MASAAG Paper 123

Development, Validation, Verification and Certification of Structural Health Monitoring Systems for Military Aircraft

Prepared for:

The Military Aircraft Structural Airworthiness Advisory Group

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IMPERATIVE EXPLANATORY NOTES

Via involvement with the SAE IVHM committee, the MAA became aware of the international efforts of the SAE G-11 SHM technical committee. The Committee's first work was a published Aerospace Recommended Practice paper for civil transport aircraft. A military version of the paper is also planned with BAE Systems involvement. At present, the progress of the international work on the military paper is extremely slow. The MAA therefore suggested that a UK-specific paper could take benefit from existing work, as well as provide a good opportunity for SMEs to peer review the paper contents. MASAAG Paper 123 was therefore written and includes contents covering the UK military perspective.

MASAAG Paper 123 does not promote or endorse a technology or a system; the paper only provides guidance on best practice processes required to fit a matured system/technology into military aircraft. The aim of MASAAG Paper 123 is to provide general guidance on how to validate, verify, and certify SHM systems for military aircraft by imperative considerations of military regulations and defence standards. The paper guidance contents do not constitute a UK MOD policy or regulatory requirements. The MOD regulations and the means of compliance with these regulations are those published and updated by MAA. For aircraft products including SHM and similar systems, the UK default specifications and requirements are those stated within the UK defence standards. MASAAG Paper 123 must only be considered as a best practice guidance paper. The example background information presented in this paper covered topics such as structural design methods, architecture, derivation of generic system requirement, etc. The paper does not endorse or recommend any of these examples; it only presents them to generate multi-discipline awareness of a wide range of topics within a single paper.

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EXECUTIVE SUMMARY

BAE Systems and HAHN Spring Limited are executive members of an international Aerospace Industry Steering Committee (AISC), which operates as the SAE G-11 SHM technical committee to establish guidelines for developing, validating, verifying, and certifying SHM systems. The committee assembles leasing representatives from key aerospace organizations including Airbus, Airbus Defence and Space, Boeing, Bombardier, Embraer, BAE Systems, GE, Honeywell, the National Aeronautics and Space Administration (NASA), Sandia National Laboratories, UTC Aerospace Systems, Messier-Bugatti-Dowty, the US Air Force, the US Navy, the Federal Aviation Administration (FAA), the European Aviation Safety Agency (EASA), National Research Council Canada (NRC), HAHN Spring Limited, Stanford University, Cranfield University, Japan's RIMCOF, University of Tokyo, and Delft University of Technology (TU Delft). The Committee's first work was to develop SHM guidelines for civil transport aircraft, which were finalized as the Aerospace Recommended Practice ARP6461 and published by SAE in September 2013.

Some of the guidelines of ARP6461 can be used for military applications. However, by definition, ARP6461 does not address specific military considerations and does not cover the wider spectrum of military aircraft types. Furthermore, the scope of ARP6461 did not include guidance on the integration of SHM within the aircraft and its military support systems. Therefore, the committee decided to develop another document, ASE ARP6245, which would provide guidance on military SHM applications that were not addressed in ARP6461.

Via involvement with the SAE IVHM committee, the MAA became aware of the international efforts of the SAE G-11 SHM technical committee. To date, it has been noticed that the progress of the committee on the military paper is slow, perhaps, because the military regulations, standards, and processes required for designing and managing aircraft structures can vary between nations and between the military operators of one nation. Therefore, the MAA suggested that a UK-specific paper could take benefit from existing work and provide a good opportunity for peer reviews. MASAAG Paper 123 was therefore written and includes contents covering the UK military perspective.

The approach adopted in developing the validation, verification, and certification guidelines presented in this paper was to maintain harmony with existing applicable regulations, standards and guidelines, and to augment them with specific interpretations and best practice guidance pertaining to SHM as necessary. For the purpose of this paper, certification is considered as the processes required for obtaining approval from the appropriate Regulatory Authority (FAA, EASA, MOD, DOD, etc.) that the applicable airworthiness regulations, operating rules, and system requirements are met.

The guidelines of this paper were derived through the following activities:

- Comprehending aircraft design and maintenance philosophies;
- Reviewing regulations that govern the design and maintenance of airworthy structures;
- Summarising regulations and processes required to overcome the threats to structural integrity;
- Discussing the evolution and acquisition phases of aircraft products including SHM;
- Presenting the architectural ingredients and choices that can deliver required SHM intended functions;
- Establishing the requirements of SHM systems that are essential for compliance with airworthiness regulations and industry accepted standards;
- Presenting the methods required to validate and verify aircraft products with interpretations and extensions specific to SHM systems;
- Presenting the certification phases, their outputs, and associated approval forms, and reviewing the UK MAA Regulatory Articles for new aircraft designs and for aircraft with major changes.

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GLOSSARY

AAA	Ageing Aircraft Audit
ac	Alternating Current
AC	Advisory Circular
AC	Airworthiness Certificate
ACS	Active Control System
AD	Accidental Damage
AD	Airborne Delivery
ADC	Analogue to Digital Converter
ADCC	Approved Design Change Certificate
ADE	Aerial Delivery Equipment
ADN	Aircraft Data Network
AR	At Risk of AD/ED
ADE	Aerial Delivery Equipment
ADS	Aircraft Document Set
ADS	Aircraft Design Standards
AE	Acoustic Emission
AE	Airborne Equipment
AEA	Aircrew Equipment Assemblies
AERC	Airborne Equipment Release Certificate
AFDX	Avionics Full Duplex Switched Ethernet Network
AFE	Airborne Forces Equipment
AFERC	Airborne Forces Equipment Release Certificate
AFRL	Air Force Research Laboratory
AG	Advisory Generation
AI	Artificial Intelligence
AIA	Aerospace Industries Association
AISC	Aerospace Industry Steering Committee
ALARP	As Low As Reasonably Practicable
ALW	Air Launched Weapons
AMC	Acceptable Means of Compliance
AMM	Aircraft Maintenance Manual
AMC	Acceptable Means of Compliance
AMC	Acceptable Means of Compliance
AOA	Aircraft Operating Authority
AOAs	Aircraft Operating Authorities
AOF	Acquisition Operating Framework
APU	Auxiliary Power Unit
ARINC	Aeronautical Radio, Inc
ARP	Aerospace Recommended Practice
ASAMG	Air Support Airworthiness Management Group
AS&FC (Air)	Autonomous Systems and Future Capability (Air)
A-SHM	Automated-SHM
ASIP	Aircraft Structural Integrity Program
ASMS	Air Safety Management System
ASTRAEA	Autonomous Systems Technology Related Airborne Evaluation & Assessment
ATC	Advance Technology Centre
BCA	Business Case Approval
BIT	Short for Binary Digit
CAAMG	Combat Air Airworthiness Management Group
CAE	Continuing Airworthiness Engineering Regulations
CAM	Capability, Airworthiness and Maintenance
CAWG	Civil Aircraft Working Group

CBM	Condition Based Maintenance
CBM+	Condition Based Maintenance plus Prognostics
CDM	Chief of Defence Materiel
CG	Centre of Gravity
CLE	Clearances with Limited Evidence
CM	Configuration Management
CofU	Certificate of Usage
COO	Costs of Ownership
COTS	Commercial off The Shelf
CP	Certification Programme
CPI	Certification Process Improvement
DA	Data Acquisition
DA	Design Authority
DAL	Development Assurance Level
DAOS	Design Approved Organization Scheme
dc	Direct Current
DCF	Discounted Cash Flow
DE&S	Defence Equipment and Support
Def Stan	Defence Standards
DET or DI	Detailed Inspection
DH	Duty Holder
DLO	Defence Logistics organization, DLO is merged in DE&S
DLOD	Defence Lines of Development
DM	Data Manipulation
DME	Design and Modification Engineering
DO	Design Organization
DOD	Department of Defense
DPA	Defence Procurement Agency, DPA is merged in DE&S
DRTSA	Delegated Release To Service Authority
DUST	Dual Use Science and Technology
e-CFR	electronic Code of Federal Regulations
ED	Environmental Deterioration
EHF	Extremely High Frequency
ELF	Extremely Low Frequency
EMI	Electro Magnetic Interference
ES	End System
ESA	European Space Agency
ESVRE	Establish, Sustain, Validate, Recover and Exploit
EWIS	Electrical Wiring Interconnection Systems
FAA	Federal Aviation Administration
FAR	Federal Aviation Regulations
FCR	Flight condition Recognition
FD	Fatigue Damage
FDR	Flight Data Recorder
FHA	Functional Hazard Assessment
FLC	Front Line Command
FMEA	Failure Mode and Effects Analysis
FMECA	Failure Mode, Effects, and Criticality Analysis
FMF	Fatigue Meter Formula
FOAS	Future Offensive Air System
FRP	Fibre Reinforced Plastics
FTA	Fault Tree Analysis
FTR	Fatigue Type Record
FTS	Flight Test Schedule

FUM	Fatigue and Usage Management
GAMA	General Aviation Manufacturer's Association
GARP	Generic Aircraft Release Process
GFRP	Glass Fibre Reinforced Polymer
GPS	Global Position System
GV or GVI	General Visual Inspection
HA	Health Assessment
HAMG	Helicopter Airworthiness Management Group
HCF	High Cycle Fatigue
HOC	Head of Capability
HSWA	Health and Safety at Work Act
HUM	Health and Usage Monitoring
HUMS	Health and Usage Monitoring System
HUSLE	Helicopter under Slung Load Equipment
IAT	Individual Aircraft Tracking
ICA	Instructions for Continued Airworthiness
ICD	Interface Control Document
IGA	Initial Gate Approval
ILS	Integrated Logistic Support
IPT	Integrated Project Team
IRT	Internal Rate of Retune
ISO	International Organization for Standardization
ITE	Independent Technical Evaluator
ITEAP	Integrated Test, Evaluation and Acceptance Plan
ITS	Introduction-to-Service
IVHM	Integrated Vehicle Health Management System
JAP	Joint Air Publication
JAR	Joint Airworthiness Requirements
JSF	Joint Strike Fighter
JSP	Joint Service Publication
LCF	Low Cycle Fatigue
LoAA	Letter of Airworthiness Authority
MAA	the UK MOD Military Aviation Authority
MAA Cert S & ADS	MAA Certification Structure and Aircraft Design Standards
MAC	Media Access Control
MACP	Military Air Systems Certification Process
MAE	Military Air Environment
MAP	Maintenance and Airworthiness Processes
MASAAG	Military Aircraft Structural Airworthiness Advisory Group
MAWG	Military Aircraft Working Group
MDRE	Manual Data Recording Exercise
MFTP	Military Flight Test Permit
MGA	Main Gate Approval
MIMOSA	Machinery Information Management Open Standards Alliance
MLA	Manoeuvre Load Alleviation
MMP	Major Military Products
MMS	Master Maintenance Schedule
MOD	Ministry of Defence
MRBR	Maintenance Review Board Report
MSG	Maintenance Steering Group
MSI	Maintenance Significant Items
MTBF	Mean Time between Failures
MTC	Military Type Certificate
MTCDS	MTC Data Sheet

NAR	Not At Risk of AD/ED
NASA	National Aeronautics and Space Administration
NATO	North Atlantic Treaty Organization
NAVAIR	The US Naval Air Systems Command
NAVSEA	The US Naval Sea Systems Command
NDE	Non-Destructive Evaluation
NDI	Non Destructive Inspection
NDT	Non Destructive Test
NPV	Net Present Value
NRC	National Research Council
ODR	Operational Data Recording
OEC	Operational Emergency Clearances
OEM	Original Equipment Manufacturer
OLM	Operational Load Monitoring
OLM	Operational Loads Measurement
OMP	Obsolescence Management Plan
OSA	Open System Architecture
OSA-CBM	Open Systems Architecture for Condition-Based Maintenance
OSA-EAI	OSA for Enterprise Application Integration
OSDEP	Out of Service Date Extension Programme
PA	Prognostics Assessment
PC	Production Certificate
PHM	Prognostic Health Management
PI	Propulsion Integrity
PM	Preventive Maintenance
PMA	Parts Manufacturing Approval
POD	Portability Of Detection
PoE	Power over Ethernet
PSCP	Project Specific Certification Plan
PSE	Principal Structural Element
PSP	Partnership for Safety Plan
PSS	Products, Services and/or Systems
PSSA	Preliminary System Safety Assessment
PT	Project Team
PTL	Project Team Leader
RA	Regulatory Article
RA	Regulatory Authority
RAT	Ram Air Turbine
RCM	Reliability Centred Maintenance
RDC	Remote Data Concentrator
RF	Radio Frequency
RFI	Radio Frequency Interference
RI	Regulatory Instruction
RIMCOF	R & D Institute of Metal and Composites for Future Industries
RIU	Remote Interface Unit
RPA	Remotely Piloted Aircraft
RPAS	Remotely Piloted Air Systems
RPAV	Remotely Piloted Air Vehicle
RTB	Rotor Track and Balance
RTS	Release to Service
RTSA	Release to Service Authority
RTSR	Release to Service Recommendation
Rx	Receiver

SAE	SAE International, formally the Society of Automotive Engineering
SC	Safety Case
SCP	Structural Control Point (SCP is not defined by MAA and guessed here)
SD	State Detection
SDH	Senior Duty Holder
SDI or SI	Special Detailed Inspection
SEAL	Safety Evidence Assurance Level
SEP	Structural Examination Programme
SHF	Super High Frequency
SHM	Structural Health Management
SI	Structural Integrity
SIP	Structural Inspection Procedure
SIWG	Structural Integrity Working Group
SL	Safe Life
SME	Subject Matter Expert
SMP	Safety Management Plan
SOI	Statement of Operating Intent
SOIU	Statement of Operating Intent and Usage
SPC	Sortie Profile Code
SRD	System Requirements Document
SSA	System Safety Assessment
S-SHM	Scheduled SHM
SSI	Structural Significant Item
SSIs	Structural Significant Items
STC	Supplemental Type Certificate
STANAG	Standardization Agreement
STDA	Statement of Type Design Assurance
STFFT	Short Term Fast Fourier Transforms
STR	Static Type Record
TAA	Type Airworthiness Authority
TC	Type Certificate
TCB	Type Certificate Basis
TCE	Type Certification Exposition
TCH	Type Certificate Holder
TLMP	Through Life Management Plan
TDM	Time Division Multiplexing
TRL	Technology Readiness Level
TRU	Transformer Rectifier Units
TSO	Technical Standard Order
TSOA	Technical Standard Order Authorization
TTW	Transition to War
Tx	Transmitter
UAV	Unmanned Air Vehicle
UCS	UAV Control Station
Uhf	Ultra High Frequency
UK	The United Kingdom
UOR	Urgent Operational Requirement
URD	User Requirements Document
USAF	US Air Force
V	Volt
V&V	Validation and Verification
VHM	Vehicle Health Management
VL	Virtual Link

DEFINITION

Most of the definitions of the following terms are followed by the references from which they are quoted. Those definitions without references are either: (a) definitions introduced for the purpose of this paper, or (b) definitions believed to be in agreement with the generally accepted understanding of the terms.

Term	Definition
Accuracy	The degree of closeness of agreement between a measured quantity value and a true quantity value of a measurand; the true quantity value is obtained by a device that has been widely accepted as being accurate with high degree of confidence.
Active Sensor	A sensor system that emits energy (excitation) and then measures changes caused by the measured subject as a result of the excitation. The main elements of the sensor system are (b) a basic actuator that delivers excitation/energy and (b) a basic sensor that observes changes caused by the measured subject.
Air Safety	The state of freedom from unacceptable risk of injury to persons, or damage, throughout the life cycle of military air systems. Its purview extends across all Defence Lines of Development and includes Airworthiness, Flight Safety, Policy, Regulation and the apportionment of Resources. It does not address survivability in a hostile environment. MOD MAA02, Reference [30].
Aircraft Document Set	The documents that have a prime airworthiness function for each aircraft type. They include the Release To Service (RTS), Aircraft Safety Case, Aircraft Maintenance Manual (AMM), Operating Data Manual (ODM), Flight Reference Cards (FRCs), Support Policy Statement, Engineering Air Publications (including the Flight Test Schedule (FTS)) and the Statement of Operating Intent and Usage (SOIU). The documents comprising the ADS may be held electronically. MOD MAA02, Reference [30].
Aircraft Electrical Wire	In both the Military and Industry, this term is interchangeable with EWIS. For ease, the definition is repeated here: The Electrical Wiring Interconnect System (EWIS) includes any wire, wiring device or combination of these including terminations installed in any area of the aircraft for the purpose of transmitting electrical energy or data between two or more termination points. MOD MAA02, Reference [30].
Airworthiness	The condition of an aircraft, aircraft system, or component in which it operates in a safe manner to accomplish its intended function. ARP4754A, Reference [26].
Airworthiness	The ability of an aircraft or other airborne equipment or system to be operated in flight or on the ground without significant hazard to aircrew, ground-crew, passengers or to third parties; it is a technical attribute of materiel throughout its lifecycle. MOD MAA02, Reference [30].
Airworthiness Limitation Item (ALI)	A mandatory-maintenance action identified in the Airworthiness Limitations section of a design-approval holder's Instructions for Continued Airworthiness. These items may contain mandatory modification or replacement times, mandatory inspection thresholds, intervals, and inspection procedures. AC 25-571-1, Reference [13].
Analysis	An evaluation based on decomposition into simple elements. ARP4754A, Reference [26].
Anti-Deterioration Maintenance	Preventive maintenance required to maintain the condition of aircraft or equipment being operated under adverse conditions, at below-average utilization rates, or which are in limited storage at operational units. MOD MAA02, Reference [30].
Approval	That which permits something to be done. Note: Approval may be granted to an individual or an organization verbally or in writing by an appropriately authorized person or authority. MOD MAA MAA02, Reference [30].
Approval	The act of formal sanction of an implementation by a certification authority. ARP4754A, Reference [26].
Assessment	The use of detection and/or monitoring results along with design information and structural properties to determine the current structural status and generate, if required, instructions including inspection, repair and replacement instructions.
Assurance	The planned and systematic actions necessary to provide adequate confidence and evidence that a product or process satisfies given requirements. RTCA DO-178B.
Authority	The organization or person responsible within the State (Country) concerned with the certification of compliance with applicable requirements. ARP4754A, Reference [26].
Automated SHM (A-SHM)	A task that can automatically inform maintenance personnel that action must take place instead of having a pre-determined interval at which the maintenance action must take place. Issue Paper No: 105 by Airbus, Boeing, Bombardier, Embraer, and Gulf-stream submitted as Joint Industry Proposal for MSG-3.

Term	Definition
Basic Actuator	A transducer that influences the environment and enables the delivery of a required action
Basic Sensor	A transducer that observes the environment and enables the detection of a physical condition.
Catastrophic Failure Conditions	Failure conditions that would result in multiple fatalities, usually with the loss of the airplane. (Note: Catastrophic failure conditions are also defined as a failure condition that would prevent the continued safe flight and landing of the airplane. AC 25-19, Reference [15].
Certification	The legal recognition that a product, service, organization or person complies with the applicable requirements. Such certification comprises the activity of technically checking the product, service, organization or person, and the formal recognition of compliance with the applicable requirements by issue of a certificate, license, approval or other document as required by national laws and procedures. ARP4754A, Reference [26].
Certification	Processes to obtain the approval of the appropriate Regulatory Authority (RA) that the applicable functional requirements, airworthiness regulations and operating rules are met.
Certification	A procedure by which a third party gives written assurance that a product, process or service conforms to a specified requirement (BS 3811). Note: Certification may be provided manually by means of applying a signature to an official document, or electronically. MOD MAA02, Reference [30].
Certification	A form of FAA approval where a certificate is issued, such as Type Certificate (TC), Supplemental Type Certificate (STC), Production Certificate (PC), or Airworthiness Certificate (AC). The FAA and Industry Guide, Reference [11].
Certification Authority	Organization or person responsible for granting approval in accordance with applicable regulations. ARP4754A, Reference [26].
Common Cause Analysis	Generic term encompassing zonal safety analysis, particular risk analysis, and common mode analysis. ARP4754A, Reference [26].
Component	Any self-contained part, combination of parts, subassemblies or units that perform a distinctive function. ARP4754A, Reference [26].
Condition-Based Maintenance (CBM)	Maintenance performed as governed by condition monitoring programmes. ISO 13372.
Condition-Based Maintenance (CBM)	The application and integration of appropriate processes, technologies, and knowledge-based capabilities to improve the reliability and maintenance effectiveness of DoD systems and components. At its core, CBM+ is maintenance performed based on evidence of need provided by Reliability Centered Maintenance (RCM) analysis and other enabling processes and technologies. CBM+ uses a systems engineering approach to collect data, enable analysis, and support the decision-making processes for system acquisition, sustainment, and operations. DOD, Reference [56].
Condition-Based Maintenance (CBM)	Preventive maintenance initiated as a result of knowledge of the condition of an item gained from routine or continuous monitoring. MOD MAA CAE 4000 MAP 01, Reference [33].
Configuration Control	The maintenance of effective control of the approved configuration of materiel (Def-Stan 05-57). MOD MAA02, Reference [30].
Corrective Maintenance	The maintenance carried out after fault recognition and intended to put an item into a state in which it can perform its required function (BS 3811). MOD MAA02, Reference [30].
Criticality	Indication of the hazard level associated with a function, hardware, software, etc., considering abnormal behaviour (of this function, hardware, software, etc.) alone, or in combination with external events. SAE ARP4754, Reference [25].
Damage tolerance	The attribute of the structure that permits it to retain its required residual strength for a period of use after the structure has sustained a given level of fatigue, corrosion, or accidental or discrete source damage. AC No: 25.571-1D, FAA, Reference [13].
Damage tolerance	A design philosophy which leads to a structure that can retain the required residual strength for a period of use after the structure has sustained specific levels of detectable fatigue damage, AD or ED. MOD MAA Guidance Material 5720(2), Reference [36].
Damage tolerance	The ability of the airframe to resist failure due to the presence of flaws, cracks, or other damage for a specified period of unrepaired usage. DOD JSSG-2006, Reference [47].
Depth support	In the context of 'Forward' and 'Depth' maintenance support, 'Depth support' is defined as those logistic processes and functions that underpin the support of platforms and associated equipment, or by their nature, are optimized best in Depth, and includes all logistic elements not in Forward support. MOD MAA02, Reference [30].
Derived Requirements	Additional requirements resulting from design or implementation decisions during the development process which are not directly traceable to higher-level requirements. ARP4754A, Reference [26].

Term	Definition
Detection	Finding with pre-defined quality the existence, type, location and/or extent of structural faults (FD, ED or AD) such as crack, delamination, corrosion, erosion and moisture absorption.
Development Assurance	All of those planned and systematic actions used to substantiate, at an adequate level of confidence, that development errors have been identified and corrected such that the system satisfies the applicable certification basis. SAE ARP4754, Reference [25].
Diagnostics	The examination of symptoms and syndromes to determine the nature of faults or failures (kind, situation, extent). ISO 13372.
Diagnostics	The determination of the nature of a diseased condition; the identification of a disease by investigation of its symptoms and history; diagnostic is the art of distinguishing diseases. The Oxford English Dictionary.
Defence Lines of Development (DLOD) (equipment)	The provision of military platforms, systems and weapons, (expendable and non-expendable, including updates to legacy systems) needed to outfit/equip an individual, group or organization. MOD AOF.
Dynamic Range	The difference between the range maximum and minimum values.
Fail-safe	The attribute of the structure that permits it to retain its required residual strength for a period of unrepaired use after the failure or partial failure of a principal structural element. FAA, AC No: 25.571-1D, Reference [13].
Failure Condition	A condition caused or contributed to by one or more failures or errors, that has either a direct or consequential effect on the airplane, its occupants, and/or other persons. In identifying failure conditions, the flight phase, relevant adverse operational or environmental conditions, and external events should be considered. AC 25-19, Reference [15].
Failure Modes Effects and Criticality Analysis (FMECA)	A qualitative method of reliability analysis that involves fault modes and effects analysis, together with a consideration of the probability of their occurrence and the ranking of the seriousness of the fault (BS 4778). MOD MAA02, Reference [30].
Forward Support	Forward Support provides world-wide support for tri-Service aviation customers in order to restore, maintain or enhance capability. FS comprises 42 (Expeditionary Support) Wing (42(ES) Wing) and Fleet Forward Support (Air) 1710 Naval Air Squadron. MOD MAA02, Reference [30].
Forward support	In the context of 'Forward' and Depth' maintenance, 'Forward support' is defined as those logistic processes and functions that are focused on, and/or provide immediate support to, the operating environment or are optimized effectively best forward. MOD MAA02, Reference [30].
Functional Hazard Assessment (FHA)	A systematic, comprehensive examination of aircraft functions to identify and classify Failure Conditions of those functions according to their severity. SAE ARP4754, Reference [25].
Generic Aircraft Release Process	A generic aircraft release process which uses the as flown standard as a basis for the safety Assessment, MOD MAA02, Reference [30].
Hazardous Failure Conditions	Failure conditions that would reduce the capability of the airplane or the ability of the crew to cope with adverse operating conditions to the extent that there would be: (a) A large reduction in safety margins or functional capabilities, (b) Physical distress or higher workload such that the flight crew cannot be relied upon to perform their tasks accurately or completely, or (c) Serious or fatal injury to a relatively small number of the occupants. AC 25-19, Reference [15].
Independence	1. A concept that minimizes the likelihood of common mode errors and cascade failures between aircraft/system functions or items; 2. Separation of responsibilities that assures the accomplishment of objective evaluation e.g. validation activities not performed solely by the developer of the requirement of a system or item. ARP4754A, Reference [26].
Instructions for Continued Airworthiness (ICA)	Documentation that sets forth instructions and requirements for the maintenance that is essential to the continued airworthiness of an aircraft, engine, or propeller. AC 25-571-1, Reference [13].
Integrity	Qualitative or quantitative attribute of a system or an item indicating that it can be relied upon to work correctly. ARP4754A, Reference [26].
Item	A hardware or software element having bounded and well-defined interfaces. SAE ARP4754A, Reference [26].
Limit of validity	(of the engineering data that supports the structural maintenance program): the period of time (in flight cycles, flight hours, or both), up to which it has been demonstrated by test evidence, analysis and, if available, service experience and teardown inspection results of high-time airplanes, that widespread fatigue damage will not occur in the airplane structure. AC 25-571-1, Reference [13].

Term	Definition
Major Failure Conditions:	Failure conditions that would reduce the capability of the airplane or the ability of the crew to cope with adverse operating conditions to the extent that there would be, for example: (a) A significant reduction in safety margins or functional capabilities, (b) A significant increase in crew workload or in conditions impairing crew efficiency, or (c) Discomfort to occupants, possibly including injures. AC 25-19, Reference [15].
Management	The use of detection, monitoring and assessment results combined with information about available resources to plan fleet utilization or plan maintenance activities.
Minor Failure Conditions	Failure conditions that would not significantly reduce airplane safety, and which involve crew actions that are well within their capabilities. AC 25-19, Reference [15].
Monitoring	Maintaining regular surveillance over factors that can lead to or indicate structural faults; these factors include, for example, loads, usage, impact events, fatigue and/or environments.
Passive Sensor	A basic sensor or a sensor system that can directly respond to a measured subject or the energy of an active subject.
Precision	The degree to which repeated measurements under specified conditions show the same results.
Preliminary System Safety Assessment	A systematic evaluation of a proposed system architecture and its implementation, based on the Functional Hazard Assessment and Failure Condition classification, to determine safety requirements for systems and items. ARP4754A, Reference [26].
Preliminary System Safety Assessment	A systematic evaluation of a proposed system architecture and its implementation, based on the Functional Hazard Assessment and failure condition classification, to determine safety requirements for all items in the architecture. SAE ARP4754, Reference [25].
Preventive Maintenance	Maintenance carried out at predetermined intervals or according to prescribed criteria and intended to reduce the probability of failure or the degradation of the functioning of an item (BS 4778). MOD MAA02, Reference [30].
Primary Structure	The structure that significantly contributes to the carrying of flight, ground, or pressure loads. It is also known as a Structurally Significant Item (SSI). AC-1529-1A, Reference [16].
Principal Structural Element (PSE)	An element that contributes significantly to the carrying of flight, ground, or pressurization loads, and whose integrity is essential in maintaining the overall structural integrity of the airplane. Principal structural elements include all structure susceptible to fatigue cracking, which could contribute to a catastrophic failure. AC 25-571-1, Reference [13].
Process	A set of interrelated activities performed to produce a prescribed output or product. DO-254/ED-80.
Product	Hardware, software, item or system generated in response to a defined set of requirements. ARP4754, Reference [25].
Product	Product" is used to identify aircraft, aircraft engines, propellers, as well as appliances and components or parts throughout the FAA & Industry Guide, Reference [11].
Product Certification	The complete certification cycle that includes type certification (design approval), production certification (production approval), airworthiness certification (airworthiness approval) and continued airworthiness management. The FAA and Industry Guide, Reference [11].
Prognostics	The analysis of the symptoms of faults to predict future condition and remaining useful life. ISO 13372.
Prognostic	A forecast of the probable course of a case of disease; prognostic is defined as a prediction; a forecast; an advance indication or omen, especially of the course of disease. The Oxford English Dictionary.
Prognostics	A technique which allows data to be collected and analysed on the operational status of an entity so that predictions can be made as to when failures are likely to occur. Prognostics can be considered as a subset of testability, but the storage of data and the instantaneous analysis of data can be highly complex, so it is usually only applied to critical performance attributes. MOD, Def Stan 00-42 Part 4.
Prognostic	A forecast of the health state of a subject over a future period given the expected operational conditions of the subject over the future period and based on a progression relationship established between the health state and the subject operational conditions.
Qualification	A verification process to verify through tests that a product complies with a specified set of requirements including airworthiness requirements.
Range	The maximum and minimum values of an applied parameter that can be measured.
Redundancy	Multiple independent means incorporated to accomplish a given function. ARP4754A, Reference [26].
Regulated Structural Rules	Rules or requirements recommended by regulator to guide the development of maintenance and management programmes as means of compliance with applicable airworthiness regulations.

Term	Definition
Regulatory Authority (RA)	An organization, institution or governmental agency authorized to perform tasks including but not limited to: (a) publishing, maintaining and updating safety regulations and operating rules (b) monitoring the correct implementation of these regulations and rules, and (c) approving the design, production, installation and use of products that comply with these regulations and rules. Examples of such regulators for civil applications are: the Federal Aviation Administration (FAA) in the United States (US) and the European Aviation Safety Agency (EASA) in the European Union (EU). Examples of such regulators for military applications are: the Ministry of Defence (MOD) in the United Kingdom (UK) and the Department of Department of Defense (DOD) in the US.
Release to Service	The release document that authorizes Service flying on behalf of the Service Chief of Staff. The RTS refers to the Safety Assessment documentation for the aircraft or equipment, including the limitations and aircraft description, and defines the as-flown standard of the aircraft. For legacy aircraft that have yet to move to the Generic Aircraft Release Process (GARP), it also contains Service Deviations (SD) for the aircraft. The limitations of the RTS are the definitive limits for the aircraft in Service-regulated flying. Release to Service Authority (RTSA) is the authority that issues the RTS. MOD MAA02, Reference [30].
Release to Service Authority	The competent authority issuing the Release to Service for an aircraft type. MOD MAA02, Reference [30].
Reliability	The probability of repeatedly and successfully observing a desirable outcome from an entity under prescribed conditions. The entity can be any observable subject such as structure, system, sensor, mission, or event. In other words, the reliability is measured by the probability of the success, failure-free, desirable performance of the entity. A common example of a desirable outcome as sited in system engineering literature is the ability of a system or component to perform a required function under stated conditions for a specified period.
Reliability	The probability that an item will perform a required function under specified conditions, without failure, for a specified period of time. SAE ARP4754, Reference [25].
Reliability Centred Maintenance	The systematic approach for identifying preventive maintenance tasks for an equipment or item in accordance with a specified set of procedures and for establishing intervals between maintenance tasks [with minimum expenditure of resources] (Def Stan 00- 40). Note: Outside the MAE, RCM is commonly known as Maintenance Steering Group (MSG) logic. MOD MAA02, Reference [30].
Residual Strength.	The strength capability of a structure after fatigue, corrosion, or a discrete source has damaged the structure. The residual strength capability includes consideration of static strength, fracture, and stiffness. AC No: 91-82, FAA, Reference [14].
Resolution	The smallest change in a quantity being measured that causes a perceptible change in the corresponding indication.
Risk	The word “risk” encompasses both the probability of undesirable event (failure) and the consequence of the event if it happens.
Risk	Is a measure of exposure to possible loss and it combines the severity of loss (how bad) and the likelihood of suffering that loss (how often). MOD MAA02, Reference [30].
Safety Assessment	A systematic, comprehensive evaluation to identify all safety features of the system design, including hardware and software, and to identify all hazards and safety factors cross-DLOD that may be present in, or required for, the system being acquired, and then operated, including specific procedural controls and precautions that are to be followed. The Safety Assessment (SA) contains the structured argument that the system is safe for its intended use and that a specific DLOD has been considered in the context of the overarching Air System Safety Case. MOD MAA02, Reference [30].
Safe Life	For an aircraft designed to be retired before fatigue cracking occurs, this is the period of operation during which the risk of cracking occurring is acceptably low. MOD MAA02, Reference [30].
Safe-life	The number of events, such as flight cycles, landings, or flight hours, within which the structure strength has a low probability of degrading below its design ultimate value due to fatigue cracking. AC 25.571-1D, FAA, [13].
Safety Case	A structured argument, supported by a body of evidence that provides a compelling, comprehensible and valid case that a system is safe for a given application in a given operating environment. MOD MAA02, Reference [30].
Safety Evidence Assurance Level (SEAL)	A category of required evidence needed to assure that a given system is sufficiently safe (i.e. it has achieved its required safety integrity level). SEAL is the US government equivalent to DAL.
Sensitivity	The quotient of change in the indication of a measuring system and the corresponding change in the value of a quantity being measured.

Term	Definition
Scheduled SHM (S-SHM)	The act to use/run/read out a SHM device at an interval set at a fixed schedule. ATA, MSG-3, Reference [24].
Structural Integrity	The ability of an aircraft structure to retain its strength, function and shape within acceptable limits, without failure when subjected to the loads imposed throughout the aircraft's service life by operation within the limitations of Release To Service (RTS) and to the usage described in the Statement of Operating Intent (SOI) or the Statement of Operating Intent and Usage (SOIU). MOD MAA02, Reference [30].
Structural Significant Item	Any detail, element or assembly, which contributes significantly to carrying flight, ground, pressure or control loads and whose failure could affect the Structural Integrity necessary for the continued safe and controlled flight of the aircraft. MOD MAA02, Reference [30].
Structural Health Monitoring (SHM)	The process of acquiring and analyzing data from on-board sensors to evaluate the state of a structure. SAE ARP6461.
Safety	The freedom from unacceptable risks of personal harm. MOD MAA02, Reference [30].
Structural Health Monitoring (SHM)	The concept of checking or watching a specific structural item, detail, installation or assembly using on board mechanical, optical or electronic devices specifically designed for the application used. SHM does not name any specific method or technology. ATA MSG-3, Reference [24].
System Safety Assessment	A systematic, comprehensive evaluation of the implemented system to show that the relevant safety requirements are met. (ARP4754A), Reference [26].
Type Certification Basis	The list of design Standards and other requirements and Special Conditions against which the design will be certified. MOD MAA02, Reference [30].
Target Reliability	The reliability associated with a low probability (risk) of failure condition that would have adverse consequence on safety, airworthiness, economy, environment or performance through the equation "Reliability = 1 - Probability of such a Failure".
Transducer	A device that converts one form of energy to another; examples of the energy forms are: kinetic, electrical, mechanical, magnetic, chemical, acoustic, thermal and light energy
Uncommanded Flying Control Movement	Any unexplained change of aircraft in-flight attitude without a legitimate flying control input, or any movement of flying control input controls when there should be none, or any movement of flying control surfaces or systems without a corresponding legitimate input. MOD MAA02, Reference [30].
Validation	Confirmation by examination and provision of objective evidence that the particular requirements for a specific intended use are fulfilled. The UK MOD Def Stan 00-970 Part0, Reference [38].
Validation	Validation is the quality assurance process by which the DO for the materiel concerned confirms and certifies that all the information contained within a Technical Information and Data (TID) suite is accurate, safe in application and suitable for its intended purpose as defined in the contract. MOD MAA02, Reference [30].
Validation	The determination that the requirements for a product are sufficiently correct and complete. [Are we building the right aircraft/ system/ function/ item?]. SAE ARP4754, Reference [26].
Verification	Confirmation by examination and provision of objective evidence that the specified requirements have been fulfilled. The UK MOD Def Stan 00-970 Part0, Reference [38].
Verification	Verification is the process by which the Service user is satisfied that the validated TID meets the Service requirement (e.g. the maintenance policy) and can be used for its intended purpose by Service personnel under normal Service conditions. MOD MAA02, Reference [30].
Verification	The evaluation of an implementation of requirements to determine that they have been met. [Did we build the aircraft/ system/ function/ item right?] (ARP4754A), Reference [26].

1 INTRODUCTION

1.1 Background

The functional duty of defence industry poses a requirement for continuous enhancement of defence products in terms of safety, performance, and Cost of Ownership (COO). These products are designed to safely deliver superior performances; they are maintained to preserve reliabilities and deliver the desirable level of availability at affordable costs. Existing design, maintenance, and management processes have already been implemented to maintain acceptable levels of safety, performance, and reliability throughout the lifetimes of the products. However, emerging military requirements place on the defence industry requirements for further product improvements and for new products capable of effectively defending peace and overcoming the emerging forms of threats to national security. The defence industry is addressing these requirements by introducing new materials, implementing better manufacturing techniques, and exploring advanced technologies. The affordability and safety requirements do not only require advanced designs but also require a revolutionary effective approach to through-life maintenance practices. Integrated Vehicle Health Management (IVHM) of which Structural Health Monitoring (SHM) constitutes a significant component is a key enabler of affordable aircraft capability and improved maintenance support for existing and future weapon systems.

The United Kingdom (UK) Ministry of Defence (MOD) and the UK Industry have pioneered the development of SHM systems that monitored and managed usage damage: the pioneering effort was witnessed by the introduction of the Tornado Operational Load Monitoring (OLM) capability in the 1980s; another recent example is the development of the Typhoon SHM system. Currently, the UK is leading the development of an SHM system for the three variants of the Joint Strike Fighter (JSF): along with comprehensive usage monitoring, the JSF SHM system will provide advanced OLM using parametric models driven by recorded aircraft parameters; the JSF SHM system will also monitor environmental damage using advanced corrosion sensors.

In order to further advance the IVHM/SHM technologies and address their challenges, the UK Government and Defence Industry continue to sponsor SHM related projects at universities and Small Medium Enterprise (SME) companies. For example, the UK Industry has led, and is one of the key partners in, the Autonomous Systems Technology Related Airborne Evaluation & Assessment (ASTRAEA) Programme. ASTRAEA involves some of the largest defence companies in Europe, research associations, dedicated regional bodies, and the UK Government: the aim is to develop existing technologies, regulations, systems, and procedures to bring routine, non-segregate operations of Uninhabited Air Vehicles (UAVs) to UK airspace. The programme key areas cover: ground operations, communications, handling, adaptive routing, collision avoidance, multiple air vehicle integration, prognostic and health management, decision modelling, good airmanship, a route to compliance, operating rules and procedures, integration with operating environment, propulsion, and affordability. The technologies developed under the ASTRAEA programme have been demonstrated using the BAE Systems' Jetstream, which has been developed as 'flying test bed' to be flown by pilots or as a UAV controlled by: ground-based pilot, airborne computers, and satellite communications. The demonstrated technologies include autonomous weather avoidance system, 'sense and avoid' technologies, and an autonomous emergency landing system. The Jetstream successfully completed a 500-mile trip in shared UK airspace under the command of a ground-based pilot and control of air traffic controllers. The next phase of the UK ASTRAEA programme will be targeted at consolidating regulatory work and addressing certification issues with the UK Civil Aviation Authority (CAA).

At present, BAE Systems and HAHN Spring Limited are executive members of an international "Aerospace Industry Steering Committee" (AISC) working on various SHM aspects within SAE International; AISC assembles powerful representations from key aerospace industries including: Boeing, Airbus, Airbus Defence and Space, Bombardier, Embraer, BAE Systems, GE, Honeywell, UTC Aerospace Systems, the National Aeronautics and Space Administration (NASA), Sandia National Laboratories, Messier-Bugatti-Dowty, the US Air Force, the US Navy, the Federal Aviation Administration (FAA), the European Aviation Safety Agency (EASA), National Research Council Canada (NRC), HAHN Spring Limited, Stanford University, Cranfield University, Japan's R & D Institute of Metal and Composites for Future Industries (RIMCOF), University of Tokyo and Delft University of Technology (TU Delft).

The efforts of AISC since January 2009 have been concluded by developing guidelines on validation, verification, and certification of SHM for civil transport aircraft; in September 2013, these SHM guidelines were published by SAE as an Aerospace Recommended Practice (ARP) document numbered ARP6461. Motivated by considerations specific to military applications, which are briefly discussed in Section 1.2, the primary focus of BAE Systems and HAHN Spring Limited since July 2012 has been military applications.

This paper presents a UK perspective on SHM guidelines for military aircraft working in compliance with UK military regulations.

1.2 Motivation for Military SHM Guidelines

The main motivation is to provide guidelines covering the following aspects which were beyond the scope of the civil SHM guidelines ARP6461:

1.2.1 A Wider Range of Military Aircraft Types

The military forces operate a wide range of aircraft. The operations, missions, and sizes of aircraft vary between Remotely Piloted Aircraft (RPA), autonomous Unmanned Air Systems (UAS), fast jets and transport airplanes. SHM requirements should be tailored for each aircraft in sympathy with variations in size, operation, and mission. For example, because of size and weight constraints, and because adding new sensors and systems to fighter airplanes could be more challenging than adding them to large transport airplanes, requirements for sensor weight reductions would be needed. Therefore, the guidelines should consider SHM requirements for three military aircraft types: aircraft performing autonomous operations, remotely piloted aircraft and, manned aircraft.

1.2.2 Varying Requirements Across Nations and Operators

The military regulations, standards, and processes required for designing and managing aircraft structures can vary between nations and between the military operators of one nation. For example, a military operator may require the aircraft structures to be designed and managed using a safe life approach, a damage tolerance approach, or a combination of the two approaches. The structural design should adhere to the requirement of the military operator with careful considerations of differences in structural loads between military and civil aircraft. Since major aircraft manufacturers supply their aircraft products not only to one nation, generic guidelines, which can meet varying requirements across nations and operators, would lead to reductions in the costs of aircraft products. Therefore, the military guidelines should scrutinize the requirements of a number of key military operators and extract from them a set of SHM guidelines applicable across nations and operators.

1.2.3 A Need for Concise Information for SHM Military Stakeholders

The evolution of SHM involves a number of stakeholders and a wide range of disciplines such as: fundamental research, structural & system engineering, software, avionics, inspection, sensing, testing, manufacturing, and certification. The experience of each stakeholder can't cover all disciplines. However, the awareness of each stakeholder of these disciplines through clear concise information would help accelerating the evolution of SHM. With common concise information, the efforts of a stakeholder would adequately consider the requirements associated with other disciplines and stakeholders. Furthermore, the major aircraft manufacturers integrate into Major Military Products (MMP) advanced technologies developed by various stakeholders. Concise information provided to each stakeholder is needed and would lead to:

- awareness of military trends and associated potential MMP requirements and their planned timescales,
- technology maturation efforts from the stakeholder meeting the MMP requirements within timescales, and
- awareness of the experiences of other stakeholders to facilitate integration.

Therefore, the SHM guidelines should provide concise information about relevant military requirements, standards and processes along with concise information about potential MMP requirements and stakeholders' disciplines.

1.2.4 A Need for Guidance on Integrating SHM within Aircraft Support Systems

The guidelines should address the regulations, standards, and processes required for integrating SHM into the various types of military aircraft and their support systems. For new aircraft designs, SHM may be considered at early design stages, and hence, SHM may influence the aircraft design and their support systems. The support systems to be considered are:

- Maintenance/management support systems that maintain airworthiness by exploiting advanced damage and deterioration detection systems, and by optimally planning maintenance actions and fleet utilization.
- Flying support systems: a Flight Management System (FMS) mainly influences how the aircraft can fly a pre-planned route; a Mission Management System (MMS) manages a large number of tactical sensors and interfaces with systems such as FMS to, for example, optimally deliver weapons, provide situational awareness, and plan/perform missions.

1.2.5 Specific Military Considerations

The military guidelines should scrutinize the existing practices that address special military considerations and collate, extend, and present concise information about them. Examples of key considerations that should be addressed are:

- The Configurations of Military Aircraft: The same structural design can undergo a number of configurations. For example: a number of aircraft having the same structural design may be equipped with stores for ferry missions; a number of aircraft may be fitted with weapons for air-to-air combat missions; a number of aircraft may be fitted with weapons and releasable stores for ground attack missions; configuration modifications may be introduced to the same airframe in response to operational requirements. Therefore, the military guidelines should show how the structural integrity management approach, which would include SHM, could carefully consider effects of varying configurations across airframes or across the same airframe during its lifetime.
- Military Operational Conditions: Various airframes having the same structural design can be subjected to a wide range of operational conditions; the aircraft can operate at remote locations with varying degrees of operational conditions and maintenance support; it can be

chosen for carrier operations; it can operate in harsh erosive or corrosive environments; it can perform different missions; it can be deployed to locations having low freezing temperatures or locations having extremely high temperatures.

- Security and Interoperability Considerations: For SHM systems that exchange data with other military assets or share resources with secure aircraft systems (e.g. mission systems), security and interoperability considerations should be carefully addressed and should cover all data transfer and communication means. Military systems should identify, transfer and exchange data in a reliable secure way. The military guidelines should show how SHM would identify each individual aircraft, structural component conditions, and operational environments without broadcasting the extent of the military capability and posing military risks on the nation. Considerations of reliable and secure data transfer methods between changing remote locations should also be addressed.
- Other Special Military Considerations: In emergency, especially for deployed aircraft, ad-hoc repairs or movements of components between aircraft may be essential for successful military operations and may introduce structural modifications. Other special considerations include structures with low observable coatings, wireless communications in hostile terrains, etc. The military guidelines should scrutinize the existing military practices that address such special considerations and collate, extend, and present concise information about them.

To address these considerations, the major aircraft manufacturers would evolve products that can meet the associated demanding military requirements at reduced costs whilst providing superior military capabilities**.** Such products would be realized by enhancing existing design, management, and operational concepts, or by adopting new concepts. Therefore the military guidelines should provide information about these advanced concepts and the processes required for the transition to them from existing concepts.

1.3 The Rationale, Aim and Approach of This Paper

Rationale: A wide range of engineering disciplines are required to evolve SHM technologies, and to integrate these technologies into aircraft systems, and into maintenance and operational support systems for military aircraft. These disciplines are required from a variety of stakeholders with varying experiences: military operators, aircraft manufacturers, system suppliers, regulatory agencies, academia, etc. Therefore, clear concise information and guidelines are needed to make each stakeholder aware of the disciplines and requirements of the other stakeholders. The concise information should present, in a clear common language, the guidelines, recommended practices, requirements, processes, standards and regulations required for the evolution, integration, and approval of SHM.

