was that there was no space on the flight deck to provide tactical information to the pilots, mainly because of the proliferation of single-function indicators and control panels. This meant that the flight deck crew were only made aware of tactical situations by the intercom. This had an impact on their effectiveness and the safety of the aircraft for the following reasons:

- Their reaction time to commands from the tactical crew was delayed.
- Lack of exterior view in poor light conditions reduced the effectiveness of rescue efforts.
- If aircraft manoeuvres, especially violent manoeuvres or in poor weather conditions,

were commanded by the rear crew with the flight deck crew ‘hands off’, the pilots tended to suffer disorientation or air sickness (rather similar to car passengers compared with the driver of a rally car.

The availability of modern multifunction display suites as described in Chapter 7, allows information from the mission system to be more readily displayed within the existing display suite. This means that the flight deck crew can see the tactical map and can also see InfraRed (IR) images that will enable them to conduct rescue missions more effectively. This also means that they can make a major contribution to the mission by flying the aircraft as if they were part of the mission crew. In this case the pilots will elect to alternate the role of flying pilot and tactical monitoring at regular intervals. This technique keeps both pilots alert and reduces the probability of both getting immersed in the tactical scenario and neglecting the flying task. The ability to switch the tactical picture from one side of the cockpit to the other is an advantage.

The flight deck displays can also display weather radar pictures obtained from the mission radar and are also able to display threat information produced by the defensive aids system. This enables the pilots to take evasive action rapidly if they believe that the threat is serious and endangers the aircraft.

Communications

The aircraft must have a set of communications to fly in controlled airspace – VHF and HF are part of the commercial aircraft fit, and both are essential for military use. However, most military communications use UHF so that additional UHF sets will be installed or combined V/UHF radios fitted. It may also be necessary to include encryption devices into the aircraft to provide secure communications, in which case the majority of the radios will need to be replaced. Maritime patrol aircraft that operate in close cooperation with naval and marine assets may need short-wave or marine band radios.

For most secure communication the military use satellite communications and data links. These systems allow encrypted standardized format messages to be sent and received that contain data as well as speech. This allows the aircraft to download its tactical information to other aircraft or to headquarters with a high probability of sound reception and minimum risk of interception. The most common data link in use in NATO is Link 16 or the Joint Tactical Information Distribution System (JTIDS), although naval operations employ Link 11. Both links enable the mission crews to compile messages using preferred, defined formats and to transmit and receive using their normal communications transceivers.

Military applications may include the intermittent use of high-power transmitters which may interfere with, and may damage, the aircraft communications and radio navigation aids. To reduce the likelihood of this, the sites of the antennae are carefully selected to prevent mutual interference. An active form of interference prevention known as blanking and suppression often reinforces this method of careful antenna location. This is accomplished by ensuring that equipment transmitting and receiving in the same frequency band signal are connected so that receiving equipment temporarily stops receiving while other equipment is transmitting at high power.

Aircraft systems

The aircraft electrical systems will be compatible with military avionics, although the civil and military specifications for power supply quality may differ in their method of defining quality. There may be a need to provide additional load conditioning such as filtering to ensure compatibility. Another issue is that some military applications may impose transient and intermittent high power loads that may cause power surges and induced noise. In the event that the military loads exceed the capacity of the aircraft generation system, it will be necessary to install larger generators, and this will have an impact on the engine off-take loads and may lead to a recertification programme.

The aircraft cooling systems will be designed to deal with a large passenger heat load, so their use in a military system is usually adequate – although the equipment cooling requirement may be higher, the number of passengers (crew) is considerably smaller. Very high power loads such as radars or jamming transmitters may need their own integral liquid cooling systems.

One complication, however, is the need to operate and survive in conditions where biological and chemical contaminants may have been encountered. Rather than modify the environmental conditioning system of the aircraft, the usual method is to provide the crew with their own personal survival equipment in the form of a survival suit and a portable filtration unit. This can be complemented by installing an On-Board Oxygen Generation System (OBOGS) if the aircraft role demands that it needs to remain on station for long periods of time in contaminated conditions (**1**), but the installation of an OBOGS is a costly option.

The impact on the aircraft hydraulics systems is usually minimal, although some sensors may need hydraulic power for rotation or for extension and retraction.