Aim: The aim of this paper is to collate and develop information and guidelines that can assist UK military stakeholders in the development, validation, verification, and certification of SHM technologies, and the integration of these technologies into encompassing maintenance and operational support systems.

Approach: The approach adopted in developing the guidelines was to maintain harmony with existing applicable regulations, standards and guidelines, and to augment them with specific interpretations or extensions pertaining to SHM as necessary. Where existing key definitions and guidelines were available from relevant publications, they have been adopted without modifications; interpretations specific to SHM have been added for clarity. Additional definitions and information have been collated and developed only where their equivalent did not clearly exist. In this way, consistency is maintained with existing regulations, standards, and industry accepted practices. Therefore, this document does not replace any existing standards or regulations. Applicable documents from which the SHM key guidelines and definitions were collated are listed in Section 1.6.

1.4 SHM Definition and Introductory Remarks

For the purpose of this document, SHM is defined as "the process of acquiring and analyzing data from onboard sensors to evaluate the state of a structure", ARP6461.

- Unlike NDI equipment, the main inputs required to perform SHM functions are data acquired from onboard sensors installed in each aircraft specifically for SHM or acquired by other aircraft systems, e.g. flight parameters acquired by Flight Data Recorder (FDR).
- The words "evaluate the state of a structure" are used to indicate that SHM would perform structural evaluation functions, which should improve structural integrity tasks.

SHM can be a component of an encompassing IVHM system and a structural health management system; the latter would use information obtained from SHM, crew, operators, maintainers, and other ground-based systems or equipment to optimally maintain the structural integrity and readiness of the aircraft, Figure 1.

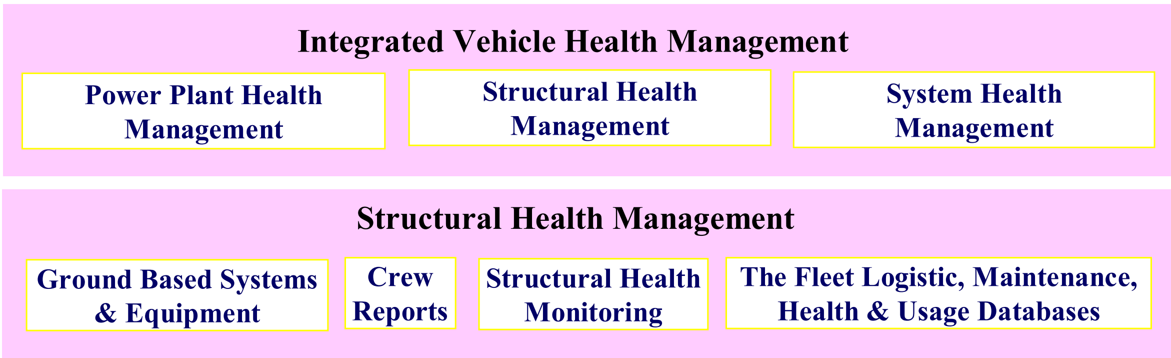


Figure 1: Structural Health Management

1.5 The Paper Applicability to Fixed-wing Aircraft and Rotorcraft

The paper is applicable to SHM systems for fixed-wing aircraft and rotorcraft. Although the paper might appear to be focused on fixed-wing applications, the SHM validation, verification, and certification guidelines presented in the paper are equally applicable to rotorcraft structures.

In applying these guidelines to rotorcraft, considerations should be made to the specific structures, operations, and environments of the rotorcraft along with the vibratory loads induced on the structures by rotors and transmissions.

Furthermore considerations should be made to the fact that the majority of the UK military helicopters have been fitted with Health and Usage Monitoring Systems (HUMS). Such systems have been already validated, qualified, and approved. Generally HUMS offer some or all of the following capabilities: Rotor Track and Balance (RTB), drive-train diagnostics, exceedance monitoring, engine power assurance, and usage monitoring. The capabilities of HUMS are based on vibration sensing, oil debris analysis, and aircraft data recording. The HUMS information have been approved and used as advisory information that has significantly enhanced safety. Some of the HUMS capabilities have been approved and used to trigger maintenance tasks. Also, approved HUMS data can be used to support structural integrity management tasks, see Section A.6.3. So, HUMS recorded flight data can by used to provide SHM functionality. If this functionality is a new extension to existing HUMS capabilities, it should be validated, verified, and approved following the guidelines of this paper. Also, the guidelines should be applied to SHM systems designed to share resources with other rotorcraft systems or designed as standalone systems.

1.6 Relevant Publications

The ultimate aim of the paper is to provide guidance for SHM compliance with applicable airworthiness regulations, aircraft standards, and operating rules such as those relevant parts of the following publications:

1.6.1 Publications form Civil Aircraft Regulatory Authorities

- The electronic Code of Federal Regulations (e-CFR) Title 14, Parts 21, 23, 25, 26, 27, 29, 39, 43, 91 and 121, References [1] to [10].
- The FAA and Industry Guide to Product Certification, Reference [11]
- Maintenance Review Board Report, Maintenance Type Board, and OEM/TCH Inspection Program Procedures, FAA AC 121-22B, Reference [12].
- Damage Tolerance and Fatigue Evaluation of Structure, FAA AC 25.571-1-D, Reference [13].
- Fatigue Management Programs for In-Service Issues, FAA AC No: 91-82A, Reference [14].
- Certification Maintenance Requirements, FAA AC No: 25-19, Reference [15].
- Instruction for Continued Airworthiness of Structural Repairs on Transport Airplanes, FAA AC No: 25.1529-1A, Reference [16].
- The equivalent parts of the European Aviation Safety Agency (EASA) regulations; e.g., References [17] to [21].
- Canadian Aviation Regulations (CARs) 2012-1, Part I - General Provisions, Subpart 1 – Interpretation (CARs 101.01), Reference [22].
- Canadian Aviation Regulations (CARs) 2012-1, Part V - Airworthiness, Subpart 21 - Approval of the Type Design or a Change to the Type Design of an Aeronautical Product, Division IV - Changes to a Type Design (CARs 521.151 to 521.161), Reference [23].

1.6.2 Internationally Recognized Standards, Processes, and Best Practices

- The Maintenance Steering Group 3 (MSG-3), Operator/Manufacturer Scheduled Maintenance Development, Reference [24].

This paper recommends, wherever applicable, adopting internationally recognized standards including the following SAE and EUROCAE/RTCA publications:

- SAE ARP4754 "Certification Considerations for Highly-Integrated or Complex Aircraft Systems" [25], which is developed in the context of EASA/FAA regulations and EUROCAE/RTCA standards.
- SAE ARP4754 Rev.A "Guidelines for Development of Civil Aircraft and Systems", Reference [26].
- SAE ARP4761 "Guidelines and Methods for Conducting the Safety Assessment Process on Civil Airborne Systems and Equipment", Reference [27].
- SAE ARP5150 "Safety Assessment of Transport Airplanes in Commercial Service", Reference [28].
- RTCA/DO-160D, Environmental Conditions and Test Procedures for Airborne Equipment.
- RTCA/DO-178 "Software Considerations in Airborne Systems and Equipment

Certification" for the development of software".

- DO-200A "Standards for Processing Aeronautical Data".
- DO-201A "Industry Requirements for Aeronautical Information".
- RTCA/DO-254 "Design Assurance Guidance for Airborne Electronic Hardware Considerations in Airborne Systems and Equipment Certification" for the development of hardware.

1.6.3 *The UK MOD Publications*

The main aim is to provide guidance for SHM compliance with UK military regulations and standards such as those applicable parts of the following publications:

- The UK Military Aviation Authority (MAA), References [29] to [37].
- The UK Ministry of Defence (MOD) Defence Standards (Def Stan), References [39] to [46].
- The UK MOD Joint Services Publication, JSP 815, Defence Environment and Safety Management, Reference [47].
- The UK MOD Joint Services Publication, JSP 886, Volume 7, Part 2, Defence Logistics Support Chain Manual, Supportability Engineering, Integrated Logistic Support Management, Reference [48].
- The UK MOD Joint Services Publication, JSP 886, Volume 7, Part 8.03A, Defence Logistics Support Chain Manual, Supportability Engineering, Maintenance Planning, Reference [49].
- The UK MAA Joint Air Publication JAP(D) 100C-20, Military Aviation Engineering, Preparation and Amendment of Maintenance Schedules, Reference [50]
- The UK MAA Joint Air Publication JAP(D) 100C-22, Military Aviation Engineering, Guide to Developing and Sustaining Preventive Maintenance Programmes, Reference [51]
- The UK MOD Acquisition Operating Framework (AOF), Acquisition business, Capability Management, MOD websites.

The UK defence standards explicitly quote international standards such as DO-178. The MAA publications explicitly quote standards such SAE ARP4761. Furthermore, MAA guidance materials indicate that aircraft accepted into UK military service may be designed to satisfy one of a number of different design requirements or standards such as Def Stan 00-970 and international military or civil standards. MAA may accept, in part or in full, certificates issued by civilian or foreign military authorities as evidence of compliance; EASA and the FAA are automatically recognised by MAA. The publications of EASA and FAA include the above internationally recognized standards, processes, and best practices. The validation and verification approach presented in this paper builds on these internationally recognised publications. Therefore, the approach developed in this paper would be applicable to UK military aircraft.

1.6.4 *The DOD Publications*

- The DOD Joint Service Specification Guide, Aircraft Structure, JSSG-2006, Reference [52].
- The US Air Force (USAF) Aircraft Structural Integrity Program (ASIP), Reference [53].
- Environmental Engineering Considerations and Laboratory Tests, MIL-STD-810,

Reference [54].

- Requirements for the Control of Electromagnetic Interference Characteristics of Subsystems and Equipment, MIL-STD-461 [55].
- Condition Based Maintenance Plus DOD Guidebook, Reference [56].
- Nondestructive Evaluation System Reliability Assessment, MIL-HDBK-1823A, Reference [57].
- The US Naval Air (NAVAIR) Systems Command, Guidelines for the Naval Aviation Reliability-Centered Maintenance Process, NAVAIR 00-25-403, Reference [58].
- The US Naval Sea (NAVSEA) Systems Command, S9081-AB-GIB-010, Revision 1, Reliability-Centered Maintenance (RCM) Handbook, Reference [59].

1.6.5 The North Atlantic Treaty Organization (NATO) Publications

The NATO Standardization Agency publishes, maintains and regularly updates the NATO regulations within a series of Standardization Agreement (STANAG) documents.

- The NATO STANAG 4671, Unmanned Aerial Vehicles System Airworthiness Requirements (USAR), Reference [60].

1.7 An Overview of the Remainder of This Paper

1.7.1 Designing and Maintaining Airworthy Structures, Section 2

Section 2 reviews the main regulations that govern the design and maintenance of airworthy structures. The section summarises the main regulations and processes required to overcome the threats to structural integrity, which can arise from the damaging effects of fatigue, environments, and accidents. SHM should provide improved means for maintaining airworthy structures and for assisting in overcoming the threats to structural integrity without violating relevant airworthiness regulations. Section 2 gives details about these regulations and associated processes for civil aircraft applications; for military aircraft applications, the section focuses more on the UK MOD regulations and standards. Section 2 also summarises specifications for the use of SHM systems in aircraft maintenance, which have been recommended and proposed by the Air Transport Association (ATA) MSG-3 in recent and pending document revisions (Revision 2009.1, Reference [24], and Issue Paper 105).

1.7.2 The Evolution and Acquisition of Aircraft Products, Section 3

Having highlighted key regulations and standards for civil and military aircraft applications, and having highlighted the ATA MSG-3 recommended specifications, Section 3 briefly discusses the evolution and acquisition phases of aircraft products including SHM. For aircraft products, the evolution phases span maturation, development, production, installation/integration, certification, utilization, and maintenance. The main differences between the civil and UK military evolution phases are the approval/certification processes and their outputs. The evolution phases start with identifying intended functions that address emerging demands. The SHM technology maturation efforts are motivated by demands for improving maintenance practices, underwriting new aircraft materials/designs, and improving aircraft operational practices. When the SHM technologies required to deliver the intended functions are sufficiently matured, the development process starts. The section presents brief descriptions of the essential processes required to develop an approved SHM for civil and military aircraft. Section 3 also discusses the six acquisition phases carried out by the UK MOD to acquire aircraft products.

1.7.3 The SHM Intended Functions, Section 4

The demands for potential intended functions to address emerging needs trigger the evolution phases. Section 4 presents detailed discussions about the SHM intended functions.

1.7.4 SHM System Architectures, Section 5

Section 5 emphasizes that any given SHM intended function can be realized by a variety of different system architectures. The architecture is constructed from hardware and software entities: the SHM data entities are acquired from airborne sensors; other SHM entities can be located onboard the aircraft or within ground-based systems. Depending on the chosen architecture, some or all of these entities may be designed specifically for SHM or can be shared entities from other pre-existing systems. An optimum architecture can be chosen by assessing factors such as the technology readiness levels of entities, the associated safety requirements, development timescale, life cycle costs, and weight. Section 5 also describes high level architectural choices for three military aircraft types: aircraft performing autonomous operations, remotely controlled aircraft and manned aircraft.

1.7.5 SHM System Requirements, Section 6

Section 6 focuses on allocation of SHM requirements. In this section, it is noted that the development of the system architecture and the allocation of requirements are tightly-coupled, iterative processes. The requirements of two SHM architectures that provide the same intended function may be different because of differences between the items of the two architectures: items shared with existing aircraft systems versus items designed specifically for SHM, ground-based items versus airborne items, similar airborne items installed into different aircraft location and exposed to different environmental conditions. Section 6 presents more than 30 SHM requirements along with guidance notes supported by relevant regulations and standards. The section mainly covers: intended function/performance requirements, survivability/environmental requirements, operational requirements, and interface and installation requirements.

1.7.6 Validation & Verification, Section 7

Section 7 presents the typical methods required to validate and verify aircraft products, and introduces interpretations and extensions specific to SHM systems. The validation methods ensure the correctness & completeness of requirements. The verification methods ensure correct implementation of validated requirements. Section 7 recognizes that the rigour, and hence, the assurance level required for validation and verification, depends on the chosen architecture and the consequence of the failure conditions of each SHM item on safe operations. The more severe the consequence, the more rigour required. For SHM systems that have negligible effects on aircraft structures and systems, the validation and verification methods presented in Section 7 may not be required to support certification; such systems may be approved by a manufacturer qualified representative. However, SHM systems that have major effects can be approved only by the appropriate regulator after witnessing development, validation, and verification activities conducted with rigour proportionate to the consequence of SHM failure conditions.

1.7.7 Certification, Section 8

For the purpose of this document, Section 8 considers certification as the processes required for obtaining approval from the appropriate Regulatory Authority (FAA, EASA, MOD, DOD, etc.) that the applicable airworthiness regulations, operating rules, and system requirements are met. Formal definitions of certification taken from ARP4761 and MOD publications are presented in the definition section. The certification efforts involve the regulator, the product developer, and the aircraft manufacture along with any other stakeholders.

For civil aircraft, the certification efforts are often initiated through an application made by the product developer to the appropriate regulator, typically before the start of the development phase; these efforts can be performed in parallel to the other evolution phases and can overlap with the maturation phase. The main differences between the civil and UK military evolution phases are the approval/certification processes and their outputs. Section 8 presents more details about the certification phases, their outputs, and associated approval forms for both civil applications and for UK military aircraft complying with the MAA Regulatory Articles for new aircraft designs and for aircraft with major changes.

1.7.8 Appendices

Appendix A collates information about the design and maintenance philosophies of military and civil aircraft. The appendix presents introductory information about the safe life and damage tolerance approaches. The two approaches deliver aircraft products and maintain their target reliabilities throughout their in-service lives. Therefore, an interpretation of the words "target reliability" is presented and MOD regulations that are relevant to this interpretation are referenced. The appendix also discusses key military standards and regulations that form the perspective of the UK MOD on these two approaches. The appendix briefly discusses key UK military standards and regulations that are used for maintaining aircraft structural integrity. Since the target reliability is achieved by minimizing the risks of failures to very low levels, the appendix compiles brief notes about the failure causes, and introduces the Reliability Centred Maintenance (RCM) approach, which is implemented to determine the most appropriate maintenance tasks that would overcome the threats of functional failures, and to develop scheduled maintenance plans. The appendix also quotes, from UK MOD publications, brief notes about preventive and corrective maintenance tasks.

Appendix B presents definitions of Technology Readiness Levels (TRLs), which are used to assess the maturity of evolving aerospace technologies and, systematically, incorporate them into aerospace systems when they reach a high TRL.

Appendix C discusses the widely used quality attributes of system measurements including accuracy, resolution, precision, repeatability, reliability, sensitivity, dynamic range, and bandwidth. The appendix provides interpretations of these attributes extended, when appropriate, to SHM measurements.

2 DESIGNING AND MAINTAINING AIRWORTHY STRUCTURES

2.1 Aircraft Design and Maintenance Philosophies

For military aircraft in the UK, the MOD requires demonstrating by safety analysis that the aircraft design meets the required level of safety for the proposed flights. A number of MOD regulations require that measures should be taken to counter sources of threats to safety, airworthiness, and aircraft integrity, reducing the risks to “As Low As Reasonably Practicable” (ALARP). According to Reference [32], a risk can be said to be reduced to a level that is ALARP when the cost of further reduction is "grossly disproportionate" to the benefits of risk reduction. This cost may include more than financial cost and must consider the time and trouble involved in taking measures to avoid risk. Therefore, an ALARP argument should balance the "sacrifice" (in time, money, or trouble) of possible further risk reduction measures with their expected safety benefit (incremental reduction in risk exposure). The balance should be weighted in favour of safety, with a greater "disproportion factor" for higher levels of risk exposure”, Reference [32]. ALARP is essentially the "stopping condition" for risk reduction, so justifying and recording how this is reached is an important and vital step in safety management.

Generally, aircraft are designed and their safety, airworthiness, and integrity maintained during their in-service life by adopting a Safe Life (SL) approach or a Damage Tolerance (DT) approach. Whilst DT is the dominant philosophy for civil transport aircraft, military aircraft are designed and maintained using SL or DT philosophies. The SL philosophy requires that sufficient fatigue tests and analysis have been conducted to establish confidence that there will be no failures caused by expected operational conditions during promulgated in-service safe lives. The damage tolerance philosophy achieves and maintains a target reliability level through (a) designs allowing the presence and growth of damage during determined service periods, (b) planned inspections capable of assessing the levels of damage and its effects on the target reliability, and (c) planned repairs capable of restoring the target reliability, and assuring operational safety, during a following service period.

According to Def Stan 00-970 Part 1 Section 3 Clause 3.2, the SL approach is the standard approach in the UK; the evident advantage of this approach is to minimize the need for in-service inspections; then, a “clear by inspection” approach may be adopted to enable life extension beyond the safe life, and is used to overcome the threats of accidental damage, which may be induced by events such as impacts, reported overloads, or reported break of corrosion protection systems.

A Remotely Piloted Air Systems (RPAS) consists of several elements that are critical to engineering and flight safety including not only the flying Remotely Piloted Air Vehicle (RPAV) and all its associated flight safety-critical elements, but also elements such as the ground-based control unit and the ground-launch system. Def Stan 00-970 Part 9, Reference [42], presents the UK design and airworthiness requirements for RPAS.

Appendix A collates and presents more information about the design and maintenance philosophies of military and civil aircraft. Extended introductory information about the SL and DT approaches are presented in Sections A.1 to A.3. Section A.4 presents an interpretation of the words “target reliability” and references MOD regulations that are relevant to this interpretation. Section A.5 discusses key military standards and regulations that form the perspective of the UK MOD on SL and DT. Section A.6 briefly discusses key UK military standards and regulations that are used for maintaining aircraft structural integrity. Since the target reliability is achieved by minimizing the risks of failures to very low levels, Section A.7 presents brief notes about the main failure causes of structural and mechanical components.

Maintaining the aircraft is achieved by identifying potential risks and taking appropriate preventive and corrective actions to mitigate the risks and their consequences on safety and structural integrity. Section A.8 introduces the RCM approach, which is implemented to determine the most appropriate maintenance tasks that would overcome the risks of functional failures, and to develop scheduled maintenance plans. Based on the UK MOD classification and definitions of Reference [33], Sections A.9 and A.10 quote brief notes about preventive and corrective maintenance tasks.

2.2 Designing Airworthy Structures

Emerging SHM systems could influence the structural design of future aircraft. Aircraft structures are designed according to airworthiness regulations to safely withstand all foreseeable loads and operational conditions including gusts, lightning, manoeuvre loads, engine loads, unsymmetrical loads due to engine failure, pressurized compartment loads, ground loads, etc. The compliance with these regulations must be demonstrated by analyses or tests, and by post design evaluations (e.g. fatigue tests).

For example, FAA specifies strength requirements in terms of limit loads (the maximum loads to be expected in service) and ultimate loads (limit loads multiplied by prescribed factors of safety, e.g. 1.5). For transport aircraft structure, FAA mandates that: the structure must be able to support limit loads without detrimental permanent deformation that interfere with safe operations; the structure must be able to support ultimate loads without failure; the structure must withstand any vibration, and buffeting that might occur in any likely operating condition up to demonstrated flight diving speed (VD/MD); the structure must be designed to withstand any forced structural vibration resulting from any, excluding extremely improbable, failure, malfunction, or adverse condition in the flight control system; etc. The detailed sets of FAA regulations for structures are presented in Part C of Reference [3] (14CFR Part 25: §25.301 to §25.581). The FAA airworthiness regulations for the structures of other aircraft types are presented in 14CFR Part 23: §23.301 to §23.575 of Reference [2], 14CFR Part 27: §27.301 to §27.573 of Reference [4], and 14CFR Part 29: §29.301 to §29.573 of Reference [5].

The structural design of the UK MOD fixed wing aircraft must consider all forces imposed on the structure during its operation, which should include, but not be limited to: ground loads, gust and turbulence loads (including ground gusts), aerodynamic loads, inertia loads, systems-induced loads (e.g. from control surfaces, undercarriage deployment), engine-induced loads (and auxiliary power unit induced loads, where appropriate), and loads from the associated thermal environment, stores carriage and release, flight condition (for example, buffet), maintenance, environmental effects (such as heating including that from aerodynamic effects, cooling, and moisture). The UK MOD standard requirements for fixed wing structures are presented in Def Stan 00-970 Part 1 Section 3, Reference [39] ; the standard requirements for rotorcraft structures are presented in Def Stan 00-970 Part 7 Section 2 Supplement C and Supplement D, References [40] and [41]. These requirements are among the acceptable means of compliance with the UK MAA Regulatory Articles (RA): RA 5203(2) “Contract Specifications”, RA 5203(3) “Sub-Contract Specifications”, and RA 5105(1) “Requirement for Re-qualification”, Reference [36].

The mandatory requirements for RPAS are presented in Def Stan 00-970 Part 9, Reference [42]. Part 9 mandates a group of NATO airworthiness standards with minor UK national reservations; the mandated NATO standards include those standards set out in STANAG 4671 “Unmanned Aerial Vehicles System Airworthiness Requirements (USAR)”, Reference [60]. Part 9 “Subpart C – UAV Structure” mandates a number of STANAG 4671 clauses which cover: loads, canard/tandem wing configurations, factor of safety, structural performance, strength and deformation, proof of structure, flight loads, (symmetrical and asymmetrical flight conditions, load factors, gust load factors, engine torque, side-load on engine mount, pressurised compartment loads, asymmetrical loads due to engine failure, gyroscopic and aerodynamic loads, control surface/system loads, ground loads, fatigue evalaution, etc. Def Stan 00-970 Part 9 requires that reference should also be made to the other parts of Def Stan 00-970 and to Def Stan 07-85 which, although relating to the design requirements for manned aircraft and guided weapons respectively, can also be applicable to RPAS.

2.3 Threats to Structural Integrity

The main three threats to airworthy structures are Fatigue Damage (FD), Environmental Damage (ED), and Accidental Damage (AD). In order to overcome these threats and take timely maintenance actions against them before they cause failure, the state of the structure should be regularly evaluated. Emerging SHM systems would provide improved methods for evaluating the condition of structures. FD, ED, and AD occur as a result of usage, hazardous-operations, exposure to environments, and/or component interactions with other objects. Damages caused by repeated stresses, thermal cycles, or overloads are examples of the usage and hazardous-operation damage. The exposure to salty water and sandstorms can cause corrosion/erosion leading to loss of material strengths and damage caused by environments. Interaction damage can be induced as a result of rubbing, accidental tool drops, foreign object strikes, and battles.

2.3.1 Fatigue Damage

FD is an inevitable consequence of usage (age). The exposure of metallic structures to a large number of repeated stress/strain cycles eventually initiates and grows cracks; the presence of tiny defects accelerates the initiation and growth of cracks. For composites, FD occurs as a result of repeated cycles causing growth of faults such as fibre breakage, micro-cracking of matrix, delamination between layers, and de-bonding (separation of fibres and matrix). Laboratory tests of metallic and composite structures are conducted to determine fatigue characteristics (effects of repeated cycles), and to validate models describing defect growth to failure as a function of repeated cycles; to date however, the composite cyclic failure is not as fully understood as the metallic cyclic failure. The exposure of aircraft components to cyclic loads is an inevitable consequence of normal flight operations, manoeuvres, gusts, landing impacts, pressurisation, and ground operations including taxiing, braking, towing, and starting-up/shutting-down the engines; for military aircraft, cyclic loads can also be induced by operations such as air re-fuelling, aerial delivery, and release of stores or ammunitions. Safeguarding structural integrity requires monitoring the sources of cyclic loads and frequently assessing their potential damaging effects.

2.3.2 Environmental Damage

ED is degradation of structures as a result of their exposure to environments that can cause corrosion, erosion, and/or fluid/gas absorption. ED can be induced by spillage of corrosive fluids, salty water, erosive sand storms, hail, thermal cycling, electro-magnetic radiation, significant changes in atmospheric temperature combined with salty environments, and conditions causing fluid/moisture penetration in composite materials. Integrity of composite structures can be compromised if excessive moisture is absorbed leading to softening of composite matrices, delamination of plies, and disbonding; prolonged exposure to hot-wet conditions causes ED. It is worth mentioning that some publications classify erosion damage caused by environmental events such as rain, hail, lightning, sand storms, freezing, and thawing as accidental damage. Not only continued exposure to corrosive/erosive environments causes continued loss of mass and leads to failure, but also exposure to stress cycles accelerates growth of environmental faults such as corrosion pits and leads to fatigue failure. Corrosion can occur in metallic and composite structures. Metallic airframes are treated with corrosion prevention layers and paint systems designed to last for specified periods. These preventive layers can breakdown as a consequence of accidental damage or misuse.

2.3.3 *Accidental Damage*

AD occurs as a consequence of random discrete hazardous-operation events or component interactions with other objects.

- The hazardous-operation events include operations outside design envelopes that cause overloads and plastic deformation, i.e. operations that induce loads exceeding design proof loads, e.g. exceeding design limit loads factored by 1.125, Def Stan 00-970. For example: over-speed can expose control surfaces to damaging loads; severe hard-landings can induce plastic deformations; excessive manoeuvres can overload aircraft structures.
- Interaction events that can induce damage include: aircraft mishandling on the ground, accidental impact of cargo trucks, accidental maintenance tool drops, freight mishandling, mid-air collision, bird strike, wire strike, lighting, hail, battle damage, foreign object strikes, weapons release ricochet damage, galley spillage, toilet spillage, interactions with cleaning fluids, runway debris, wear caused by rubbing between surfaces deprived from lubricants, and effects of vibration induced by highly unbalanced rotating components.

Overload is considered AD event, which may be induced by, for example, excessive gusts or during emergency operations.

Excessive wear can cause reductions in material strength and ultimately failure. Fretting is a wear process that occurs at moving/rubbing contacting surfaces of two objects; under load cycles, fretting can initiate cracks and cause fretting fatigue; in the presence of an aggressive environment, fretting causes material loss at the contact surfaces followed by oxidation; the oxidized debris can act as an abrasive causing degradation called fretting corrosion. Hence, the hazardous-operation and interaction events can initiate structural faults; then, the exposure to operational stress cycles or environmental conditions can accelerate the growth of these faults and eventually cause FD or ED.

A third class of AD can be induced as a result of development, manufacturing, and supply errors. The threats of these errors are mitigated by implementing relevant processes at levels of rigour commensurate with the consequences of failures; see Section 7. Examples of these errors are: producing incorrect assembly, using inappropriate materials, using wrong parts, providing inadequate or wrong material information and instructions, errors in software, etc.

2.4 Maintaining Structural Integrity

Emerging SHM systems would provide improved methods that support overcoming the threats of FD, ED and AD. Currently these threats are controlled and mitigated through approved maintenance and structural integrity programmes complying with relevant airworthiness regulations.

For example, for civil aircraft, industry and regulatory authorities generate Maintenance Review Board Report (MRBR) using MSG-3 logic; the report outlines the minimum scheduled interval/tasking requirements to be used in the development of an approved maintenance/inspection programmes, Reference [12]. The approved MRBR is a basis from which each operator develops maintenance/inspection programmes. The main regulations that govern the maintenance and structural integrity programmes include: 14CFR Part 25: §25.571 "Damage tolerance and fatigue evaluation of structure", 14CFR Part 25: §25.1529 "Instructions for Continued Airworthiness", 14CFR Part 43 "Maintenance, Preventive Maintenance, Rebuilding and Alteration", and 14CFR Part 91 Subpart E, See References [3], [6], [8], [9], [24], [12], [13] and [14].

For UK military aircraft, the maintenance and structural integrity programmes are governed by UK MAA regulations including for example the following: RA 5723 "Ageing Aircraft Audit", RA 5724 "Life Extension Programme", RA 5720(1) to RA 5720(6) that cover Structural

Integrity Management, and RA 4200 to RA 4974 that cover various maintenance aspects for the aircraft and its subsystems; see References [29] to [36].

Figure 2 summarises existing civil and military regulations and processes that are currently used to design airworthy structures and maintain the structural integrity of the aircraft. Emerging SHM systems must comply with such mandated regulations and provide improved methods for maintaining the integrity of the structure.

Figure 2: Designing and maintaining airworthy structures

A RPAS consists of several elements that are critical to engineering and flight safety including not only the flying RPAV and all its associated flight safety-critical elements, but also elements such as the ground-based control unit and the ground-launch system. It is MOD policy that all military RPASs are operated and maintained in accordance with the same policy and procedural requirements applicable to manned aircraft with minor exceptions that apply to the maintenance of RPASs waiving regulations regarding continuous charge and indicating that RPAS flight servicing is not to be waived, see CAE 4000 - MAP-01, Reference [33]; the aircraft type is considered to be on continuous charge if its concept of operation demands that the aircraft land, stop their engine/s, change crews, possibly refuel, restart engine/s and take off again in order to complete a particular mission.

Regulatory and industry publications such as References [12] describe the requirements (rules) that should be considered during the development of maintenance and management programmes to comply with the mandated regulations; the following sections refer to these requirements as "regulatory structural rules" to avoid confusing the words "SHM requirements" with the "requirements" that have been set to guide the development of maintenance and management programmes. Emerging SHM applications must be guided by existing regulatory structural rules and must comply with mandated airworthiness regulations such as those of References of [1] to [39]. Therefore, details about key regulations and rules are presented in the following sections.

2.5 Key Airworthiness Regulations

2.5.1 Key Civil Aircraft Regulations

SHM must comply with the FAA regulations for "Equipment, systems, and installations" given in Sections §25.1301 and §25.1309 of Reference [3]: it is required that equipment, systems, and installations whose functioning is required by §25.1309, must be designed to ensure that they perform their intended functions under any foreseeable operating condition; they must be designed so that: (1) The occurrence of any failure condition which would prevent the continued safe flight and landing of the airplane is extremely improbable, and (2) The occurrence of any other failure conditions which would reduce the capability of the airplane or the ability of the crew to cope with adverse operating conditions is improbable. The following two practical considerations are essential for SHM that can realistically comply with the above regulations:

- Often, the failure of a credited SHM function does not directly cause structural failures that prevent the continued safe flight or reduce the capability of the airplane and crew; in other words, the failure of a credited SHM function does not directly cause "major", "hazardous/severe major", or "catastrophic" structural failures; such failures happen because the structure continues to degrade as a result of usage, exposure to environments and/or accidental damage in absence of timely indication of the degree of degradation by the failed SHM system.
- The SHM system must survive all its local environmental conditions for a reasonable specified service period; the system does not necessarily need to perform its intended functions under all these environmental conditions, it must perform its intended functions only under specified operating conditions. Therefore, the SHM developers must clearly discriminate between operability and survivability; they must not confuse the entire SHM environmental conditions with the subset of specified operational environments; for example, one of the SHM systems may only operate when the aircraft is on the ground. The operational environments vary between SHM systems; the environmental conditions that a system must survive vary between designs and depend on the locations of the onboard system components.

The FAA regulation for "damage tolerance and fatigue evaluation of structure" given in 14CFR Part 25 §25.571 of Reference [3] states the following requirement: "Based on the evaluations required by this section, inspections or other procedures must be established, as necessary, to prevent catastrophic failure, and must be included in the Airworthiness Limitations Section of the "Instructions for Continued Airworthiness" required by 14CFR Part 25 §25.1529. Inspection thresholds for the following types of structure must be established based on crack growth analyses and/or tests, assuming the structure contains an initial flaw of the maximum probable size that could exist as a result of manufacturing or service-induced damage: (i) Single load path structure, and (ii) Multiple load path "fail-safe" structure, and crack arrest "fail-safe" structure, where it cannot be demonstrated that load path failure, partial failure, or crack arrest will be detected and repaired during normal maintenance, inspection, or operation of an airplane prior to failure of the remaining structure."

2.5.2 Key Military Aircraft Regulations

2.5.2.1 Acceptable and As Low As Reasonably Practicable (ALARP) Risks

The purpose of the Def Stan 00-56: "Safety Management Requirements for Defence", Reference [43], is to provide requirements and guidance for the achievement, assurance and management of safety. It can be applied to any MOD project and in any phase of a project's life. Def Stan 00-56 states the following: "Under UK law, all employers have a duty of care to their employees, the general public and the wider environment. For the MOD this includes, but is not limited to, an obligation to manage the Risk to Life associated with operation of military systems. In accordance with general guidance provided by the Health and Safety Executive, and as defined in Joint Service Publication (JSP) 815, MOD will discharge this duty by ensuring that all identified risks to life are reduced to levels that are As Low As Reasonably Practicable (ALARP) and tolerable, unless legislation, regulations or MOD Policy imposes a more stringent standard". Def Stan 00-56 also states the following:

- "The ultimate responsibility for accepting and operating a system lies with the MOD, and a decision to deploy a System which is not ALARP can be made only by the duty holder who has considered the Risk to Life and referred the risk to an appropriate level for acceptance".
- Contractors who supply Products, Services and/or Systems (PSS) to the MOD are subject to legal duties, which may vary with the place of manufacture and supply or operation. MOD shall have regard to the needs of Contractors to discharge their legal duties when interpreting and applying the requirements of this Standard".
- "Contractors must apply ALARP principles through generating and satisfying Derived Safety Requirements, applying mitigation strategies and possibly using Cost-Benefit Analysis to justify decisions".
- "The Contractor shall undertake the design of the PSS so as to meet all Safety Requirements. The Contractor shall identify mitigation strategies to minimise safety risk and meet Safety Requirements. The Contractor shall select and implement a combination of mitigation strategies for hazards or failure modes that contribute to a hazard, according to the following precedence: (a) Elimination, (b) Reduce the Risk to Life by engineering means, and (c) Reduce the Risk to Life by means based on human factors, incorporating requirements from Defence Standard 00-250, as appropriate".
- "The Contractor shall demonstrate the effectiveness of the process for identifying and selecting mitigation strategies, and shall record the rationale, including the application of the ALARP principle, for the selection of each mitigation strategy in the information set".
- "Although the Contractor may not be judgements directly, they will be in a position to support the decision process by identifying new technology options which may enable more cost-effective mitigations, or which make previously discarded design options practicable. The Safety Management Plan (SMP) is reviewed and agreed by the Safety Committee, and is the appropriate mechanism to proposed design enhancements arising out of compliance with this clause. This is to enable the MOD to judge which improvements can be implemented in order to comply with its obligations, and meet its operational commitments".

The ALARP principle imposes rigorous safety requirements and enables effective operations in emergency and hostile environments.

2.5.2.2 The MOD MAA Regulatory Publications

As published by MOD, Figure 3 shows the structure of the MAA regulatory publications, which reference the ALARP principle. Along with "Def Stan 00-970 Part 1, Section 3 - Structure", the most relevant publications to SHM are the 4000 Series and 5000 Series. Key regulations are extracted from these documents and presented in this paper.

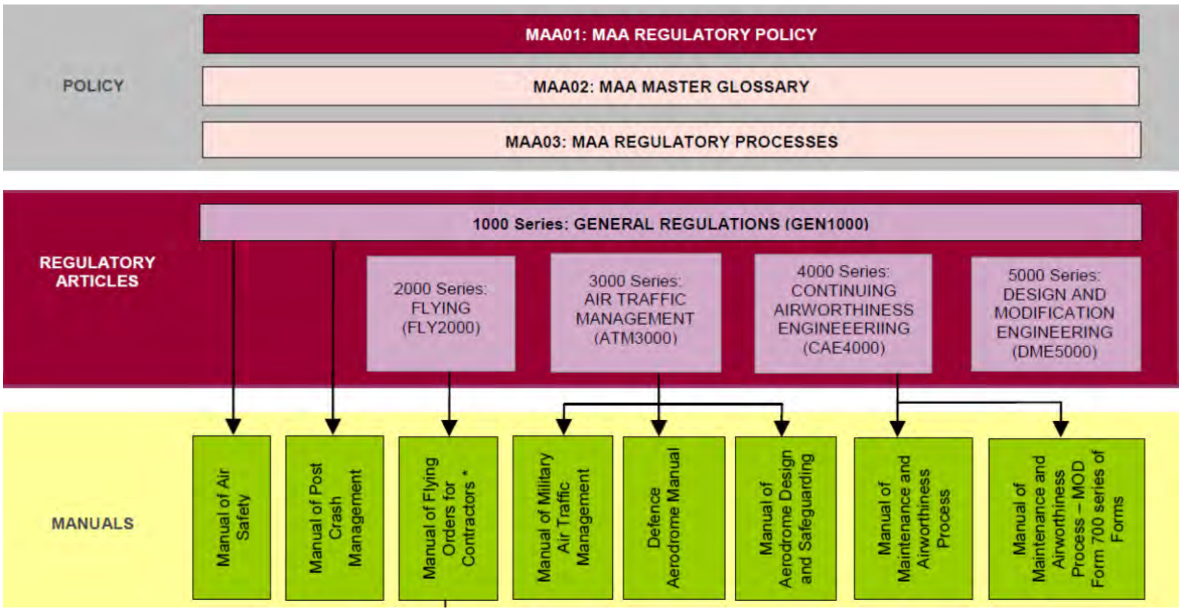


Figure 3: The structure of the MAA Regulatory Publications (MRP)

2.6 Regulatory Structural Rules

2.6.1 Rules from Maintenance Review Board Report (MRBR) [12]

The MRBR is generated by industry and regulatory authorities as a coordinated effort of achieving timely compliance with the applicable certification regulatory requirements. The MRBR outlines the minimum scheduled interval/tasking requirements to be used in the development of an approved maintenance/inspection programmes. Reference [12] presents recommendations to guide the development of the MRBR contents; some of these key recommendations are:

- The requirements in the report should be developed using MSG-3 decision logic.
- The report should contain the analysis of all Maintenance Significant Items (MSI) and Structural Significant Items (SSI) on a task-by-task basis. If a task is determined to be a safety task or applicable cost effective task, an appropriate tasking interval should be selected.
- The report should contain sections covering “maintenance requirement rules”, “system/power plant requirement rules”, “structural program rules”, and “zonal program rules”.
- In the “structural program rules” section, FAA states the following quoted structural requirements to be considered by the Original Equipment Manufacturer (OEM) and the Type Certificate Holder (TCH): “The OEM/TCH develops structural inspection procedure (SIP) requirement rules to meet the inspection requirements for damage tolerance. The types of damage considered during structural requirement development are environmental deterioration (ED) (e.g. corrosion and stress corrosion), accidental damage (AD), and fatigue damage (FD). Some forms of ED are age related; therefore, calendar intervals control inspections for this type of deterioration. The required structural requirements section incorporates these calendar inspections, plus the requirements for detecting other types of ED, AD, and FD”. Reference [12] also requires provisions to the quoted requirements: “external and internal inspections, structural sampling and age-exploration programs, corrosion prevention and control programs, and additional supplemental structural inspections that may be required for fatigue-related items. Calendar time, flight cycles, or flight hours express the initial check intervals for the SIP”.

- The regulator approves the MRBR for use by certificated operators; the requirements of the approved report are a basis from which each operator develops maintenance/inspection programmes.

2.6.2 Rules from UK Defence Standards [39]

The UK MOD publishes structural rules for military aircraft in MAA documents and defence standards, References [35] to [39]. An example of these rules is the safe life approach that requires comparison of in-service usage with design assumptions. The approach requires monitoring simple usage parameters, but equally requires making provision for recording other usage data, which impact airframe and component fatigue lives; for example, pressurisations, landings, undercarriage cycles and engine acoustics in flight and during ground runs can all affect lives. The in-service usage is assessed by comparison with information relating to the operating intent and the original design, preferably the design spectrum; the information is compiled in a SOIU document. Every fixed-wing aircraft must be provided with instrumentation to enable in-service monitoring ranging from a comprehensive sensor suite on highly manoeuvrable combat types to a basic suite on aircraft being used in a passenger role; however, limit load exceedance monitoring is considered essential for all types. Helicopter Health and Usage Monitoring (HUM) Systems can provide comprehensive operational usage data for most fleets. The MOD structural monitoring approach is underwritten by OLM programs on fixed-wing aircraft or by Operational Data Recording (ODR) programmes on helicopters: a number of aircraft is comprehensively instrumented to capture a sufficient proportion of fleet's usage data across all usage types and roles to enable an accurate assessment; continuous or periodic OLM/ODR programmes are mandatory to comply with MOD airworthiness policy.

Where a component is exposed to impact damage during maintenance or operations, it must be also the subject of an inspection-based substantiation. Normally, the safe life of components which are to be cleared by inspection must be demonstrated to be at least half of the specified life under the design spectrum subject to conditions specified in the Def Stan 00-970 such as: the presence of fatigue cracks can be identified with acceptable confidence; any crack that remains undetected after an inspection will not grow, under the service spectrum, to an unacceptable size before the next inspection or before scheduled replacement or retirement; in-service incident reports, if any, are taken into account; the inspection penalty is acceptable on operational and economic grounds.

Def Stan 00-970 also presents the rules required for using non-adaptive prediction methods for in-service monitoring; the methods use a fixed set of transformation equations that operate on flight parameters such as normal acceleration and roll rate to produce outputs that approximate a target value such as strain, load, or fatigue damage.

2.6.3 Rules from Military Aviation Authority Publications

2.6.3.1 Regulation RA 4200 - Maintenance Philosophy - General

- Regulation 4200(1) - Maintenance: "Aircraft and associated equipment shall be subject to preventive and corrective maintenance, supported by appropriate sustainment or enhancement modification action".
- Regulation 4200(2) - Type Airworthiness: "PTs shall ensure the type airworthiness of their platform type by using a system of assessment that ensures the ongoing analysis of the fault management system, a fundamental part of which is the Fault Reporting And Corrective Action System (FRACAS)."

The AMC with these regulations is contained within CAE 4000 - MAP-01 Chapter 5.1, Reference [33], which are summarised in Sections A.9 and A.10.

2.6.3.2 RA 4201 - Maintenance Policy - Composite Materials

- 4201(1) - Composite Materials Maintenance: "In order that the design properties of Composite Materials (CM) are retained or recovered in a cost-effective and efficient manner throughout the service life of the aircraft, aircraft CM structures and components shall be subject to specific maintenance activity".
- 4201(2) Composite Materials Awareness and Husbandry: "To ensure the continued structural integrity of aircraft structure and components constructed using Fibre Reinforced Plastics (FRP), and to reduce maintenance costs, Project Teams, FLCs and user units shall put in place procedures to establish and maintain appropriate levels of awareness and husbandry".
- 4201(3) - Recording of Composite Materials Related Maintenance: "A database shall be used to record all structural concessions, repairs, modifications and accidental damage and environmental damage to CM. Changes to the configuration of FRP structure and components are also to be included in this record".

The AMC with these regulations is contained within CAE 4000 - MAP-01 Chapter 5.1.1, Reference [33].

2.6.3.3 Maintenance of Remotely Piloted Air Systems (RPAS)

- RA 4050(1) - Maintenance of RPAS: "All RPAS operated within the Military Air Environment shall be maintained in accordance with the same policy and procedural requirements applicable to manned aircraft".
- RA 4806(10) AMC- "Non-engineering tradesmen required to undertake maintenance tasks on Remotely Piloted Air Systems, including assembly, pre-flight checks and user-level maintenance, should have completed the appropriate formal specific-to-type training prior to authorization. The process for authorization detailed in MAP-01 Chapter 2.1 should be followed."

The AMC with these regulations is contained within CAE 4000 - MAP-01 Chapter 2.1, Reference [33].

2.6.4 Rules for USAF [44]

In the US, the military structural rules are published by DOD. For example, Reference [44] presents the USAF Aircraft Structural Integrity Program (ASIP) that provides the general approach for establishing and sustaining structural integrity throughout the life of any aircraft type; the program main objective is the cost effective prevention of structural failures without losing mission capability. The ASIP document is tailored for individual programs covering particular aircraft types with each individual document includes the aircraft designator in its name (e.g. the C-130 ASIP).

The intended functions of an SHM system should be sought to enable an improved means of compliance with such regulatory structural rules.

2.7 The Use of SHM as Introduced by MSG-3

In 2009, the Air Transport Association Maintenance Steering Group (ATA MSG-3) has revised its recommended specifications to incorporate operation uses of SHM in aircraft maintenance, Reference [24]. In their definition of SHM, ATA MSG-3 have clearly discriminated SHM from NDI equipment by the distinctive onboard sensors of SHM; ATA MSG-3 have defined SHM as "The concept of checking or watching a specific structural item, detail, installation, or assembly using on board mechanical, optical or electronic devices specifically designed for the application used; SHM does not name any specific method or technology". ATA MSG-3 have also defined the term Scheduled SHM (S-SHM) as "the act to use/run/read out a SHM device at an interval set at a fixed schedule". Then, S-SHM has been added to a group of scheduled tasks to be accomplished at specified intervals; namely, the "inspection/functional check" group has been expanded to become:

- General Visual Inspection (GV or GVI)
- Detailed Inspection (DI or DET)
- Special Detailed Inspection (SI or SDI)
- Scheduled Structural Health Monitoring (S-SHM)

Reference [24] has indicated that SHM may be an option to check or watch for AD, ED, and FD where demonstrated to be applicable and effective; the reference states the following:

- The manufacturer may propose a validated S-SHM application(s) as long as it satisfies the detection requirement(s) of AD and ED;
- Details of the fatigue related task requirements based on the manufacturer's damage tolerance evaluations, including validated S-SHM application(s), are presented to the Structural Working Group (SWG) (or equivalent body) who determines if they are acceptable.

Following from the MSG-3 work, an Issue Paper (IP No: 105) has been submitted by Airbus, Boeing, Bombardier, Embraer, and Gulf-stream as Joint Industry Proposal to further advance the definition of SHM and include automated applications. The provisional definition of Automated SHM (A-SHM) is "a task that can automatically inform maintenance personnel that action must take place instead of having a pre-determined interval at which the maintenance action must take place".

3 THE EVOLUTION AND ACQUISITION OF AIRCRAFT PRODUCTS

3.1 Overview

The word "product" is used in a FAA/Industry guide, Reference [24], and in this document, to identify aircraft, aircraft engines, propellers as well as systems, appliances, components, or parts. Therefore, the word "product" covers aircraft structural components such as wings and aircraft systems such as SHM systems.

Figure 4 illustrates the evolution phases of civil aircraft product, which span maturation, design, production, installation/integration, certification, and utilisation. Whilst this section briefly describes these phases, the following sections give more detail about the key aspects of the SHM evolution phases, which include SHM architecture, requirements, validation, verification, and certification; the sections either give concise information and extensions applicable to SHM or refer the reader to existing industry accepted publications for more information.

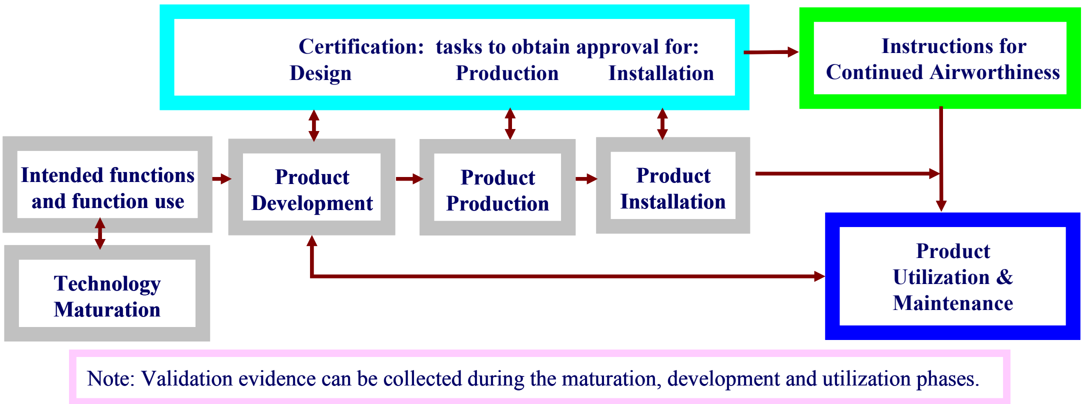


Figure 4: The evolution of civil aircraft products including SHM systems

For UK military aircraft products, Figure 5 shows similar evolution phases.

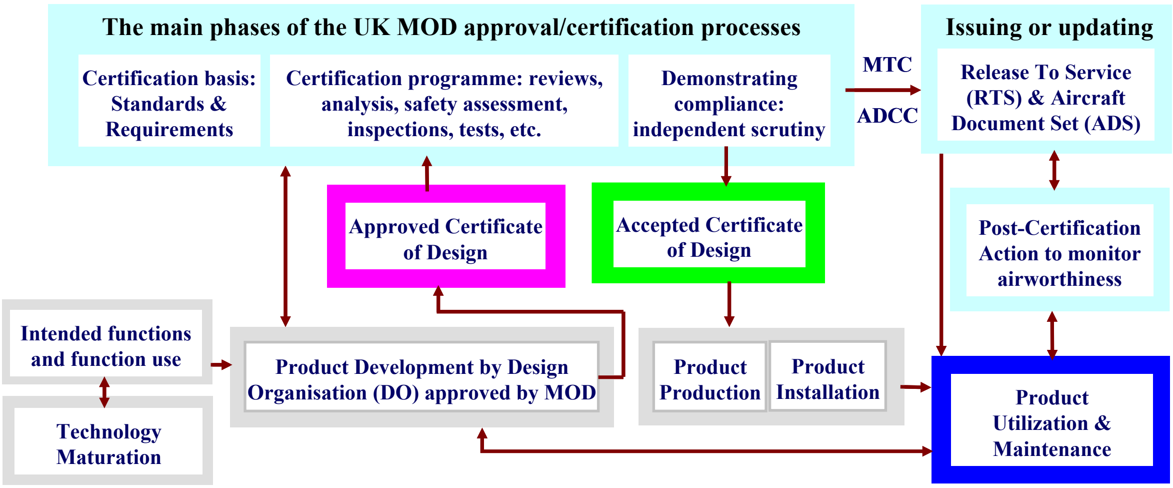


Figure 5: The evolution of the UK military aircraft products

The main differences between the two evolution approaches are the approval/certification processes and their outputs noting the following:

- The DO concludes the development phase by preparing a certificate of design signed by an approved member of the DO. The certificate of design should be submitted to the MOD before the first flight of the airborne product.

- The certificate of design can be accepted by MOD after establishing compliance with specifications through inspection, demonstration, analysis, and test.
- Acceptance of the Certificate of Design does not imply acceptance of responsibility for the design, which remains with the DO.
- The DO retains the original signed copy of the certificate of design and distributes the certificate of design in accordance with the instructions of the MOD.
- The Military Type Certificate (MTC) or Approved Design Change Certificate (ADCC) are held and used by the MOD in support of the RTS Recommendations made for a new air system or a major change to an air system.
- Changes other than major changes may be self-certified in accordance with extant procedures by the MOD Type Airworthiness Authority (TAA).

3.2 Intended Functions

The evolution phases of a new aircraft or an aircraft system start with high level intended functions that address emerging demands. Examples of key emerging demands are: (a) the large capacity of the Airbus A380 or the Boeing 747-400, and (b) the affordability and common designs of the three Joint Strike Fighter (JSF) variants.

3.2.1 SHM, Why?

The moral duties of care pose everlasting demands on the aircraft industry for improved airworthiness; driven by market forces, functional duties pose requirements for continuous reductions in the Costs Of Ownership (COO) and enhancements of aircraft performance. The SHM technology maturation and development efforts are driven by these general requirements and specifically address the following demands:

- Demands for reductions in maintenance costs through reductions in inspection times, reductions in inspection efforts, improvements in inspection quality, and/or generating information that can be used to defer maintenance actions,
- Demands for certifiable SHM systems that enable underwriting new aircraft materials and designs, and
- Demands for improvements in operational management, affordability, and performance.

The SHM maturation and development efforts start with identifying the intended functions that can address these demands.

3.2.2 Intended Functions

In order to address the above demands, the potential intended functions of an SHM system should provide improved knowledge that enable, for example, improved techniques for: fatigue monitoring, exceedance monitoring, environmental monitoring, FD detection, ED detection, and AD detection. The SHM functions should provide maintenance, operational, and/or ownership benefits such as: reducing inspection costs and times, improving repair planning, optimising inspection intervals, reducing aircraft downtime, increasing residual values of used aircraft, reducing weight, and enabling life extensions. In other words, SHM must provide improvements in one or more of the structural functions cited in civil/military publications, which include for example: inspection methods to detect FD, ED and AD, structural sampling and age-exploration programs, corrosion prevention and control programs, ASIP techniques, in-service usage monitoring and OLM methods. Figure 6 shows the potential intended functions of SHM and briefly describes how they can be used.

Intended Functions:
• **Detect** damage:
Fatigue Damage (FD), Environmental Damage (ED) & Accidental Damage (AD)
• **Monitor** factors that lead to or indicate damage
• **Assess** the impact on structural integrity
Function use:
• Generate advisory information
• Improve maintenance /operational practices
• Realize new designs

Figure 6: The SHM intended functions and their use

3.3 Maturation

After determination of potential intended functions that address emerging demands, maturation efforts are initiated. Usually, the maturation phase starts before the development and certification phases, and can overlap them. Often, the maturation phase starts with relatively small Research and Development (R&D) programmes that may involve a wide range of academia, technology developers, and aerospace organizations. As advances are made, the R&D activities become more focused and can grow to major maturity programmes led by key manufactures; the JSF Concept Demonstration (CD) phase can be considered as a large scale maturity phase involving two independent programmes led by Lockheed Martin and Boeing.

3.3.1 Maturation Efforts

For SHM, the maturation efforts are often guided by technology and product roadmaps: efforts are allocated to develop sensing technologies, algorithms, and software for SHM, and to enhance the performance of SHM in terms of increased accuracy, reduced weight, improved reliability, advanced communication, and efficient data transfer. Technology gaps and risks are identified and efforts are allocated to fill the gaps and to mitigate the risks. During the maturation phase, the potential benefits and credits of SHM are assessed and validation evidence is gathered. Efforts can also be allocated to develop and flight-test SHM prototypes, and to develop efficient production processes and reliable installation techniques. Perhaps, some of the key factors that would accelerate the maturity of SHM systems include:

- Consideration of an SHM modular distributed architecture,
- Consideration of wireless communications, where light weight sensor nodes and processing units powered by energy harvesting techniques are distributed over the required structures and integrated via wireless network communications,
- Consideration of an SHM system where the aircraft onboard sensors are powered and interrogated by ground-based equipment,
- Consideration of indirect SHM techniques where recorded aircraft data such as speed and acceleration are used to provide OLM, usage monitoring, fatigue monitoring, and exceedance monitoring capabilities, and
- Sufficient collection of validation evidence regarding the SHM benefits/credits; the presence of conclusive evidence would encourage the initiation of the system development, production, and installation phases.

3.3.2 Maturity Assessment

Technology Readiness Levels (TRL 1 to TRL 9) are used to assess the maturity of evolving aerospace technologies, and systematically incorporate them into aerospace systems when they reach a high TRL. Several TRL definitions exist, e.g. the NASA definitions, the DOD definitions, and the European Space Agency (ESA) definitions. These levels range from reporting basic principles to flying and proving the actual system through mission operations. The NASA, DOD, and ESA definitions are presented in Appendix B to make this report self-contained. The use of one of these sets to assess the maturity of SHM technologies is recommended; the introduction of a new set of TRL definitions specific to SHM is discouraged.

The challenges, gaps, and risks of a SHM technology should be re-assessed at the end of each level. The SHM system development activities may start when the maturity reaches a level where the risks are significantly mitigated and challenges/gaps addressed; such a level needs not to be TRL 9; in other words, the activities required to achieve a high TRL level may overlap with system development activities.

3.4 Certification

Certification involves efforts to obtain the approval of the appropriate regulator that the applicable functional requirements, airworthiness regulations, and operating rules are met. The certification efforts involve both the regulator and the product developer along with any other stakeholders such as the aircraft manufacturer. The certification efforts are initiated through an application made by the product developer to the appropriate RA; they are often performed in parallel to the other evolution phases; the degree of the regulatory authority engagement at early evolution stages varies from one authority to another.

For civil aircraft, the outputs of successful certification phases include:

- design approval, production approval, installation approval, and
- Instructions for Continued Airworthiness (ICA).

For UK military aircraft, the main outputs of successful certification phases include:

- Acceptance of a certificate of design prepared and signed by an Approved Design Organization (DO).
- Military Type Certificate (MTC) for a new air system or Approved Design Change Certificate (ADCC) for a major change to an air system; the certificate is held and used by the MOD in support of the Release To Service (RTS) Recommendations.
- Release To Service (RTS).

The RTS is a central source document for the Aircraft Document Set (ADS); ADS includes the RTS, Aircraft Safety Case, Aircraft Maintenance Manual (AMM), Operating Data Manual (ODM), Flight Reference Cards (FRCs), Support Policy Statement, Engineering Air Publications including Flight Test Schedule (FTS), and SOIU.

Section 8 presents details about the certification phases, their outputs and associated approval forms for both civil aircraft and UK military aircraft.

3.5 SHM System Development

Given the intended function(s) and the intended use of a product, the development phases of the product involve the following main activities: development of product architecture, allocation of safety requirements, allocation of detailed requirements, design, implementation, test/evaluation, integration, test/evaluation of the integrated product, flight tests, and development of instructions for production, installation, maintenance, and operation. The essential processes required during the development phases cover: safety assessment, validation, verification, and configuration management.

Figure 7 describes the main development activities and presents brief definitions of the essential processes required to develop an approved SHM.

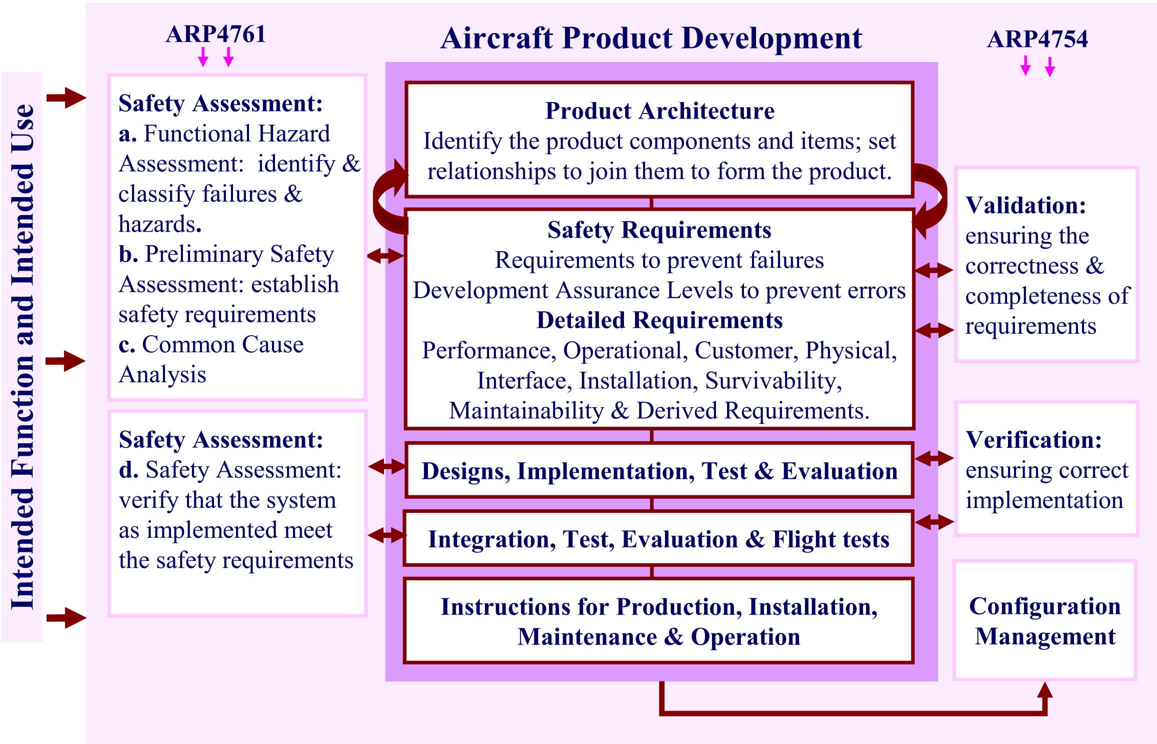


Figure 7: Key system development activities

3.5.1 *Safety Assessment*

The safety assessment process consists of Functional Hazard Assessment (FHA), Preliminary System Safety Assessment (PSSA), Common Cause Analysis (CCA), and System Safety Assessment (SSA). SAE ARP4761 gives detailed guidelines on the SSA process and describes methods for conducting the process, Reference [27].

3.5.1.1 Functional Hazard Assessment

The FHA examines the functions of each product item, identifies potential functional failures and classifies the hazard associated with each failure condition. The classification of each failure condition is based on evaluating the failure consequence on safety; the consequence of a failure is classified as catastrophic, hazardous/severe-major, major, minor, or no safety effect. The FHA is developed early in the development process and is updated when new functions or failure conditions are identified.

3.5.1.2 Preliminary System Safety Assessment

By examining the product architecture and the results of the FHA, the PSSA establishes the safety requirements of the product and its items and provides a preliminary indication that the anticipated product architecture can meet these safety requirements. The safety requirements are introduced to prevent failures and eliminate development errors; more accurately, they are introduced to significantly reduce the rates of failures and errors to specified low values consistent with the classification severity of failures/errors. The PSSA may identify the need for alternative protective strategies such as partitioning, built-in-test, monitoring, independence, and safety maintenance task intervals. So, the PSSA can be updated throughout the system development process.

3.5.1.3 Common Cause Analysis

The CCA identifies failures or external events that can lead to a catastrophic or hazardous/severe-major failure condition. The CCA validates that these failures and events are independent; i.e. they are not common between systems, items, or functions. To satisfy safety and regulatory requirements, it is necessary to ensure that such independence exists, or that the lack of independence is acceptable. The CCA examines the effects of potential development, manufacturing, installation, maintenance and crew errors that can defeat the independence and cause common failures in multiple systems/items and lead to a catastrophic or hazardous/severe-major failure condition. The results of a preliminary CAA are essential for the assignment of the development assurance levels and the determination of any additional safety requirements that reduce the probability of such failures to acceptable levels.

3.5.1.4 System Safety Assessment

The SSA collects, analyzes, and documents verification that the product, as implemented, meets the safety requirements established by the FHA, PSSA, and CCA processes. The SSA integrates the results of these processes and verifies that the implemented product meets all of the specified safety requirements and identified safety considerations including the safety of the overall aircraft. The difference between SSA and PSSA is that the PSSA is a validation process to evaluate the product architectures and derive safety requirements; the SSA is a verification process to verify that the implemented design meets the safety requirements as determined from the PSSA and CCA processes.

3.5.2 Validation and Verification

Validation is a process applied to ensure the correctness and completeness of requirements. Verification is a process applied to ensure correct implementation of the requirements. SAE ARP4754 Rev.A, Reference [26], gives generic guidelines for validation and verification, and presents example models for them. Based on this SAE Aerospace Recommended Practice (ARP), Section 7 presents validation and verification guidelines developed for SHM.

3.5.3 Configuration Management

The Configuration Management (CM) process identifies, tracks, and documents the development process data that includes:

- The functional and physical characteristics of all items that make up the product,
- Any changes made to these characteristics,
- Sufficient data about tests, facilities, and tools used during the development processes,
- Product requirements and design information, and
- Safety assessment, validation, verification, and certification data.

The development process data should be retrievable for later reference. The source of the data and the methods used to generate the data should be sufficiently controlled to allow regeneration of the same data and to provide archived evidence for future enhancements, problem resolution, or review by certification authorities.

The CM process facilitates the management of product baseline information along with any changes from the baseline that may be introduced to improve performance, reduce cost, or correct defects. The CM process facilitates systematic evaluation of any proposed changes and their anticipated impact on the entire system so that the product maintains its integrity over its lifetime.

3.5.4 *Product Architecture*

The product architecture provides a description of how entities join together to form a system; the entities can be items, components, and subsystems: an item is a hardware or software element having well-defined interfaces. The development of the product architecture and the allocation of requirements are tightly-coupled iterative processes. Section 5 discusses the main items from which a large number of SHM architectures can be constructed; Section 5 also presents examples covering architecture extremes including a very simple SHM architecture and an integrated safety critical one.

3.5.5 *Detailed Requirements*

The developer should state the SHM system requirements that include: performance, operational, customer, physical, interface, installation, maintainability, survivability, and derived requirements arising from design choices. The SHM requirements must be sufficiently correct and complete, satisfy the intended functions, adhere to integrity attributes derived from determined criticality levels and, comply with applicable airworthiness regulations and operating rules. Section 6 presents more details pertinent to SHM system requirements.

3.5.6 *Design, Test & Evaluation*

Existing aerospace standards and processes for the design, test, evaluation, and integration of aircraft avionics and equipment are applicable to SHM systems. If the sensors of an SHM system are embedded within a structural component, the standards and processes for the design, test, and evaluation of structures should be applied to the structural component; in this case, additional tests should be conducted to ensure that both the qualified avionics and structural component, when integrated, perform as intended. Flight tests may be required to demonstrate compliance with applicable airworthiness regulations.

3.5.7 *Instruction for Production, Installation, Maintenance & Operation*

After successful design, integration, and tests demonstrating complete and correct requirement implementations, the manufacturer develops instructions for production, installation, maintenance, and operations. The installation instructions can incorporate both generalized installation guidelines and specific instructions. The generalized installation guidelines could reference standard practices used in the installation such as electrical wire selection. The specific installation instructions would address more critical elements; these instructions should include procedures for determining the placement, installation, and post installation checkout of the avionics equipment; they should include instructions for commissioning tests on the installed system that ensure correct functionality and prove that the system does not unfavourably influence other aircraft systems.

3.6 SHM Production and Installation

During the development phase, only small number of systems are build, qualified, installed and flight tested to obtain design and installation approvals. Having approved the type design, the type designer may apply for a production certificate; also any manufacturer may apply for a production certificate if he holds appropriate approval certificate and right to its benefits under a licensing agreement with the type designer. Usually, the RA conducts progressive evaluation of the applicant's quality and production systems to verify that the applicant is capable of maintaining a quality assurance system, which ensures that only products and parts conforming to the approved design are released to service. Each produced system is subjected to acceptance and functional tests before the system delivery and installation to ensure manufacturing quality and correct functionality. The tests can include profile tests, e.g. a small number of temperature and vibration cycles, to ensure manufacturing quality and correct functionality under typical environmental conditions.

3.7 Instruction for Continued Airworthiness

The instructions include a plan to ensure continued airworthiness of those parts that could change with time or usage and include the methods used to ensure continued airworthiness. 14CFR Part 25 §25.1529 of Reference [3] describes the FAA regulatory requirements for the "Instructions for Continued Airworthiness" which must be written in English as a manual or manuals containing: maintenance manual, maintenance instructions, information to facilitate maintenance, airworthiness limitation section, and a section for Electrical Wiring Interconnection Systems (EWIS), see Section 8.2.4.

3.8 The Phases of the MOD Acquisition Process

Within the MOD Acquisition Operating Framework (AOF), the term CADMID is composed from the initial letters of the six acquisition phases shown in Figure 8, which are: Concept, Assessment, Demonstration, Manufacture, In-Service, and Disposal. Each phase involves executing a plan agreed in the previous phase, reviewing the outcome, and planning for the remaining phases.

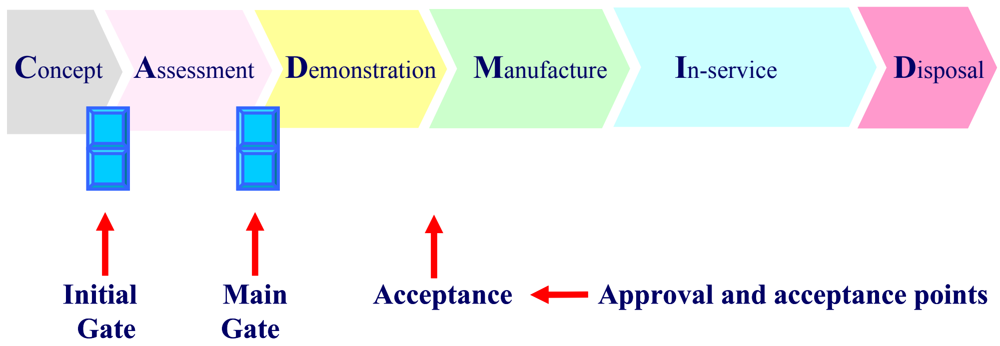


Figure 8: The UK MOD six acquisition phases

3.8.1 Phase 1: Concept

This phase starts by capturing, within a User Requirements Document (URD), the output (capability) that the military user requires from the system. The remaining activities of this phase include: form a delivery team; involve industry; identify technology and procurement options for meeting the requirement; obtain funding and agree a detailed assessment plan; outline subsequent stages, identify performance, cost and time boundaries within which it is to be conducted; initiate a Through Life Management Plan (TLMP); continuously monitor maturity and, when appropriate, submit an Initial Gate Business Case seeking approval for the Assessment Stage.

3.8.2 Phase 2: Assessment

This phase starts with the production of an initial System Requirements Document (SRD), defining what the system must do to meet the military user needs as stated in the URD. The remaining activities of this phase include: establish and maintain the linkage between user and system requirements; identify the most cost-effective technological and procurement solution; develop the SRD; assess trading time, cost and performance to identify the technological solution, reducing risk to an acceptable level; refine the TLMP, including detailed plans for the demonstration phase; continuously monitor maturity and, when appropriate, submit a Main Gate Business Case seeking approval for the project within tightly defined performance, time and cost boundaries.

3.8.3 Phase 3: Demonstration

The activities of this phase include: eliminate progressively the development risk and fix performance targets for manufacture, ensuring there is consistency between the final selected solution, and the SRD and URD; place contract(s) to meet the SRD; demonstrate the ability to produce integrated capability.

3.8.4 Phase 4: Manufacture

The activities of this phase include: deliver the solution to the military requirement within the time and cost limits; conduct system acceptance to confirm that the system satisfies the SRD and the URD, as agreed at Main Gate; transfer the lead customer function to the User, for equipment.

3.8.5 Phase 5: In-Service

The main activities of this phase include: confirm the availability of the system capability provided for operational use, to the extent defined at Main Gate, and declare the In-Service Date; provide effective support to the front line; maintain levels of performance within agreed parameters, whilst driving down the annual cost of ownership; carry out any agreed upgrades or improvements, refits, or acquisition increments.

3.8.6 Phase 6: Disposal

Carry out plans for efficient, effective, and safe disposal of the equipment.

3.9 The Decision Points of the MOD Acquisition Process

The early stages of a project lifecycle contain two major decision points – Initial Gate Approval (IGA) and Main Gate Approval (MGA). Sponsors and project teams are required to present Business Cases at both these gates to justify the project proceeding to the next stage. IGA is the first decision point, which occurs before the assessment phase and is considered to be a relatively low hurdle in the process. Industry must not be engaged formally prior to IGA. MGA can be obtained after the successful completion of the assessment phase and is the major decision point at which the solution and 'not to exceed figures' are approved. No manufacture or service contracts can be signed prior to approval. The approval categories of projects are divided into four categories, A, B, C, and D. Categorisation is based primarily on value, although other factors (novel and contentious issues for example), can lead to a project being moved into a higher category.

4 INTENDED FUNCTIONS OF SHM SYSTEMS

The potential intended functions of an SHM system can be stipulated from the definition of SHM and by examining the relevant regulatory structural rules. For example, the SHM intended functions should provide improved techniques for fatigue monitoring, exceedance monitoring, environmental monitoring, fatigue damage detection, environmental damage detection, and accidental damage detection. The SHM functions should provide maintenance, operational, and/or ownership benefits such as: reducing inspection costs and times, improving repair planning, optimising inspection intervals, reducing aircraft downtime, increasing residual values of used aircraft, reducing weight, and enabling life extensions. In words cited in the civil/military regulatory structural rules, the SHM intended functions should provide improvements in: inspection methods to detect FD, ED and AD, structural sampling and age-exploration programs, corrosion prevention and control programs, ASIP techniques, in-service usage monitoring, OLM/ODR methods, etc.

Figure 9 shows three groups of SHM potential functions, namely: damage detection, usage monitoring, and damage/usage monitoring +; the figure also indicates the outputs of these functions.

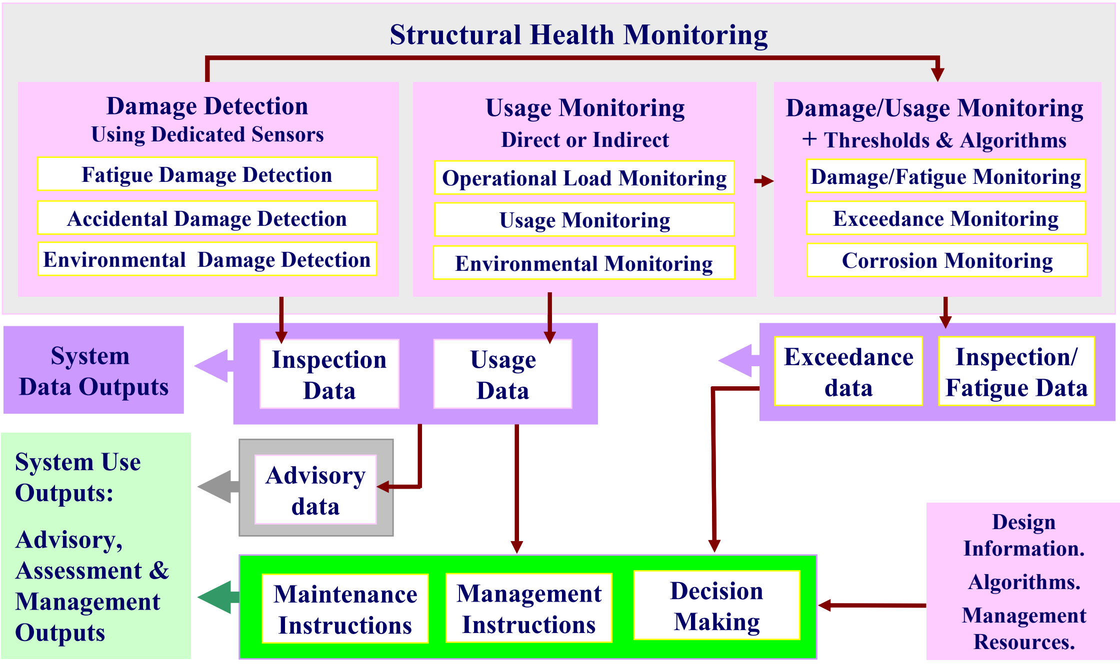


Figure 9: Potential Intended Functions of Structural Health Monitoring Systems

4.1 Damage Detection

The realization of damage detection functions requires implementing dedicated sensors that directly indicate the type, location, and/or the size of the damage.

Reference [61] has reviewed a wide range of sensor technologies that can provide damage detection capabilities; examples of these sensors are:

- Piezoelectric transducers that can hear Acoustic Emission (AE) resulting from growing metallic cracks or composite damage such as broken matrix and fretting of delaminated surfaces,
- Strain or fibre optic sensors that detect changes in strain fields caused by metallic or composite damages,
- Active piezoelectric transducers acting as transmitters and receivers: the receivers can

identify composite and metallic damages from changes in ultrasound bursts emitted by the transmitters at frequencies exceeding 100 kHz; the transmitters can emit Lamb waves or surface acoustic waves,

- Comparative Vacuum Monitoring (CVM) with polymer patches containing independent air-tight galleries bonded to a surface; as a crack propagates, the air leaking from one gallery to the next will indicate the presence of cracks, and
- Resistive sacrificial gauges that corrode away at the same rate as an exposed metal structure in the same location; the gauges detect corrosion from changes in their resistance caused by mass-loss.

The damage detection functions can provide the following three capabilities:

- Fatigue Damage Detection
- Accidental Damage Detection
- Environmental Damage Detection

4.2 Usage Monitoring

The usage monitoring functions can provide three main capabilities:

4.2.1 Operational Load monitoring

The 'operational load monitoring' function directly measures, or indirectly estimates, forces, stresses, or strains:

- Direct operational load monitoring involves determining the loads, stresses, or strains directly from measurements of sensors such as strain gauges and fibre optic sensors.
- Indirect operational load monitoring involves using algorithms to indirectly compute the loads, stresses, or strains from flight parameters such as speed, acceleration, and altitude.

4.2.2 Usage monitoring

Aircraft usage can be evaluated not only by monitoring operational loads, but also by monitoring significant usage events such as flight duration, start/end weight and number of pressurisations, and also by computing usage indices summarising, for example, the usage of control surfaces during landing operations.

4.2.3 Environmental Monitoring

Environmental monitoring involves using sensors to measure local environmental contents (parameters) that may have been produced by corrosion or would lead to corrosion; examples of these parameters are acidity (pH), humidity, local temperature, oxides, and time in wetness. The parameter monitored can include contextual information such flight routes and weather records, in the air and on the ground.

4.3 Damage\Usage Monitoring +

By using configuration data and specialized algorithms, if made available by the manufacturer, the information collected by the damage\usage monitoring capabilities can be converted to information providing refined capabilities such as fatigue monitoring, exceedance monitoring, and corrosion monitoring. Examples of the configuration data are material properties such as limit, proof and ultimate loads, and fatigue data such as crack growth curves and numbers of stress/strain cycles causing failure (S-N data).

4.3.1 Fatigue Monitoring

Fatigue monitoring involves using material characteristics and specialized algorithms operating on monitored loads, stresses, or strains to estimate accumulated fatigue damage, remaining safe life, crack sizes, or remaining times to critical crack sizes, see Appendix A for more details.

4.3.2 Exceedance Monitoring

Exceedance monitoring involves using thresholds and algorithms to capture events such as overloads, over-speeds, hard landings, parameter values outside design envelopes, and durations spent outside these envelopes.

4.3.3 Corrosion Monitoring

Corrosion monitoring involves using algorithms (corrosion prediction models) working on monitored environmental data to indicate corrosion, if it occurs, and determine the severity of corrosion.

4.4 The Use of SHM As Introduced by MSG-3

ATA MSG-3 has revised its recommended specifications to incorporate operation uses of SHM in aircraft maintenance, Reference [24]. ATA MSG-3 has introduced the term Scheduled SHM (S-SHM) and added S-SHM to a group of scheduled tasks to be accomplished at specified intervals; namely, the “inspection/functional check” group has been expanded to include S-SHM along with GV, DI, and SI. Reference [24] has indicated that SHM may be an option to check or watch for AD, ED, and FD where demonstrated to be applicable and effective. Following from the MSG-3 work, an Issue Paper (IP No: 105) has been submitted to further advance the definition of SHM and include automated applications; see Section 2.7 for more details.

4.5 The General Use of SHM Intended Functions

Figure 9 shows potential outputs of SHM systems and indicates how the system can be used. The damage detection capability produces inspection data such as crack sizes; the usage monitoring capability produces usage data; the damage\usage monitoring + capability uses specialized algorithms and thresholds taken from design envelopes to output information about fatigue, exceedance events, and corrosion. These outputs can be used as advisory information. If further design information, maintenance information, and management information are available, these outputs can be transformed to maintenance instructions, management instructions, or improved advisory information; for example, the availability of information about the maximum allowable crack sizes would allow triggering repair instructions if the crack sizes contained in the inspection data reach values close to critical. In other words, SHM systems can be used as advisory systems or as systems providing maintenance, operational, and/or design credits/benefits.

The following subsections define the words “credits” and “benefits” along with four categories of system use.

4.5.1 Credits and Benefits

Following from MOD work, the interpretations of the words “Credit” and “Benefit” can be generalized for SHM as follows:

- A benefit is a gain from an SHM application approved to improve maintenance practices, flight operations or designs, the failure of which will not result in “major” failure, "hazardous/severe major" failure, or "catastrophic" failure. A benefit does not constitute a credit.

- A credit is a given approval to an SHM application that alters or intervenes in industry accepted maintenance practices, flight operations or designs, the failure of which can result in a "major" failure, "hazardous/severe major" failure, or "catastrophic" failure. Such an application impacts the Safety Case (SC), requires safety analysis and regulatory/industry approval.

If SHM performs a task as an alternative equivalent of an approved task performed by existing equipment, then, the use of SHM may not require regulatory approval. Therefore, the following general definitions of the word benefits and credits are proposed for IVHM applications:

- A benefit is a gain from an application that (a) does not significantly change existing maintenance practice, flight operations, or design, (b) provides detection, monitoring, assessment, or management functions equivalent to those provided by existing approved applications or systems, and (c) delivers these functions at improved performance (e.g. faster and more accurate at reduced costs).
- A credit is a gain from an SHM application that (a) alters accepted maintenance practice, flight operations, or design (b) is approved by demonstrating compliance with airworthiness regulations and operating rules for each altered function, and (c) delivers the altered functions at improved performance.

By adopting these proposed definitions: (a) a system providing a benefit may only require the approval of manufacturers or their representatives, who would ensure that the system performs a task equivalent to that of approved equipment at a better quality; (b) a system providing a credit would require a certification project for approval.

4.5.2 *Advisory Systems*

An advisory system provides advisory information that can improve existing maintenance, management, airworthiness, and/or design practices; the advisory system does not change any existing maintenance, airworthiness, structural integrity, and design processes.

Advisory systems can be fitted on the entire fleet or on a fleet sample (a small number of aircraft) to achieve, for example, the following:

- Collect validation evidence to prove that an SHM task is an alternative replacement of existing task (benefits);
- Generate the evidence required to approve favourable changes in existing processes (credits);
- Generate requirements for advanced systems and new designs;
- Demonstrate that a system has attained the highest technology readiness level through in-service flights; and
- Gather evidence to conclude the development phase.

Advisory systems fitted on the entire fleet can generate information to assist in investigating incidents and undesirable events. The available information can facilitate systematic approval of tasks that deliver credits and benefits.

4.5.3 *Systems Providing Maintenance Credits/Benefits*

A system providing maintenance credits/benefits enables specific improved maintenance tasks that can replace existing tasks or can be considered as AMC with existing maintenance tasks such as inspections. Examples of such tasks are:

- Detect a falsely reported hard-landing event and eliminate the inspection required after the event;
- Detect structural damage at fixed scheduled intervals; and
- Indicate a need for planning preventive maintenance activities.

4.5.4 Systems Providing Operational Credits/Benefits

A system providing operational credits/benefits enables improved operational, management and structural integrity tasks that can replace existing tasks or can result in maintenance planning benefits such as increasing inspection intervals. For example, load and fatigue monitoring functions can be used to adjust scheduled task intervals or trigger crack inspection tasks.

4.5.5 Systems Providing Design Credits/Benefits

A system providing design credits/benefits can be sought to underwrite new designs and materials.

4.6 The SHM Elementary Functions

The SHM intended functions should be broken down into elementary functional tasks; by decomposing the SHM intended functions into elementary functional tasks, these guidelines would cover the wide spectrum of SHM systems and avoid the exclusion of an SHM system that only performs a part of an intended function or that performs SHM tasks across two functions. For example: the detection of damage location in a composite structure can be considered as an elementary task; the detection of the damage size can be another elementary task. Furthermore, practical consideration may impose requirements for performing other tasks such as producing maintenance or management instructions to directly act upon them without the need for consulting other systems; examples are inspection and repair instructions or instructions to change aircraft routes between short and long hauls. Each SHM elementary task, whether offered or imposed, should be clearly identified, its criticality assessed, validated, and verified.

Although SHM is an abbreviation of the words “Structural Health Monitoring”, these words have been used worldwide to indicate not only a monitoring elementary function but also other functions such as detection, assessment, and management tasks as defined in the following sections. One SHM system can be targeted at performing one elementary function; another system can be targeted at performing a number of elementary functions.

4.6.1 Detection

Detection involves finding with pre-defined quality the existence, type, location, and/or extent of structural faults (FD, ED or AD) such as crack, delamination, corrosion, erosion, and moisture absorption. The detection tasks require data from dedicated onboard sensors. A detection intended function can involve a number of elementary intended functions:

- Detection of fault existence: this elementary function only indicates whether a fault exists or not with pre-defined quality where the detection quality can be defined in terms of a detection probability along with acceptable “false positive”/“false negative” probabilities; “false positive” occurs when the system trigger a false alarm indicating the presence of a damage that does not exist; “false negative” occurs when the system fails to detect a damage that does exist.

- Detection of a fault location: this elementary function detects the damage location with pre-defined quality where the detection quality can be defined in terms of a pre-defined zonal and co-ordinates accuracies to guide, for example, subsequent fault repairs.
- Detection of a fault extent: this elementary function detects the fault size with a pre-defined quality where the detection quality can be defined in terms of Portability of Detection (POD).
- Detection of a fault type where the SHM system is targeted at detecting more than one fault type and correctly discriminates between them. However, the optimization of an SHM sensor technology is often targeted at detecting a specific fault type.

4.6.2 Monitoring

Monitoring involves maintaining with pre-defined quality regular surveillance over factors that can lead to or indicate structural faults; these factors include, for example, loads, usage, impact events, fatigue, and/or environments. The monitoring functions can be classified as direct and indirect monitoring functions:

- The direct monitoring functions use inputs from sensors that directly measure with pre-defined quality the factors of interests; for example, the loads can be monitored using data directly acquired from strain gauges or fibre optic sensors.
- The indirect monitoring functions implement algorithms that operate on a set of parameters acquired by onboard sensors to indirectly compute with pre-defined quality the factors of interest; for example, the loads can be accurately estimated from FDR parameters such as speed, acceleration and temperature.

Monitoring events such as impacts or landings that may be excessive and lead to overloads or damage can include a number of elementary intended functions:

- Event occurrence: the monitoring quality of occurrence can be defined in terms of occurrence probability along with acceptable “false positive”/“false negative” probabilities.
- Event location: the monitoring quality of location can be defined in terms of a pre-defined zonal and co-ordinates accuracies.
- Event magnitude (e.g. impact energy or overload value): the monitoring quality of event magnitude can be defined in terms of accuracy.

The threats of the monitored events to structural integrity are evaluated through assessment tasks.

4.6.3 Assessment

Assessment involves the use of detection and/or monitoring results along with design information and structural properties to determine the current structural status and generate, if required, instructions including inspection, repair, and replacement instructions. The design information and structural properties include, for example: design flight envelopes, fatigue data and algorithms, crack growth algorithms, design usage spectra, strength, toughness, and critical crack sizes.

In other words, the information required for assessment includes information directly generated by SHM (detection and monitoring information) and information held by other stakeholders (design/structural information). Only if the latter set of information is made available to the SHM system provider, the design of an SHM system having assessment functionality will be possible.

Assessment can also include prognostics where the future conditions of the structure are forecasted and/or the remaining life of critical components estimated.

4.6.4 Management

Management involves the use of detection, monitoring and assessment results combined with information about available resources to plan fleet utilization or plan maintenance activities.

It is worth mentioning that the management function is not an elementary intended function for SHM; it is anticipated that most of the management plans and instructions are issued by a structural health management system not SHM. However, practical considerations may impose on SHM systems requirements for performing management elementary functional tasks to directly act upon them without the need for consulting other systems. The validation and verification activities should carefully cover these management elementary functions if they are imposed on SHM systems. The extent of the elementary management functions depends on the availability of information held by the operator and maintainer. If this information is held within dedicated management and maintenance systems, the imposed requirements would define the interfaces that would allow access to the information; the imposed requirements would also define the optimum place/system from which a management task is triggered.

5 SHM SYSTEM ARCHITECTURE

5.1 Introduction

The word “architecture” is used to indicate a general description of how entities join together to form a system; it indicates an abstract description of the entities of a system and the relationships between these entities necessary to satisfy constraints and requirements; the system architecture establishes the structure and boundaries within which designs of entities are implemented to meet established requirements including functional, safety, technical, and performance requirements. The system entities can be items, components, and subsystems: an item is a hardware or software element having bounded and well-defined interfaces; each component is made of a number of items and each subsystem is made of a number of components.

The most important aspects of the system architecture are physical and functional aspects: the physical architecture describes the system by showing how it is broken down into items, components, and subsystems (a representation of physical items and their interconnections); the functional architecture identifies the allocated functional and performance requirements (a partially ordered list of activities or functions that are needed to accomplish the system requirements). In practice, the development of system architecture and the allocation of requirements are tightly-coupled, iterative processes. The other aspects of architecture are technical and dynamic operational aspects. The former is an elaboration of the physical architecture that comprises a minimal set of rules governing the arrangement, interconnections, and interdependence of the architecture entities. The latter is a description of how the entities operate and interact over time.

System entities: items, components and subsystems

- **An item is a hardware or software element having bounded and well-defined interfaces.**
- **A component is made of a number of items.**
- **A subsystem is made of a number of components.**

Architecture

A description of how entities join together to form a system; a description of the entities and the relationships between them necessary to satisfy constraints and requirements.

Physical architecture

How the system is broken down into items, components and subsystems (a representation of physical items and their interconnections).

Functional architecture

Allocated functional and performance requirements (a partially ordered list of activities or functions that are needed to accomplish the system requirements).

Figure 10: The meaning of architecture

The various SHM intended functions can be delivered through a large number of potential system architectures. For a required intended function, an optimum system architecture can be selected by assessing the feasibility of meeting safety requirements and evaluating factors such as technology readiness level, development timescale, through life costs, and weight. Constraints on the choice of the SHM architecture can be imposed by established architectural features of other aircraft systems. More constraints are expected for legacy aircraft. Therefore, the determination of the optimum SHM architecture would also require careful examination of the existing architectural features of other aircraft systems. Selecting the SHM system architecture is an essential task that allows sufficient and correct allocation of requirements. Many SHM intended functions can be considered and various architectures exist for each intended function. Therefore, the following sections discuss the main components from which a large number of architectures can be constructed. Then, examples covering architecture extremes are presented: a very simple architecture, and an integrated safety critical architecture.

5.2 Potential Main Physical Components of SHM Systems

Figure 11 shows the main physical entities of candidate SHM architectures. Following from the definition of SHM, the sensors and their connections (e.g. connectors and wires) are physical entities that must be built into each individual aircraft. The remaining physical entities can be elements of either an airborne system or ground-based equipment; examples are: power components, signal conditioning and processing components, storage media and communication components. A number of SHM systems would require storing their current and historical data on ground-based databases and performing further tasks through SHM ground stations. For such systems, the development of the ground stations should be seriously considered and should not be obscured by other important entities. The main entities of SHM are described in the following subsections.

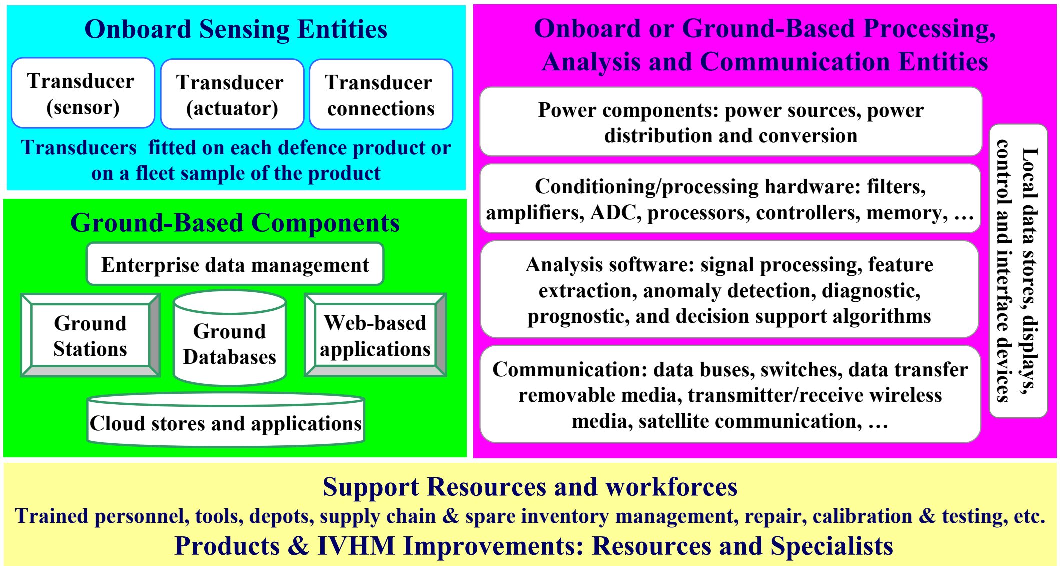


Figure 11: Some of the main physical components of IVHM systems

5.2.1 *Airborne Transducers and Connections (Sensors and Actuators)*

A transducer is a device that converts one form of energy to another. Examples of the energy forms are: kinetic, electrical, mechanical, magnetic, chemical, acoustic, thermal, and light energy. A basic sensor is a transducer that observes the environment and enables the detection of a physical condition. A basic actuator is a transducer that influences the environment and enables the delivery of a required action. Often, the sensors and actuators contain additional components packaged with the basic transducers to directly detect conditions or deliver actions.