Applications

Personnel, matériel, and vehicle transport

For large-scale military operations requiring troops or matériel to be transported into a war zone, specially designed aircraft such as C-130 or C-17 are used. However, there are instances where a less sophisticated platform will suffice. Typical candidates for such roles are passenger-carrying aircraft such as the Boeing 707, L1011 Tristar, VC10, or Airbus series. For personnel or troop transportation, very little change to the basic aircraft is required. Typical additional military avionics includes:

- Communications systems may need to include military IFF with its own cryptos as well as HF or satellite communication.
- TACAN is used to gain access to the military chain of homing and navigation beacons.
- If the aircraft is to be modified to enable paratroops to be carried and dispatched in large numbers, then the flight deck needs to be informed of door or ramp opening to avoid the impact of pressurization changes.
- Station keeping – to enable the aircraft to fly safely in large formations and to maintain a safe distance from others in the fleet, especially during inclement weather conditions.
- Landing aids such as the Microwave Landing System (MLS) or differential GPS – to allow the aircraft to fly safely into remote military landing zones using mobile ground stations set up in the field.

- Defensive aids system – to detect threats and to provide a defensive system using chaff and flare dispensing equipment.

Air-to-air refuelling

Air-to-air refuelling has been used to extend the range and endurance of military aircraft and, in some instances, to extend test flying, as successfully demonstrated in the Lockheed Joint Strike Fighter X-35 demonstration aircraft. The cost of designing and building special-to-type refuelling aircraft is extremely high since the number used is relatively small. The majority of aircraft in use today are developed from commercial airliners:

- The RAF uses tankers based on the VC10 and L1011 Tristar with drogues fitted to the tanker and probes fitted to the receiving aircraft – the typical UK and European refuelling method.
- The USAF uses tankers based on the Boeing 707, known as the KC-135, with a boom fitted to the tanker and a socket fitted to the receiver – the typical US refuelling method.
- Future candidates for conversion include airframes from Airbus (A330) and Boeing (767). The cost of maintaining a fleet of tankers at operational readiness at all times is causing air forces to consider an alternative in the form of the provision of a service by industry. Serious consideration has been given to modification kits to allow rapid conversion of commercial aircraft in times of tension.

The tanker is equipped with fuel tanks and modifications to its fuel system to allow pressurized refuelling of up to three receiver aircraft. The tanker must be able to navigate to remote locations, to loiter, and to maintain station with its receiving aircraft. The difference in speed between the large tanker and fast jets can be problematic, with the tanker flying fast and the fighters flying slow at high angles of incidence. The differences in weight resulting from the transfer can alter this speed differential, resulting in a high workload task. Typical additional military avionics include the following:

- Communications systems may need to include military IFF with its own cryptos as well as HF or satellite communication.
- TACAN is used to gain access to the military chain of homing and navigation beacons. TACAN also includes a function that enables receiving aircraft to home onto the tanker.
- Data links in order to maintain secure communications with headquarters and the receiving aircraft and their controllers – the location of these aircraft and the refuelling zone is of vital tactical significance. Link 16 also includes a facility to allow aircraft to home in on the tanker.
- Defensive aids system – to detect threats and to provide a defensive system using chaff and flare dispensing equipment.
- A means of monitoring the receiving aircraft on station and controlling the flow of fuel.
- A means of monitoring the correct deployment of the drogues or booms.
- A means of cutting and jettison if the drogue/boom fails to retract.

Maritime patrol

Maritime Patrol Aircraft (MPA) are used for a number of purposes by nations with extensive coastlines, those that may be dependent on commercial shipping for import and export, or those that have extensive naval fleets. Such aircraft are most often used in support of the homeland, although remote deployments are not uncommon in times of tension or war. Typical roles are a mixture of civilian, police, and military duties:

- Fisheries protection and monitoring of illegal fishing.
- Off-shore assets protection (e.g. oil installations).
- Anti-drug/armaments/smuggling/terrorist operations.
- Search and rescue.
- Protection against hostile forces – antisubmarine/surface vessel threats.
- Command and control for emergency operations.

To perform these roles, the aircraft requires a number of sensors and a mission crew to gather and interpret data, identify vessels, report their location, and prosecute an attack if necessary (see Fig. 13.5 for a summary of aircraft roles).

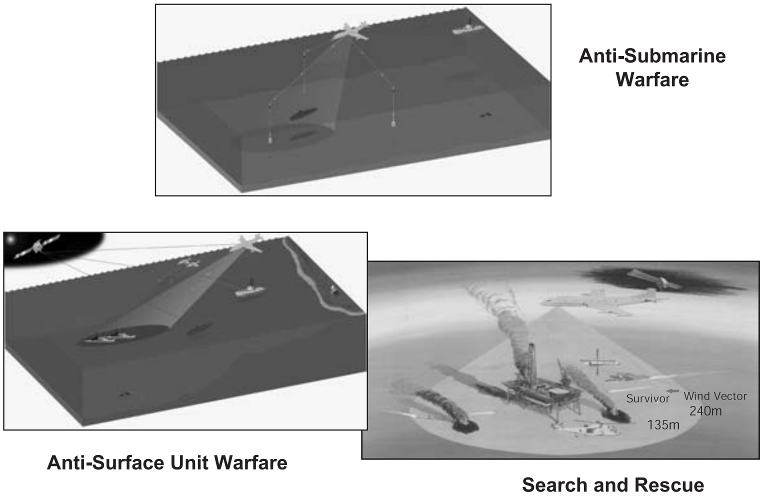


Fig. 13.5 *Maritime Patrol Aircraft roles (BAE SYSTEMS)*

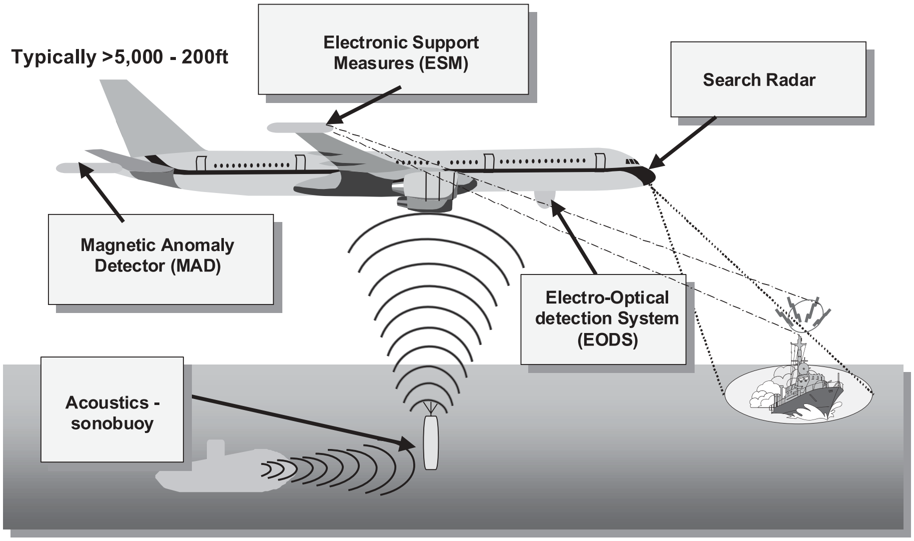
Sensors

The sensors necessary to perform these roles are shown in Fig. 13.6.

Maritime search radar

The maritime search radar beam is optimized to conduct a wide area surface search to discriminate small and fleeting targets against a background of sea clutter. The range of targets that an MPA must detect includes large capital ships, small surface craft, the periscope or antenna of a submarine, or human survivors and life rafts. The MPA must be able to detect and track such targets in sea states ranging from flat calm (sea state 0) to rough waves (often sea state 4). This is no easy task, and the typical search radar

Fig. 13.6 Maritime patrol aircraft – sensors



must be designed to detect such targets, and also to minimize side lobes so that the MPA is not detected by enemy sensors.

The radar is typically located in the nose of the aircraft so that the downward looking search mode is complemented by weather mode and some degree of look-up mode to detect missiles, enemy aircraft, or cooperating aircraft. This location restricts the search to about 270˚ about the nose of the aircraft, and search patterns are designed to make maximum use of this limitation. A 360˚ search coverage can be achieved by installing a radar in the tail of the aircraft and fusing the two signal returns. This is expensive and complicated. An alternative method is to install the radar beneath the aircraft, which maximizes the circular search at the expense of the look-up capability and is also a drag penalty.

Electro-optics

In conditions of poor visibility or night-time operations, electro-optical sensors are used to provide an image of targets that can be sensed, detected, and tracked manually or automatically by a turret located on the ventral surface of the aircraft. Such sensors will include IR, low-light TV, and digital cameras. They can be used singly or in combination to provide an image to the mission crew or the flight crew. This is especially helpful in locating survivors in the sea, while for military and policing operations the images can be annotated with position and time and may be used as evidence in legal proceedings.