The sensors are classified by the energy forms they observe, e.g. acoustic sensors and optical sensors. They are also classified by their components that are sensitive to changes in the observed energy; examples are: resistive sensors, capacitive sensors, and inductive sensors. They can be classified by combination of their responses, sensitive components, and observed energy; examples are: resistive displacement sensors and thermoelectric sensors. The common actuator types are hydraulic, pneumatic, and electromechanical actuators. Devices such as displays and electrical converters are also actuators.

Consider measurements taken to monitor and/or detect subjects such as loads and cracks: a subject can be a phenomenon, a body, or a substance; it can be a passive subject or an active subject. A passive measurand is a quantity value of a passive subject that does not notably change or emits/absorbs measurable energy during a finite measurement period. Such a subject can be measured on the ground while the aircraft is stationary and the engines are shut-down; examples are: cracks, corrosion damage, erosion damage, plastic deformations, and impact damage. A subject such as corrosion may interact with the environment and absorb/emit chemical energy leading to slow changes in the measurand after a long period that far exceeds the measurement finite period; therefore, corrosion can be considered as a passive subject. An active measurand is a quantity value of an active subject that does change or emits/absorbs measurable energy during a finite measurement period. For aircraft applications, such a measurand (quantity value) can be adequately portrayed and measured only when the aircraft is operated; examples are loads, strains, usage parameters, exceedance events, and growing cracks.

Sensors can be designed as passive or active devices. A passive sensor can directly respond to a measured subject; it can feel the energy of an active subject; examples are thermocouple and photodiode that sense temperature and light. Passive sensors can measure subjects activated by aircraft operational loads (e.g. cracks growing under varying load cycles). An active sensor emits energy (excitation) and then measures changes caused by the measured subject as a result of the excitation; the active sensor requires more input power to generate the required excitation. So, the active sensor is made of two elements: (a) a basic actuator that delivers excitation/energy and (b) a basic sensor that observes changes caused by the measured subject. For some SHM active sensors, the main two elements are separated by the expected locations of the target subject (e.g. cracks). The basic actuators and sensors can also be packaged as one unit that sends an excitation and measures the reflection of the excitation from the same position.

Figure 12 illustrates the potential differences between the wide ranges of sensors that can be used for SHM applications: these differences can include physical differences (shape, size, etc.), functional differences, installation requirements, etc.

- **The available SHM sensors differ in size, material, functionality and shape.**
- **Most of the sensors require power; few of them do not.**
- **The majority of the sensors require media, often wires, to channel their data.**
- **The sensors may be fastened to the structure using brackets/bolts; they may be glued to the structure, embedded within the structure or integrated with the structure.**
- **It is possible to develop a sensor that does not need power or data channels; changes in the condition of such a sensor, i.e. cracks developed in the sensor or changes in its colour and shape, may indicate the health of neighbouring structural components.**
- **Connectors would be required to interface the power and data channels with airborne or ground based subsystems such as data bus terminals/switches, processing units, etc.**

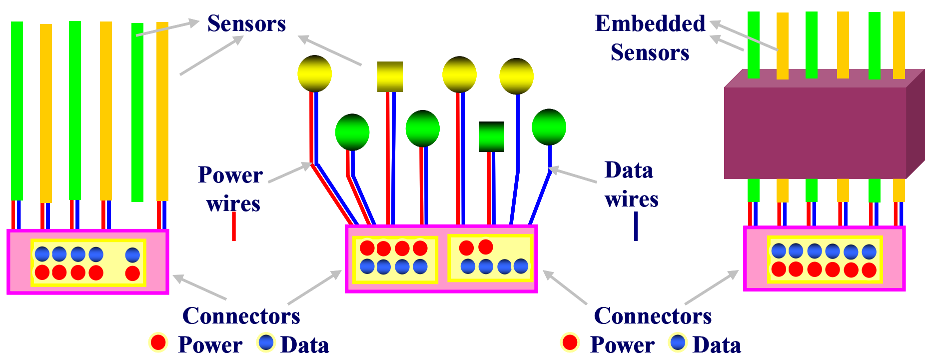


Figure 12: Potential differences between wide ranges of SHM sensors

The airborne sensors and their connections should survive their local environments during any foreseeable aircraft operation over specified failure-free periods. During these periods, they should perform their allocated functionality with quality consistent with the system intended functions. If the sensors/connections are embedded in the structure, their failure-free life should exceed the economic life of the structure or should be repairable without the need for replacing expensive structural items (e.g. they might be a part of a replaceable, inexpensive structural assembly).

Ground-based equipment may power SHM sensors and transfer, process, interrogate, and store the sensor data. In this case, only the sensors along with connection wires and connectors would be fitted in the aircraft as illustrated in Figure 13. However, it is likely that the airborne system would also include components such signal conditioning units, transmitters, and receivers.

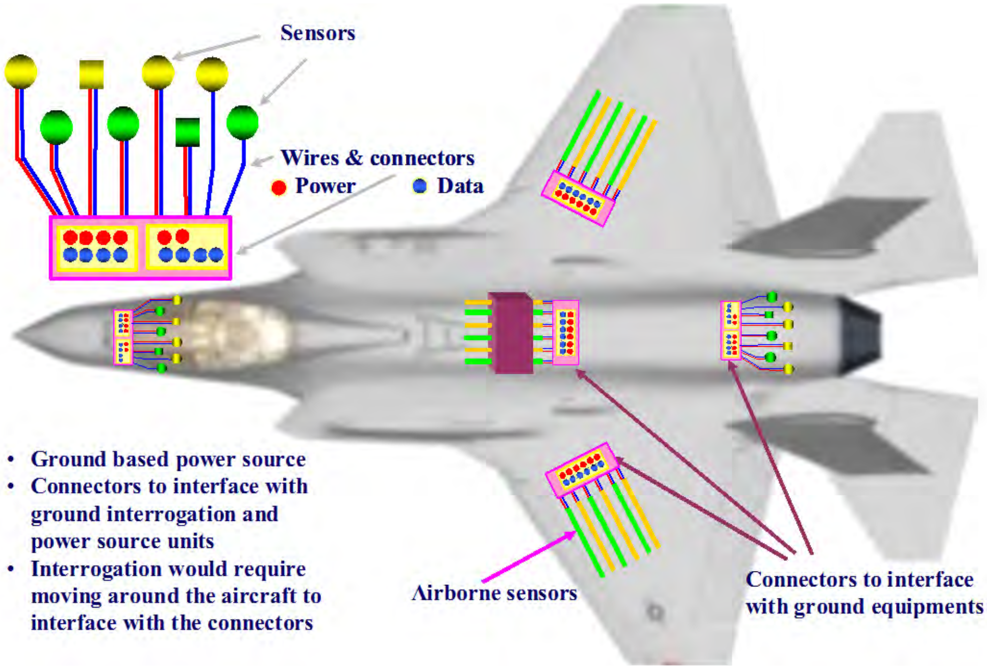


Figure 13: Airborne sensors and connections to ground-based equipment

5.2.2 Power Components

The potential power sources of SHM systems can be one of the following: aircraft power sources, ground-based power sources, batteries, and energy harvesting devices.

The aircraft primary power distribution system consolidates the inputs from power sources that can include: (a) main generators driven by the engines (b) a ground power source that supplies power to the aircraft systems during maintenance prior to departure and after arrival (c) an Auxiliary Power Unit (APU): a small independent gas turbine that supplies engine starting power and emergency power to primary electrical systems in the event of main power system failures (d) a Ram Air Turbine (RAT): an air-driven turbine is deployed to furnish the crew with limited power to fly the aircraft while attempting to restore failed primary generators. Batteries are used to assist in damping transient loads in DC systems, provide power in system start-up modes when other power sources are not available, and provide a short-term power source during emergency conditions while alternative sources of power are being brought online.

Typically, the main power generators feed a power bus with a 3 phase (3Ø) 115V alternating current (ac) 400 Hz power. Devices such as inventors and Transformer Rectifier Units (TRU) convert 3Ø ac to 1Ø ac and direct current (dc). Typical converted power sources can include: a 115/200Vac 400Hz bus, a 115Vac 60Hz bus, a 28Vdc bus, and 270Vdc bus. Power distribution/switching are achieved by devices such as distribution panels, thermal circuit breakers, electro-mechanical relays, solid state power controllers, and electromagnetic contactors with built-in current sensing and control electronics. Generally switching devices such as contactors and relays work in similar fashions: for example, a contactor is magnetically held in a preferred position until a signal is applied; alternatively, a signal may be continuously applied to the contactor to hold the contact closed, and removal of the signal causes the contacts to open. SHM system components may draw their required power from available aircraft buses. The SHM components may require dedicated conversion devices to transform available power levels to levels specific to them. If the SHM components receive power from the aircraft power resources, a distribution and protection network to the terminals of the components should be provided and the characteristics of the aircraft power resources should be maintained according to standards such as MIL-STD-704F and MIL-STD-461E.

The SHM system components may acquire their required power from their own power sources, for example: ground-based power sources, batteries, or energy harvesting devices. Several energy harvesting devices are emerging: these devices capture the energy dissipated in the environment as vibration, heat, light, or particle flow; some devices store the captured power in media such as batteries.

5.2.3 Signal Conditioning, Processing, Storage and Display

Usually, devices including filters, amplifiers, and Analogue to Digital Converters (ADCs) condition the SHM signals: the filters mitigate noise effects and remove from the signals undesirable frequency bands; the amplifiers and ADCs magnify and digitize the signals. Then, software including signal processing and SHM algorithms transforms the digitized signals into SHM detection and monitoring information. The software and related configuration data can be loaded into non-volatile memories and executed by a computer processor to produce the required information; alternatively, the software can be embedded into special-purpose electronic chips (computer chips) capable of executing the software and computing the required information. The SHM information can be stored or transferred (transmitted) to a required destination/media.

Components such as filters, amplifiers, ADCs, computer chips, storage media, embedded software, transmitters, and receivers (data ports) can be integrated into a subsystem including a control unit that act upon commands sent to the controller to route/switch the power and instruct the subsystem to start/stop acquiring data. The integrated subsystem can interface with a number of sensors and is often called Remote Data Concentrator (RDC) or Remote Interface Unit (RIU), Figure 14. .

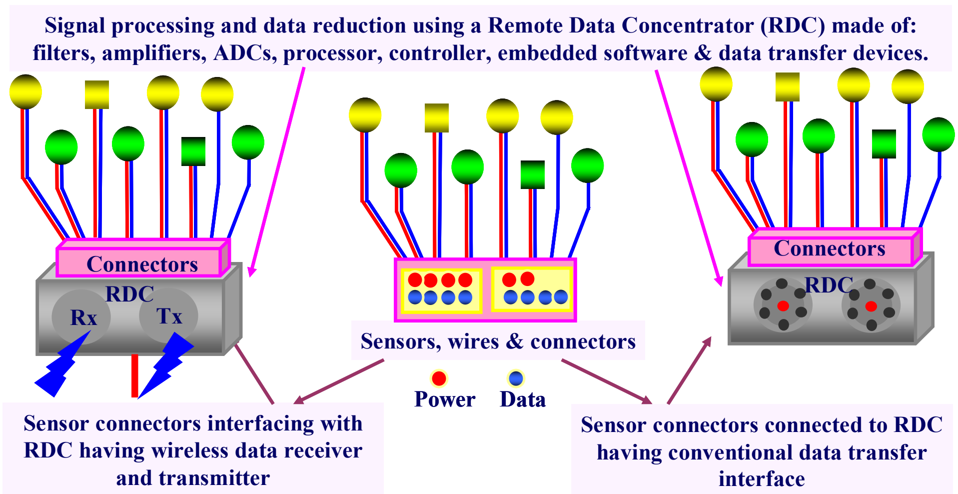


Figure 14: Interfacing sensors, their wires & connectors with RDC

The design of such a subsystem entails a trade-off between: (a) the extent of processing and, hence, the amount of data to be transferred on one hand, and (b) the weight of the subsystem and the complexity of its embedded software on the other hand.

The RDC may only perform basic signal processing and data reduction tasks; in this case, a ground station or an aircraft central processor would perform the remaining data processing tasks. Interface devices can display processed SHM results; they can also be used to upload the SHM software and its configuration data, or to upgrade the system with enhanced software and updated configuration data.

5.2.4 *Communication*

The communication media between two devices can be wires with two interface ends compatible with the data ports of the two devices; the ends of the interface of wireless communication are a transmitter (Tx) and a receiver (Rx). The SHM developers can be forced to follow existing standard specifications to interface with existing airborne or ground-based devices that comply with these specifications. Even if the SHM system is entirely independent of these existing devices, following industry accepted specifications would allow interoperability and facilitate cost effective integration of components manufactured by different providers. The following paragraphs briefly describe examples of industry accepted specifications.

5.2.4.1 ARINC 429

The most common standard specifications used in legacy aircraft communication are those of Aeronautical Radio, Inc (ARINC) 429. ARINC 429 provides specifications for a unidirectional data bus; it defines hardware and data formats (word structures and protocol) necessary to establish the communication. The hardware is a twisted shielded pair data bus (point-to-point wiring) connecting a single transmitter (source) with 1 to 20 receivers (sinks); each receiver continually monitors for its applicable data, but does not acknowledge receipt of data. If a transmitter requires the receiver to acknowledge receipt of data, another twisted pair data bus (channel) will be required; a receipt handshaking is performed using a particular word style communicated via the second data bus. ARINC 429 words are 32 bits in length; most messages consist of a single data word containing 24 bits for the actual information and 8 bits for a label describing the data. Transmission of data may be at either a low or high speed (12.5 or 100 kHz).

5.2.4.2 MIL-STD-1553

Published by the US DOD, the standard MIL-STD-1553 defines the characteristics of a serial data bus that uses Time Division Multiplexing (TDM) for the transmission of information from several sources through a single transmission media: by staggering sampled data from the sources and forming a pulse train, the communications between the different avionic boxes takes place at different moments in time. MIL-STD-1553 has been applied to several civil and military applications. MIL-STD-1773 is a version of MIL-STD-1553 that uses optical cabling. The UK had issued Def Stan 00-18 (Part 2) and NATO had published STANAG 3838 AVS, both of which had been versions of MIL-STD-1553B. The standard defines four elements: transmission media, remote terminals, bus controllers, and bus monitors, Figure 15.

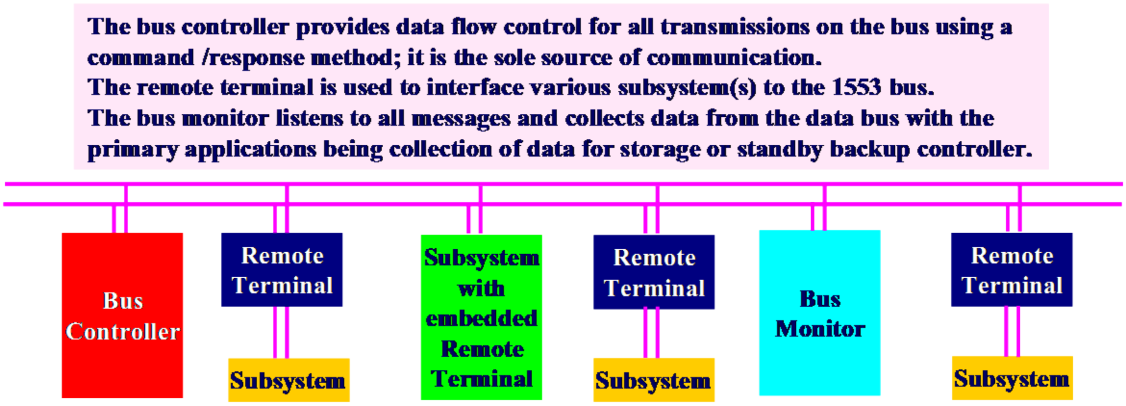


Figure 15: MIL-STD-1553 communications

The transmission media is a twisted shielded pair line consisting of a main bus and a number of stubs, one stub for each terminal connected to the bus. The main data bus is terminated at each end with a resistance. A remote terminal has all the electronics necessary to transfer data between the bus and data source(s)/subsystem(s). It is capable of receiving and decoding commands from the bus controller and responding accordingly. It is also capable of buffering a message, detecting transmission errors, performing validation tests, and reporting the status of the message transfer. It only responds to commands received from the bus controller within a very small time (speaks only when spoken to); often, modern remote terminals are capable of providing status information to the originator. An embedded remote terminal consists of interface circuitry located inside a sensor or subsystem directly connected to the data bus. The bus controller issues commands for the transfer of data or the control of the bus. Typically, the bus controller is a function that is contained within a computer such as a mission computer. A bus monitor is a terminal that listens to the exchange of information. A monitor may collect all or some of the bus data. It can act as a recorder; in this case, the subsystem is typically a recording device. The bus monitor can also act as a terminal functioning as a back-up bus controller; in this case the subsystem is the computer.

Figure 16 illustrates how SHM sensors can be connected to aircraft resources including a standard data bus, power sources, and central computing platform.

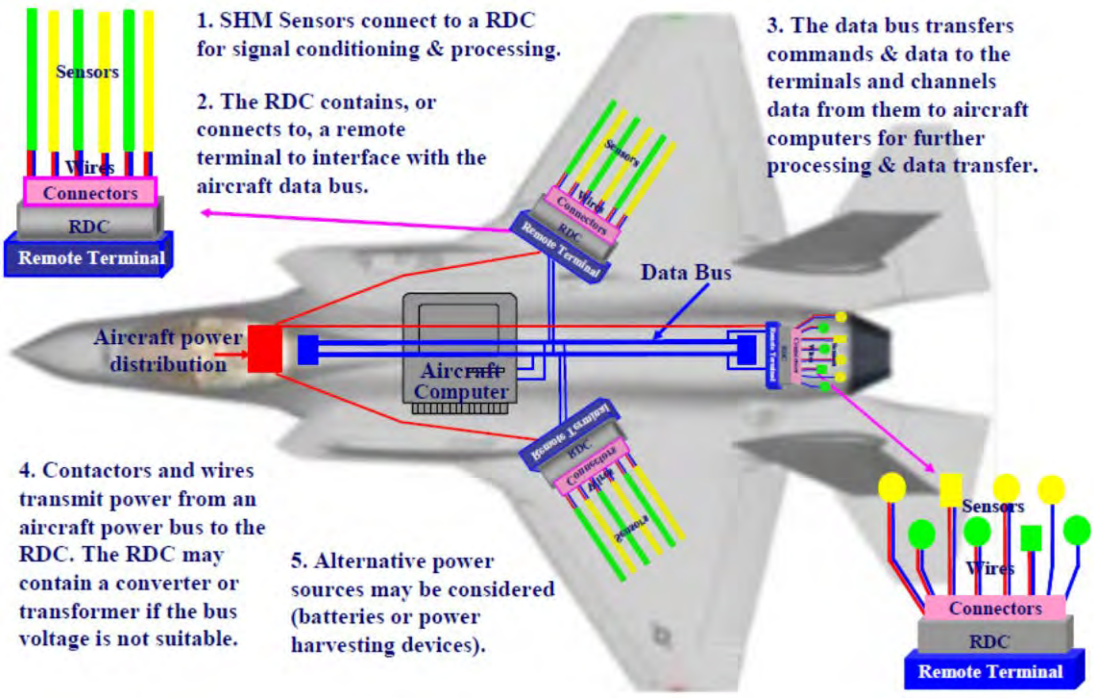


Figure 16: SHM communications via 1553 data bus

5.2.4.3 AFDX/ARINC 664

The Avionics Full Duplex Switched Ethernet Network (AFDX) standard was originally defined by Airbus; meanwhile, the same standard existed as the ARINC 664 and Boeing has based the B787 Aircraft Data Network (ADN) on ARINC 664 with some minor extensions. AFDX is also used in modern aircraft such as A400M, A350, AW101, AW149, and Bombardier CSeries. Prior to the advent of AFDX, the three main ADN standards were ARINC 429, MIL-STD-1553, and ARINC 629 with a maximum bandwidth of 100Kbps, 1Mbps and 2Mbps, respectively. ARINC 629 was introduced by Boeing for B777 and required custom hardware at more costs; therefore other manufactures did not openly accept the ARINC 629. AFDX is an ADN for safety-critical applications that allows for transfer rates of either 10 or 100Mbps between End Systems (ESs) over either a copper or fibre transmission medium using switches. Unlike a network hub that transmits received signals to all connected ESs, a switch receives a message from any connected ES and then transmits the message only to those connected ESs for which the message was meant. AFDX is based on the conventional non-deterministic Ethernet standard IEEE802.3 with extensions to ensure a deterministic behaviour through traffic control and, ensure a high reliability through redundancy allowing the transmission of the same data at the same time through two channels by duplicating the connections (wires) and switches. The configuration of the switch establishes the logical communication links between ESs and allows the switch to police the bandwidth allocated to each communication link; if the switch detects that the bandwidth of a communication link is exceeded, data is discarded and not forwarded until the bandwidth regains its specified limits. End-Systems exchange data through Virtual Links (VLs): a VL defines a unidirectional (logical) connection from one source ES to one or more destination ESs. An AFDX network can define a huge number of VLS (64k), each identified by a 16-Bit identifier in the Media Access Control (MAC) destination field of Ethernet frame; each ES can support multiple VLs, a switch performs traffic policing on each VL; see the illustration of Figure 17.

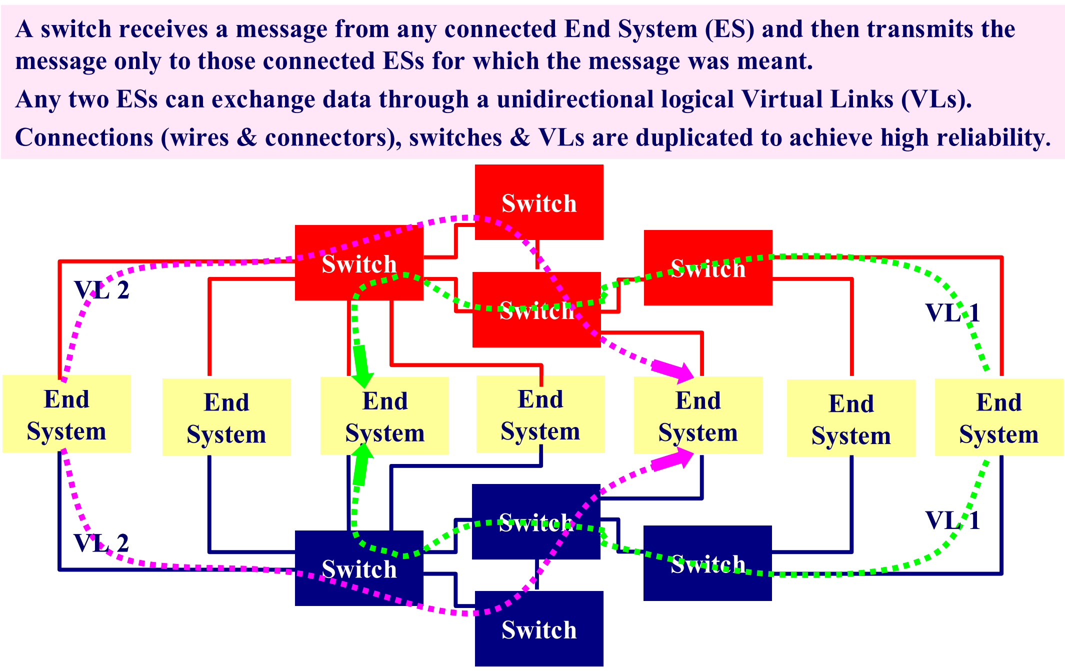


Figure 17: Avionics Full Duplex Switched Ethernet Network communications

For SHM applications, the SHM systems are the end systems that would, for example, have RDCs with Ethernet interfaces; see the illustration of Figure 18.

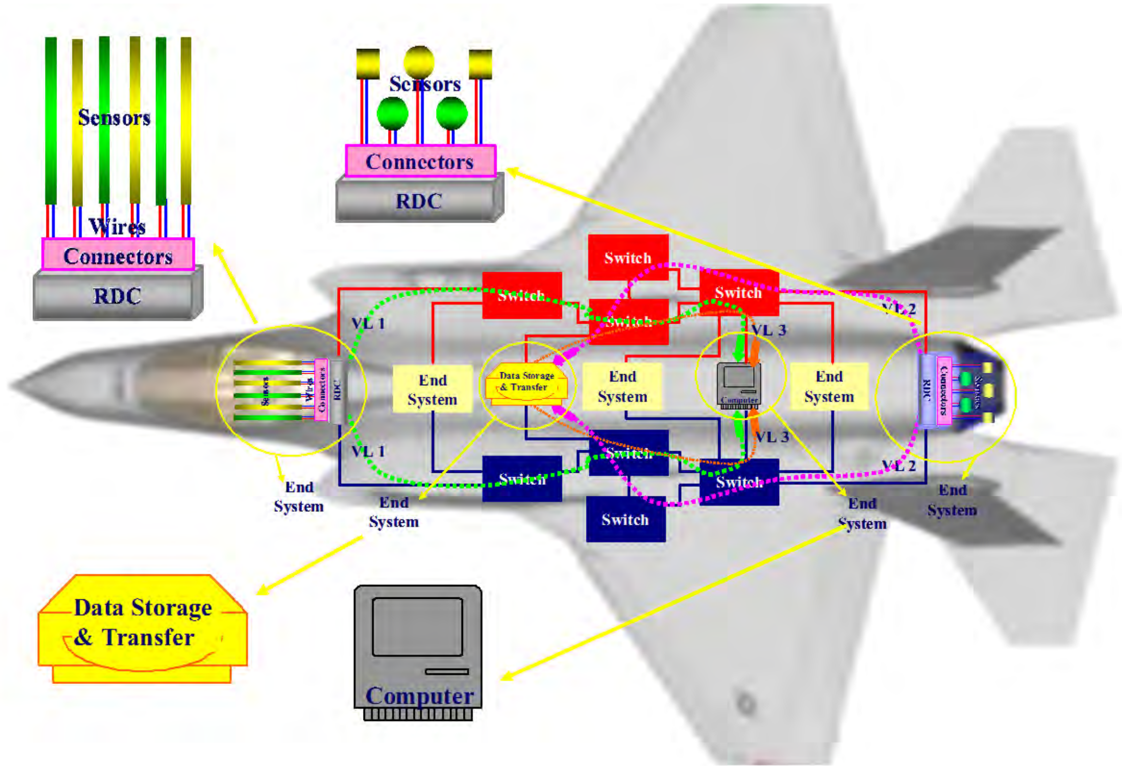


Figure 18: SHM communications via AFDX

IEEE has also introduced standards for Power over Ethernet (PoE) technologies; the standards describe a system that pass electrical power safely, along with data, on Ethernet cabling; for example, the IEEE 802.3af 2003 and 2009 PoE standards provide up to 15.4W and 25.5W of DC power.

5.2.4.4 Wireless Communications

A typical passenger aircraft such as B747 contains over 220,000m of wire weighing 1600kg. Alternative solutions to conventional copper wiring include aluminium which is used for over 50% of the A380 500,000m of wiring. Wireless alternatives would further reduce weight and complexity. Wireless transmitters and receivers can establish communications between aircraft SHM components and/or between airborne and ground-based components. Wireless technologies have faced performance and certification challenges including aircraft electromagnetic interference, wireless system coexistence issues, and wireless system security challenges. Boeing abandoned its plan to install a wireless in-flight-entertainment system on 787 because of technology performance problems and the lack of bandwidth spectrum in some parts of the world.

The Radio Frequency (RF) spectral bands range from 3-30Hz Extremely Low Frequency (ELF), to 300MHz-3GHz Ultra High Frequency (UHF), to 3-30GHz Super High Frequency (SHF) to 30-300GHz Extremely High Frequency (EHF). Parts of the RF spectrum are referred to as unlicensed, license-free, or unprotected spectrum. The unlicensed spectrum has rules pre-defined for the RF systems and their deployment methods: potential interference is mitigated by these technical rules; any entity (person or organization) that does not infringe upon the rules can develop and deploy RF applications at any time for either private or public purposes. The other parts of the RF spectrum are protected: they are controlled for military use and for public safety and commercial services; only an authorized entity can use the licensed spectrum. Countries have varying rules for unlicensed spectrum applications and they allocate different frequency bands for protected spectrums. So, the wireless system coexistence issues arise from the following: worldwide spectrum allocated specifically for fly-by-wireless systems does not exist; unlicensed spectrum is shared in uncontrolled manner; if unlicensed spectrum is used for fly-by-wireless systems, they must coexist with other less critical systems. The S band as defined by IEEE ranges from 2 to 4GHz: a large number of Aircraft S band transmitters operate from 2.1 to 2.45GHz; sharing this frequency band may lead to serious interference problems. "Line of sight" is another issue when implementing a low power wireless data link from one part of the aircraft to another; if the line of sight from transmitter to receiver is interrupted, the quality of the data link may suffer. Furthermore, existing regulations and guidance do not explicitly address system security; however, some wireless systems may require addressing security issues by identifying security threats and introducing risk mitigation techniques that prevent unauthorized access or modification of data or software.

Whilst considerable development efforts have been targeted at addressing the above issues, there are many certified aircraft wireless RF systems; most of these are aircraft communication, navigation and surveillance radio systems, primarily air-to-ground or air-to-satellite radio systems; some of these certified systems support critical aircraft functions, and therefore, operate in an internationally protected frequency spectrum. FAA and EASA have also certified less critical aircraft wireless RF systems such as wireless smoke and fire detection systems, passenger wireless network systems, cabin emergency lightning systems with wireless controls; typically, these systems operate in unlicensed spectrum. The development and certification of these systems have been achieved by complying with existing regulations and following existing standards and guidelines; in other word, there are no specific regulations for aircraft wireless systems. Generally, the wireless system developers should demonstrate that: the wireless systems do not affect other aircraft systems; existing aircraft transmitters or passengers electronic devices do not interfere with the wireless systems.

Examples of potential SHM airborne wireless communication solutions include: (a) IEEE 802.15.1 (Bluetooth) having 1-100m range, 100mW power, 24Mbps, and 2.4GHz (b) IEEE 802.11 (WLAN or WiFi) having 1-300m range, 100mW power, 11Mbps and 2.4GHz, (c) IEEE 802.15.4 (ZigBee) having 1-400m range, 30mW power, 0.25Mbps and 1-2.4GHz. Figure 19 illustrates wireless communications between self powered RDCs and central computing platform; the power sources can be batteries or energy harvesting devices.

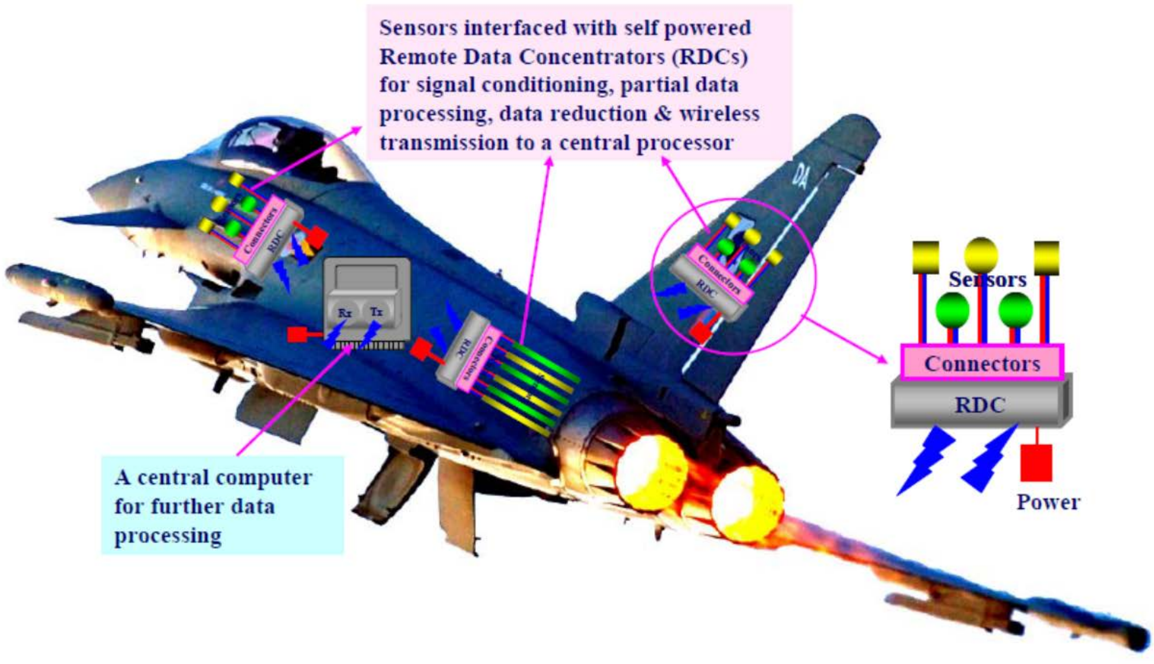


Figure 19: Wireless transmission of RDC data to a central processor

5.2.5 *Open System Architecture (OSA)*

OSA would allow qualified third parties to add, modify, replace, remove, or provide support for a component of a system, based on open standards and published interfaces for the component of that system. Generally, OSA main objective is modular designs based on standards and published interfaces, with loose coupling and high cohesion that allow for independent acquisition of system components; any changes to one module should not necessitate extensive changes to other modules, and hence, facilitate module replacements; the modules should be characterized by assignments of identifiable discrete functionality (high cohesion).

The International Organization for Standardization (ISO) has published ISO-13374 "Condition monitoring and diagnostics of machines": ISO-13374 Part 1 presents general guideline covering "Data interpretation and diagnostics techniques"; ISO-13374 Part 2 & Part 3 cover data processing and communication. ISO-13374 defines the following six functional blocks for condition monitoring and diagnostics: Data Acquisition (DA), Data Manipulation (DM), Health Assessment (HA), State Detection (SD), Prognostics Assessment (PA), and Advisory Generation (AG).

Open Systems Architecture for Condition Based Maintenance (OSA-CBM) is an implementation of ISO-13374 specification. OSA-CBM was initially developed in 2001 by an industry led team partially funded by the US Navy through a Dual Use Science and Technology (DUST) program. The team included industrial, commercial, academic, and military members from organizations including: Boeing, Caterpillar, Rockwell Automation, Rockwell Science Center, Newport News Shipbuilding, Oceana Sensor Technologies, the Penn State University, and the standard body "Machinery Information Management Open Standards Alliance" (MIMOSA) who has published the OSA-CBM standards and its updates ever since.

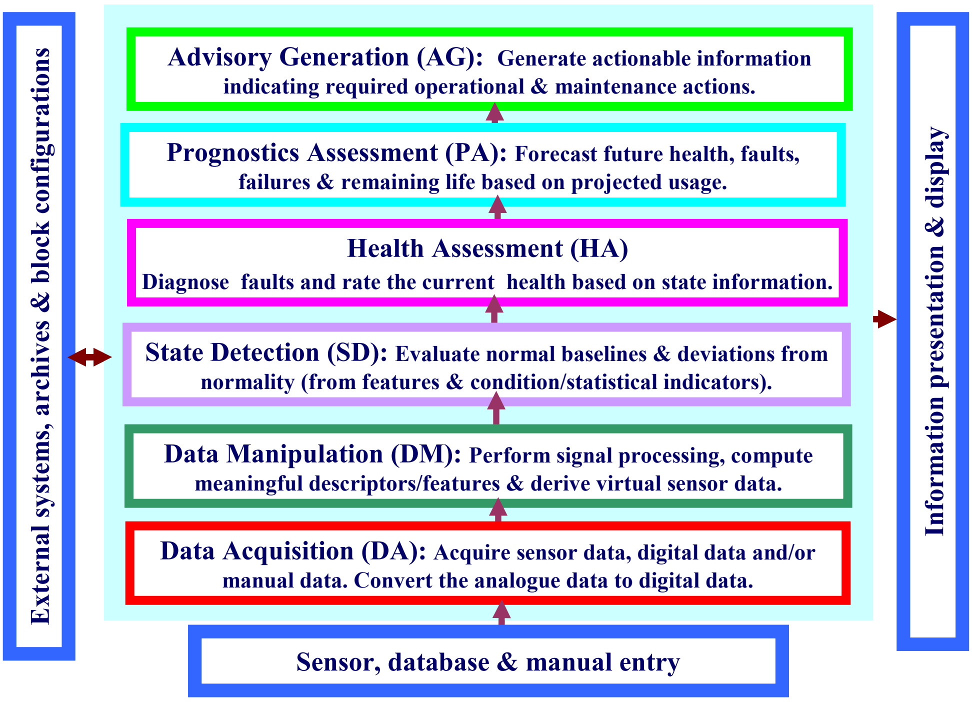


Figure 20: ISO-13374 definitions used for OSA-CBM

The OSA-CBM implementation of ISO-13374 specification adds data structures and defines the interfaces between the functional blocks of ISO-13374. Another standard published by MIMOSA is OSA for Enterprise Application Integration (OSA-EAI): a standard that defines data structures for storing and moving collective information about all aspects of platforms, systems, and subsystems including information about their health, physical configuration, reliability, condition, and maintenance.

5.2.6 *OSA Consideration for SHM Systems*

It is worth emphasizing that the functional blocks of OSA-CBM are implementation blocks and should not be confused with the SHM intended functions or elementary functions. The design of SHM systems should, as far as possible, adopt an OSA approach which is not necessarily the OSA-CBM approach: the SHM system should consist of modules designed based on industry accepted standards and interfaces that allow for the following:

- Independent acquisition of modules,
- Independent affordable means for module improvements,
- Independent access and use to the system data and results, and
- Improved obsolescence management.

For OSA-CBM, the following remarks should be considered:

- As mentioned above, the functional blocks of OSA-CBM are implementation blocks and should not be confused with the SHM intended functions or elementary functions.
- OSA-CBM is not a hierarchy: SHM elementary and intended functions can be implemented over a subset of the OSA-CBM functional blocks.
- OSA-CBM standard does not specify data repository. Therefore, SHM data can be archived in any repository including those of OSA-EAI.
- The HA module of OSA-CBM can provide error codes to external applications.
- The SHM software and hardware components need not to be broken down to exactly six items that correspond to the OSA-CBM blocks. They can combine more than one block and form open module with interfaces that comply with standards including those of OSA-CBM specifications.

5.3 Integration of SHM within the Aircraft and its Support Systems

This section considers the architectural choices that would enable the integration of SHM within the aircraft and its support systems. The section considers three military aircraft types: aircraft performing autonomous operations, remotely controlled aircraft, and manned aircraft. Whilst the section focuses on military applications, the high level architectural choices for military manned aircraft are almost the same as those for civil aircraft.

For new aircraft designs, SHM may be considered at early design stages and, hence, SHM may influence the aircraft design and their support systems, Figure 21. The extent of the SHM airborne items/functions would directly influence the designs of aircraft structures and airborne systems.

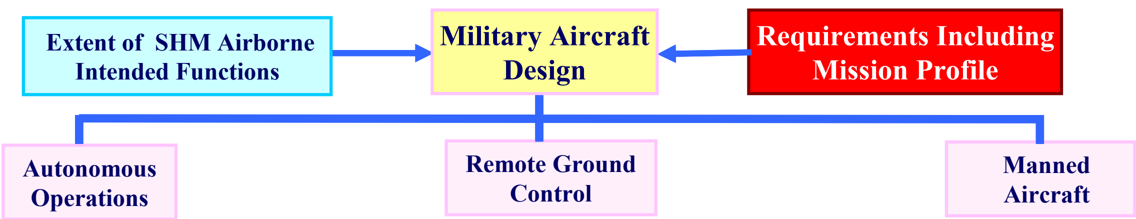


Figure 21: Military Aircraft Types

The successful integration of SHM within military support systems requires not only the analysis of the aircraft architecture, but also requires the analysis of the architecture of the support systems. Then, each SHM elementary function can be integrated with the optimum number of architectural items, airborne and ground-based items, that may include Remote Interface Units (RIU), data buses, central computing units, and other avionic or IVHM items. The SHM elementary functions are sensing, monitoring, detection, assessment, and decision making/management:

- Sensing involves collecting data from airborne sensors.
- Monitoring involves the use of sensed data to maintain with pre-defined quality regular surveillance over factors that can lead to or indicate structural faults. These factors include, for example, loads, usage, impact events, fatigue, and/or environments.
- Detection involves the use of sensed data to find with pre-defined quality (diagnose) the existence, type, location, and/or extent of structural faults such as crack, delamination, moisture absorption, corrosion and erosion.
- Assessment involves the use of detection and/or monitoring results along with design/structural information to determine the current structural status.
- Decision Making/Management involves the use of detection, monitoring, and assessment results combined with information about missions or available resources to reason and make decisions about aircraft flight operations, plan fleet utilization, or plan maintenance activities.

The military and civil, guidelines should not only cover the processes, standards, and regulations required for sensing, monitoring, and detection functionality, the guidelines should also cover the processes, standards, and regulations required for assessment and decision making, which are essential for integration into maintenance, mission, and flight support systems.

Generally, an elementary function can be hosted in airborne systems or ground-based systems. Figure 11 presents the main architectural entities from which an SHM system can be made. As mentioned above, architecture is physical and functional descriptions of entities and how they join together to form a system; an entity can be a software/hardware item, a component, or a subsystem. The required development rigor of an integrated SHM system and the associated certification challenges and cost benefits can be only evaluated when adequate analysis of the chosen architecture is made to identify where each SHM entity is hosted-in or interfaced-with airborne systems or ground based support systems. Figures 6, 7 and 8 show SHM architectural choices for the three military aircraft types and for legacy and new design applications, noting the following:

- SHM can be integrated into existing aircraft platforms or influence new aircraft designs.
- An architectural choice can be identified by following one path terminated with: (a) flight/mission instructions, (b) maintenance and management instructions, or (c) both types of instructions.
- The ground-based decision making and instruction functionality can be interfaced-with or hosted-in maintenance, mission, and logistic support systems.
- The components delivering elementary functions can be specifically designed for SHM, share resources with other aircraft systems, or integrate with other systems such as IVHM.
- Different architectures can deliver the same required SHM intended function (e.g. crack detection). An optimum architecture can be selected by assessing the feasibility of meeting military requirements and evaluating factors such as technology readiness level,

development timescale, through life costs and weight. Constraints on the choice of the SHM architecture can be imposed by architectural features of other airborne systems and ground support systems.

- Differences between architectural choices that deliver the same instructions arise from whether each elementary function is performed by an airborne component or a ground-based component.
- The architecture chosen to deliver flight/mission instructions is different from the one chosen to deliver maintenance/management instructions; some of the architectural items of the former can be shared with those of the latter.
- The development rigor of the integrated SHM depends on the SHM intend function, the function use, and the architecture chosen to deliver the intended function. The development rigor, and hence, the development assurance level of each architectural item is determined from safety assessments at system and aircraft levels that classify the consequences of the item failure conditions.
- For each SHM architectural item, the development efforts and certification challenges also depend on the integrity levels of the airborne/ground support items to which the SHM item will interface or integrate. The highest certification challenge would be encountered when autonomous operations are required over both friendly and hostile terrain.

5.3.1 *SHM Architectural Choices for Potential Autonomous Operations*

For autonomous operations, the delivery of flight/mission instructions requires the allocation of all the elementary functions to airborne components with two high level architectural choices: (a) interfacing the decision making component with FMS/MMS; or (b) integrating the decision making component within FMS/MMS, Figure 22.

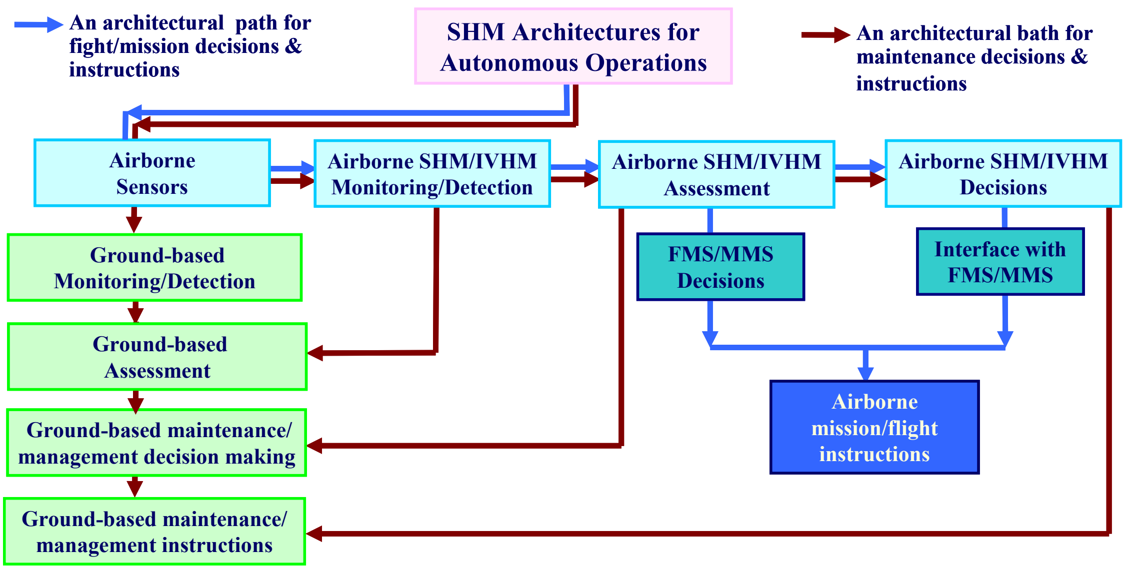


Figure 22: SHM architectures for autonomous operations

Figure 23 illustrates how two architectural choices can be derived from Figure 22.

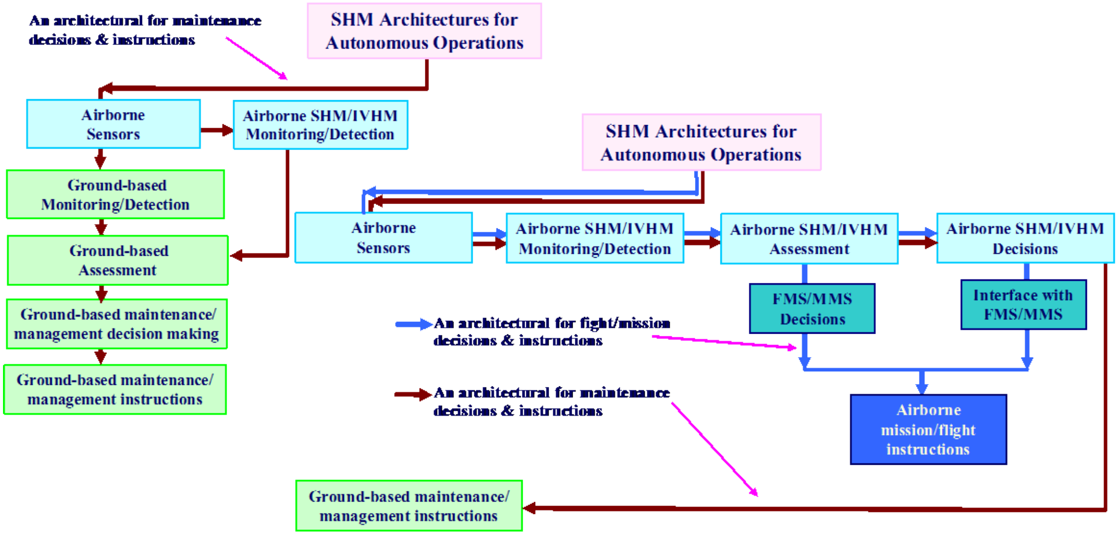


Figure 23: Architectural examples derived from Figure 22 for autonomous aircraft

5.3.2 SHM Architectural Choices for Potential Remotely Piloted Aircraft

Figure 24 shows the potential, high level, architectural choices for remotely piloted aircraft.