Electronic support measures

The Electronic Support Measures (ESM) system is able to intercept, locate, record, and analyse radiated electromagnetic energy for the purpose of gaining tactical advantage. It is a completely passive system, its function being to ‘listen’ to signals radiated from other assets. The system is capable of covering the whole frequency spectrum of interest – radars, communications, missile guidance signals, laser emissions, and IR emissions.

For monitoring radio and radar signals, the intercept receiver is a versatile conventional radio receiver operating over a wide bandwidth. A signal analyser enables an operator to determine those parameters of interest to the mission. A direction finding facility is provided so that range and bearing of the signal can be determined. A library of typical threat profiles allows an experienced operator to identify an emission to a particular emitter type, and also to identify the type of platform most likely to be equipped with that emitter.

Acoustics

A key role of the MPA is to detect and track submarines. Its main tool for doing this is the sonobuoy, which it dispenses into the water in patterns to enable the range, track, and bearing of the submarine to be detected. The sonobuoys listen for acoustic signals from the submerged vessel and transmit the signal to the aircraft. Submarines are operated to avoid detection, maintaining silence by reducing the operation of machinery. Various types of sonobuoy are used:

- Passive (listening only).
- Active (using a sonar emission or 'ping' and monitoring the return echo).

Magnetic Anomaly Detector (MAD)

The Magnetic Anomaly Detector (MAD) is used to detect large metallic objects in water, such as a submarine. The MAD is usually mounted at the end of a tail boom to keep the sensitive detector head away from magnetic objects in the airframe that could affect its performance. Once a maritime patrol aircraft has detected and pinpointed what it believes to be a large underwater mass that could be a submarine, it will confirm the target by overflying and looking for the MAD to signal that the mass is magnetic. This provides an extremely accurate fix and incidentally ensures that the target is not a shoal of fish or a whale.

The range of the MAD is very limited, so the aircraft must fly very low over the suspected target, often as low as 200 ft. To do this, the pilot will usually fly manually, disconnecting the autopilot to avoid pitch runaways causing a catastrophic descent. To do this automatically would require an autopilot designed and cleared to operate at very low levels – this is not a normal operation for a commercial aircraft.

Stores

The MPA is equipped as a hunter-killer and carries weapons to enable it to attack and destroy surface and subsurface vessels. For this purpose it will carry antiship missiles and torpedoes. The weapons are often carried internally in a bomb bay. It is also possible to carry missiles on wing-mounted pylons. For rescue operations the aircraft will carry flares, life rafts, and survival stores which can be dropped near to survivors.

Mission crew

Modern navigation systems based on inertial navigation with satellite-aided global navigation systems and flight management systems have led to the situation in which the flight deck crew members are able to perform all the navigation aspects of a mission. This has led to the replacement of the routine navigator in the Nimrod MRA4, leading to a reduction in the mission crew.

The mission crew act as a team, although each member may have a specific role such as radar operator, communications manager, acoustics operator, and ESM operator. The data they each gather and refine is passed to the TACtical COmmander (TACCO) who compiles a composite picture of the surface, annotating friendly and enemy assets together with their tracks and identification if known. The avionic systems provide navigation data such as present position, heading, track, altitude, height above the surface, and speed. This enables the TACCO to determine the position of the MPA with respect to other assets. Tactical information is also sent to the flight crew so that they are aware of any potential manoeuvres required and can take rapid action. The flight deck crew must monitor height above the surface at all times to ensure that the TACCO does not command an attack manoeuvre that would endanger the aircraft.

Airborne early warning

In many conflicts it is essential to gain air superiority to protect air, ground, and maritime forces. A vital element of this strategy is constantly to monitor the hostile air threat. The airborne early warning aircraft can provide this capability by patrolling at height and observing aircraft movements beyond the sight of ground radars. The monitoring systems can provide range, bearing, and speed of approaching air threats, and associated databases or libraries of threat profiles are able to provide intelligence on the type of threat. The aircraft crew are able to direct fighter interceptor aircraft to intercept targets of interest.

A typical aircraft is the E-3 Sentry Airborne Warning and Control System (AWACS) which is a modified Boeing 707 aircraft. A radar scanner is installed in a 9 m diameter rotating dome mounted on struts above the aircraft rear fuselage. The radar is able to search for targets between the land or sea surface and up to the stratosphere. The radar has a range of over 250 miles for low-flying targets, and in excess of this for high-flying targets. The radar has an IFF interrogator which enables the crew positively to discriminate between friendly and hostile targets, discriminating low-flying targets from ground clutter returns. The aircraft is also equipped with an electronic support measures system to allow detection and identification of radio frequency transmitters. Data link allows secure communication with external agencies and other forces. A mission computing system and a number of operator workstations provide facilities for a large crew of operators and analysts to work comfortably for long periods of time. The aircraft also includes facilities for rest and sleep.