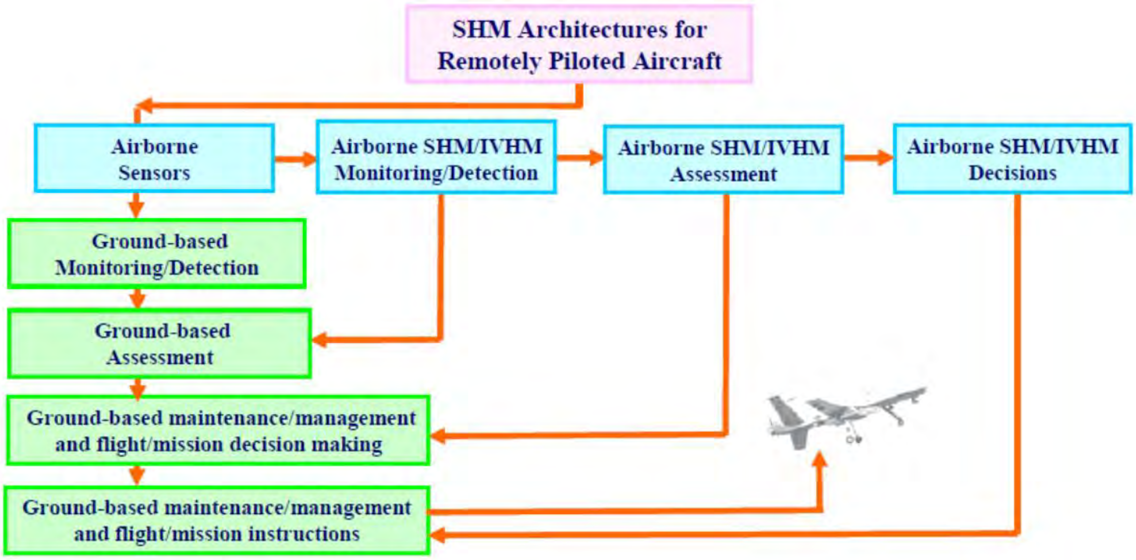


Figure 24: SHM architectures for remotely piloted aircraft

Figure 25 illustrates how two architectural choices can be derived from Figure 24.

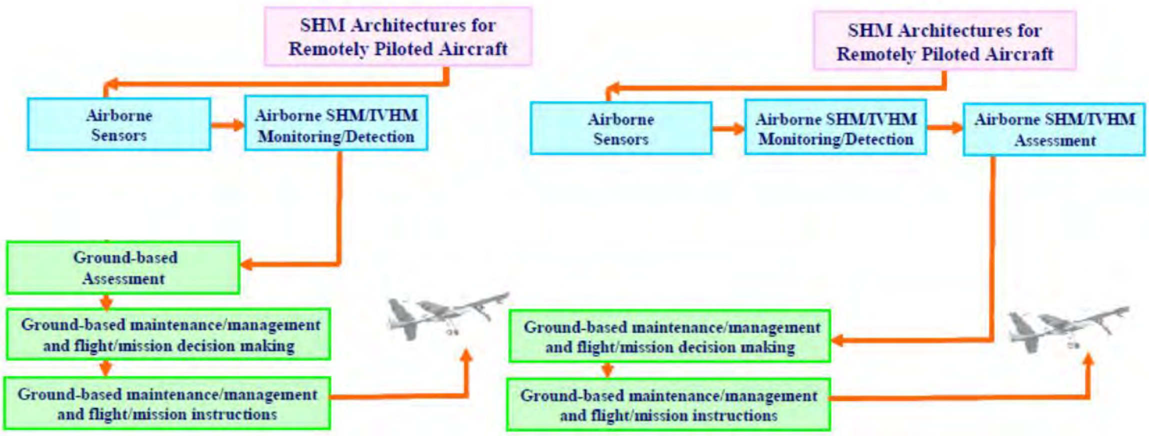


Figure 25: Architectural examples derived from Figure 24 for remotely piloted aircraft

5.3.3 *SHM Architectural Choices for Manned Aircraft*

Figure 26 shows the potential, high level, architectural choices for manned aircraft.

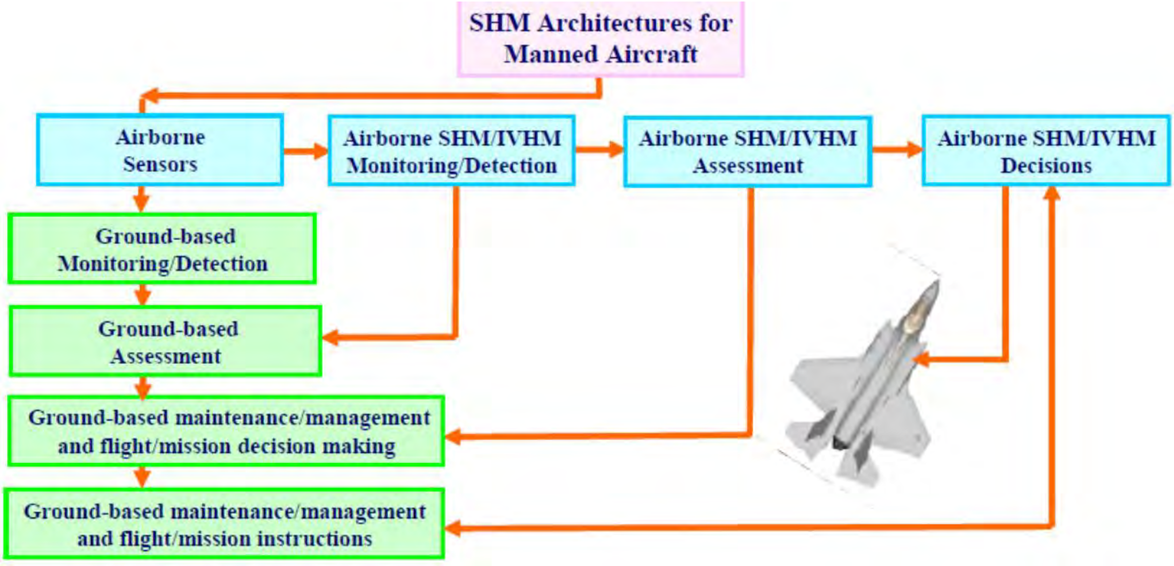


Figure 26: SHM architectures for manned aircraft operations

Figure 27 shows some of the potential SHM architectural choices for the three aircraft types.

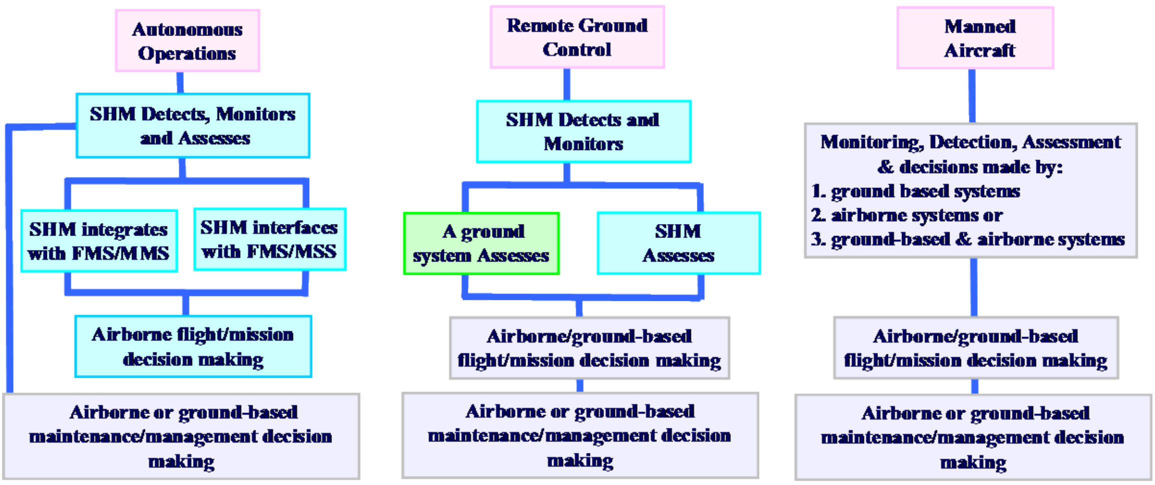


Figure 27: Some of Potential SHM Architectural Choices for Military Aircraft

6 SHM REQUIREMENTS

6.1 Introduction

SHM is defined as "the process of acquiring and analyzing data from onboard sensors to evaluate the state of a structure"; this definition directly points to a high-level set of requirements specifying intended functions that would facilitate the evaluation of the state of the structure; see Section 4. Having declared a set of intended functions, the architecture of a system that would deliver the required set can be developed. The architecture identifies hardware items, software items, items' functions and items' relationships necessary to deliver the declared intended functions. For each item and function, safety requirements and detailed requirements are established.

Safety assessment methods are applied to identify the potential failures of the items and functions of the developed architecture; the methods also classify the hazard associated with each failure condition and determine safety requirements: the hazard is classified as catastrophic (A), hazardous/severe-major (B), major (C), minor (D), or no safety effect (E); the safety requirements should reduce the probabilities of development errors and system failures to acceptable low levels that satisfy applicable airworthiness regulations and operating rules:

- To reduce potential development errors, a Development Assurance Level (DAL) is assigned to each item and function based on the classification of related failure conditions (e.g. A, B, C, D, or E) and based on the intended functions and the intended use of these functions along with the detailed requirements of the item. A DAL defines the rigour of all planned and systematic actions used to substantiate, at an adequate level of confidence, that errors or omissions in requirements, design, and implementation have been identified and corrected to satisfy the applicable certification requirements. The more severe the failure condition classification, the higher the level of development assurance necessary to mitigate the errors that could lead to this failure condition.
- The probabilities of failures can be reduced, to adequate low levels, by adhering to the determined DAL requirements or by introducing additional requirements that identify the need for alternative protective strategies; examples are: (a) safety maintenance task intervals, (b) partitioning, (c) functional independence where two sets of different requirements are employed to deliver the same function, e.g. a navigation function delivered by a Global Position System (GPS) and by an inertial reference system, and (d) development independence where the likelihood of a common development error is minimized through the development of two items using different teams/processes, different technologies such as hydraulic and electrical actuations, different software languages, different operating systems, etc.

The detailed requirements include operational, physical, interface, installation, survivability, maintainability, customer, and derived requirements arising from design choices. Figure 28 shows the three sets of SHM requirements, which are: intended functions, safety requirements, and detailed requirements; Figure 28 also illustrates the iterative nature of the requirement allocation processes.

It is worth emphasising that SHM would be targeted at performing or improving structural health functions, structural integrity functions, or structural management functions. The SHM functions can influence future designs and can be used to trigger or plan maintenance and management tasks. Most of these structural functions have already been introduced and performed using existing maintenance and management tasks including Non Destructive Inspection (NDI). By complementing or replacing existing tasks with equivalent SHM tasks, significant improvements can be achieved leading to maintenance, operational, and ownership benefits in terms of quality, performance, and cost. For example, SHM inspections can be more accurate and faster at reduced costs. Only SHM systems complying with mandated airworthiness regulations and regulatory structural rules can be approved to deliver these improvements.

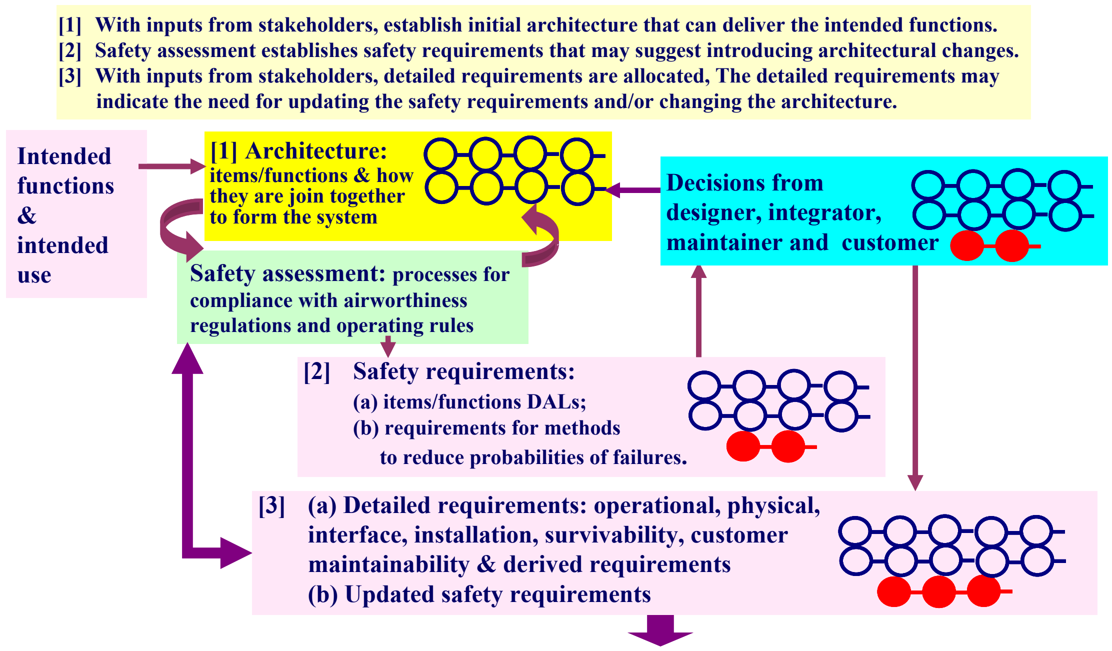


Figure 28: Iterative requirement allocation processes

The following sections compile sets of generic requirements common to most SHM systems. These generic requirements are associated with intended functions, safety, and detailed requirements; the latter set includes functional, physical, installation, survivability, and maintainability requirements along with additional certification requirements and derived requirements. The specific and low level requirements for an SHM system and its associated technologies should consider, as appropriate, these sets of generic requirements bearing in mind that each requirement should be traceable to a parent or rationale, unambiguous, not redundant, has a unique interpretation, can be validated, and can be physically implemented and verified. It is worth emphasising that the aim of this paper is to provide general guidance on how to evolve SHM systems for military aircraft by imperative considerations of military regulations and defence standards. The paper guidance contents do not constitute a UK MOD policy or regulatory requirements. The MOD regulations and the means of compliance with these regulations are those published and updated by MAA. For aircraft products including SHM and similar systems, the UK default specifications and requirements are those stated within the UK defence standards. The reminder of this section must only be considered as a best practice guide on how to identify SHM generic requirements from which a complete set of high and low level requirements can be generated and validated.

6.2 Intended Function/Performance Requirements

<u>Requirement 1:</u> *The SHM intended function(s) should be clearly stated along with clear identification of the purpose of each function and how it will be used.*

<u>Requirement 2:</u> *Each intended function should be decomposed to its elementary functions.*

<u>Requirement 3:</u> *A Development Assurance Level (DAL) should be assigned to each elementary function based on the adverse consequences of the failure of the elementary function on airworthiness and operational reliability.*

<u>Requirement 4:</u> *Quality characteristics should be assigned to each elementary function in consistent way with the purpose of the intended function and its DAL level.*

<u>Requirement 5:</u> *A system architecture that can deliver the intended function(s) with the required qualities at acceptable costs should be developed to enable the allocation of requirements, the application of safety analysis, and the assignment of development assurance levels to all the hardware and software items of the architecture.*

Guidance Notes:

- The SHM performance requirements define the elementary intended function (i.e. the type of performance required) and its quality characteristics in terms of quality attributes and quantitative attribute values that make SHM useful as intended. An elementary function example is the detection of accidentally induced cracks as required by MAA RA 5720 of Reference [36] or the detection of cracks as required by the 14CFR Part25 §25.571 of Reference [3]; the detectable crack size should be determined through damage tolerance analysis, Reference [36] and Reference [13]. Examples of the quality attributes are: (a) the probability of successfully detecting cracks having certain sizes and (b) the size accuracies of the detected cracks.
- In addition to defining the type of performance required (i.e. an intended function such as crack detection or load monitoring), the performance requirements should specify the quantitative values of all the attributes that make SHM useful as intended. The quality characteristics of SHM are often established based on tests, analyses, and engineering knowledge. Appendix C highlights some of the potential SHM quality characteristics including: accuracy, resolution, precision, repeatability, reliability, sensitivity, dynamic range, and bandwidth.
- Often, the quality requirements are specified by the Design Organization (DO) and operators. For example, Def Stan 00-970 Part 1 Section 3 Clause 3.2 provides requirements for data integrity, accuracy, and reliability of monitoring systems targeted at estimating the fatigue damage accumulation. For systems detecting flaws such as crack, delamination, and corrosion, the minimum size of flaw to be reliably detected should be specified based on analyses taken into account the expected flaw growth rates between inspection intervals. Therefore, the relevant DO should be consulted and existing regulations examined when specifying quality attributes such as the minimum detectable flaw size, the flaw location accuracy, and the minimum distance between independently detectable adjacent flaws.
- The environments in which the system should perform with the specified quality should be clearly described; the SHM system should withstand all its operational environments, but may be only required to perform with the specified quality in a subset of these environments, e.g. on the ground with the engine switched off.
- Since interrogating an SHM system can be more frequent than interrogating currently used equivalent systems (e.g. NDI applications), the individual quality requirements of an SHM system need not to exceed those of the equivalent systems; however, they should lead to overall performance better than, or at least similar to, the performance of the equivalent systems in terms of, for example, maintenance costs, inspection times, and failure risks.
- In evaluating the consequences of the intended function failure, considerations should be given to the allowable length of time after which actions should be immediately taken in response to the SHM information; the allowable length of time could be sufficient enough for (a) signalling faulty SHM components, e.g. through built-in tests, and (b) taking corrective actions. Considerations should also be given to any architectural capabilities introduced to significantly reduce the probabilities of the intended function failure and mitigate the adverse consequences of failure on airworthiness and operational reliability. The mitigation capabilities would ensure correct functionality and could be achieved through, for example, organizational procedures, spare equipment to replace faulty SHM systems, or a design allowing for functional independence.

- In establishing and choosing the SHM system architecture, considerations should be given to the aircraft design criteria, the architecture of the aircraft, the architecture of the aircraft support systems, and the cost benefits of the chosen architecture: as mentioned previously, the SHM intended functions can be delivered through a number of potential system architectures; an optimum architecture should integrate the SHM sensors/components into the aircraft without any appreciable adverse effects on the integrity of the structures or other aircraft systems; the optimum architecture can be selected by assessing the feasibility of meeting safety requirements and evaluating factors such as technology readiness level, development timescale, through life costs, and weight; the determination of the optimum architecture would also require careful examination of any existing architectural features of aircraft airborne and ground support systems; using items already developed for these systems may reduce the SHM development costs; the development rigor, certification challenges, and cost benefits of an integrated SHM system can be only assessed when adequate analysis of the chosen architecture is made; the analysis should clearly identify the SHM items that are independent of other systems and the items that are integrated or interfaced with aircraft structures or systems.
- For flaw detection applications, the SHM architecture may consider a local sensing arrangement, a global sensing network, or a combination of local and global sensor arrangements. A local SHM system only detects the flaws at the local location of the sensors or at a hot-spot location very close to the sensors. A global SHM system covers a large area of the structure; the number of network sensors required to cover the required area would depend on the used sensing technology and the required quality attributes of flaw detection. Therefore, in establishing the DAL of sensors and in determining any required mitigation methods, not only the consequences of failed sensors should be assessed, but also the adequacy of the sensor arrangements noting that some system designs may allow for graceful failure: after acceptable maintenance free period, a sensor may fail causing minor degradation in the performance of a network of sensors; afterwards, another sensor may fail causing further degradation; meanwhile, a built-in self-test capability would signal out the failed sensors and indicate whether the quality of flaw detection remains acceptable.

6.3 Safety/Airworthiness Requirements

The safety requirements of an aircraft system are determined by identifying and classifying the system functional failure conditions using safety assessment methods. The failure conditions of the functions and items of SHM should be identified and their effects on the aircraft safety classified even if the classification is "no safety effect". Safety related functional failure may have either contributory or direct effects upon aircraft safety. The safety requirements include development assurance levels, which are introduced to reduce the probabilities of development errors; they can also include requirements such as independence, which are introduced to further reduce the probabilities of failure conditions to very low values consistent with the consequence of failure conditions.

The safety requirements are functionally decomposed from an aircraft level to an item level in a hierarchical structure. At the aircraft level, the safety requirements are those requirements generated from the aircraft Functional Hazard Assessment (FHA) based on aircraft level functions e.g. directional control, deceleration on ground, etc. At a system level, the safety requirements are all those system level requirements generated from the system FHA which are decompositions of the aircraft level safety requirements. Requirements that are defined to prevent failure conditions or to provide safety related functions should be traceable through the levels of development at least to the point of allocation to hardware and software. This will ensure visibility of the safety requirements at the software and hardware design level.

6.3.1 Overarching Requirements

Requirement 6: *The SHM system must comply with applicable airworthiness regulations for equipment, systems, and installation: the SHM system should not have any appreciable effects on the safety of the aircraft, passengers, crew members (operators and maintainers), or civilians; the SHM system installation, operation, and maintenance should not adversely affect the performance, reliability, or maintainability of structures or other systems.*

Guidance Notes:

- The main objective of the applicable airworthiness regulations is to ensure that the equipment, systems, and installations should not adversely influence the aircraft safety and they should perform their intended functions under any foreseeable operating conditions without adverse effects on other systems. Compliance with such regulations should be demonstrated by analysis and tests. The regulations also require demonstrating compliance with Electro Magnetic Interference (EMI) requirements.
- The details of these regulations for each civil aircraft category are published by FAA in 14CFR Part 23 to Part 29, regulations: §23.1309, §25.1309, §27.1309, and §29.1309. They are published by EASA in the Certification Specifications: CS 23.1309, CS 25.1309, CS 27.1309, and CS 29.1309. Examples of such details as required by §25.1309 are: (1) equipment, systems and installations must be designed to ensure that they perform their intended functions under any foreseeable operating condition; they must be designed so that: (2) the occurrence of any failure condition which would prevent the continued safe flight and landing of the airplane is extremely improbable; (3) the occurrence of any other failure conditions which would reduce the capability of the airplane or the ability of the crew to cope with adverse operating conditions is improbable.
- The UK MOD standards for equipments can be found in Def Stan 00-970 P1 S6, which quotes EASA regulations set out in "CS 25 subpart F – Equipment" (e.g. CS 25.1309). The MAA RA 1200 to RA 1230 state the UK MOD regulations for airworthiness aspects such as air safety management, management of operating risk, and airworthiness strategy along with the requirements for safety case and independent assessment. The UK MOD manages the safety risks associated with military systems and their operation not only because of the duty of care to employees, the general public, and the wider environment, but also because safety is a vital characteristic of defence systems as it often has a significant impact upon operational effectiveness. MOD requires that insofar as risks are not judged to be unacceptable, they are reduced to a level which is As Low As Reasonably Practicable (ALARP); contractors who undertake the design, development, manufacture, supply, and support of equipment and defence systems for MOD are obliged to apply the ALARP principles as described in Def Stan 00-56, see Section 2.5.2.1 for examples.
- A RPAS consist of several elements that are critical to engineering and flight safety including not only the flying RPAV and all its associated flight safety-critical elements, but also elements such as the ground-based control unit and the ground-launch system. The mandatory requirements for RPAS are presented in Def Stan 00-970 Part 9, Reference [42]. Part 9 mandates, with minor UK national reservations, the NATO standards set out in STANAG 4671, Reference [60], which are derived from EASA CS-23. The airworthiness requirements of STANAG 4671 have been prepared such that they correspond, as closely as practicable, to a comparable minimum level of airworthiness requirements for fixed-wing aircraft as embodied in documents such as 14 CFR Part 23 and EASA CS-23 whilst recognising that there are certain unique features of UAV systems that require particular additional requirements or subparts. Therefore, the designs of RPAS equipment should comply with airworthiness regulations such as those of "CS 23 subpart F - Equipment" (e.g. CS 23.1309). Furthermore, the operation and maintenance of the RPAS and its equipment should comply with the MOD policy: it is the MOD policy that all military RPASs are

operated and maintained in accordance with the same policy and procedural requirements applicable to manned aircraft with minor exceptions that apply to the maintenance of RPASs waiving regulations regarding continuous charge and indicating that RPAS flight servicing is not to be waived, see CAE 4000 - MAP-01, Reference [33].

6.3.2 Personnel Health, Safety, and Performance

Requirement 7: The SHM system should not adversely affect the health of crew members, passengers, maintainers, or public personnel under any foreseeable conditions during which they can be directly in contact or exposed to the system or any of its components. Adequate protection measures must be in place to prevent harmful effects of materials or components, if any, during the manufacturing, installation, operation, and disposal of the SHM system.

Requirement 8: The SHM system should not adversely affect the performance of the personnel using the system according to the system operating procedure within the system intended operational environments.

Requirement 9: The SHM should not require either directly or by implication the use of substances which could create an adverse environmental effect during the manufacture, use, or disposal of the materiel. The system should not require the use of substances controlled by the Montreal Protocol and associated European Community regulations, AMC to MAA RA 5203(1).

6.3.3 Safety Analysis Process

Requirement 10: A safety analysis process should be adopted to identify the potential failure conditions of SHM functions and items, classify each failure based on its effects, and introduce, if required, SHM safety requirements to ensure that the SHM architecture meets the aircraft safety requirements.

Guidance Notes:

- Def Stan 00-56 P1 documents the UK MOD standards and guidance on safety management requirements for defence equipment, which include requirements for safety analysis, safety case, etc.
- ARP4671 provides industry accepted guidance on methods for conducting the safety assessment process. MAA publications explicitly quote ARP4761.
- The safety assessment process consists of FHA, Preliminary System Safety Assessment (PSSA), Common Cause Analysis (CCA), and System Safety Assessment (SSA). The FHA examines the functions of each product item, identifies potential functional failures, and classifies the hazard associated with each failure condition based on the failure consequence on safety; the consequence of a failure is classified as catastrophic, hazardous/severe-major, major, minor, or no safety effect. By examining the product architecture and the results of the FHA, the PSSA establishes the safety requirements of the product and its items and provides a preliminary indication that the anticipated product architecture can meet these safety requirements. The safety requirements are introduced to significantly reduce the rates of failures and errors to low values consistent with the classified severity of failures/errors. The PSSA may identify the need for alternative protective strategies such as partitioning, built-in-test, monitoring, independence, and safety maintenance task intervals. The CCA identifies failures or external events that can lead to a catastrophic or hazardous/severe-major failure condition. The CCA validates that these failures and events are independent; i.e. they are not common between systems, items, or functions. To satisfy safety and regulatory requirements, it is necessary to ensure that such independence exists, or that the lack of independence is acceptable. The CCA examines the effects of potential

development, manufacturing, installation, maintenance, and crew errors that can defeat the independence and cause common failures in multiple systems/items and lead to a catastrophic or hazardous/severe-major failure condition. The results of a preliminary CAA are essential for the assignment of the development assurance levels and the determination of any additional safety requirements that reduce the probability of such failures to acceptable levels. The SSA collects, analyzes, and documents verification that the product, as implemented, meets the safety requirements established by the FHA, PSSA, and CCA processes. The SSA integrates the results of these processes and verifies that the implemented product meets all of the specified safety requirements. The safety analysis process implements methods needed for conducting acceptable FHA, PSSA, and SSA. These methods include FTA, DD, MA, FMEA, and FMES, Reference [27].

6.3.4 Development Assurance Process

Requirement 11: *A Development Assurance process should be adopted to establish levels of confidence that development errors contributing to or causing failure conditions have been minimized to acceptable low levels with sufficient degrees of rigour.*

Guidance Notes:

- The UK MOD acceptable low levels are those giving confidence that airworthiness risks are at least tolerable and As Low As Reasonably Practicable (ALARP). A risk can be said to be reduced to a level that is ALARP when the cost of further reduction is "grossly disproportionate" to the benefits of risk reduction. This cost may include more than financial cost and must consider the time and trouble involved in taking measures to avoid risk. Therefore, an ALARP argument should balance the "sacrifice" (in time, money, or trouble) of possible further risk reduction measures with their expected safety benefit (incremental reduction in risk exposure). The balance should be weighted in favour of safety, with a greater "disproportion factor" for higher levels of risk exposure. ALARP is essentially the "stopping condition" for risk reduction; therefore, justifying and recording how this is reached is an important and vital step in safety management.
- ARP4754 provides industry accepted guidance on Development Assurance Level (DAL) assignments to functions and items based on classifications of failure condition effects. Def Stan 00-970 P0, recognises ARP4754 as a document among those related to the UK defence standards.
- A, B, C, D, or E is the DAL assigned to a function or an item if its failure effect is classified as Catastrophic, Severe-Major/Hazardous, Major, Minor, or No Safety Effect respectively. DAL should be assigned to each function/item whether it is airborne, off-board, hardware, or software function/item.
- The DAL assigned to a function or an item determines the degrees of development rigour required for the function/item.
- DO-178 and DO-254 are industry accepted means to implement the required development assurance rigour for airborne software and electronic hardware items.
- The guidelines and processes of DO-178 and DO-254 can be adapted to achieve the required development assurance rigour for off-board functions/items noting that more choices are available for assurance considerations such as independence and mitigation methods, and fewer restrictions exist on features such as weight, size, and severity of environments.
- DO-278 provides assurance level considerations for ground-based software that can be adapted for SHM.
- Existing Commercial off The Shelf (COTS) hardware and software (computers, operating

systems, database engines, etc.) may be used if they are developed at the DALs assigned to their functions and items. If the COTS DALs are not known, they should be assessed based on service history, inspections, sufficient tests, and demonstrations. If the COTS DALs are lower than the assigned DAL levels, the levels can be increased through mitigation and independence methods.

- DO-200 provides means to implement development assurance rigour for aeronautical data items and can be used for similar SHM data items.
- The regulatory authorities of civil and military aircraft recognise DO-178, DO-254, DO-278, and DO-200 as acceptable means for implementing the development assurance rigour; for example refer to Def Stan 00-970 P0, AC 20-115, AC 20-152, and AC 20-153.
- The DAL assigned to an SHM item should be high enough to reduce the probability of development errors and system failures to acceptable low levels that satisfy applicable airworthiness regulations. For example, the DAL levels assigned to sensors embedded in structures are expected to be higher than those of sensors fitted in the cabin. High assurance levels can also be achieved by introducing additional requirements that identify the need for alternative protective strategies such as functional independence and safety maintenance task intervals.

6.4 SHM Survivability/Environmental Requirements

<u>Requirement 12:</u> *Each SHM component should survive its manufacturing, repair, and installation environments even if the severities of these environments exceed those of the aircraft operational environments.*

<u>Requirement 13:</u> *Over a specified survivability period (failure-free period), each SHM airborne component should survive its surrounding environments during all foreseeable aircraft operational conditions, and should survive the expected cyclic variations of these environments.*

<u>Requirement 14:</u> *During the failure-free period, and within the intended system operational environments, each airborne component should perform its allocated functionality with quality consistent with its intended function and its DAL level.*

<u>Requirement 15:</u> *The failure-free period should be long enough to maintain aircraft safety and operational reliability at acceptable costs taken into account whether independence and mitigation methods would be implemented: if sensors or other items are embedded in the structure or another system, their failure-free period should exceed the economic life of the structure/system, or should be repairable without the need for replacing expensive items (e.g. they might be a part of a replaceable inexpensive structural assembly); a shorter failure-free period may be accepted if: (a) stand-by alternative methods are prepared to replace faulty inaccessible sensors/components, and (b) the availability and maintenance cost benefits gained during the shorter failure-free period exceed the costs of fitting the sensors/components in each individual aircraft.*

<u>Requirement 16:</u> *Depending on the severity of the failure effects on safety, not only the "mean time between failures" but also the "minimum time between failures" should be equal to or greater than the failure-free period.*

<u>Requirement 17:</u> *The SHM ground-based equipment should survive their environments including transportation and handling environments, and should survive potential operations in extreme weather conditions and sand storms.*

Guidance Notes:

- The surrounding environments of a component are not necessarily the entire aircraft operational environments; it can be a small subset of the aircraft environments. The various components of an SHM airborne system can be surrounded by different environments; e.g. the environments of a component located near an engine is at a great variance with the environments of a component located in the cabin. The SHM system may be required to perform only in a subset of its surrounding environments; e.g. accurate measurements may be only required on the ground.
- For military aircraft, the SHM system should survive potential hostile military environments. The SHM system may be exposed to these hostile environments for a long period; for example, the system may be exposed to humid corrosive environments onboard of a carrier, sandy erosive environments during desert operations, and high strain fields or excessive vibration environments during aggressive manoeuvres or recovery with battle damage to safe locations.
- The compliance with the survivability/environmental requirements should be managed and demonstrated according to standards such as Def Stan 00-35, DO-160, MIL-STD-810, and MIL-STD-461; the latter three publications are among the related documents of the UK Def Stan 00-970, which are listed in Part 0.
- The compliance with the survivability/environmental requirements should be demonstrated through qualification tests. The tests should not only demonstrate that each airborne component can survive its surrounding environments over a failure-free period, but should also demonstrate that the component continues to correctly perform its function during this period under a specified subset of environments in which SHM measurements will be acquired. The tests may cover environmental cycles/profiles of parameters such as temperature, altitude, humidity, shock, crash, vibration, explosion, water/fluids susceptibility, resistance to salt, sand, dust, fungus, g, emission, lightning, icing, radiation, electrostatic discharge, and fire. The tests may cover, if applicable, representative structures with embedded or bonded SHM sensors. Special wires, connectors, and bonding materials should be also tested.
- Each SHM item should be tested with a severity level commensurate with the consequences of the item failure on the safety and operational reliability of the aircraft; for example, items developed at DAL A should be subjected to the most severe test conditions (categories) presented in DO-160.
- Specific SHM system requirements (e.g. manufacturing and some special operational environments) may need specifications supplementary to standards such as DO-160.
- Any required manufacturing, assembly, and repair processes should be reproducible without any adverse effects on the performance and integrity of the SHM sensors/components. MAA RA 5102 requires assurance that the design will be suitable for production in facilities agreed between the contractor and MOD. The SHM quality should remain as specified after embedding, integrating, or surface mounting the SHM sensors through the manufacturing, assembly, or repair processes; the SHM system should withstand any mechanical, thermal, and chemical environments encountered during these processes; for example, components installed during aircraft manufacturing or repair processes should withstand any high temperatures induced during such processes, e.g. temperatures up to 180 °C during curing Glass Fibre Reinforced Polymer (GFRP).
- Often, the aircraft manufacturers/operators specify the environmental conditions to be met by the system.
- According to the UK MOD regulations RA 5101 to RA 5103, a certificate of design signed by an approved member of the DO would be required before the first flight of the SHM airborne system. The Certificate of Design can be accepted by MOD after establishing

compliance with specifications through inspection, demonstration, analysis, and test. According to RA 5105, The DO must consider the need to repeat qualification tests (re-qualification), in whole or in part, when a change in method of manufacture, or change of material or source of material of a component or equipment would invalidate the current issue of a Certificate of Design or when the place of manufacture of the component or equipment is changed.

- Although the SHM components could survive their surrounding environments, the desirable characteristics of some of these components, specially bonding and synthetic materials, may degrade if subjected to these environments for a long period. Therefore, additional time-resilience tests may be required to demonstrate that such desirable characteristics can survive, during a failure-free period, the exposure to environmental conditions including: humidity, icing conditions, sand, dust, fungus, vibration, and fluids such as: water, hydraulic fluid, kerosene, lubrication oils, cleaning fluids, de-icing fluids, anti-icing fluids, insecticides, disinfectants, and coolant dielectric fluids.

- High levels of performance qualities and survivability attributes can be achieved through a rigours design at high DAL levels or through a damage-tolerant design where SHM components including sensors, or the entire system, can be duplicated to achieve the required performance over the specified failure-free period.

6.5 Operational Requirements

The SHM operational requirements should describe how and when the system would be used, interrogated, accessed, and maintained through friendly interfaces with various users; this information should be included in an operational manual that contains a full system description supported by illustrations, tables, and drawings. Generally, the operational requirements define the interfaces between the flight crew and each functional system, the maintenance crew and each aircraft system, and various other aircraft support personnel and related functions or equipment. Actions, decisions, information, and timing requirements constitute the bulk of the operational requirements. Both normal and abnormal circumstances need to be considered when defining operational requirements.

6.5.1 Concept of Operations

Requirement 18: *The system concept of operations should be declared and fully described in clear documentations that: (a) explain and illustrate when, where, and how the system should be used and by who, (b) describe the format and flow of system accessible data from raw sensed forms to final actionable forms, and (c) describe how stored data can be accessed.*

Guidance Notes:

- As illustrated in Figure 9, the potential output of an SHM system can be inspection data, usage data, or instructions for maintenance and management purposes; the output can also be a combination of these types of data and instructions. The usage data and associated information are acquired and computed during the operations of the aircraft. The inspection data does not necessarily require acquisitions during the entire operational conditions of the aircraft; for example, the data can be acquired only at specified inspection intervals with the aircraft engine(s) switched off. It is also possible to acquire a number of inspection datasets during the aircraft operations and use abnormal datasets to timely generate warnings or store these datasets for ground-based investigations. The system data and instructions may be stored, displayed, transmitted to other systems, or made accessible/transferable via suitable media or equipment.

- The concept of operations should be clearly illustrated in documentation supported by drawings describing as applicable: the process of switching the system on and off, the access points to system data/instructions, the methods or tools required to access the system data/instructions, the required skills of system users, the time taken to use the system as intended (i.e. the time required to operate the system, interrogate the system, view/retrieve actionable instructions, and/or download/transfer system data). The documentation should describe all the interfaces with crew members and system users. The information received via these interfaces should be described; any briefly displayed information should be explained and the actions required in response to this information documented. An accurate Interface Control Document (ICD) should be provided to enable independent use of system data in any investigation supporting the structural integrity and airworthiness of the aircraft; the ICD should clearly describe the structures of all accessible SHM data; i.e. it should state the type and format of each data item and the organization of various data items within data-files, data messages, or data streams.

6.5.2 Electro-magnetic Compatibility

Requirement 19: *The system should operate, with adequate safety margins, without malfunction or degradation of performance, in the electro-magnetic environment corresponding to the operational one.*

- MAA RA 5106(4) requires that it must be demonstrated to the satisfaction of the Project Team Leader (PTL) by ground and flight trials as agreed that all systems will operate, with adequate safety margins, without malfunction or degradation of performance, in the electro-magnetic environment corresponding to the operational one.

6.5.3 Data Requirements

Requirement 20: *The system data items should be generated, transmitted, and stored with integrity consistent with the purpose of intended function and the assigned DAL levels.*

Requirement 21: *The system data items should be tagged with identifiers indicating: the structural components they monitor, the aircraft from which the data acquisitions are made, the system configuration used to generate the data, and the times/dates of acquisitions along with any specific operational conditions at the time of acquisitions.*

Requirement 22: *If system data items are transmitted between airborne and/or ground-based systems, the data items should be securely transmitted without broadcasting the extent of the military capability and posing military risks on the nation.*

Guidance Notes:

- Integrity requirements commensurate with the purpose and DAL of each data item should be carefully considered. The integrity requirements should identify the quality attributes required for each data item, e.g. accuracy, resolution, dynamic range, and bandwidth. The requirements could call for sufficient automation or other procedures to avoid the loss or corruption of data. Synchronising some of the data items before further fusion/processing could be also required.
- The system should manage the data with integrity assuring that all data items required to support the intended functions are timely generated, processed, transmitted, and stored as required without any significant corruption or loss. The system should timely transmit/deliver alerting data items well ahead of the time taken for the associated monitored condition to reach a critical level, i.e. before a condition such as a growing crack or accumulated fatigue reaches a value that impairs structural integrity.
- The system configuration used to generate each data item along with any specific operational conditions required during the acquisitions of the data item should be stored in a uniquely identifiable file. The file should contain all of the configuration details including

information about gains, filtered frequencies, calibration factors, software identifiers, etc. Any changes in these details should result in a new uniquely identifiable configuration file.

- For some SHM applications, the integrity requirements should specify the period over which the data items should be available. For example, if the SHM sensors are sealed or embedded in expensive structures, the data availability period should exceed the expected time for the formation of structural faults such as corrosion and cracks. The availability period can be assured by high sensor integrity, sensor redundancy, or other methods capable of filling-in lost data items due to an inoperable SHM system. Generally, methods for filling-in lost data should be considered for most of the data items to ensure optimum structural health management. For example, lost usage data from an air vehicle performing certain operations during SHM maintenance periods can be estimated from the usage data of air vehicles that performed similar operations.
- Security measures commensurate with the criticality of the data items (data encryption, aliasing, passwords, etc.) should be in place to provide assurance of sufficient access control and secure data transmission.

6.5.4 System Configuration, Calibration, and Self-Diagnostics

Requirement 23: An access controlled interface between the SHM system and its users should be designed to upload approved software applications and configuration data.

Requirement 24: A built-in self-test capability should be designed with diagnostic coverage and rigour commensurate with the purpose of the intended function and its assigned DAL level.

Requirement 25: A process should be designed to gather, process, and store baseline data and, as applicable, use the processed data to check, initialize, offset, or calibrate system sensed or processed data.

- A subsystem should be designed to upload approved software applications and configuration data to a target airborne system without any adverse effects on the data, software, or performance of other aircraft systems. The interface to the upload-subsystem should be user-friendly and secure through an access-control capability; the interface should facilitate configuration management providing, for each uploaded software item, assurance of validity, traceability, and sufficient identification of, for example, the version number, date, and supplier of the item.
- The self-test should be capable of diagnosing the overall status of the system. For complex systems having many sensors and components, the self-test should identify failed sensors or components to facilitate maintenance targeted at the faulty parts of the system; data from the healthy parts may be used whilst maintaining the system.
- The upload-subsystem and/or the self-test facility should be capable of checking that any newly uploaded software or configuration data will not result in consuming more resources than the specified original resources, which include data transfer resources, computational power, and data storage resources.
- A baseline process should regularly, or on user demands, gather, process, and store baseline data during specified system operational conditions. As applicable, the process should use the processed data to check, initialize, offset, or calibrate system sensed or processed data. The process should be operated by approved personnel through secure access-controlled interface under configuration management.
- For some SHM applications, the gathered data should be collected from healthy aircraft during specified system operations. Then, normality baseline models (or reference

thresholds for healthy components) could be automatically established using the gathered data; the deviation from normality would indicate component faults. The normality models may change because of, for example, age and changes in operational environments; in this case, the normality models should be updated using a new set of data acquired from healthy aircraft. For such applications, the healthy status of the aircraft should be confirmed before gathering the baseline data; the confirmation may require the use of NDI applications.

- For some SHM applications, offsets or gain deviations should be automatically detected in data gathered during specified system operations. In this case, the baseline process should initialize and calibrate the system using the observed offsets and deviations. The validity of the gathered data could be also checked to indicate whether the system is operable or not in support of the self-test diagnostic process.

6.5.5 *Maintainability Requirements*

Requirement 26: *Interfaces between the system and its maintainer should be designed and documented to allow for accessing system components for maintenance, replacement, and repair purposes at expected maintenance intervals.*

Guidance Notes:

- Like other systems, maintainability requirements should be established for SHM to define how the system components will be accessed, replaced, repaired, or serviced. This information should be included in a maintenance manual supported by installation and drawings. The maintenance manual should include a list of all repairable or replaceable components and should include sufficient details describing how to access these components and maintain them along with the tools and skills required for maintenance. Generally, the maintainability requirements could include scheduled and unscheduled maintenance requirements. Factors such as the percent of failure detection or the percent of fault isolation may also be important. Provisions for external test equipment and connections should be defined in the manual along with the skills, efforts, and times required to perform maintenance tasks.
- If the SHM sensors or other components are embedded in structures or other systems, the failure-free period of the sensors/components should be longer than the economic failure-free life of the structures/systems; a shorter failure-free period may be accepted if replacing the entire structures/systems is not expensive and will not pose unexpected adverse effects on planned availability and maintenance costs; a shorter failure-free period may be also accepted if: (a) stand-by alternative methods are prepared to replace faulty inaccessible sensors/components, and (b) the availability and maintenance cost benefits gained during the shorter failure-free period exceed the costs of fitting the sensors/components in each individual aircraft.

6.5.6 *Personnel Qualification and Training Requirements*

Requirement 27: *Any special qualification and training requirements should be specified and supported by training and user manuals covering system installation, configuration, calibration, maintenance, and general use including responding to system alerts and instructions, or handling, analysing, and interpreting system data.*

6.6 Physical, Interface and Installation Requirements

Aircraft design and performance constraints are likely to influence the physical and installation requirements. These requirements relate the physical attributes of the SHM system to the aircraft environment; they may include: size, mounting provisions, power, cooling, environmental restrictions, visibility, access, adjustment, handling, and storage.

The SHM architecture describes the SHM items and functions. The interfaces of each item with other SHM items should be clearly specified. Compatible interfaces should be specified for SHM items interfacing with other aircraft items. Generally the interface requirements include the physical item interconnections along with the relevant characteristics of the specific information communicated. The interfaces should be defined with all inputs having a source and all output destinations defined.

6.6.1 The Weight, Size, and Power of SHM

Requirement 28: *The weight of the SHM airborne components, their dimensions, and locations should be declared through sufficient drawings and documentation ensuring compliance with the physical and performance constraints of the target aircraft.*

Requirement 29: *The power resource and the power consumed by each SHM component should be declared, determined, and documented. If a component requires power from an aircraft power resource, protection to the power distribution terminal should be provided to eliminate any adverse effects on the aircraft power resource.*

Guidance Notes:

- The SHM weight and size requirements should be tailored for each aircraft in sympathy with variations in size, operation, and mission; for example, because of size and weight constraints, and because adding new sensors and systems to fighter airplanes could be more challenging than adding them to large transport airplanes, requirements for sensor weight reductions should be considered.
- The SHM weight should not significantly reduce the aircraft payload or increase the fuel consumption; furthermore, the size of an externally mounted SHM component should not adversely influence the aerodynamic characteristics of the aircraft.
- If the SHM components receive power from the aircraft power resources, a distribution and protection network to the terminals of the components should be provided and the characteristics of the aircraft power resources should be maintained according to standards such as MIL-STD-704F and MIL-STD-461E.
- The SHM system components may acquire their required power from their own power sources, for example: ground-based power sources, batteries, or energy harvesting devices. Several energy harvesting devices are emerging: these devices capture the energy dissipated in the environment as vibration, heat, light, or particle flow; some devices store the captured power in media such as batteries.