Ground surveillance

An important aspect of air surveillance in times of tension or during conflict is to enable staff officers to gain an understanding of the disposition of assets, ground forces, installations, and communications sites of opposing forces. This information completes the picture presented by maritime patrol, AEW, and ground intelligence.

Surveillance of the battlefield can be conducted by aircraft flying high and deep inside friendly airspace. The aircraft carries a Sideways Looking Aperture Radar (SLAR) or a Synthetic Aperture Radar (SAR) which has a fixed or ‘staring’ antenna. The aircraft flies a fixed track that allows the best view of the area under surveillance. This enables the radar picture to be built in slices, which can be represented by a computer-generated picture. This can be supported by ESM to capture well-camouflaged or hidden transmitters that the radar may not detect.

An example of such a system is the Airborne Stand-Off Radar (ASTOR) system developed by Raytheon. The system is based on the Bombardier Global Express long-range executive jet fitted with a derivative of the Raytheon ASARS-2 Sideways Looking Radar (SLR). The radar is an upgrade of the SAR used on the U-2 aircraft and is capable of operation in all weathers at high altitudes to provide high-resolution images. The ASARS-2 includes dual Synthetic Aperture Radar (SAR) which provides photographic quality images of the area being surveyed and a Moving Target Indicator (MTI) radar which tracks moving vehicles over wide ranges, penetrating cloud and rainy conditions.

The SAR can operate in spot mode to identify and track specific targets or can be switched to swath mode which provides a large number of strips of pictures which can be joined together to form a detailed image of the battlefield. The SAR/MTI combination identifies the location of hostile forces and their quantity, direction, and speed. Additional imagery can be provided by the use of electro-optical devices.

The image data are transmitted in real time by secure data links to ground stations, other forces, or a headquarters command. The links are interoperable with existing U-2s and other surveillance platforms. The data can also be analysed and used by the on-board mission crew to direct ground operations and supporting land and air vehicles. At command headquarters the information is used by commanders and tacticians to understand the total battlefield scenario.

The aircraft is also equipped with a defensive aids system for self-protection which includes missile warning, radar warning receiver, towed radar decoy, and chaff/flare dispensers.

Electronic warfare

Electronic warfare is a term that encompasses a number of aspects of intelligence gathering and active jamming of enemy transmitting assets. Commonly used techniques of electronic warfare are:

- Electronic CounterMeasures (ECM).
- Electronic Support Measures (ESM).
- SIGnals INTelligence (SIGINT).
- Electronic Counter-CounterMeasures (ECCM).

The gathering of information is usually conducted by all available means, including airborne, space satellite, naval, and ground-based systems. Information gathered is transmitted to a headquarters intelligence cell where the data are analysed and used to conduct military operations. Part of the airborne element of this activity can be conducted by fast jets equipped with purpose-built photographic, receiving, or jamming pods, making rapid transitions over enemy territory. However, for long-term intelligence gathering together with the ability to act as an airborne command post, a large long-endurance platform is preferred. Such a platform has the ability to fly at high altitude in friendly territory with its sensors observing targets many miles inside enemy territory. This operation is relatively safe, since the aircraft can ‘stand off’ out of the range of surface-to-air missiles.

Flying classroom

Mission crews or back-seat navigators will usually do their training in ground-based simulators. An additional degree of realism can be introduced by overflying ranges and operating in a live range environment. Although this can be performed in operational aircraft, the cost is high, especially in fast jets. An alternative that has been used is to install a number of crew workstations in the rear of a large aircraft. This allows a single instructor to train a number of crew members on a single mission. An example of this is the BAE SYSTEMS Jetstream equipped to train Tornado navigators.

Range target/safety

Military aircraft can conduct a lot of peacetime exercises on a military range. Targets can be simulated by specially equipped commercial aircraft equipped with radio frequency emitters that emulate enemy aircraft so that crews can perform search and interdiction missions.

An example of this is the Falcon aircraft fleet operated on a commercial basis by Flight Refuelling. These aircraft can also patrol the range boundaries to provide safety and observation. This is especially useful for very large desert or maritime range exercises where the range area may have to be patrolled for long periods of time to restrict access to vessels or aircraft that may stray onto the range.

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