6.6.2 Interface and Installation Requirements

Requirement 30: *The installation, functional, and environmental requirements of the interfaces of SHM components with aircraft systems/structures should be documented and agreed with the Design Organizations (DOs) of the systems/structures concerned, MAA RA 5106(2) and RA 5204(1).*

Requirement 31: *An installation process that efficiently and repeatedly meets the specified requirements should be designed, documented, and demonstrated; see MAA RA 5106(3).*

Guidance Notes:

- The requirements of sensors (including wires and connectors) embedded in structures should be agreed with the DO responsible of the integration of the sensors and structures.

The embedded sensor should not have any appreciable effects on specified structural characteristics; they should have negligible effects on the required carrying-load capabilities, strength, fatigue limit, toughness, structure integrity, etc. The failure-free period of the embedded sensor-systems should exceed the economic life of the structure, or should be repairable without the need for replacing expensive structural items. The integrated structural design of the embedded sensor systems should be approved by the DO and should be substantiated by tests and analyses from stress and material specialists.

- The installation and functional requirements of the interfaces of SHM components with aircraft systems should be documented and agreed with the DOs of the systems. These interfaces should not adversely affect the aircraft systems or degrade their specified functionality in any way. Neither the SHM components attached to structures nor the attachments methods (e.g. holes, bolts, and brackets or bonding materials) should have any appreciable adverse effects on specified structural and aerodynamic characteristics. The designs of attachments should be approved by the DO and should be substantiated by analyses from stress and material specialists. Any resources required for SHM from aircraft systems (e.g. processing, computing, data transfer, storage, power, and data-bus communication resources) should be declared, quantified and agreed with the DOs of the associated aircraft systems to ensure that the resources required for SHM are available and will not consume the resources required for other aircraft systems. Analyses and tests should be conducted to demonstrate that the declared quantified resources of SHM will not be exceeded; alternatively the interfaces of SHM should have protective measures that prevent SHM from using excessive resources.

- Installation process should be developed and documented. The process should result in reproducible SHM characteristics across the target fleet of aircraft. The process should be efficient so that installing the SHM systems into an existing fleet of aircraft can be achieved without unacceptable interruptions to the fleet operations.

- The installation process should include practical clear instructions for positioning, placing, bonding, drilling, bolting, curing, protecting, or treating sensors or other components along with any required instructions for surface preparation, installation brackets, special tools, and handling requirements to protect the integrity of the installed components and its surrounding structures/systems. Such instructions should constitute an efficient reproducible process for installing SHM and replicating its desired performance across the aircraft fleet. The instructions should include installation procedures for each component and each sensor including its wires and connectors; the instructions should specify (a) any required sensor application conditions (e.g. pressure, torque, and temperature) along with the means of providing these conditions (b) any required protecting layers (e.g. sealant, top coat, copper foil, or GFRP layer), (c) sufficient information/datasheets about, all substances, devices, tools, and consumables needed during installation, (d) health and safety instructions including clothing, protective equipment, and handling instructions. The installation process should have no adverse effect on the aircraft structures, engines, or systems.

- A suitable protection layer for a sensor could be chosen to prevent the sensor from being damaged without degrading the sensor performance; the protection layer should endure its local environments and loadings. The surface or installation location of a sensor should be treated to ensure optimum sensor performance, bonding quality, etc. The treatment method and substances should not have any appreciable effects on the specified characteristics of aircraft systems or structures. The treatment methods can include: (a) smoothing the surface by overcoming irregularities induced by factors such as primer crazing and pitting, (b) removal of contamination such as metal oxides, oil, grease, primer dust, chemical products of corrosion, sealants, humidity, or dust; contamination should be removed by approved tools and substances to avoid damaging the structure, and (c) cleaning the surface/location from the deposits resulting from smoothing or decontamination.

- The bonding between a sensor and a structure should withstand its local environments and stress fields generated from the expected operational loading conditions without affecting the sensor performance.
- The interface documentation should make reference to compliance with relevant requirements. The documentation, where appropriate, should include, but not limited to, the following: (a) description of the equipment including the main parameters and the list of the items constituting the equipment along with any related documents, (b) sizes, shapes, masses, and ranges of cg position of all separate units, (c) forces and moments exerted on the aircraft including dynamic effects due to moving parts, (d) locations, methods of attachment and clearances required for installation and maintenance access, (e) any requirements for heating, cooling, sealing, bonding, anti-corrosion, etc., (f) information about interconnections and power supplies including, for example, recommended cables/connectors, input/output levels, voltage, current, impedance, and frequency of power supply lines, etc., (g) EMC information, (h) pre/post-installation testing and recommended test equipment along with any special test facilities, (i) drawings including layouts of separate units, mounting details, functional block diagrams, and interface diagrams showing test points and signal levels.

6.7 Customer Requirements

Aircraft operators (customers) may require specific SHM features driven by their operating requirements and maintenance practices. These requirements would vary between operators and aircraft types and could include for example: SHM system weight, SHM MTBF, required structure/format of SHM data for potential integration with other maintenance systems, etc.

6.8 Additional Certification Requirements

Additional functions, functional attributes, or implementations may be required by airworthiness regulations or may be necessary to show compliance with airworthiness regulations. Requirements of this type should be defined and agreed upon with the appropriate certification authorities.

6.9 Derived Requirements

At each phase of the development activity, decisions are made as to how particular requirements or groups of requirements are to be met. The consequences of these architectural and design choices become requirements for the next phase of the development. Since these requirements result from the design process itself, they may not be uniquely related to a higher-level requirement and are referred to as derived requirements. For example, derived requirements may result from the decision to select the source of power; the chosen power source may introduce a set of derived requirements. Another example is the derived requirements associated with material choices for hardware or development tools for software.

7 VALIDATION AND VERIFICATION

The evolution of an SHM system begins with desirable intended functions, e.g. crack detection, corrosion detection, operational load monitoring, etc. When the SHM technologies required to deliver the intended functions are sufficiently matured, the development process starts with a proposed architecture that can deliver the intended functions. The most important aspects of architecture are physical and functional aspects: the physical architecture describes the system by showing how it is broken down into subsystems, components and items (a representation of the system physical items and their interconnections); the functional architecture identifies the functions allocated to each item (a partially ordered list of activities or functions). Based on architectural analyses and safety assessment methods, requirements are allocated to items/functions. The safety assessment process consists of FHA, PSSA, CCA and SSA, which have to be considered at aircraft and system levels. ARP4761 gives detailed guidelines on the safety assessment process and describes methods for conducting the process. In practice, the development of system architecture and the allocation of requirements are tightly-coupled, iterative processes. Validation encompasses the efforts required to ensure that the allocated requirements are sufficiently correct and complete. Verification encompasses the efforts required to check the correct implementation of the system requirements. As a simple example, a validation task could be checking that a requirement for a specific weight exists; weighing the system is a verification task assuring that the specified weight is not exceeded.

7.1 Development Rigour and Use of SHM Systems

A safety assessment process consisting of FHA, PSSA, and SSA is used to identify and classify the failure conditions of SHM and its items; the consequence of a failure is classified as catastrophic (A), hazardous/severe-major (B), major (C), minor (D), or no safety effect (E); then, safety requirements are established to minimize the probabilities of development errors and system failures to acceptable low values that satisfy applicable airworthiness regulations and operating rules:

- To reduce potential development errors, a DAL is assigned to each item and function based on the classification of related failure conditions (e.g. A, B, C, D, or E) and based on the intended functions and the intended use of these functions along with the detailed requirements of the item. A DAL defines the rigour of all planned and systematic actions used to substantiate, at an adequate level of confidence, that errors or omissions in requirements, design, and implementation have been identified and corrected to satisfy the applicable certification requirements. The more severe the failure condition classification, the higher the level of development assurance necessary to mitigate the errors that could lead to this failure condition. ARP4754A regards the activities described in DO-178B and DO-254 as a means to implement the determined development assurance rigour for software and electronic hardware items. Thus, the levels of rigour of the development processes of SHM items and functions are established by assigning appropriate DALs to the items and functions.
- The probabilities of failures can be reduced, to adequate low values, by adhering to the determined DAL requirements or by introducing additional requirements that identify the need for alternative protective strategies; examples are: (a) safety maintenance task intervals, (b) partitioning, (c) functional independence where two sets of different requirements are employed to deliver the same function, e.g. a “fuel quantity” function delivered by engine fuel flow devices and tank fuel probes, and (d) development independence where two items are developed using different teams/processes, different technologies such as hydraulic and electrical actuations, different software languages, different operating systems, etc.

Mainly, DALs are assigned depending on the classification of failure conditions considering the possible independence between items and functions that can limit the consequences of development errors. For example, if a catastrophic failure condition could result from a possible development error of an item, then at least DAL A is assigned to the item. If a catastrophic failure condition could result from a combination of possible development errors between two or more independently developed items then, either at least DAL A is assigned to one item, or at least DAL B is assigned to two items; no lower than DAL C is assigned to the other independently developed items; DAL A is assigned to the process required to establish that the two or more independently developed items are truly independent. Thus, the DAL assignments to functions and items determine the levels of rigour required for development processes including validation and verification processes. For DAL A and B, all of the validation/verification data and methods described in this document would be required to support certification. For DAL E, all of these data and methods may not be required to support certification; however, the development of a system at DAL E should follow structured validation and verification processes, perhaps at a minimum effort, to support the development of a usable useful SHM product. In other words, the validation, verification, and certification processes described in this document are not necessarily required for each SHM system; they may or may not be fully required depending on the determined development rigour. For comprehensive guidelines on DAL assignments refer to ARP4754A.

The validation, verification, certification, and use of SHM systems would require one or more of distinct development disciplines covering structural items, airborne equipment, and system use. The following subsections briefly discuss the development rigour associated with these disciplines.

7.1.1 Modified Structures

The first development discipline covers any structural items that might have been modified by the SHM system or its sensors. Some SHM systems may require "minor" or "major" structural changes in the type design:

- According to the FAA regulation 14CFR § 21.93, "a minor change is one that has no appreciable effect on the weight, balance, structural strength, reliability, operational characteristics, or other characteristics affecting the airworthiness of the product. All other changes are major changes", Reference [1].
- EASA, EC 748/2012, 21.A.91 defines the minor and major changes as follows: "a minor change is one that has no appreciable effect on the mass, balance, structural strength, reliability, operational characteristics, noise, fuel venting, exhaust emission, or other characteristics affecting the airworthiness of the product, all other changes are major changes", Reference [17].
- The Canadian Aviation Regulations CARs 101.01 define the major changes as follows: "major modification - means an alteration to the type design of an aeronautical product in respect of which a type certificate has been issued that has other than a negligible effect on the weight and centre-of-gravity limits, structural strength, performance, power plant operation, flight characteristics or other qualities affecting its airworthiness or environmental characteristics", Reference [22]. A minor change to the type design is a change other than a major change", AC 521-004.
- In the UK, MAA lists the changes to military air systems that must be classified as major. Examples are those changes that: result in any mark number change; involve multiple systems and areas; involve structural changes that could invalidate previous airworthiness assessments; introduce a new engine; modify air-to-air refuelling systems; modify fuel systems; modify hot air systems; modify weapons release/firing systems; extensively modify cockpit instrumentation. In Reference [37], MAA presents the detailed list of the changes that must be classified as major changes.

Generally, the major changes can be approved only by the appropriate regulator after witnessing validation, verification, and certification activities conducted at a level of rigour depending on the magnitude of the structural changes. It is anticipated that these activities will be performed by the type designer, mainly as structural development activities, not SHM development activities, because the aircraft structural carrying load functions are primarily performed by the structures not by SHM; this is also the case for structural items with embedded sensors. In other words, structural specialists working for the type designer or his license holder must perform these activities with rigour proportionate to the most severe failure condition of the modified structural items not SHM. For example, according to EASA CS 25.302, aircraft equipped with systems that affect structural performance, either directly or as a result of a failure or malfunction, the influence of these systems and their failure conditions must be taken into account when showing compliance with the requirements of CS 25 Subparts C and D, which are equivalent to Subparts C and D of 14CFR Part 25; these subparts cover airworthiness requirement for “Structure” and “Design and Construction”. In Canada, CARs 521.151 to CARs 521.161 indicate that a change to a type design that has other than a negligible effect can be approved only by the regulator after submitting an appropriate application and performing specified certification steps, Reference [23].

Generally, minor changes do not require regulatory approval. They require approval under procedures agreed with the regulator. Delegates authorized by the regulator can decide whether the changes are minor or not. A delegate can be an organization or an engineer working for the manufacturer. The regulator specifies the qualifications, experiences, and responsibilities of the delegates.

- According to 14CFR § 21.95, minor changes in a type design may be approved under a method acceptable to the regulator before submitting to the regulator any substantiating or descriptive data. According to FAA Order 8110.37E, the acceptable methods can include approvals given by the manufacturer Designated Engineering Representative (DER) without prior authorization by the Aircraft Certification Office (ACO). The decision as to whether changes and/or modifications are major or minor must be reviewed with the ACO if the decision is controversial or if the DER needs guidance. According to FAA Order 8100.8D, A Designated Airworthiness Representative (DAR) may perform examination, inspection and testing services necessary to the issuance of certificates for the company.
- In Europe, EASA, EC 748/2012, 21A.95 states that minor changes in a type design shall be classified and approved either (a) by the Agency, or (b) by an appropriately approved design organization under a procedure agreed with the Agency. Therefore, minor changes can be approved by the manufacturer under an agreed procedure and, the substantiating data maintained for potential audits; in this case, the certification process described in this document will not be required.
- In Canada, CARs 521.154 indicates that the holder of a design approval document who proposes to make a negligible change to an aeronautical product, other than a major change, shall establish procedures to ensure that the changed aeronautical product continues to conform to its certification basis and make the change after the Minister accepts the procedure. AC 521-004 gives details about the procedures required for such minor modifications. Among other requirements, AC 521-004 indicates that these procedures must include a process that the persons authorized to approve these changes must use to assess the proposed change and the means by which the change is classified as minor. Such negligible/minor modifications are deemed to have insignificant effects on airworthiness. Authorized delegates, e.g. Design Approval Designee (DAD) or Design Approval Organization (DAO) decide whether the modifications are truly negligible. Although minor changes do not require regulatory approval, they require the approval of the manufacturer under controlled procedures.

- In the UK, changes other than the major changes listed in Reference [37] may be self-certified by the Type Airworthiness Authority (TAA) in accordance with extant procedures. The TAA is the individual, often an aircraft Project Team Leader (PTL), who on behalf of the Secretary of State for Defence oversees the airworthiness of specified air system types. As the TAA, the PTL responsibilities are as laid down and agreed in their Letter of Airworthiness Authority from their respective Director. During a transition phase to updated regulations, the TAA was also authorized to approve major changes that did not result in a change of mark number and was expected to achieve RTS before 1st April 2012.

7.1.2 Airborne Equipment

The second development discipline covers any SHM airborne functions and items including sensors, avionics and software. The second discipline must include validation, verification, and certification activities for each airborne function/item at a level of rigour depending on the interaction level of the function/item with other aircraft systems.

The SHM airborne item(s) may require airborne resources for power management, data acquisition, data processing, data storage, data transfer, or alert displays. If such resources are taken from, or affect, other aircraft systems, the airborne item(s) must be developed with rigour proportionate to the effect of its most severe failure conditions on the aircraft systems. If the aircraft systems support aircraft-level functions and have failure modes with the potential to affect the safety of the aircraft, the development of the SHM item(s) should strictly follow guidelines such as those of ARP4754, ARP4761 and ARP5150; examples of aircraft-level functions are: flight controls, ground operation controls, engine controls, communication, passenger safety, navigation and guidance, and collision avoidance.

If the installed equipment does not significantly affect aircraft systems or structures (e.g. DAL E), the validation, verification, and certification processes described in this document may not be required; however, an appropriate DAL should be proposed by the aircraft manufacturers to ensure their approval of the installation and use of SHM.

7.1.3 System Use

The third discipline addresses the processes required to issue instructions/manuals on how to use the SHM system. Often, issuing instructions does not require certification; i.e. the issued instructions do not require the approval of a regulatory authority (FAA, EASA, etc.). However, they must be approved by the aircraft manufacturers or their license holders. Similar to NDI processes, it is anticipated that the manufacturers will test and calibrate the SHM equipment to ensure compliance with, for example, the approved inspection requirements. In other words, existing processes can be used to issue such SHM instructions including inspection instructions, usage monitoring instructions, etc.

It is worth emphasizing that the rigour of the development processes, and hence, the assignment of DAL to SHM functions and items, does not only depend on the classifications of related SHM failure conditions, but also depend on the intended functions and the intended use of functions. SHM can be used to (a) generate advisory information, (b) provide favourable improvements or changes of maintenance practices, (c) provide favourable improvements or changes of operational practices, and (d) enable favourable changes in design methods by underwriting new materials and designs.

An advisory SHM system does not change any existing maintenance tasks, operational processes, structural integrity approaches, or design approaches. The use of an advisory system can generate evidence required to approve favourable improvement or changes to existing tasks and processes; it can also generate requirements for advanced SHM systems and new designs. The development of an advisory SHM system would require the least development efforts. The development rigour and efforts required for SHM systems that change existing practices are expected to be more than those required for SHM systems that improve these practices: an example of the former is a system that defers inspection or alters inspection intervals; an example of the latter is a system that does not change inspection tasks but performs them faster with better quality. SHM systems that would change existing design methods by underwriting new materials and designs would require the highest development rigour and efforts.

7.1.4 *Implementation of Determined Rigor and Assurance Levels*

The previous sections describe how the rigour of SHM development activities can be determined and how the associated development assurance levels can be assigned to items and functions. In order to implement a required level of rigour, the associated process activities described in industry accepted standards should be applied. For example, ARP4754A regards the activities described in DO-178B and DO-254 as a means to implement the determined development assurance rigour for software and electronic hardware items including sensors. Furthermore, more extensive qualification and environmental tests would be required to demonstrate a high degree of performance rigour following standards such as DO-160. It is worth mentioning that existing SHM systems, HUMS, and Engine Monitoring Systems (EMS) use sensors such as strain gauges and accelerometers, and sensors that measure parameters such as speed, temperature, engine spool speeds, etc. The design, production, and installation of these systems (hardware, software, and sensors) have been approved and certified after demonstrating that the appropriate degrees of rigour are correctly determined, implemented, and demonstrated using standards such as ARP4754, ARP4761, DO-178, DO-254, DO-160, etc. The previous sections and the following sections highlight the considerations specific to SHM.

Def Stan 00-970, which is used as a baseline in establishing appropriate design and airworthiness requirements, references standards such ARP4754, ARP4761, DO-178, and DO-160, and indicates that maximum use has been made of EASA regulations and certification specifications where these are applicable to both military and civil roles.

The previous sections indicate that for SHM systems that have no appreciable (or negligible) effects on aircraft structures and systems, the design, production, and installation of the systems may be approved by the holder of the type design documents (often, the aircraft manufacturer) through procedures approved by the regulator; for the other SHM systems, only the regulator can approve the design, production, and installation of the systems through projects that must involve the holder. The aircraft manufacturer may approve, under approved procedures, the use of a system to improve existing maintenance and operational practices without changing them; an example of such a system is a system that performs the same existing inspection tasks at the same intervals faster with better quality. Only the regulator can approve the use of a system to change existing maintenance, operational, or design practices; an example is a system that alters inspection intervals or defers maintenance tasks; however, the use of a system to introduce minor changes may be approved by the manufacturer under approved procedures with varying involvement levels from the regulator.

7.2 Validation

Validation is defined as “the determination that the requirements for a product are sufficiently correct and complete.” The validation process provides answers to the following questions: Do the requirements of the desired product fulfil its function? Are we making the right thing? Will the product, if produced in accordance with the requirements, do what the user needs and requires? Does the product definition correspond to its need and function? So, the validation processes ensure and evidence that “you are building the right thing”

The input to the validation process does not only include the SHM system requirements, but can also include information such as a definition of the system architecture, a description of the system operating environment and the development assurance levels allocated to each subsystem or item. The objective of the validation process is examining these inputs to ensure the following:

- The requirements are correct: The proof of correctness requires checking that each requirement is: traceable to a parent or rationales, unambiguous, not redundant and verifiable, has a unique interpretation, and can be physically implemented. The proof of correctness also requires ensuring that the requirements reflect safety analyses; e.g. all system failure conditions are correctly identified and classified; the development assurance levels are correctly assigned to functions and items.
- The requirements are complete: A list of all of possible types of requirements can form a basis for performing a completeness check. The proof of completeness requires checking that the requirements: fully satisfy parent requirements; fully include functional requirements traceable to system architecture and allocated to system items; include all of the safety requirements; fully adhere to regulatory and industry standards; cover all potential operational and maintenance scenarios; include all of the interface requirements to other systems/users; and adequately address design assumptions.
- The requirements are sufficient and necessary, and the probability of the presence of unintended functions is significantly reduced.
- The requirements comply with relevant airworthiness regulations and operating rules.
- The requirements address the needs of various SHM stakeholders: the developer, supplier, integrator, regulator, aircraft manufacturer, crew, operator, and maintainer. These needs can be addressed by firstly identifying the interfaces with aircraft structures and other systems as well as the interfaces with aircraft maintenance and operational processes. Then, the needs are addressed by identifying the relevant technical disciplines for each interface and the individuals that have the primary interest in the interface along with their development and review responsibilities.

Ideally, requirements should be validated before design implementation commences. However, adequate validation of requirements may not be possible until the system is implemented and tested. In other words, validation can be a staged process continuing through the development cycle; at each stage the validation activity provides increasing confidence in the correctness and completeness of the requirements.

Differences in the format of the validation processes of various organizations are expected. However, each development organization should clearly define and adopt a structured validation process to support certification. Key aspects of a structured validation process are presented in the following paragraphs.

7.2.1 Validation Planning

The structured validation process should pivot on a clear plan. The plan should define the methods to be used for requirement validation and describe how any development assumptions will be managed; in other words, the plan should outline how the requirements will be shown to be complete and correct. The plan should state the roles and responsibilities of the individuals required to perform identified validation activities; the plan should estimate the efforts and timescales required to perform these activities. The plan should also identify the required validation data to be generated and collected; it should describe how validation data, information and results will be managed stored and accessed; it should specify how reviews and investigations will be performed and their results recorded; etc.

7.2.2 Validation of Assumptions

Assumptions are introduced as a substitute for more explicit knowledge that will be available later or not directly provable at the time the information is needed. The processes used to validate assumptions may include: reviews, analyses, and tests.

Some requirements may be based on assumptions about operations, environments, reliabilities, and/or human factors rather than on traceable requirements. Examples of such assumptions are the assumption made about operational flight envelopes, exposure times to various environmental conditions, traffic densities, failure rates, potential failure latency, adequacy of scheduled maintenance tasks and their frequency, completeness of failure modes analyses, adequacy of durability data to demonstrate MTBF predictions, provisions for service and repair that do not degrade safety, response times of crew or maintenance personnel, and interpretation accuracy of system information under various environmental conditions and emergency conditions. These assumptions can be accepted based on reviews against existing industry experience, common practice, historical data, service and maintenance procedures, and industry standards.

Some requirements may be based on assumptions arising during the SHM iterative development process as a substitute for precise knowledge from concurrent interfacing systems underdevelopment. The validation process should check the reasonableness of these requirements, track them, and ensure that they are eventually adjusted based on the precise knowledge when becomes available. The adjusted requirements may require re-validation.

Installation assumptions such as isolation, environment, sources of contamination, mount integrity, grounding, and shielding should be validated by review against industry standards/practice, selective testing and/or inspections of mock-up, prototype, or production drawings/hardware.

7.2.3 Validation Rigour

The levels of validation rigour are determined by the DAL assignments to system functions and items. For a high DAL (A and B) a more rigorous validation process can include independent reviews of requirement data and supporting rationale to determine if there is sufficient evidence to argue the correctness and completeness of requirements; the reviews can include engineering reviews and reviews by customers, users, maintainers and certification authorities, and can also involve independent organizations.

For DAL A and B, the validation data and methods recommended to support certification usually include the following: PSSA, validation plan, validation tracking data and summary, traceability of requirements, rationale of derived requirements, similarity, and engineering reviews/inspections along with modelling, analysis, and/or tests.

For DAL E, all of these data and methods may not be required to support certification; however, the development of a system at DAL E should follow structured validation process, perhaps at a minimum effort, to support the development of usable useful SHM product. For comprehensive guidelines on DAL assignments, refer to ARP4754.

7.2.4 Validation Tracking and Summary

The status and data of the requirement validation process should be tracked with a level of detail proportional to the DAL assignments to functions and items. The tracked status and data should include references to: requirements, derived requirements, assumptions, sources of requirements, rationales, environmental and operational considerations, associated functions, hardware/software performance, DAL assignments, validation supporting evidence, validation methods, validation results (valid or invalid), etc. The tracked status/data should be updated regularly during the development and included in the validation summary.

The validation summary should provide assurance that the requirements were adequately validated. The summary should include: a reference to the validation plan and a description of any significant deviations from the plan, DAL assignments, validation tracking data, any supporting data, and validation results.

7.3 Verification

Verification is defined as "the evaluation of requirement implementation to determine that they have been met". The verification process provides answers to the following activities: Have we made what we were trying to make? Does the product conform to specifications? Specifications include product specifications, relevant regulations or any conditions imposed on the implementation processes. So, the verification processes ensure and evidence that "you built it right". So, the objectives of a successful verification process are:

- Confirm that each level of implementation meets its specified requirements;
- Confirm that the requirements are satisfied;
- Ensure that the safety analysis remains valid for the system as implemented;
- Confirm that the implemented system can correctly deliver the intended functions.

The inputs to the verification process include documented requirements and complete descriptions for each subsystem and items to be verified.

The verification activities do not only ensure correct implementation that delivers the intended functions, but also uncover and report any anomalies or unintended functions so that they can be rectified. The verification activities should be planned tracked and performed with rigour determined from the DAL assignments to items/functions.

7.3.1 Verification Planning

A structured verification process should pivot on a clear plan. The plan should identify all system configurations and items including any hardware and software items to be verified. The plan should identify the verification methods required to show compliance with each requirement for each identified item. The plan should identify any special test equipment and facilities required for verification. The plan should clearly define the success criteria required to judge the results of each applied verification method. The plan should organize and sequence key verification activities. The plan should estimate the efforts and timescales required to perform these activities taken into account the timescales of design activities. The plan should also identify the required verification data to be generated and collected; it should describe how verification data, information, and results will be managed stored, and accessed; it should specify how reviews and investigations will be performed and their results recorded; etc.

7.3.2 Verification Rigour

The levels of verification rigour are determined by the DAL assignments to system functions and items. Minimizing implementation errors can be achieved through rigorous verification methods; minimizing implementation errors can also be achieved through independence. The most common means of achieving independence in verification is independent specification and execution of verification methods such as tests, analysis and reviews; e.g. teams not involved in the system design independently generate the details of the verification methods.

For DAL A and B, the verification data and methods recommended to support certification usually include the following: SSA, verification plan, verification tracking data, verification procedures, verification summary, service experience, and reviews/inspections along with tests and/or analysis; the verification methods should involve some form of test and should also include tests targeted at minimizing the probability of the presence of unintended functions. The extent to which each method needs to be applied or data developed should be agreed with the certification authority. For DAL E, all of these data and methods may not be required to support certification; however, the development of a system at DAL E should follow a structured verification process support the development of usable useful SHM product. For comprehensive guidelines on DAL assignments, refer to ARP4754.

7.3.3 Verification Tracking and Summary

The status, data, methods, and procedures of the verification process should be tracked and documented with a level of detail depending on the DAL of the system or item being verified. The tracked information should include clear references to: requirements, associated functions, applied verification methods, verification procedures and results, verification conclusion (valid or invalid), etc. The tracked status/data should be updated regularly during the development and included in the validation summary.

The verification summary should provide assurance that the implementation of the system/items met the requirements; the summary should include: a reference to the verification plan and a description of any significant deviations from the plan, DAL assignments, verification tracking data, descriptions of any open problems, safety assessment results of these problems along with any supporting data, verification results, and verification coverage summary.

7.4 Validation and Verification Methods

For each requirement, the structured validation process should determine and apply a combination of methods necessary to validate the requirement and establish a required level of validation confidence. The validation methods include traceability, analysis, similarity, service experience, modelling, reviews, inspections, and test. The main verification methods include analysis, similarity, service experience, modelling, reviews, inspections, and test. The purpose of these methods is to verify satisfactory implementation of requirements in all intended operating environments. More than one verification method may be necessary to substantiate compliance with requirements and assure correct implementation under worst case scenarios.

Thus, the above methods can serve the purposes of both verification and validation; however, the emphasis of a method used for validation should be demonstrating the correctness and completeness of requirements; the emphasis of the same method when used for verification should focus on demonstrating satisfactory implementation of requirements.

7.4.1 Traceability

Each requirement should be traceable to one of the following (a) a parent requirement, (b) a missing parent requirement that should be added to the system requirements (c) rational of an architectural choice or a design decision resulting in a derived requirement that may not be uniquely related to a higher-level requirement, or (d) assumptions arising during the SHM iterative development process as a substitute for precise knowledge that will be available later from, for example, concurrent developments of interfacing systems.

7.4.2 Analysis

The analysis of an aspect, such as functionality, performance and safety involves an evaluation based on decomposing the aspect into simple elements to provide unambiguous validation and verification results.

Like any aircraft system, safety analysis methods should be used for requirement validation and verification. The safety analysis methods include the methods needed for conducting an accepted safety assessment encompassing FHA, PSSA, and SSA. These methods include Fault Tree Analysis (FTA), Dependence Diagram (DD), Markov Analysis (MA), Failure Modes and Effect Analysis (FMEA), Failure Modes and Effects Summary (FMES), and Common Cause Analysis (CCA). These safety-related analysis methods are described in ARP4761, Reference [27].

For some SHM systems, analysis may be required to demonstrate that the load carrying characteristics of the structures are not degraded by SHM; more specifically, rigorous development efforts at a high DAL may require analysis involving, for example: stress analysis, finite element analysis, and dynamic analysis that may take into account steady and unsteady aerodynamic characteristics as well as all significant structural degrees of freedom including rigid body motions and elastic modes.

Coverage analysis can be performed through traceability examination to determine the degree to which the requirements are sufficiently and correctly allocated and implemented. Furthermore, models, simulations, and tests can be implemented for analysis purposes.

Careful analysis efforts do not only check the correctness and completeness of requirements, but also provide compliance evidence of correct implementation capable of delivering target functionality, performance and safety objectives.

SHM analysis results can influence the required ingredients of the other validation and verification methods. Therefore, the analysis details should be carefully considered; Section 7.5 highlights some of these details.

7.4.3 Similarity/Service Experience

Similarity involves using gained experiences to validate requirements by comparing them to the requirements of similar certificated systems or items, or by comparing them to the requirements of existing acceptable applications. Verification evidence (credit) for systems/items may also be derived from satisfactory service experience or verification evidence previously gained from similar systems/items that have been successfully implemented for other aircraft. The relevant service experience and similarity data along with engineering and operational judgment should be well documented to show that all potential implementation and installation failures have been identified, classified and resolved.

For emerging SHM systems, such service experiences may have not been adequately accumulated. However, for an item of a SHM system, the similarity argument may be used if sufficient previous experiences are gained from an existing similar item. Validation and verification by similarity may be claimed if the two items have the same function and failure condition classification, and operate in the same environment with similar usage. Existing acceptable applications that may contain items similar to those of some SHM systems are NDI applications and ground-based data management systems. Certified systems such as HUMS contain airborne items such as processors, memories, and data transfer devices, which may be similar to airborne items of some SHM systems.

Service history may be used to support validation, verification, and certification of a new item or system: samples of a new SHM item or system can be shown to be airworthy; then, the samples can be fitted into a number of service aircraft; data collected from the samples can support the validation of specific requirements such as intended functions and intended use. The similarity argument gains strength as the applicable period of service experience increases.

7.4.4 Modelling

Models provide representations of given aspects of systems or items; the models are used for analysis, simulation, and/or code generation; they should be developed in structured way and should have unambiguous well defined characteristics. A system model can consist of a combination of software and hardware (computation and test article); a model of a deterministic system may be based only on computational software. The models of a desired system/item may be used to validate requirements, evaluate system parameters, and generate some of the verification evidence.

A proposed system can be also modelled by hardware and/or software prototypes. Furthermore, development versions of the proposed system can be used as prototypes. Prototypes permit interaction with the modelled system to (a) prove the correctness and completeness of requirements (b) provide some of the evidence of satisfactory implementation, or (c) highlight missing requirements, undesirable behaviours, and potential problems. A model of system environments can be developed and interfaced to a prototype to validate applicable requirements and provide a high degree of functional coverage. By exercising the prototype, missing requirements may be identified and the prototype updated by introducing the identified requirements. Thus, the prototypes are powerful validation tools that aid demonstrating the completeness of requirements.

Methods such as state diagrams can be used to construct scenarios that model the operational aspect of a system. These scenarios can describe, in detailed steps, how a system should function to accomplish a desired goal in response to inputs from users in all possible operational conditions. Exercising these scenarios is a powerful means for identifying any missing requirements and eventually demonstrating the completeness of requirements.

7.4.5 Test

Requirements may be validated by testing articles such as structural specimens instrumented with sensors, mock-ups of systems/items, prototypes, simulations, or actual hardware/software of systems/items.

Verification tests may also be used to support validation; in other words, testing may simultaneously serve the purposes of verification as well as validation. The verification tests provide repeatable evidence that verifies satisfactory implementation of requirements. Test readiness reviews establish the applicability of the test cases to system or item requirements.

The validation tests of an article can be conducted at anytime during the development phases when the article becomes available. Each article should be developed and tested using procedures documented in sufficient detail so that the article test results can be independently reproduced. Generally, an article is developed and tested to validate a group of requirements or to verify some of the implementation aspects.

By exercising an article, testing provides repeatable evidence of correctness and verifies that the requirements are satisfied. The purposes of testing a system/item are (a) to check that the requirements are met by the implemented system/item, (b) to demonstrate that the system/item implementation performs its intended functions, and (c) to provide adequate confidence that the implemented system/item does not perform unintended functions that impact safety. It should be noted that complete absence of unintended function can never be established by test. Problems uncovered during testing should be reported, tracked, corrected, and the corrected system/item retested.

The specifications of each test should include: the required input variability, the sequence of test actions required, the test rationales, the requirements covered by the test, and the expected results/measurements and their qualities. These results should be tagged with the specification version of the test and the design version of tested article; the results should be recorded and concluded with a clear statement about the success or failure in achieving the test objectives.

7.4.6 Reviews/Inspections

Reviews or inspections involve applying experiences of engineers through visual examinations of process documents, drawings, hardware, or software; they also involve witnessing tests, simulations, and demonstrations. The reviewers and their roles should be identified and their reviews structured and documented (e.g. through check lists). The reviewers should examine whether properly justified rationale or logic was applied and documented through the development phases including, for example: rationale for the allocation of requirements to hardware or software with appropriate safety objectives and development assurance levels, rationale for the classification of each failure condition through FHA, rational for any assumptions, and rational for each test, simulation, or demonstration. The rigour of a review depends on the review scope and details, the care taken with the review, the degree of independency of the reviewers, and their experience levels.

Reviews/inspections should be structured and performed to support the validation of requirements and the determination of their completeness and correctness. The reviewers should challenge the assumptions and interpretations of captured requirements to ensure that they have not caused deviations from the meanings of the original source of requirements. Prior experiences of reviewers that cover similar systems or items, if available, should be implemented as an effective means of validating derived requirements.

Reviews/inspections could also be structured and performed to verify that requirements are satisfied and to establish that the physical implementation of the requirements of a system or an item are met.

Problems uncovered through the review/inspection activities should be reported, tracked and corrected.

7.5 Detailed Validation & Verification Analysis for SHM

As defined in ARP4754, analysis is an evaluation based on decomposition into simple elements. Hence, decomposing the SHM intended functions to elementary functions (simple elements) is an essential analysis step. Furthermore, by decomposing the SHM intended functions into elementary functional tasks, these guidelines would cover the wide spectrum of SHM systems and avoid the exclusion of an SHM system that only performs a part of an intended function or that performs SHM tasks across two functions.

Therefore, the SHM intended functions should be broken down into elementary functional tasks as defined in Section 4.6. Each SHM elementary task, whether offered or imposed, should be clearly identified, its criticality assessed, validated and verified through the steps described in the following section.

7.5.1 Validation and Verification Steps

The validation efforts must check the completeness and correctness of requirements adequate to reliably perform the identified elementary functions. The validation and verification efforts should consider the following steps:

- Decompose the SHM intended functions into elementary functions; see Section 4.6;
- For each elementary function, identify from the system architecture, all physical items (and their functions) required to deliver the elementary function;

- Scrutinize the safety assessment results to ensure that the failure conditions of the SHM items/functions are adequately and correctly identified, and their severity classified;
- Ensure that a complete and correct set of requirements are allocated to each item and function; Section 6 presents, in detail, all potential types of requirements that can be allocated to SHM items and functions;
- Ensure that the DAL assignments to the items/functions are consistent with the severity of failure conditions taken into account the intended use of the elementary function and the requirements of the items/functions as well as any mitigation or independence methods used;
- Ensure that sufficient measurement characteristics are specified to achieve the elementary function with the rigour required; see Section C.
- Apply a combination of the validation and verification methods described in Section 7.4 with rigour determined from the DAL assignments.
- If required extend the methods to include conclusive validation and verification plan and efforts demonstrating that the SHM system when installed into the aircraft can conclusively deliver its elementary functions with the specified quality measures; see Section 7.9.

7.6 Qualification

Qualification is defined as a verification process to verify through tests that a product (often, an airborne system: hardware and software), complies with a specified set of requirements. The qualification tasks of a system include airworthiness qualification tasks to verify compliance with airworthiness requirements and demonstrate “fitness for flight”; they also include additional qualification tasks to verify compliance with performance, environmental, and functional requirements, and to demonstrate “fitness for purpose”. In other words, “qualification” is a subset of “verification”; the qualification subset focuses more on testing airborne subsystems to verify their fitness for flight and fitness for purpose. The qualification tests can be conducted by the system Design Authority (DA) and/or by subcontractors to the DA; the tests can be also witnessed or conducted by independent organizations. Usually, all of the qualification results are submitted by the DA to the regulator for certification.

Only a sample representing the system type is subjected to the qualification tests to prove the compliance of the type and obtain the regulatory approval. Often, systems subjected to qualification tests are not fitted into the aircraft because the tests would have thoroughly exercised them to a status close to or exceeding their design life to demonstrate fitness for flight. After successful qualification, each production system is subjected to a small set of acceptance and functional tests before the system delivery and installation to ensure manufacturing quality and correct functionality. The tests can include profile tests, e.g. a small number of temperature cycles and vibration cycles, to ensure manufacturing quality and correct functionality under typical environmental conditions. After the system installation into the aircraft, commissioning tests are conducted to ensure correct functionality and prove that the system does not unfavourably influence other aircraft systems.

The qualification of a system performing inspection tasks should include methods targeted at evaluating the reliability and inspection capability of the system to demonstrate fitness for purpose; existing methods used to qualify NDI applications should be adopted for SHM systems. It is worth mentioning that these NDI qualification methods have considered not only the NDI equipment but also application details such as structural details, fault types, inspection environments, and human factors including inspectors’ skills; some of these details are not applicable to built-in automated SHM inspection tasks; however, the main NDI qualification efforts should be applicable to SHM.

The main qualification efforts of NDI applications are targeted at tests to demonstrate and prove that an adequate inspection capability can be achieved. The tests are used to evaluate POD curves that indicate the probabilities and confidence levels associated with detecting different damage sizes. Often, the capability of NDI is measured by the smallest damage size "a_s" found with 95% confidence that the probability of detecting this size is greater than 90% (detecting "a_s" with "90/95" capability); the determination of the inspection intervals depends on this damage size. Therefore, qualification tests must be conducted to evaluate the POD curves and demonstrate the NDI capability (e.g. the capability of detecting "a_s" with "90/95" capability). MIL-HDBK-1823A describes the most common recommended ways of determining POD, which are "â vs a" followed by "hit/miss". MIL-HDBK-1823A also indicates that the previous Berens/Hovey method is still valid: proving the POD capability in NDI using "29 out of 29 (29/29)" method requires that the inspector must find the damage 29 times out of 29 attempts in his first trial. The literature indicates a large number of investigations targeted at the evaluation of POD curves for different NDI applications and for emerging SHM systems. The existing recommended "â vs a" and "hit/miss" POD methods used to qualify structural NDI applications along with the result of these investigations should be evolved, adopted, and used to verify the capability of SHM applications that perform similar NDI tasks. In other words, SHM systems that replace specific NDI inspections will likely be required to demonstrate some equivalent capability metric. However, it is recognized that POD methodologies, as currently utilized for NDI, may not be directly applicable or implementable to SHM systems. Thus, new or evolutionary approaches to demonstrate the capability of SHM systems may be required.

7.6.1 Qualification of SHM Airborne Equipment

The entire onboard components of SHM systems including sensors and avionics should be subjected to sufficient qualification tests including environmental tests according to standards such as DO-160, DO-254, and DO-178. The rigour of tests and their coverage depend on the DAL assignments to airborne items and functions through safety assessment methods; the DAL assignments should be based on the classifications of related failure conditions and based on the intended functions and the intended use of these functions along with the relevant requirements of the airborne items/functions. The qualification tests should demonstrate that each airborne component can survive, over a specified survivability period, its environments and the expected cyclic variations of these environments. The qualification tests should also demonstrate that the component continue to correctly perform its functions during the survivability period under a specified subset of environments in which SHM measurements will be acquired. The tests should cover environmental cycles/profiles of parameters such as temperature, altitude, humidity, shock, crash, vibration, explosion, water/fluids susceptibility, resistance to salt, sand, dust and fungus, g, emission, lightning, icing, electrostatic discharge and fire. The tests should cover representative structures with embedded or bonded SHM sensors. Special wires, connectors, and bonding materials should be also tested. Tests should be performed to demonstrate compliance with Electro Magnetic Interference (EMI) requirements and hardware interface standards (MIL-STD-461) and to demonstrate build-in test capabilities and durability attributes such as required MTBF or fatigue life. The tests should also demonstrate the presence of an adequate source of electrical power with integrity attributes commensurate with the determined equipment criticality; there must be no unacceptable reduction in the level of safety or reliability for other equipment as a result of acquiring power for SHM equipment.

7.7 Validation and Verification of Ground-based Equipment and Procedures

7.7.1 General

For many designs, ground-based equipment (hardware and software items) is developed as an essential part of the entire SHM to support achieving the declared intended functions. The ground-based equipment can be used to power, acquire, process, store, display and/or manage data sensed or acquired by airborne SHM equipment.

The SHM intended function can be targeted at improvements in aircraft maintenance, management or design; it can be targeted at interventions/changes in aircraft maintenance, management or design practices. Hence, the ground-based equipment can be used to make decisions pertaining to some improvement or intervention actions or can provide data to other SHM items that enable these actions; in other words, the ground-based equipment can be an important part of the determination process of improvement/intervention actions. Therefore, the developer of SHM ground-based equipment should carefully assess at aircraft and system levels the consequences of triggering wrong actions or failing to timely trigger required actions. The degrees of rigour required for developing, validating, and verifying the off-board equipment depend on the severity of these consequences: by applying safety assessment methods to the entire SHM system, the consequences of failure conditions of ground-based items can be identified and classified; then, DALs can be assigned to each off-board item/function taking into account the consequences of failure conditions, the intended functions, and the item requirements; the DALs determine the degrees of rigour required to avoid introducing errors during the development phases. The safety assessment may also introduce additional requirements that reduce the probability of failures to small values consistent with the consequences of failure; an example of such requirements is a requirement for independent equipment performing the same task.

7.7.2 Validation and Verification of Ground-based Hardware

Guidelines and standards such as ARP4754A, ARP4761, DO-160E, and DO254 focus on airborne systems and don't directly address off-board equipment. Nevertheless, any required assurance level and associated degree of rigour can be achieved by adapting these guidelines for ground-based applications: the guidelines can be adapted by filtering out the parts that can only be applied to airborne items; then, processes with varying degree of rigour can be compiled and used to develop, validate, and verify off-board items at any required DAL. As an example, the same materials, designs, and processes used to develop airborne items at DAL A can be used to develop ground-based items at the same DAL; for this example, the qualification tests of the ground-based items are expected to be less extensive than those of the airborne items because of environmental differences. MIL-STD-810 "Environmental Engineering Consideration and Laboratory Tests" can be consulted to define the extent of the qualification tests required to demonstrate the functionality, durability, and reliability of the ground-based hardware during specified maintenance-free period and under all the foreseeable environments in which the ground-based hardware may operate. Therefore, guidelines and standards such as ARP4754, ARP4761, DO-160 and DO-254 can be used to develop ground-based hardware items specially built for SHM. The specially built items can include, for example, computers, storage media and interrogation equipment and can be developed at DAL A, B, C, D or E. However, whenever feasible, the developers of ground-based hardware items should take advantage of COSTS products and design flexibility available to them.

The designers of ground-based hardware can implement design choices free from restrictions on features such as the weight and size of airborne hardware. Therefore, they can develop ground-based hardware with more flexibility taking into account appropriate assurance considerations which may differ from airborne hardware by using usual mitigation and independence techniques; see Section 7.7.5. As ground based systems may not have the same availability requirements compared to airborne systems (e.g. real time requirements), more development emphasis on integrity requirements through mitigation and independence techniques should be considered.

The designers of ground-based hardware can also make use of existing ground-based items including NDI equipment and COTS products such as personal computers, rugged laptops, servers, storage devices and data transfer devices. A DAL, if not known, can be assumed for each chosen COTS item (e.g. DAL E or DAL D); the assumptions about unknown DALs should be substantiated by, for example, qualification tests and service history; then, mitigation and independence techniques can be used to raise the assurance level of the entire SHM system to the required assurance level which may be higher if several independent software applications are used jointly to compute the data.

7.7.3 Validation and Verification of Ground-based Software

Although the title of DO-178 is "Software Considerations in Airborne Systems and Equipment Certification", the software development processes described in this widely accepted reference can be adopted to develop, at any required DAL, ground-based software specially written for SHM. In other words, operating systems, compilers, database engines, software development tools, processes, etc. used to develop and host airborne software applications can be used to develop and host ground-based software applications at any required DAL (A, B, C, D or E). A ground-based software application could be developed with assurance considerations less severe than those of the entire SHM system that hosts the application if the integrity of the application is raised by mitigation and independence techniques; see Section 7.7.5.

Furthermore, applicable processes can be extracted from guidelines such as: AC-29-2C-MG 15 "Certification of Transport Category Rotorcraft, Airworthiness Approval of Rotorcraft Health Usage Monitoring Systems (HUMS)" and DO-278 "Guidelines for Communication, Navigation, Surveillance and Air Traffic Management (CNS/ATM) System Software Integrity Assurance," which include some considerations for Commercial off the Shelf (COTS) items. DO-278 provides Assurance Level (AL) considerations for ground-based software. It defines assurance levels ranging from AL1 to AL6: AL1 is used for software that could cause or contribute to the failure of the ground-based system resulting in a catastrophic failure condition; AL6 is for software that could cause or contribute to the failure of the ground-based system resulting in no effect on the system. If such processes are adopted for SHM, mapping should be established and substantiated between the two sets of assurance levels: the required assurance levels A, B, C, D or E and the levels AL1 to AL6.

The developer of ground-based software can make use of 'user definable variables' approaches, which allow changes under configuration control procedures rather than software changes and, hence, can reduce the cost and time required for system modifications and upgrades. The developer of such a system have to ensure, with the assigned degree of rigour, that the system can function correctly for the range of all possible values of each variable and for any combination of them. The variables may include external algorithms that can be plugged into the system to, for example, calculate strain life. The software of such algorithms must be qualified with the same degree of rigour. Furthermore, both the configurable system and the added algorithms must be qualified to ensure that the added algorithms do not exhaust the memory or storage resources, or degrade the system performance.

7.7.3.1 The use of COTS software

Often, the data of existing ground-based structural health management systems are stored, analyzed and managed by COTS software and application programs; most systems employ two types of software: one type is COTS operational software including database engines and operating systems; the other type is application programs including data retrieval and analytical algorithms.

The SHM developers can make use of existing operational COTS software (operating systems, database engines, word processing tools, etc.). Such software is difficult to validate and verify using processes similar to those of DO-178. However, DALs, if not known, could be assumed for the used operational COTS software items; the DALs assumptions should be substantiated by, for example, service history and sufficient validation and verification activities (inspections, tests, and demonstrations) including independent validation and verification activities.

The COTS vendor may regularly update the operational software. Therefore, a transition and configuration management processes should be established to provide sufficient validation and verification data demonstrating that the application programs are correctly retrieved and compiled under the updated COTS versions; the validation and verification data should also demonstrate that all relevant data items are unchanged and correctly migrated to the updated versions.

The application programs operate on structural data to provide maintenance and management information. The application programs are hosted within COTS hardware (computers) and run under a COTS operating system (e.g., a Microsoft Windows operating system). The application software should be developed to the required development assurance level. For example, the software could be developed at a lower assurance level and its integrity is raised by various mitigation and management techniques (e.g. another software independent application performing the same computation). For some applications, the software should consider security features for authentication and access control to prevent unauthorized changes to software or data. The correctness of the results of an application program can be validated and verified by domain experts and, sometimes, by independent means of validation and verification. The application software can be developed, validated and, verified using the same processes used for equivalent existing software (e.g., NDI software). Generally, by adapting processes similar to those of DO-178, a high level of integrity can be assured for the application software.

7.7.4 Validation and Verification of Ground-based Data

The ground-based data includes: data downloaded from airborne SHM components, source codes of ground-based software, software configuration data, and data resulting from ground-based application programs. The integrity of the ground-based data should be validated and verified using processes such as those of DO-178, DO-200 "Standards for Processing Aeronautical Data," and DO-201 "Industry Requirements for Aeronautical Information". DO-200 defines the data assurance level as "the degree of confidence that a data element is not corrupted while stored or in transit". DO-200 categorizes the assurance levels into three levels: 1, 2, and 3 with 1 being the highest degree of confidence". So mapping between these three levels and the required level A, B, C, D, and E should be established and substantiated. The levels of rigor of the adopted processes should be consistent with the determined assurance levels of the data items including software; the assurance considerations should include with acceptable rigor, but not limited to, the following:

- Sufficient validation and verification activities commensurate with the data item's intended use and demonstrating that each item complies with its intended requirements,
- Configuration management providing, for each data item, assurance of validity, traceability, sufficient identification (labels, dates, suppliers, validity periods, etc.), and ability to recover the item if it is lost or corrupted,
- Security measures providing assurance of sufficient access control commensurate with the criticality of the data item,
- Sufficient automation of data transfer processes to avoid errors or losses due to manual transfer or manual inputs of data items, and
- Transition processes providing sufficient validation and verification evidence that the data

items are unchanged and correctly migrated to any updated versions of operating systems or COTS tools such as databases and word processors.

If the failure of a data item causes catastrophic failure, a high assurance level should be considered. In this case, the data item should be developed, validated and verified under strict considerations including rigorous considerations for configuration control, backup, storage, security, access control, authentication and any other features that prevent loss, unauthorized intervention or change to the data item.

7.7.5 *Validation and Verification by Similarity and Failure Mitigation Methods*

It is worth emphasizing that ground-based tools (software and hardware) have already existed and implemented within structural health management systems to monitor and manage aircraft structures and maintain their integrity. The existing tools include NDI equipment and COTS products such as personal computers, rugged laptops, servers, storage devices, data transfer devices, database engines, and operating systems. Often, the hardware of existing ground-based equipment, including NDI equipment, is developed with assurance considerations less severe than those of airborne hardware because the failure of the ground-based equipment does not pose immediate risk on passengers, crew, or civilians. Then, the integrity level of the equipment is raised by mitigation techniques to ensure correct functionality and mitigate failure risks. The mitigation techniques may be achieved through a combination of people, procedure, equipment or design (partitioning, redundancy, safety monitoring, and data integrity checking): for example, risks of failure can be mitigated by:

- People: sufficient training can be provided to engineers to optimally use the equipment and to check their functionality;
- Procedure: well defined processes can be introduced to ensure functional integrity;
- Equipment: a spare system can be made available to replace a faulty one;
- Redundancy: an alternative system can be used to confirm diagnosis or fault detection.

In other words, methods such as build-in tests and checking procedures are implemented to indicate whether the equipment is functioning or not; when the equipment fails, alternative stand-by equipment (mitigation methods) are used to deliver the same functionality; the integrity of the data can be raised by using independent equipment.

Therefore, the SHM ground-based equipment (software and hardware items) can be validated and verified using methods similar to those used for equivalent existing items as described in the above paragraph; the validation and verification processes should provide assurance that the integrity and quality requirements of the various equipment items are the same as those of existing or known equivalent items. The developed SHM ground-based system including the established mitigation methods should timely deliver the required functional quality with rigour commensurate to the DAL assignments; for example, a SHM ground-based system targeted at crack detection should enable the delivery of the required detection capability (e.g. 90/95 POD capability).

7.8 Remarks about End-to-End Validation and Verification Methods

For new SHM system, it is essential to demonstrate that the end-to-end system (airborne and ground-based components and procedures) provides at least capabilities equivalent to those of existing equipment and procedures (e.g. detection or monitoring capability) with the same quality (e.g. reliability and accuracy). Validation and verification at a high DAL should not be based only on equivalence, but should also consider other acceptable methods including independent activities.

Depending on the rigour required, equivalence or other validation and verification methods should be considered not only for the ground-based equipment, but also for the changes of procedures and actions arising from using data processed by SHM airborne or ground-based subsystems. These methods may include:

- Satisfactory service history,
- Physical inspection(s) by domain experts,
- Processing by alternative independent equipment, e.g. with dissimilar COTS processor,
- A combination of physical inspection(s) and independent processing.
- Comparisons between the actions arising from SHM results (which can lead to favourable maintenance, management, or operation improvements/changes) and existing equivalent actions (e.g. actual maintenance performed as a result of an existing NDI application): The approval required to use SHM and introduce these favourable improvements/changes requires satisfactory comparisons based on data collection from SHM systems. The amount and duration of data collection should be agreed between the applicant and the relevant authority at the beginning of an approval project on a case-by-case basis. The relevant authority can be the holder of a design approval document (often, the aircraft manufacturer) or can include the regulator and the holder of a design approval document. It is worth re-emphasizing that an agreement with certifying agencies on the quantity, types of tests, and analysis needed is considered to be an essential step in allowing the use of the SHM system on the aircraft when the SHM failure could lead to a major or more significant failure condition.
- An independent means of verifying the integrity of equipment (hardware and software) and the accuracy of the equipment results.
- Any other independent methods to verify the actions resulting from SHM processed data through a satisfactory comparison to actions obtained from the independent means.
- The independent verification means and efforts should be discontinued after significant quantities of SHM processed data consistently agree with the independent means, and after the authority (the regulator or his delegate within the aircraft manufacturing organization) approves any procedural changes to allow the use of the approved SHM equipment.

7.9 Conclusive Validation &Verification of SHM Intended Functions

7.9.1 Introduction

For some elementary functions such as detecting the size of a crack, collecting validation evidence of correct functionality through laboratory tests of simple structural specimens may not provide the conclusive proof of similar correct functionality (correct crack size) for the aircraft complex structural assembly. On the other hand, if laboratory tests similar to those used to commission NDI applications were conducted, the qualification of SHM would be straightforward. Perhaps, such laboratory tests would not lead to procuring SHM systems for each aircraft because of a key difference between NDI and SHM: the NDI equipment can move around the inspection areas of a number of aircraft; the SHM sensors are permanently installed in each aircraft. Because of this difference, the approved manuals of the NDI application can be easily updated in sympathy with infield experience or arising evidence; such flexibility does not exist for permanently installed SHM sensors.

Hence, it might be argued that the costs required for the development, procurement and use of NDI equipment would be less than those costs required for an equivalent SHM system, which include development, production and installation costs along with life cycle costs for systems fitted in each individual aircraft. Therefore, for a mature SHM system, cost benefits analysis would be required to demonstrate the falsity of this argument and justify replacing conventional NDI tasks with advanced SHM tasks. The cost benefits analysis should highlight the hidden costs of NDI and quantify the savings of SHM. For example, in 2006, Boeing indicated that for a typical fleet of aircraft, 70% of all structural maintenance expenses are incurred inspecting airframes during periodic maintenance tasks; the majority of the inspection expenses are associated with effort spent to access the inspection areas; the remaining 30% of the expenses are incurred repairing fatigue cracks and other structural damage if found during inspections. In 2004 Boeing indicated that for a 747-400 commercial aircraft, over 36000 labour hours are spent inspecting the aircraft for fatigue cracks and corrosion. Of these, 25000 hours are spent inspecting for corrosion; over 21000 of these hours are spent gaining access to hard to inspect areas; only 4000 hours are spent doing the actual inspection.

Because of the initial costs required to develop, produce, and install the SHM systems in each aircraft, conclusive evidence confirming the SHM correct functionality would be required to encourage the procurement of the systems. The conclusive evidence would be, for example, structural fault detection capabilities observed during the operation of the aircraft. The occurrences of structural faults such as cracks are infrequent, and hence, years of flight tests might be required to collect validation evidence; small number of flights would be only sufficient to prove the system “fitness for flight” and would be insufficient to prove “fitness for purpose”.

Therefore, a conclusive validation plan should be developed to describe how to extrapolate the results of a detection system from laboratory tests to actual aircraft; the validation plan should adequately describe the steps required to demonstrate and witness correct detection functionality at a pre-defined quality level. As an example, a validation plan should exist for a crack sizing task; the plan should describe the steps required to demonstrate and witness the required pre-defined quality of the sizing function in terms of POD characteristics. As suggested in Reference [67], the conclusive validation approach could vary between different SHM elementary functions (tasks) and technologies. So developing this plan is a validation application specific task; executing the plan is a verification task.

7.9.2 Generic Ingredients of a Conclusive Validation and Verifications Approach

A validation plan should be developed and implemented for each SHM system (hardware and software). The main element of the hardware is the SHM sensors; the main element of the software is the algorithm(s) that would operate on the sensor data to perform a specific SHM task/application; such an algorithm(s) is referred to as the “SHM Algorithm”. The following paragraphs present potential ingredients that could be considered to validate and verify structural fault detection tasks.

7.9.2.1 Laboratory tests

The laboratory tests should progress form very simple tests to flight tests; the tests may progress through (a) testing a sufficient number of simple structural specimens, (b) testing a reasonable number of complex specimens (e.g. three dimensional specimens) (c) testing a smaller number of representative specimens or components (d) testing an assembly, and (d) flight tests. Reprehensive prototypes of the SHM system (sensors and software) should be fitted in the tested articles. Faults should be induced in the specimens, components, and assembly at various locations; different fault sizes should be induced artificially by tools or naturally by allowing small faults to grow under typical repeated stress cycles. The tests should progressively provide increased confidence that the tested SHM system can deliver its intended functions.

7.9.2.2 A Procedure for Calibration/Configuration

It would be desirable to develop a procedure describing systematic generic steps for calibrating and configuring the SHM system and algorithms. Successful use of this tool would provide increased confidence that the results obtained from simple tests can be successfully extrapolated to cover more complex tests using the same procedure without any human interventions that may unwittingly introduce biased inaccuracies.

As a result of factors such as potential installation variances, changes in aircraft configurations, normal wear, and other aging effects, the configuration and calibration of SHM should be repeated. Therefore, calibration is not only required during the development phase, it is also required for commissioning the installed SHM systems and should be regularly repeated to confirm that the system continues to perform its intended functions with the specified quality

7.9.2.3 Non-Destructive Fault Simulation Equipment

Whenever feasible for the detection technology considered, it would be desirable to design fault simulation equipment that can simulate the effects of the target structural fault as felt by the sensors. By using the fault simulation equipment, the effects of a large number of simulated faults can be induced at various locations under various measurement environments in simple and complex structures. For example, if the SHM sensors detect signals generated by growing cracks, the fault simulation equipment could be a non-destructive emitter that produces pulses similar to those of growing cracks. The effects produced by the fault simulation equipment can be optimized and verified by comparing them with the effects produced by a small number of actual faults induced in the tested structures.

7.9.2.4 Commissioning the SHM System

The ingredients described above would enable validation and verification activities that should produce conclusive evidence of correct detection functionality: they should provide evidence confirming the extrapolation capability from simple specimens to complex assembly; they would provide cost-effective wide coverage of all potential fault locations under all foreseeable measurement conditions; they should demonstrate that a non-destructive fault simulation equipment would produce effects similar to those of actual faults. Thus, after installing the SHM system in the aircraft, only generic calibration/configuration methods would be required to calibrate, test, and commission the SHM system.

8 CERTIFICATION

8.1 SHM Certification

Certification involves activities to obtain the approval of the appropriate regulator that the applicable functional requirements, airworthiness regulations, and operating rules are met. The development and certification phases start after reaching an acceptable degree of maturity. The certification phases encompass tasks to witness and confirm that applicable airworthiness regulations, operating rules, and product functional requirements are met. They involve the SHM developer, OEM/TCH, and the regulator.

For civil aircraft and their products, the certification phases are often initiated through an application made by the product developer to the regulator; they are performed in parallel to the other evolution phases, Figure 4. The degree of the regulatory authority engagement at early evolution stages varies from one authority to another: FAA is not involved in the initial development stage of any new product; EASA however, emphasizes the need for early discussion with the regulator.

The certification phases presented in Section 8.4 are those recommended by the Aerospace Industries Association (AIA), General Aviation Manufacturer's Association (GAMA), and FAA in Reference [11], which presents a vision for Certification Process Improvement (CPI) that does not change what is done; rather it changes how it is done through a Partnership for Safety Plan (PSP) and a Project Specific Certification Plan (PSCP). PSP is a written agreement between the FAA and the Applicant that defines generic procedures to plan for product certification, establish the general expectations or operating norms, and identify deliverables. PSP also defines the general discipline and methodology to be used in planning and administering certification projects. PSCP captures procedures based on the PSP generic methodologies and applies them to a specific certification project; it is used as a project management tool providing milestones, performance measures, and information unique to the project. The certification phases conclude with approvals obtained from the regulator.

In the UK, the MOD mandates that "New UK military Air Systems that will be operated in the Service Environment on the UK Military Aircraft Register (MAR), and Major Changes to the Type Design of such systems already on the MAR, shall be certified prior to their Release to Service (RTS)", see the MAA RA 1500(1) of Reference [37]. As an AMC with this RA, MAA requires that the TAA responsible for the introduction of a new UK military registered air system or major changes to the type design of an air system should ensure that the air system is certified in accordance with a Military Air Systems Certification Process (MACP) consisting of the six phases described in Section 8.4; some of these phases may run concurrently; the first two phases are completed before MGA or before Business Case Approval (BCA) for lower value programmes, see Figure 8.

The MAA RA indicated the following rules:

- For changes other than Major, and which will not therefore be assured by the MAA through the MACP, the TAA must ensure, in accordance with extant DME 5000 series regulations, that the proposed change has undergone an evaluation process in line with the intent of this RA. The appropriate categorization and approval of Type Design changes will be subject to routine MAA assurance and audit. Thus, changes other than major changes may be self-certified in accordance with extant procedures by the TAA, who is often an aircraft PTL.
- A full Military Air Systems Certification Process (MACP) should be applied to those new air systems, and to those systems with major changes.
- A tailored version of MACP should be applied to (a) New UK military registered Air Systems that were post Main Gate approval and had not achieved RTS on 1 Sep 11 and (b)

Major Changes to the Type Designs of UK military registered Air Systems, that were post Main Gate (or Business Case Approval for lower value programmes) but had not achieved RTS on 1 Apr 12.

- For Urgent Operational Requirements (UORs), the guidance material of RA 1500(1) states that "The MAA will take due note of the degree of UORs when determining the level of rigour in the independent assessment of compliance".

8.2 Certification Outputs for Civil Aircraft Products

The outputs of successful certification phases include design approval, production approval, installation approval, and Instructions for Continued Airworthiness (ICA).

8.2.1 Design Approval

The approval is granted after the applicant has demonstrated and the regulator has verified that the design meets its requirements that include intended functions, safety requirements, system requirements, environmental test requirements, etc. Usually, the approval is granted at the end of the development period during which the applicant must have agreed with the regulator the means of compliance that will be met to address the applicable regulations, e.g. the applicable FAR, which must be identified paragraph by paragraph. The compliance documentation should account for all data pertinent to defining the type design including manufacturing specifications, and to demonstrating compliance including, but not limited to, test plans, test reports, test setup schematics, test instrumentation, drawings, analyses (e.g. stress analysis and safety analysis), material/process specifications, manuals, etc.

8.2.2 Production Approval

The approval is granted after the applicant has demonstrated and the regulator has verified that the applicant has developed and is capable of maintaining a quality assurance system that assures that only products and parts conforming to the approved design are released for service use. Applicants other than the DA must seek permission from the DA to use their approved design data.

8.2.3 Installation Approval

The approval is granted after the applicant has demonstrated and the regulator has verified that the installation meets the airworthiness requirements and is applicable to the aircraft model requested; the installation instructions include, for example, standard practices such as aircraft electrical wire selection and specific instructions that address more critical elements including procedures for determining the placement, installation, and post installation checkout of the system parts including any required interface units; post installation checkout procedures could include: electrical load analysis, equipment mounting/wiring/testing/verification, EMI tests, and Radio Frequency Interference (RFI) tests.

8.2.4 Instructions for Continued Airworthiness

The instructions cover the entire aircraft and its products including structures, power plants, systems and appliances required by Appendix H to 14CFR Part 25, and any required information relating to the interface of those products with the aircraft. The instructions include a plan to ensure continued airworthiness of those parts that could change with time or usage and include the methods used to ensure continued airworthiness. Section §25.1529 of Reference [3] describes the FAA regulatory requirements for the "Instructions for Continued Airworthiness" which must be written in English as a manual or manuals with sections containing maintenance manual, maintenance instructions, information to facilitate maintenance, airworthiness limitation section, and a section for EWIS.

8.2.4.1 Aircraft Maintenance Manual

The manual information includes:

- Information about the aircraft and its features and data to the extent necessary for maintenance or preventive maintenance,
- A description of the aircraft, its products, and installations,
- Basic control and operation information describing how components and systems are controlled and operate, including any special procedures and limitations, and
- Servicing information covering servicing points, capacities of tanks, reservoirs, types of fluids to be used, pressures applicable to the various systems, location of access panels for inspection and servicing, locations of lubrication points, lubricants to be used, equipment required for servicing, tow instructions and limitations, mooring, jacking, and levelling information.

8.2.4.2 Maintenance Instructions

The maintenance instructions include:

- Scheduling information for each part of the aircraft and its products that provides the recommended periods at which they should be cleaned, inspected, adjusted, tested, and lubricated, and the degree of inspection, the applicable wear tolerances, and work recommended at these periods along with the recommended overhaul periods, necessary cross references to the Airworthiness Limitations section, and, an inspection program that includes the frequency and extent of inspections,
- Troubleshooting information describing probable malfunctions, how to recognize those malfunctions, and the remedial action for those malfunctions,
- Information describing the order and method of removing and replacing parts with any necessary precautions to be taken, and
- Other instructions including procedures for system testing, symmetry checks, weighing, determining the centre of gravity, lifting, and storage limitations.

8.2.4.3 Information to Facilitate Maintenance

Supporting maintenance information includes:

- Diagrams of structural access plates and information needed to gain access for inspections when access plates are not provided,
- Instructions for special inspections such as radiographic and ultrasonic, if required,
- Instructions for applying protective treatments to the structure after inspection,
- Structural fasteners' data such as: identification, discard recommendations, and torque values, and
- A list of special tools needed.

8.2.4.4 Airworthiness Limitations Section

The ICA must contain a section titled Airworthiness Limitations that is segregated and clearly distinguishable from the rest of the document. This section is FAA approved and specifies maintenance required under §43.16 of Reference [8] and §91.403 of Reference [9] unless an alternative program has been FAA approved. The section must include:

- Each mandatory modification time, replacement time, structural inspection interval, and related structural inspection procedure approved under §25.571 (Damage-tolerance and fatigue evaluation of structure),
- Each mandatory replacement time, inspection interval, related inspection procedure, and all critical design configuration control limitations approved under §25.981 (Fuel tank ignition prevention).
- Any mandatory replacement time of EWIS components as defined in section 25.1701.
- A Limit Of Validity (LOV) of the engineering data that supports the structural maintenance program, stated as a total number of accumulated flight cycles or flight hours or both, approved under §25.571. Until the full-scale fatigue testing is completed and the FAA has approved the LOV, the number of cycles accumulated by the airplane cannot be greater than1/2 the number of cycles accumulated on the fatigue test article.

8.2.4.5 EWIS ICA

EWIS ICA must be prepared as defined by §25.1701 and approved by the FAA. The EWIS ICA section must include maintenance and inspection requirements for the EWIS developed with the use of an enhanced zonal analysis procedure that includes:

- Identification of: each aircraft zone, each zone containing EWIS, each zone containing EWIS and combustible materials, and each EWIS zone in close proximity to primary/back-up hydraulic, mechanical/electrical flight controls and lines;
- Identification of tasks, and the intervals for performing those tasks, that will reduce the likelihood of ignition sources and accumulation of combustible material;
- Identification of procedures and the intervals for performing those procedures that will effectively clean the EWIS components of combustible material if there is not an effective task to reduce the likelihood of combustible material accumulation;
- Instructions for protections and caution information to minimize contamination and accidental damage to EWIS, as applicable, during performance of maintenance, alteration, or repairs.

The EWIS ICA section must also include:

- Acceptable EWIS maintenance practices in a standard format,
- Wire separation requirements as determined under §25.1707,
- Information about the EWIS identification method and requirements for identifying any changes to EWIS under §25.1711, and
- Electrical load data and instructions.

8.3 Approval Forms of Civil Aircraft Products

For civil applications, the approval form can be: Type Certificate (TC), Supplemental Type Certificate (STC), Technical Standard Order Authorization (TSOA), Parts Manufacturing Approval (PMA), Production Certificate (PC), and Airworthiness Certificate (AC).

8.3.1 *Type Certificate*

TC is a design/installation approval of a new product or modifications to an existing product.

8.3.2 *Supplemental Type Certificate*

STC is a design/installation approval of modifications to improve existing product for a supplier other than the holder of the TC.

8.3.3 *Technical Standard Order Authorization*

TSOA is a design and production approval granted after reviewing an applicant's statement of conformance to the requirement of Technical Standard Order (TSO).

8.3.4 *Technical Standard Order*

TSO is a minimum performance standard for specified materials, parts, and appliances used on civil aircraft.

8.3.5 *Parts Manufacturing Approval*

PMA is a design and production approval based on prior approved design and installation data such as STC.

8.3.6 *Airworthiness Certificate*

AC is an approval indicating that the owner of a registered aircraft can safely operate and maintain the aircraft; the certificate is transferred with the aircraft and can be surrendered, suspended, revoked or terminated; the certificate may be obtained by the holder or licensee of a TC upon compliance with Sections §21.173 through §21.189 of Reference [1]; the owner of the aircraft may also apply for AC.

8.4 SHM Certification Phases, a Civil Aircraft Example

In Reference [11], AIA, GAMA, and FAA have recommended five typical certification phases through PSCP for aircraft products including avionics and structural products such as wings. The following subsections briefly describe these phases. It is worth mentioning that the grouping of the certification activities may change between various regulators; the main certification activities within these phases are essentially the same (planning, meeting minutes, analyses, reviews, witnessing, inspection, etc.).

8.4.1 *Conceptual Design*

This phase is initiated when the applicant begins design concept for a product that may lead to a viable certification project. The intent is to ensure, at early stage, value added joint involvement with an expectation to highlight critical areas and related regulatory issues. The key activities and outcomes of this phase include:

- Scrutinize information about new designs, technologies, materials, and processes;
- Conduct initial safety assessment;
- Propose and agree preliminary certification basis and means of compliance;
- Formulate a preliminary PSCP including a plan for resolving critical issues associated with, for example, new designs, co-production, and foreign suppliers.

8.4.2 *Requirements Definition*

This phase clarifies the product definition and the associated risks; the phase concludes with a mutual commitment to move forward with product certification. The key activities and outcomes of this phase include:

- Formally start the project through activities including: submission of an application to FAA, acknowledgment of application, establishment of project, and establishment of FAA and applicant project certification team;
- Review applicant's data including: descriptive design data, production data, critical issues definition, refined safety assessment, and proposed schedule;
- Agree certification basis plan and, identify critical issues, specific regulatory requirements and methods of compliance;
- Develop a more formal preliminary PSCP including project milestones and related events such as program status reviews.

8.4.3 *Compliance Planning*

During this phase, a PSCP is completed. The plan is a tool to which the responsible parties commit and use to manage the product certification project. The information required to finalize the plan includes: initial FMEA/safety assessments, refined critical issues, and production processes. The outcome of this phase includes:

- Signed PSCP and agreed project schedule with established FAA/Applicant milestones for completion of applicable items affecting the completion of the project such as: analyses, test plan submission, inspection authorization, conformities, flight test and critical issues resolution plan,
- Agreed certification basis and compliance check list,
- Identification of stakeholders, including suppliers and installers, and
- Resource requirements, conformity procedures, and project evaluation measures.

8.4.4 *Implementation*

During this phase, the applicant and FAA work closely in managing, refining, and achieving their agreed PSCP to ensure that all agreed specific certification requirements are met. The information required during this phase includes: production analysis, witnessing, inspection results, and safety analysis. The key activities and outcomes of this phase include:

- Demonstrate compliance, verify conformance requirements, and convene a final certification board meeting;
- Meet milestones for completion of applicable items affecting the completion of the project such as analyses, test plan submission, inspection authorization, conformities, flight test, and critical issues resolution plan;
- Complete test plans/reports, conformity requests, inspections, and compliance documentation;
- Issue design and production approvals.

8.4.5 *Post Certification*

During this phase, project follow-up and closure tasks provide the foundation for continued airworthiness activities and certificate management for the remainder of the product's life cycle. The information required during this phase can include, as applicable: airworthiness limitations, maintenance & operation requirements, project lessons learned, relevant safety data, certificate data sheet, evaluation findings, and design change data. The phase concludes by preparing: compliance summary document, inspection report, ICA, and continued airworthiness management plan.

8.5 The Phases of the UK Military Air Systems Certification Process (MACP) [37]

The word Product in the following subsections refers to new air systems (type designs) or major changes to air systems.

8.5.1 Phase 1 - Identify the Requirement for and Obtain Organizational Approvals

Organizations with airworthiness responsibilities for the product must hold appropriate approvals: normally, the approvals should be under the MOD Design Approved Organization Scheme (DAOS); alternative approvals may be acceptable where the TAA, who must hold an appropriate Letter of Airworthiness Authority (LoAA), can demonstrate to MAA that they are appropriate and equivalent.

One of the MOD pillars of airworthiness is the use of competent organizations. The DAOS is a mechanism by which the competence of design organizations can be assured. According to RA 5101(1) “An organization shall only be included in the DAOS and awarded approval for a defined range of products when it is in the interests of MOD and when the organization has been accepted by the Military Aviation Authority (MAA)”.

Reference [37] also presents the rules for projects intending to place reliance on the approvals of other military regulators or military certification bodies.

8.5.2 Phase 2 - Establish and Agree the Type Certification Basis (TCB)

The Type Certification Basis (TCB) is the list of design standards and other requirements and special conditions against which the design will be certified. The Product TCB must be included in an airworthiness strategy, and involves selection of applicable certification specifications or specifications proposed by the TAA. The selected specification must be agreed by the MAA. The default specification is Def Stan 00-970; where Def Stan 00-970 has not been used, the TAA should demonstrate the equivalence of the selected airworthiness codes as described in Reference [37]. Normally, the most recent version of a specification will be applied. Special conditions included in the TCB, to cater for areas where extant certification specifications are judged to be inadequate, will be identified. The TCB will be effective for a period of five years from the date of agreement. Should the RTS not be achieved within that timescale, a review of the changes to the specifications that defined the TCB will be required to assess any shortfall against contemporary requirements. The MAA will agree with the TAA which of these changes need to be adopted as part of an updated TCB.

8.5.3 Phase 3 - Agree the Certification Programme

The Certification Programme (CP) will be owned and managed by the TAA and agreed with the MAA, and will usually form part of the Integrated Test, Evaluation and Acceptance Plan (ITEAP). For each element of the TCB, the CP will identify the proposed means of compliance that may include: compliance statement, design review, calculation, analysis, safety assessment, simulation, inspection, equipment qualification, laboratory test, ground test, and flight test. In the case of tests, the TAA must certify that either the test specimen conforms to the type design, or that any deviations from the type design do not influence the test. The CP will also identify when the compliance documents or evidence will be available and include periodic progress reviews between the MAA, TAA, and other relevant organizations.

8.5.4 *Phase 4 – Demonstrate Compliance*

To demonstrate compliance, the TAA must provide the MAA with the evidence identified in the CP. It is an obligatory that Independent Safety Audit (ISA) and Independent Technical Evaluation (ITE) should be considered for systems in the higher risk classes and in other cases if the aircraft, its systems or equipment are novel, complex or high risk, RA1220(3). The TAA will be expected to ensure the design is subject to independent evaluation and audit; and the design organization will be expected to have undertaken independent internal compliance verification of all evidence prior to submission. A TC or STC issued by a recognised civilian or foreign military authority may be used, in part or in full, as evidence of compliance with the TCB. EASA and the FAA are automatically recognised by the MAA. Other civilian or foreign military authorities will need to be assessed on a project-by-project basis under arrangements agreed in Phase 1. At the conclusion of this phase, the TAA must produce a Type Certification Exposition (TCE) that demonstrates compliance with each element of the TCB, identifying any airworthiness provisions not complied with and compensated for by factors that provide an equivalent level of safety. The TCE must include details of the type design, operating limitations, and a draft Military Type Certificate (MTC) Data Sheet (MTCDS).

8.5.5 *Phase 5 – Produce Final Report and Issue Certificate*

The MAA will review the TCE to confirm that the design conforms to the TCB, and to determine any areas where compliance evidence is incomplete. The outcome of the MAA's analysis will be a formal certification report that will underpin the subsequent issue of a MTC, Approved Design Change Certificate (ADCC), or Statement of Type Design Assurance (STDA) as appropriate.

8.5.6 *Phase 6 – Undertake Post-Certification Actions.*

After a new air system has been certified there will be ongoing involvement from the MAA in approving major changes to the type design and in monitoring airworthiness throughout the air system lifecycle. The latter activity will include assurance activities such as attendance at: type airworthiness reviews, safety meetings, condition surveys, AAA, and structural, systems and propulsion integrity working groups.

8.6 Certification and Approval Outputs for UK Military Aircraft

According to the MAA regulatory policy MAA01, Reference [29], for an aircraft to operate on the military register, either a RTS, Military Flight Test Permit (MFTP), or Certificate of Usage (CofU) must be issued. Any aircraft operating outside of the RTS must have either a MFTP or CofU.

8.6.1 *Release to Service (RTS)*

The RTS is a central source document for the Aircraft Document Set (ADS). This set includes the RTS, Aircraft SC, Aircraft Maintenance Manual (AMM), Operating Data Manual, Flight Reference Cards, Support Policy Statement, Engineering Air Publications including Flight Test Schedule (FTS), and SOIU. The RTS contains explicit cross-references to appropriate documents within the ADS. The RTS handles safety information and helps define the safety envelope of the aircraft with the relevant information presented without omissions. The RTS is a reference work aimed at senior supervisors, MOD level staff and HQ staff who are able to absorb, and need, a wider view of the aircraft limitations than line aircrew.

According to RA 1300(1) of Reference [32], "A RTS shall be prepared for all aircraft that are subject to MAA regulated Service flying". As AMC with 1300(1), the RTS should be an integrated document, with all clearances and associated limitations lodged in the appropriate part, and should conform to the following principles:

- The primacy of the as-flown standard of aircraft as the focus for project safety management activity, and the standard to be underpinned by the system SC.
- The RTS identifies and specifies the approved configurations for the as-flown aircraft, and reflects the procedural safety mitigations identified by the SC for the as-flown aircraft.
- All equipment authorized to be carried in or fitted to the aircraft, but not included in the basic design standard, is included in the RTS.
- For Air Launched Weapons (ALW) and Airborne Forces Equipment (AFE) with its own Equipment Release, the Release to Service Recommendation (RTSR) will cover loading, carriage, and flight-handling.
- Temporary Information should be recorded in the RTS.
- Clearances with Limited Evidence (CLE) and Operational Emergency Clearances (OEC), should be recorded in the RTS, but flagged as appropriate for management review.
- An audit trail should be detailed in the RTS.

The RTS is issued by the Release to Service Authority (RTSA) on behalf of the Senior Duty Holder (SDH) and, where appropriate, will be supported by a MTC issued by the MAA. The RTSA is responsible for the authorization and issue of the RTS for an aircraft type with due attention to continual assuring of the validity of the RTS. Where operational imperatives may result in higher levels of risk exposure or where supporting evidence is still immature, the RTSA must consider clearances beyond the certified RTSR (e.g. OEC and CLE), and advise Duty Holders (DHs) as appropriate. The Aviation DHs have a personal level duty of care; they are legally accountable for the safe operation of systems in their area of responsibility and for ensuring that risks to life are reduced to at least tolerable and ALARP.

8.6.2 *Military Flight Test Permit (MFTP)*

A MFTP is issued when an aircraft, in support of an MOD contract, is required to be operated outside the 'In-Service' environment. It must be supported by a platform SC and authorised by the TAA.

8.6.3 *Certificate of Usage (CofU)*

For contractor owned aircraft operating on the Military Register, the TAA will produce a CofU, for approval by the appropriate MOD 2* Sponsor (e.g. Rear-Admiral, Major-General and Air Vice Marshal), supported by the contractor's SC, which will specify the conditions under which the aircraft can operate on the register.

8.6.4 *Military Type Certificate (MTC)*

For a new air system, successful completion of the full MACP described in Reference [37] will result in the MAA issuing a MTC to the TAA underpinned by the production of an MAA Type Certification Report (TCR). Unlike a civil TC, a MTC covers the entire air system including engines and propellers, where applicable. The MTC certifies that the air system: (a) has been designed by an approved organization; (b) meets the approved TCB, or that any airworthiness provisions not complied with are compensated for by factors that provide an equivalent level of safety; (c) will remain airworthy in its approved roles when operated and maintained in accordance with the approved data.

The MTC will list any conditions, restrictions, or operating limitations and will be accompanied by the MTCDS describing the TCB and giving general information about the type design.

8.6.4.1 Up-Issued Military Type Certificate

Successful completion of the full MACP for a major change to a type design will usually result in the MAA up-issuing the MTC if one exists.

8.6.5 Approved Design Change Certificate (ADCC)

An ADCC is the equivalent of a MTC, but is limited to the scope of the design change. Where the change to the type design of an in-service air system is so extensive that a substantially complete investigation of compliance with the applicable TCB is required, e.g. on the introduction of a new mark, then the outcome of the MACP could be the issue of an MTC rather than an ADCC.

8.6.6 Statement of Type Design Assurance (STDA)

Tailored application of the MACP will normally result in the issue of a STDA to the TAA. The STDA will identify the extent to which the MAA has been able to assure the certification evidence provided and detail any areas where the evidence is unavailable, incomplete, or inadequate. If the MAA's certification assurance activities conclude that the requirements of the MACP have been met in full, a MTC or ADCC (as appropriate) may be issued rather than a STDA.

8.6.7 Relationship of MTC, ADCC or STDA with the RTS Recommendation (RTSR)

The MTC, ADCC or STDA, together with the underpinning TCR, will be used by the TAA in support of the initial RTSR made for the new Air System or a Major Change. The initial RTSR must be submitted to the RTSA and the MAA. For new aircraft and Major Changes that result in the Mark Number for the aircraft changing, these recommendations will be subject to independent audit by the MAA. For all other Major Changes, it will be decided by the MAA, in consultation with the RTSA and TAA, during Phase 3 of the MACP, as to whether the MAA will carry out an RTSR audit in addition to a TCR.

8.6.8 Release to Service Recommendation (RTSR)

The RTSR is the statement that an acceptable SC has been prepared for the aircraft and its equipment. It is written for the aircrew and engineers responsible for the day-to-day supervision of flying operations, and the desk officers responsible for developing policy and procedures.

According to RA 1300(2) of Reference [32], "The PTL shall prepare the RTSR, on behalf of CDM, to the satisfaction of the HOC/RTSA". Note: CDM is the Chief of Defence Materiel; HOC is the Head of Capability; CDM is the head of the Defence Equipment and Support (DE&S) organization formed by merger of the Defence Procurement Agency (DPA) and the Defence Logistics Organization (DLO).

The RTSR should be the foundation of an eventual RTS, with all clearances and associated limitations lodged in a format consistent with the RTS structure. Once content, the PTL should certify that the initial RTSR provides an acceptably safe operating envelope for subsequent approval and authorization. This certification should exclude CLE and OEC although they may be included in the document.

For military variants of transport aircraft, the RTSR may be based on the civil TC and the civil flight manual of the aircraft concerned. For foreign aircraft, the RTSR may be based on the certificate of the appropriate authority. Where flight development and evaluation is undertaken in more than one country, and where agreed joint flight clearance programmes are arranged, the RTSR may be based on the certificate of the appropriate authority.

Ideally, the aircraft or aircraft weapon system will receive a RTSR that imposes no undue restrictions on either the roles or the conditions in which the aircraft may operate. In practice, the achievement of this ideal might incur unacceptable delays. It might therefore be necessary, in the first instance, to issue an initial RTSR of restricted scope; indeed, even in the long term it might not be possible to remove all restrictions. To facilitate delivery of the aircraft, it might also be necessary to clear certain aspects of the aircraft weapon system in advance of others. In such cases, the RTSR would proceed in stages. The priority of each stage will be agreed between the HOC, the RTSA, and the PTL.

8.6.9 RTSR for Remotely Piloted Air Systems (RPAS)

The guidance material of the MAA RA 1300(2) indicates that RTSR procedures for RPAS are the same as those for manned aircraft. Achievement of the airworthiness criteria for RPAS requires an assessment of all the factors that influence the safety of the overall system. A RPAS confined to use within a safe area may not require the same level of design integrity when compared to one that is to be cleared for regular operation over an urban population. Each system design needs to be tailored to its specified operational environment. In assessing the safety of a RPAS, the PTL will take into account the following factors and any others that are relevant to the system, its intended operation, and area of flight: (a) the reliability of the air vehicle (including the structure, engine, avionics, etc.), (b) the reliability of the command and control systems and links, (c) the reliability of software, (d) the reliability of any flight termination systems, (e) the mode of operation, (f) the risk of collision with other aircraft, (g) the density of civilian population in the area which it over-flies, and the exposure time of that population, (h) any risk reduction techniques or procedures, (i) peace, tension, threat, Transition to War (TTW), and war situations. MAA has noted that the term “reliability” will be interpreted in its broadest sense: i.e. including the MTBF of the equipment together with its susceptibility to external influence and its robustness to mishandling and errors in operation.

8.6.10 Certificate of Design

Prior to the first flight of a new aircraft, aircraft weapon system, unmanned air vehicle, tethered balloon or airborne forces, the contractor should submit a Certificate of Design. The Certificate of Design should include any exceptions or limitations to the requirements specification. Contractor's inspection, demonstration, analysis, and test should establish the extent of compliance with the specification. When compliance with the requirements of the specification has been demonstrably satisfied, the Certificate of Design for the relevant design should be submitted to the PTL. The PTL should decide if modification action requires re-issuing or amending the Certificate of Design. The design configuration of the aircraft should be explicitly defined in the Certificate of Design. Reference [36] provides the regulations governing the certification of designs; a set of these regulations are quoted hereafter.

- RA 5001(1): “The Contractor/Design Organization shall certify the extent to which the design satisfies the requirements of the specification/Cardinal Point Specification (CPS) issued by or on behalf of the MOD”.
- RA 5101(1): “An organization shall be included in the DAOS and awarded approval for a defined range of products only when it is in the interests of MOD and when the organization has been accepted by the Military Aviation Authority (MAA)”.
- RA 5102(5): “As necessary, during the design, development, construction and testing of

materiel the contractor shall make available to MOD drawings, design data, calculations and reports of important tests, such as wind tunnel, structural, safety, functioning or flight tests, so that questions which may affect the safety or performance of the completed project may be discussed at an early stage".

- RA 5103(2): "Certificates of Designs (CofDs) shall be provided on the appropriate form".
- RA 5103(1): "The CofD shall be signed by approved members of the Design Organization (DO) and the Project Team Leader (PTL)".
- RA 5103(3): "The DO shall retain the original signed CofD with the master records".
- RA 5103(4): "The DO shall submit a CoD to the PTL for sub-contracted items when they are accepted by the DO".

8.6.11 The Airborne Equipment Release Certificate (AERC)

The carriage and despatch of Airborne Equipment (AE) from aircraft present risks to life additional to those from the aircraft to users, public, and military personnel. The information in the Airborne Equipment Release Certificate (AERC) underpins the airworthiness of the AE when carried in and despatched from an aircraft. It informs the platform's RTS on the carriage and operation of the equipment concerned. The MAA RA 1345(1) mandates that "All AE shall be certified by the issue of an AERC". AE is the generic term covering the wide variety of parachuting assemblies for personnel and equipment, airdrop platforms, supply dropping equipment and ancillary items that are used for the insertion of personnel and equipment onto pre-planned Drop Zones. This equipment can be split into two areas: Airborne Forces Equipment (AFE) and Aerial Delivery Equipment (ADE).

The Airborne Forces Equipment Release Certificate (AFERC) is the formal certification, by the Hercules Tri-star PTL and DRTSA(FW), that the particular item of ADE meets the criteria for safety and airworthiness. DRTSA is the Delegated Release To Service Authority. The AFERC informs the host platform's RTS of the ADE limitations and is subordinate to that RTS whilst the ADE is carried with / despatched from the aircraft. The AFERC must be supported by a safety case for the equipment concerned and appropriate certificates of design.

8.6.11.1 Non Aircraft Type Specific Equipment

The MAA RA 1340(1) mandates that "All equipment to be carried in or fitted to the aircraft shall be authorized and included in the RTS".

Equipment not basic to the aircraft is the generic term used for items such as role equipment, stores, ADE, AFE, Aircrew Equipment Assemblies (AEA), Helicopter Under Slung Load Equipment (HUSLE), and carry-on equipment. It is fundamental that all equipment authorized to be carried in or fitted to the aircraft is included in the RTS.

Because of their complexity, or because they may be carried by several aircraft types, some of the projects of the "Non Aircraft Type Specific Equipment" are sponsored separately from individual aircraft types. In these instances, the platform PTL will ensure that the equipment safety is addressed in the SC for the aircraft and reflected in the RTS for the aircraft by including all limitations applicable to the airborne carriage of stores or equipment on that aircraft. The platform PTL will liaise with the equipment PTLs to ensure that the necessary safety justification is provided covering all aspects of the proposed use of potentially hazardous equipments. Although this analysis includes an assessment of the hazard posed to the aircraft or crew by the operation of equipments, it must not include the equipments' effectiveness in role; for example, assessment of the in-flight use of medical equipment must not include an assessment of the hazard posed to the patient. In cases where weapons, aeroengines, ADE, and other airborne equipment are to be fitted to several aircraft types, the equipment PTL will produce an equipment SC, covering the safety features of the equipment, the achievement of the safety, target and the way it will be safely interfaced to the aircraft.

8.6.12 The Air Launched Weapon Release Certificate

The carriage, launch and jettison of Air Launched Weapons (ALW) from aircraft present Risks to Life, additional to those from the aircraft, to users, the public and military personnel. The platform TAA is wholly responsible for the safety of the complete weapons system. The purpose of an ALW Release Certificate (ALWRC) is to assist him to discharge this responsibility. The MAA RA 1340(1) mandates that "All ALW shall be certified by the issue of an ALWRC".

ALWs are defined as those weapons Carried, Released (including launched, fired, or dispensed) and Jettisoned (CR&J) from an aircraft or RPAV. These comprise: all bombs, missiles, rockets, aerial mines/depth charges and torpedoes which have been designed for CR&J from external or internal armament installations on fixed and rotary wing aircraft or RPAV. The definition covers both live and inert variants of the subject items, but excludes guns & ammunition up to 20mm calibre, and countermeasures.

The ALWRC is the certification by the ALW PTL that an acceptable Safety Assessment has been prepared to assess the CR&J of the ALW within defined environments and performance envelopes on the nominated platforms, provided that the ALW has been stored, transported and maintained in accordance with the ALW Safety Assessment and manufacture to target or disposal sequence.

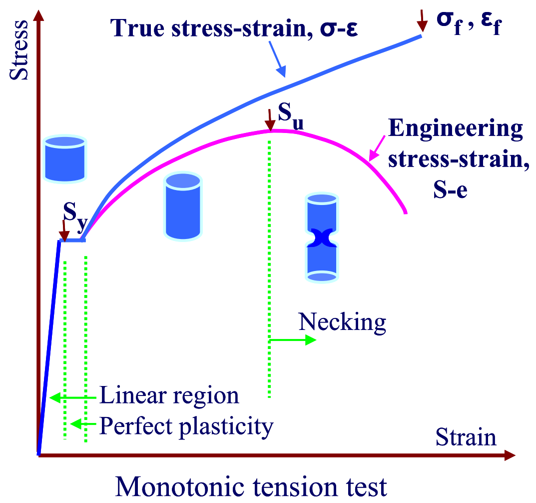
A Appendix A: Aircraft Design and Maintenance Philosophies

A.1 The Safe Life Philosophy

Whilst DT is the dominant philosophy for civil transport aircraft, military aircraft are designed and maintained using SL or DT philosophies to remain safe during their in-service life when subjected to expected operational conditions including design loading spectra and operational environments. The SL philosophy requires that sufficient fatigue tests and analysis have been conducted to establish confidence that there will be no failures caused by expected operational conditions during promulgated in-service safe lives. The tests can involve specimens, components, subassemblies, and full scale aircraft. For each significant structural item, several specimens are tested to define a safe life after which the item has to be replaced irrespective of its actual condition. To compensate for uncertainties regarding material properties, operational environments, future mission types, and severity of missions, the promulgated safe lives are assumed to be fractions of the lives demonstrated during the tests. For example, if the safe life of a component is promulgated to be 4000 flying hours, the contractual and regulatory obligations require that the fatigue tests should have demonstrated a life of B x 4000 hours, where B can vary between 3 and 5 depending on whether the component is monitored during the operational phase and the accuracy of the monitoring method, which is also based on the safe life approach. By defining the safe life in such a way, a target reliability level is achieved. The safe life approach is based on "stress life" or "strain life" data as illustrated in the following sections.

A.1.1 The Three Types of Stress-Strain Relationships

By testing material specimens having standard shapes and dimensions, three groups of material characteristics have been determined and published in specification handbooks. As illustrated in Figure 29 and Figure 30, the three groups of characteristics for a material are obtained from curves describing: a nominal engineering stress-strain relationship, a true stress-strain relationship, and a cyclic stress-strain relationship.



The nominal engineering stress = $S = P/A_o$
The true stress = $\sigma = P/A$,
$\sigma > S$ at the same strain, P: axial force
A_o, A: original & instantaneous cross sectional areas

The engineering strain = $e = \Delta\ell/\ell_o$, $\Delta\ell = \ell - \ell_o$
ℓ_o, ℓ : The original & instantaneous lengths

The true or natural strain = ε
$d\varepsilon = d\ell/\ell \rightarrow \varepsilon = \ln(\ell/\ell_o)$

S_y = yield strength, S_u = ultimate tensile strength
σ_f = true fracture strength
A measure of ductility = (Ao-Af)/Ao
ε_f = true fracture strain or ductility = $\ln(A_o/A_f)$
E = modulus of elasticity
E = the slope of the line of the linear region

Figure 29: Engineering and true stress-strain curves, monotonic tension

The nominal and true characteristics are determined by conducting standard monotonic tension tests and include yield strength, ultimate strength, true fracture strength, and true fracture strain or ductility: the nominal characteristics are obtained from the engineering stress-strain curve where (a) the stress is calculated by dividing the axial tension force by the original cross sectional area of the specimen, and (b) the strain is calculated by dividing the change in length by the original length of the specimen; the true characteristics are obtained from the true stress-strain curve where the stress and strain calculations are based on the instantaneous cross sectional area and the instantaneous length of the specimen; materials with a brittle tensile behaviour do not exhibit the necking shown in Figure 29.

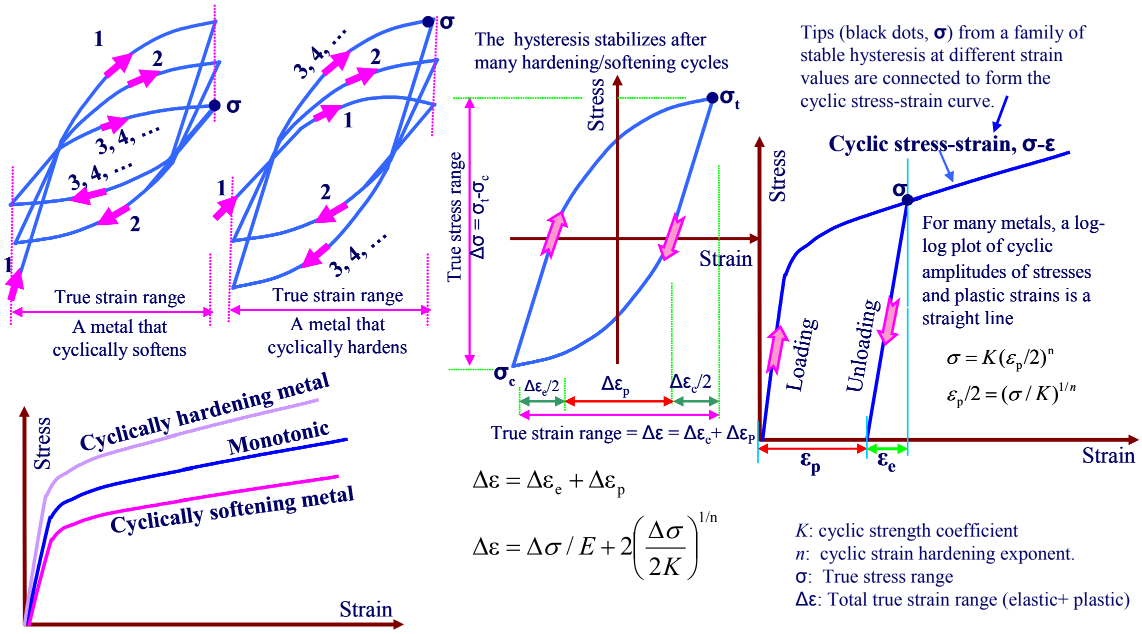


Figure 30: Cyclic stress-strain behaviour

The cyclic stress-strain curve can be obtained through strain-controlled tests of a number of material specimens: typically, a specimen is subjected to axial stress cycles at constant rate whilst controlling the specimen length to maintain constant amplitude straining cycles; the true stress-strain values are recorded periodically throughout the test and cycling is continued until fatigue failure occurs; after many cycles, the stress-strain behaviour stabilises and exhibits identical repeated hysteresis loops; such cyclic tests are repeated for other specimens to obtain a number of stable hysteresis loops having different controlled total strain values; the tips of a family of stabilized hysteresis loops with different strain amplitudes are connected to form the cyclic stress-strain curve as illustrated in Figure 30. For some metals, as the initial stress-strain cycles reach the stable states, the metals are hardened and exhibit increased resistance to deformation (same strains at higher stresses); other metals are softened and exhibit decreased resistance to deformation. The hardening/softening effects do not only depend on the type of metal but can also be influenced by how the metal is treated (e.g. cold worked, annealed, etc.); in some cases, both softening and hardening cycles occur.

As controlled strain cycles having a mean value reach a stable state, the corresponding mean stress values gradually decrease and, in some cases, can reach a zero value. Such a relaxation phenomenon is due to the presence of plastic deformation, and hence, the rate and amount of relaxation depend on the magnitude of the plastic strain amplitude; so, more mean stress relaxation occurs at larger strain amplitudes. At very high temperatures, creep causes continuous increase in the mean value of the strain cycles and hence, moves the successive hysteresis loops to the right.

At high temperatures, as the frequency of the strain cycles increases, the material strength can increase and its ductility reduce; at low temperatures, the effect of the frequency is typically very small.

A.1.2 Stress Life

The stress life design approach is based on fatigue data obtained from stress-controlled cyclic tests. The fatigue data is used to establish the well known S-N diagram, which shows the relationship between the applied stress **S**, and the number of cycles of stress **N** to cause failure: for each item, a series of tests are conducted; for each test a specimen is subjected to repeated stress cycles of constant amplitude until failure; the amplitude of the stress cycles falls with each individual test; ultimately, a stress is reached which will not cause failure. This stress is known as the fatigue limit or the endurance limit. The test results of several specimens are used to define the S-N diagram, Figure 31.

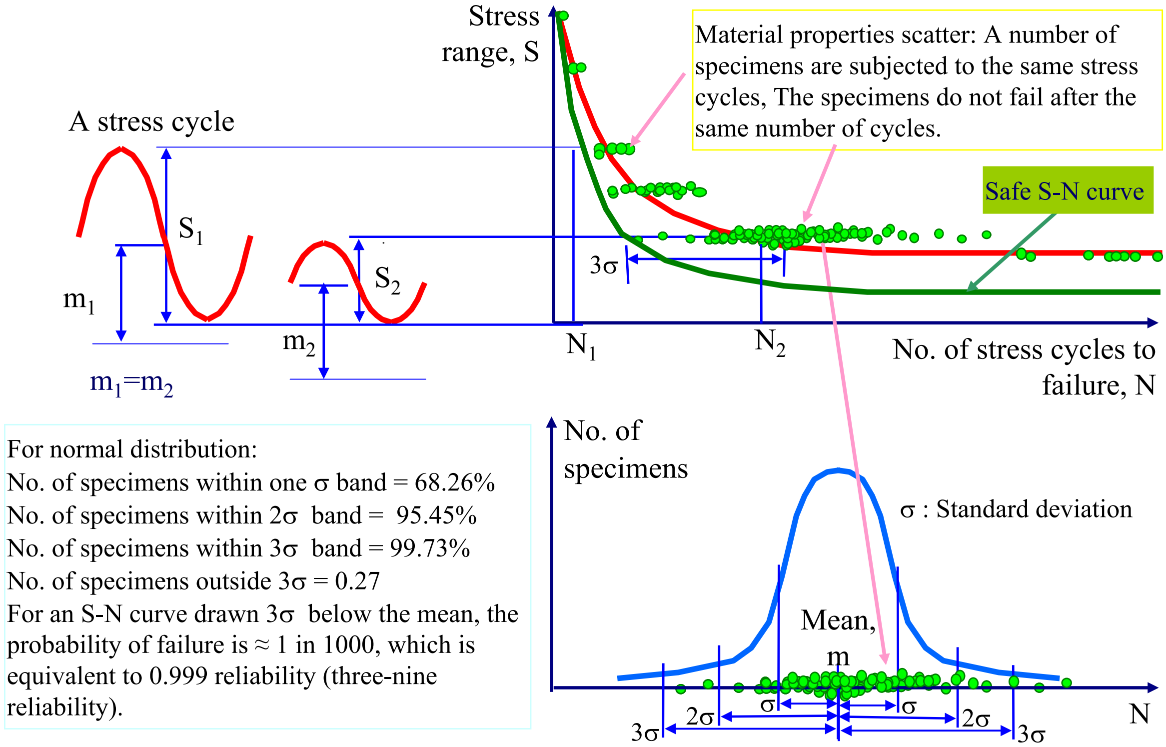


Figure 31: Derivation of a safe S-N curve achieving target reliability

Practically, with non-ferrous metals and with ferrous metals under corrosive conditions or stress concentrations and notches, the S-N curve has a more or less continuous downward trend. Therefore, the fatigue strength is used as an alternative to the fatigue limit to indicate the stress value associated with failure after a standard large number of cycles.

The S-N data evaluated from a number of similar series of tests may vary significantly. The scatter in S-N data increases as the amplitude of the applied cycles reduces. Usually, material strength safety margins are introduced to compensate for the scatter in material properties. For example, a safe S-N curve is often drawn below the mean to achieve a specified level of reliability; a curve drawn three standard deviations below the mean approximately gives a 1/1000 probability of a premature failure, which corresponds to 0.999 reliability (Reliability = 1.0 – Probability of Failure). In other words, a three-nine reliability can be considered for material strength to safely compensate for the scatter in material properties and ensures that the probability of a premature failure is 1/1000.

A.1.2.1 The Mean Stress Effects

Often, two common approaches are used to covert the S-N data to computational information. The first approach constructs a “unit damage matrix”: a finite number of matrix cells represent all possible ranges and means of ‘S’ cycles; each cell contains the damage (1/N) that is induced by a cycle having the range and mean assigned to the cell; this method directly takes into account the mean effects of any cycle by interpolating between the matrix cells. The second approach converts the S-N data to equations using curve fitting techniques supported by engineering knowledge. For example, Figure 31 shows S-N curves for cycles having different means. Using curve fitting techniques, the S-N curves can be reduced to a single reference curve along with a relationship that quantifies the effect of the varying mean on the parameters of the reference S-N equation. These parameters include the ultimate strength and the fatigue strength/endurance limit as shown in Figure 32 that presents a legacy relationship known as Goodman diagram; most of the Goodman diagrams are very close to straight lines.

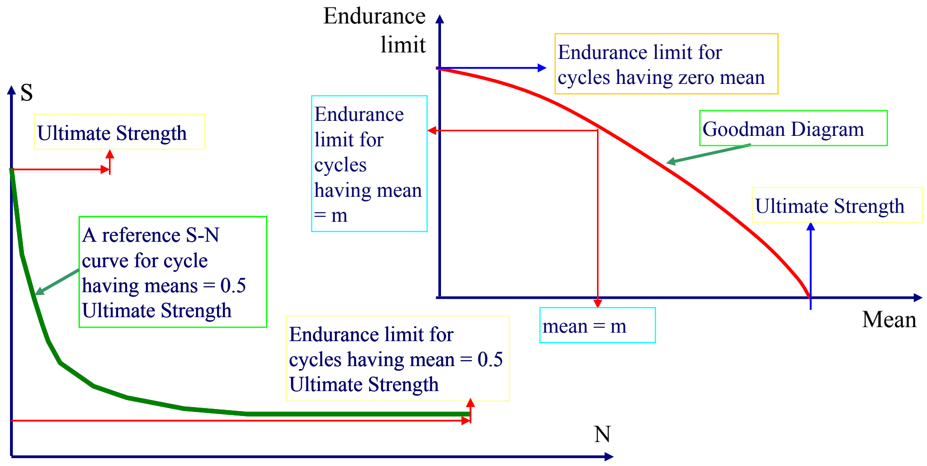


Figure 32: Fitting S-N data with a reference S-N equation and a Goodman diagram

A.1.3 Strain Life

The strain life design approach is based on fatigue data obtained from strain-controlled cyclic tests. The equipment required for strain-controlled tests are more sophisticated than those required for stress-controlled tests. However, since fatigue damage depends on local plastic strains, strain-controlled cyclic tests have become the common method for characterising the fatigue behaviour of materials. Whilst aircraft components are usually designed to withstand operational loads within elastic regions, local plastic deformations occur under elastic nominal stresses because of stress concentrations, for example, around notches, scratches, and corrosion pits. Figure 33 shows an example of fatigue data obtained from strain-controlled tests: the data are usually obtained from cyclic tests of specimens at frequencies ranging from 0.1 to 10 cycles per second for strain amplitudes causing failures after a number of cycles ranging from a handful number to about 10^6 cycles; failure at number of cycles greater than 10^6 is usually caused by small strain amplitudes dominated by elastic deformations, and hence, fatigue data can equally be obtained from stress-controlled tests at higher frequencies.

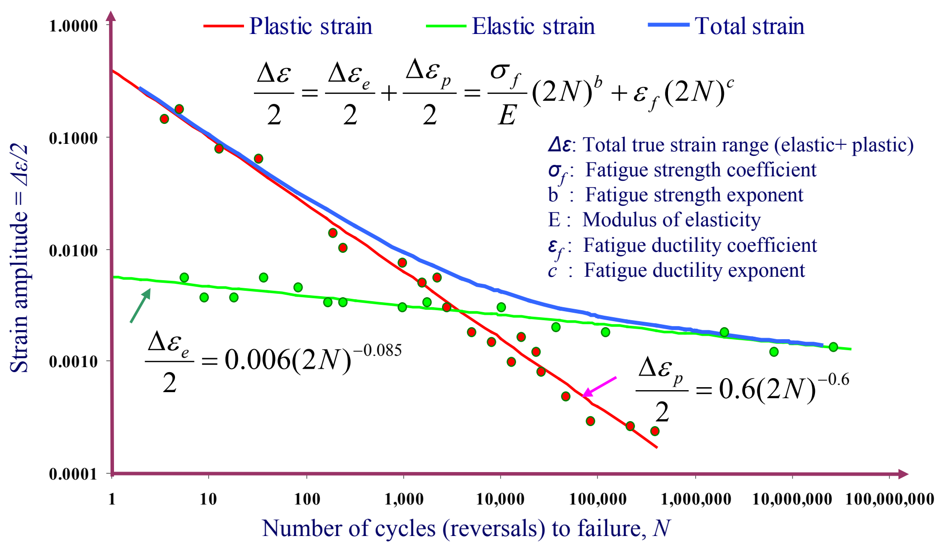


Figure 33: S-N curve obtained from strain-controlled cyclic tests

Typically, the relationship between strain and cycles to failure (ε-N) is obtained as follows: the measured total strain range is decomposed into elastic and plastic ranges; the elastic range is computed by dividing the measured stress range by the modulus of elasticity; the plastic range is computed by subtracting the elastic range from the measured total range; two straight lines (the equations of "Basquin" and "Manson-Coffin"'), are fitted to the decomposed data; the sums of the values of the two lines result in the ε-N curve as illustrated in Figure 33.

The coefficients of the two lines define the fatigue properties of the material tested, which include: fatigue strength coefficient, fatigue strength exponent, fatigue ductility coefficient, and fatigue ductility exponent.

A.1.3.1 The Mean Stress Effects

As illustrated in Figure 33, the plastic strain is dominant in the low-cycle fatigue region (the left-hand side of the point of the two line intersection); the elastic strain is dominant in the high-cycle fatigue region (the right-hand side). The mean stress effect on fatigue in the low-cycle fatigue region is smaller than the mean stress effect in the high-cycle fatigue region because the mean stress relaxes more at higher strain amplitudes due to the associated larger plastic strains. At very high strain amplitudes that fully relax the mean stress to zero value, the mean stress does not affect the fatigue behaviour.

Figure 34 shows two of the several models that have been proposed to account for the mean stress effect of the strain-life. Among these models, the Smith, Watson, and Topper (SWT) model has been found to better correlate with the test data of a wide range of materials, and therefore, has been widely used. The SWT model is based on the assumption that the product of the maximum stress, σ_{max}, and the strain amplitude, $\Delta\varepsilon=(\varepsilon_{max}-\varepsilon_{max})/2$, remains constant for a given fatigue life (N) under different combinations of strain amplitudes and mean stresses.

The cycles to failure "N" without mean effects

$$\frac{\Delta\varepsilon}{2}=\frac{\Delta\varepsilon_e}{2}+\frac{\Delta\varepsilon_p}{2}=\frac{\sigma_f}{E}(2N)^b+\varepsilon_f(2N)^c$$

Mean Effects

Morrow

$$\frac{\Delta\varepsilon}{2}=\frac{\sigma_f-\sigma_{mean}}{E}(2N)^b+\varepsilon_f(2N)^c$$

Smith, Watson, and Topper (SWT)

$$\sigma_{\max}\frac{\Delta\varepsilon}{2}=\frac{\sigma_f^2}{E}(2N)^{2b}+\sigma_f\varepsilon_f(2N)^{b+c}$$

σ_{mean}: mean stress, σ_{amp} : tress amplitude, σ_{max}: maximum stress = σ_{mean} + σ_{amp}

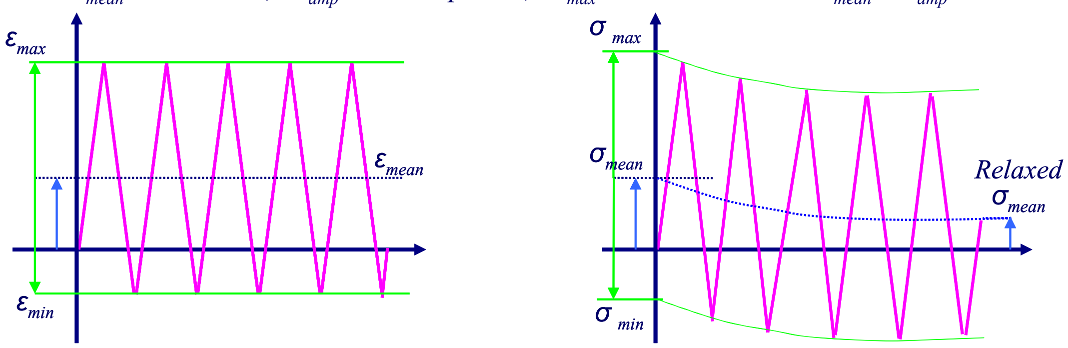


Figure 34: The mean stress effect on the ε-N relationship

A.1.4 Miner's Rule

A very important question is the influence of previous stressing on fatigue damage. One theory that has considerable acceptance is the linear damage law introduced by Miner. Miner has suggested that the damage produced by repeated stressing at any level is directly proportional to the number of cycles. Thus, if the number of cycles producing failure (100 percent damage) at certain stress is N, then the proportional damage produced by M cycles of the same stress is M/N. Cumulative damage at various stressing levels is equal to the summation of such fractional values. Failure is assumed to occur when the summation of the M/N fractional values for various stresses is equal to unity. Thus, the effect of a given number of cycles is the same whether they are applied continuously or intermittently.

A.1.5 Cycle Counting

For a stress (or strain) time history containing cycles of varying amplitudes and means, cycle counting techniques are used to identify the amplitude and mean of each stress/strain cycle within the time history, Reference [62]. The most acceptable cycle counting technique is the well known rain-flow method that identifies each cycle from the peak-trough trace of the stress time history. After extracting the mean and amplitude of each cycle, S-N or ε-N data is used to compute the fatigue damage induced by the cycle. Then, Miner's rule is used to sum the damages of all identified cycles, and hence, indicate the damage induced by the stress/strain time history.

A.1.6 Practical Considerations

Often, life prediction methods based on S-N/ε-N curves and Miner's cumulative damage rule are not accurate enough to predict fatigue under service loading of variable amplitudes. Accordingly, for each design, a standard loading sequence is introduced to provide realistic fatigue data; the standard loading sequence is often defined using peak and trough loads measured over a sufficient number of representative flights. Component and full-scale tests under such representative loading spectra can be used to correct the S-N/ε-N relationships and, provide enhanced fatigue life predictions, better safe life designs, and improved inspection techniques. Nevertheless, for each design, safety measures should be considered to compensate for usage outside the loading spectrum used during the design phase, which is referred to as the design loading spectrum; the usage safety measures can be set by evaluating a number of loading spectra from different groups of representative flights; out of these spectra, a design loading spectrum is chosen such that the probability of usage outside the chosen spectrum is, for example, 1/1000, which corresponds to 0.999 reliability; in other words, a three-nine reliability may be considered for the design loading spectrum to compensate for scatters in usage data and ensure that the probability of in-service usage outside the design loading spectrum is 1/1000. Thus, a high reliability of promulgated safe life can be achieved through the following:

- A design that considers a three-nine reliability for fatigue data (about three standard deviations below the S-N/ε-N curve) to safely compensate for scatters in material properties and ensure that the probability of a premature failure under constant amplitude cycles is 1/1000.
- A design that considers a three-nine reliability for the design loading spectrum to safely compensate for usage scatters and ensure that the probability of in-service usage outside the design spectrum is 1/1000.

Then, the probability of a premature failure under in-service usage will be the product of the above two probabilities; i.e. 1/1000 x 1/1000 = 1/100000, which corresponds to in-service promulgated fatigue life with a six-nine reliability.

In-service factors that can affect fatigue lives, and are not considered in the above discussion, include surface finish, corrosive effects, stress concentrations, erosive effects on surface conditions, accidental notches, temperature cycles, and contact damage (fretting). Therefore, promulgated safe lives have been found to vary by three orders of magnitude; such a large variance can be a consequence of the assumed safety margins that are considered to face various uncertainties regarding material properties, actual usage and in-service environments. Therefore, the probability of premature failures under all these factors during the promulgated safe life is controlled and reduced as follows:

- Inspections are scheduled at conservative time intervals to detect, evaluate and, if necessary, remove the undesirable effects of these factors through appropriate repairs.
- Periodic substantiation of promulgated safe lives is considered using in-service data acquired from a number of instrumented aircraft.

For the fleets of the UK MOD, substantiation programmes are periodically carried out and called: Operational Data Recording (ODR) programmes for helicopters, and Operational Load Monitoring (OLM) programmes for aeroplanes.

A.2 The Damage Tolerance Philosophy

The damage tolerance philosophy is enforced as a design requirement that a critical component is duplicated or its possible defects are controlled or arrested. A structure is considered to be damage tolerant when damage, if it should occur, will be discovered and repaired before residual strength falls below specified levels; thus, the structure will be damage tolerant if it is designed to withstand realistic loads despite the presence or occurrence of certain level of damage as a consequence of Manufacturing Defects, (MD), FD, AD, or ED until the damage is detected through planned inspections.

A.2.1 Existing Definitions of Damage Tolerance

FAA defines damage tolerance as “the attribute of the structure that permits it to retain its required residual strength for **a period of use** after the structure has sustained a given level of fatigue, corrosion, or accidental or discrete source of damage”, Reference [13]. For military aircraft similar definitions are used: the UK MOD refers to damage tolerance as “a design philosophy which leads to a structure that can retain the required residual strength for **a period of use** after the structure has sustained specific levels of detectable fatigue damage, AD or ED”, Reference [36]. The DOD defines damage tolerance as “the ability of the airframe to resist failure due to the presence of flaws, cracks, or other damage for a specified **period of unrepaired usage**”, Reference [52]. The periods of use are referred hereafter as the service periods as defined in the following section.

A.2.2 Service Periods

Each service period is determined by DT analysis based on fracture mechanics (Section A.3.3) such that a fault, detected or undetected, cannot grow under the expected loading and environmental conditions during this period to a level that impairs the required carrying load function of the structure. The determined service periods do not only depend on the expected service loads and environments but also depend on the capability and quality of the inspection equipment and repair methods.

A.2.3 The Three Pillars of Damage Tolerance

The damage tolerance philosophy achieves and maintains a target reliability level through the following:

- designs allowing the presence and growth of MD, FD, AD, and ED during determined service periods,
- planned inspections capable of assessing the levels of damage and their effects on the target reliability, and

- planned repairs capable of maintaining the target reliability, and assuring operational safety, during a following service period.

The "planned inspections" can include, for example, scheduled inspections or inspections dictated by accidental events such as reported hard landings. The words "planned repairs" indicate that immediate repairs are not necessarily required after detecting faults; the repairs can be planned and deferred as long as the target reliability can be maintained in the presence of detected faults. Figure 35 illustrates the pillars of a damage tolerance approach where the target reliability is maintained by ensuring that the residual strength of the structure remains above a safe level in the presence of a growing crack as described below:

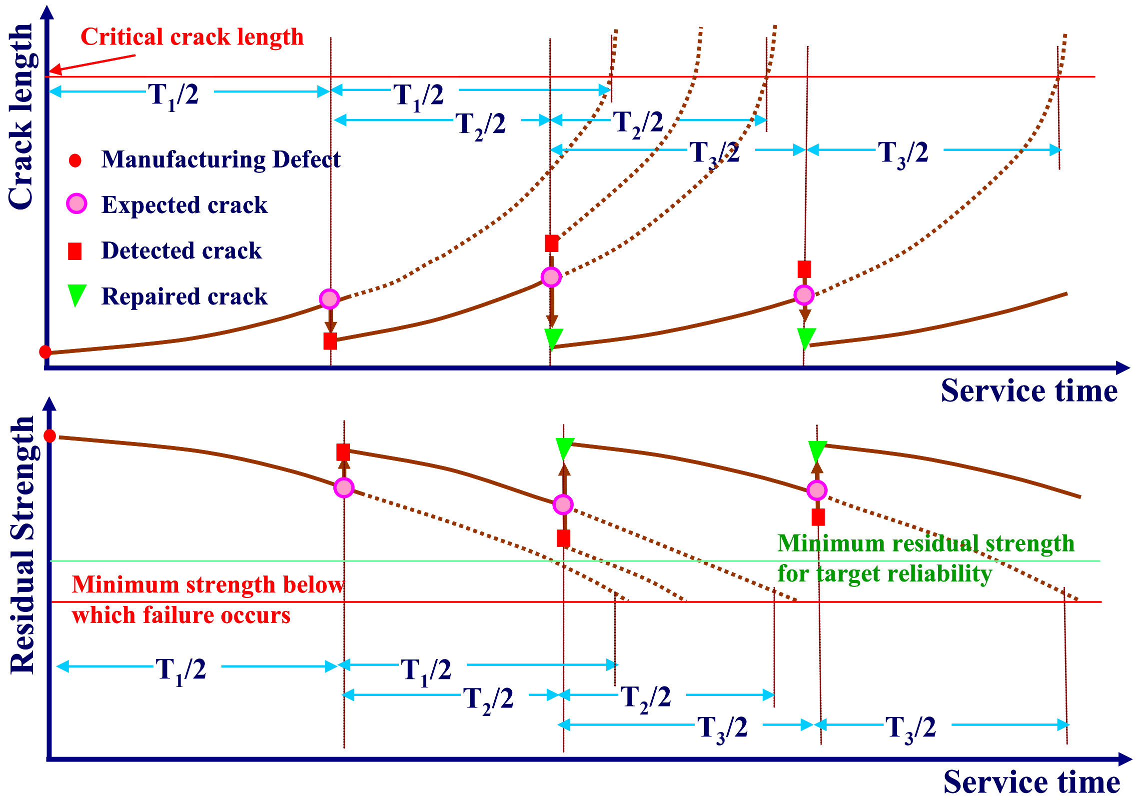


Figure 35: The three pillars of damage tolerance: designs, inspections & repairs

A.2.3.1 Damage Tolerance Design

DT design efforts resulted in a damage tolerant structure characterized by the slow crack growth curve shown in Figure 35. The structure could have a small undetectable manufacturing defect before any service usage, e.g. at time T=0. Under service loads, the crack would grow slowly and eventually cause failure after reaching a critical crack length at $T=T_1$. The growing crack would cause reductions in the residual strength of the structure; eventually a minimum residual strength would be reached below which the structure would fail.

A.2.3.2 Planned Inspections

A first inspection was set at half the expected crack growth lifetime $T_1/2$ to give two chances of crack detection before failure; the inspection was performed and a crack was found; the detected crack length was less than an expected crack length growing from an assumed manufacturing defect length. Under service loads and during a period= T_2, the detected crack would grow slowly and eventually cause failure when reaching the critical crack length. A decision was taken to defer the crack repair and to set a second inspection at half the expected crack growth lifetime $T_2/2$ because of the following:

- Inspection at $T_2/2$ would give two chances of crack detection before failure.
- $T_2/2$ constituted a reasonable time for a service period.
- The residual strengths were expected to be sufficiently higher than the minimum strength during the service period $T_2/2$.

A.2.3.3 Planned Repairs

After $T_2/2$, the inspection indicated that the crack grew faster than expected. The crack was repaired and the structure strength was restored to its original strength. Nevertheless, the structure was assumed to have undetectable repair defect longer than the assumed manufacturing defect because of potential environmental and repair tool effects. Under service loads, this defect would grow slowly and eventually cause failure after reaching the critical crack length after a period = T_3.

A third inspection was set at half the expected crack growth lifetime $T_3/2$ to give two chances of crack detection before failure; the third inspection was carried out; a crack was found and repaired.

A.3 Damage Tolerance Approaches

Damage tolerance can be achieved through a fail-safe design or slow defect growth design.

A.3.1 Fail Safe Design

A fail-safe design is accomplished by introducing multiple load paths or defect arrest structures such that propagating damage is safely contained by load shift to adjacent intact elements or by other damage arrestment features.

A.3.1.1 Multiple Load Paths

The multiple paths are either active (load carrying) or passive (unloaded). The failure of any load path still allows the remaining paths to carry the load so that a catastrophic failure can be avoided. The multiple load paths' fail-safe structure is designed and fabricated in segments; each segment can contain localized damage and prevent the complete failure of the structure; safety is assured through a slow crack growth method leading to damage detection in the remaining structure during subsequent inspections, see Section A.3.2. The multiple load design ensures that the strength and safety of damaged structure will not be degraded below a specified level for a specified period of service usage prior to planned inspections and any consequent repairs.

Depending on the source of damage and due to the nature of the assembly or manufacturing procedures, multiple load path structures are classified as dependent or independent: they are dependent if a common source of cracking exists in adjacent load paths at one location; they are independent if it is unlikely that a common source of cracking exists in more than a single load path at one location. An example of a multiple load path-dependent structure is planked tension skin where individual members are spliced in the span-wise direction by common fasteners with common drilling and assembly operations. By exploiting new composite materials, high reliability can be achieved using multiple load path designs without incurring weight penalties. Reference [63] presented a fail safe rotor blade designed with four steel spars reinforced by fibreglass spar tubes. Reference [63] demonstrated that his design can achieve a reliability of 0.999999 against catastrophic failures, a reliability equivalent to less than one failure in the life of the fleet; attempting to achieve the same reliability with safe life approach was found to cause severe penalties in weight and cost because of the need to use a working S-N curve of approximately five standard deviations below the mean; the multiple load path design required only one standard deviation below mean.

It is worth mentioning that duplicating some of the vital aircraft components are not always possible. Duplicating aircraft dynamic components such as rotating or dynamic parts of helicopters, engines, and landing gears is impractical, very difficult, or impossible. Therefore, constructing an entire aircraft according to the damage tolerance philosophy is a challenge for future designs.

A.3.1.2 Defect Arrest Structures

The stress levels of defect arrest structures are sufficiently low so that crack growth can be prevented by crack stoppers such as stringers, frames, and cut-outs. The crack arrest fail-safe structure is a structure designed and fabricated such that unstable rapid crack propagation will be stopped within a continuous area of the structure prior to complete failure; safety is assured through a slow crack growth method leading to damage detection within the remaining structure during subsequent inspections, see Section A.3.2; the strength of the remaining structure will not be degraded below a specified level for a specified period of service usage prior to planned inspections and potential consequent repairs. Usually, a fatigue test is performed to determine locations of possible failures, and to examine the potential of implementing a defect arresting philosophy.

A.3.2 Slow Defect Growth

Slow defect growth designs guarantee that defect propagation rates are slow enough to ensure detection before failure. In addition, satisfactory strength and stiffness has to be maintained prior to the onset of unstable defect propagation. In other words, slow defect designs include those design concepts where a defect is not allowed to attain the critical size after which unstable rapid crack propagation occurs; safety is assured through slow crack growth for specified periods of service usage after which the structure is inspected and repaired if necessary to maintain a target degree of reliability.

Slow defect growth design relies on an increased confidence in being able to predict both the rate of growth of a small defect and the critical crack size at which failure will occur under a specified extreme load. The overall probability of failure depends on: the probability of the crack being present, the probability of the crack being missed by inspection, the probability of the crack propagating to a critical length before the next inspection, and the probability of the critical load occurring during the inspection period. Reliability analysis that only considers the above factors would face challenges because significant problems such as corrosion and accidental damage are not taken into account.

Experience in slow defect growth philosophy stemmed from its application to simple geometry. Rigorous application of the methodology to new types of structural geometry and complex loading, which are characteristics of aircraft components, requires extensive efforts. Therefore, approaches have been developed to allow the replacement of complex structures, their loading, the geometry of crack plane, and the crack itself with simplified counterparts for whom the elements of crack growth model are known, documented, and verified. Also, significant advances in analytical tools and test capabilities have been achieved to enable improved slow defect growth models and designs.

Generally, the assessment of the slow defect capability of a component requires conducting fatigue tests and performing analytical calculations based on fracture mechanics.

A.3.3 Fracture Mechanics

The origin of three main fracture mechanics approaches is an original energy balance concept developed in the UK in 1921. The first approach is the linear elastic fracture mechanics; the approach has provided widely used analysis tools for aerospace industries. The second approach, the J-integral approach, can account for non-linear behaviour observed during fracture. The third approach is based on the assumption that the amount of crack opening influences the local behaviour at the crack tip. An elementary fracture analysis may be performed by using the theory of linear elasticity to classify the stress field surrounding a crack tip for each of the modes shown in Figure 36, which shows three main crack surface displacements.

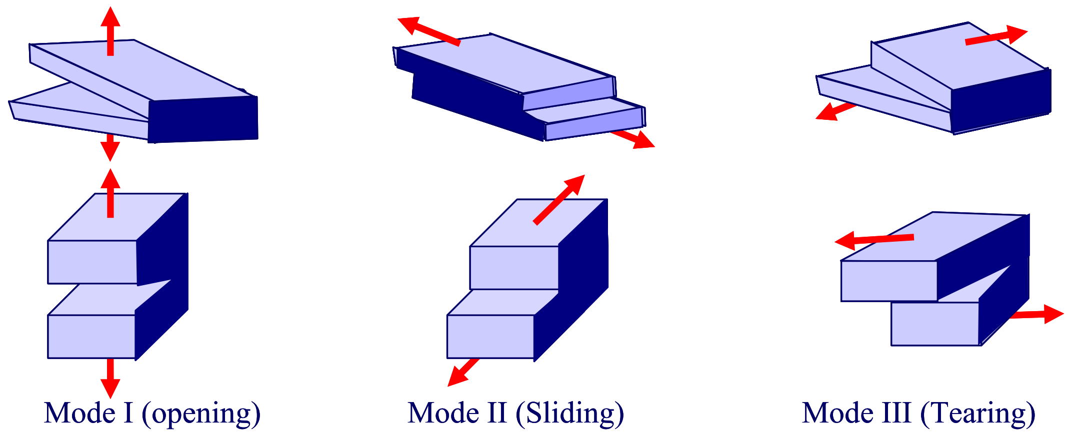
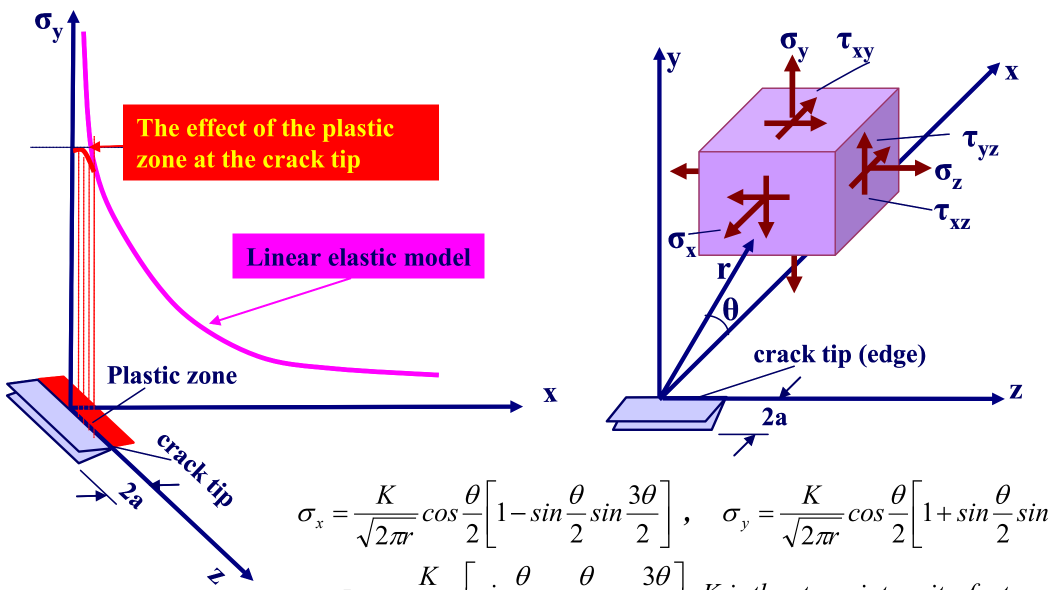


Figure 36: The modes of crack extension

Fracture mechanics, in its simplest definition, is a discipline that relates the crack extension characteristics of a material to the applied stress and part geometry. The heart of fracture mechanics is the stress intensity factor (K), which may be considered as a measure of the local stress in the region of the crack tip, and is a simple function of the applied stress, the crack length (*2a*), and a geometrical relationship. Figure 37 illustrates the theory for a plane stress problem where the stresses $\sigma_z = \tau_{yz} = \tau_{xz} = 0$.



$$\sigma_x = \frac{K}{\sqrt{2\pi r}} \cos\frac{\theta}{2}\left[1 - \sin\frac{\theta}{2}\sin\frac{3\theta}{2}\right], \quad \sigma_y = \frac{K}{\sqrt{2\pi r}} \cos\frac{\theta}{2}\left[1 + \sin\frac{\theta}{2}\sin\frac{3\theta}{2}\right]$$

$$\tau_{xy} = \frac{K}{\sqrt{2\pi r}}\left[\sin\frac{\theta}{2}\cos\frac{\theta}{2}\cos\frac{3\theta}{2}\right], K \text{ is the stress int ensity factor}$$

Figure 37: Linear elastic models for stresses in the presence of a crack

Two critical stress intensity factors are defined to be the stress at which macroscopic cracking starts (K_O) and the stress at which unstable rapid or catastrophic cracking ensues (K_C); the latter is an intrinsic property of materials and can be considered analogous to the ultimate strength; it is called the fracture toughness, Figure 38. The two critical stress intensity factors are relatively easy to measure, furthermore, the geometrical relationships can be found in textbooks (for standard geometry), derived from classical theory of elasticity, or measured.

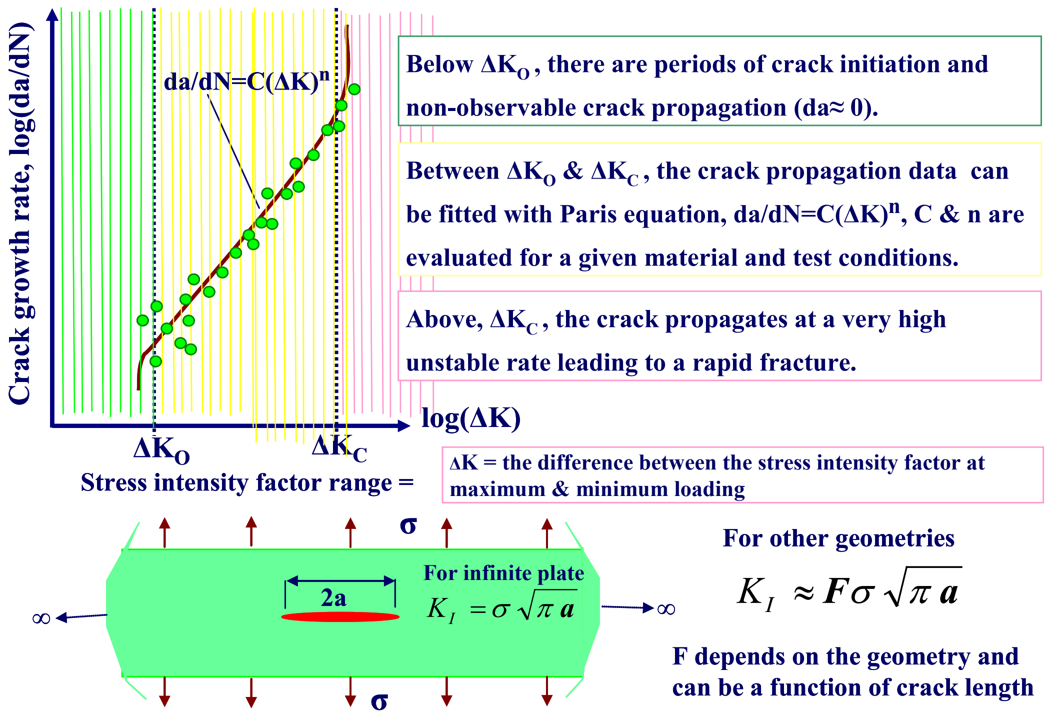


Figure 38: Paris model for crack growth rates

For known applied stress, geometrical relationship, and defect length, the stress intensity factor can be calculated and compared with the two critical stress intensity factors to decide whether the structure is safe or not. The values of the critical stress intensity factors and the applied stress along with the geometrical relationship can be utilized to predict the crack length.

Fracture mechanics can be used to predict "the time to failure" from some detectable crack size by calculating the rate of crack growth, which is a function of the stress intensity factor. Advanced tools have been developed to exploit the full potential of the theory. The above discussion focuses on Mode I (opening mode) of cracking whose importance in engineering practice far exceed Mode II and Mode III. It also focuses on the stress intensity approach which is widely considered as a more general approach to characterize crack propagation.

A.3.4 Effects of Service Loads on Crack Growth

Civil aircraft can experience moderate variations in service loads because of factors such as route and configuration differences. For example, the number of ground-air-ground load cycles and cabin pressurization cycles experienced by an aircraft dedicated to short-haul flights can be significantly higher than the number of those cycles experienced by an aircraft assigned to long-haul flights. On the other hand, variations in service loads between military aircraft can be much higher than those variations between civil aircraft because of the potential wide range of military missions and sortie types. Therefore, by considering the USAF damage tolerance inspection approach described in Reference [64], the effects of the variations in service loads on crack growth was investigated, reported in Reference [65] and presented in the following sections.

A.3.5 The USAF Damage Tolerance Inspection Strategy

Reference [64] has emphasized that the focus of the USAF damage tolerance activities has not been to develop an inspection programme, but to evolve a design that minimizes all in-service maintenance actions. After finalising a design, if it is determined that component cracks can grow from a rogue flaw (manufacturing flaw) to a critical size before two design lifetimes, then, the component should be re-designed or an inspection programme developed; the choice between the two options is based on the results of an associated business case. Reference [65] illustrated the USAF inspection approach as shown in Figure 39.

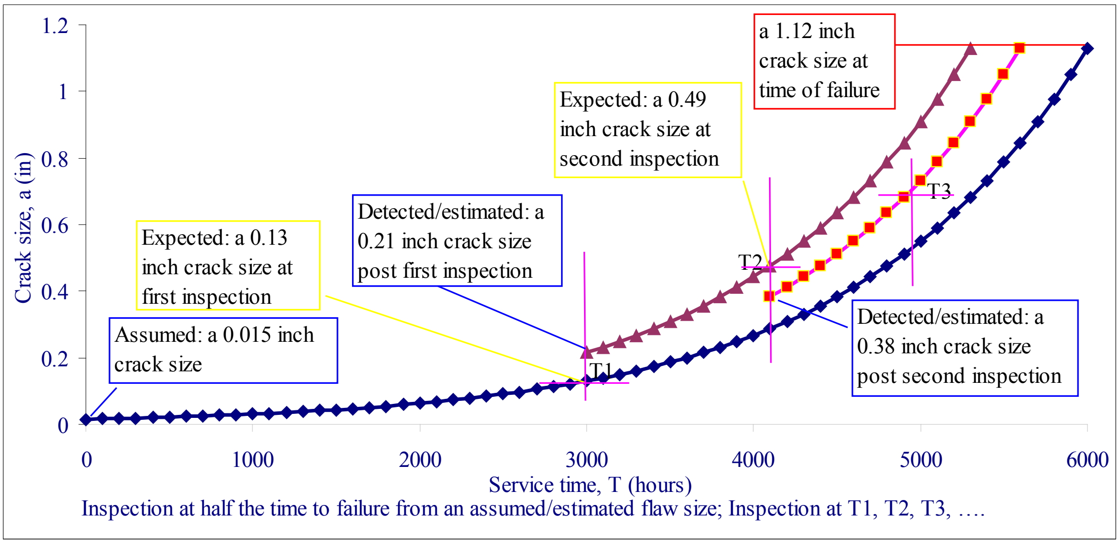


Figure 39: An inspection strategy giving two chances of crack detection before failure

The blue diamond curve shows how the crack grows under typical loading conditions from an assumed rogue flaw size (a=0.015″), which might have been missed by normal production quality control processes, and eventually causes failure at 6000 hours. The first inspection is set at half the 6000-hour crack growth lifetime to give two chances of crack detection before failure. At this midlife, a crack size a_{mid-1} = 0.21″ is detected or estimated based on the results of the inspection, taking into account the probability of missing cracks of certain sizes and allowing for the potential of inducing damage by inspection tools and processes; inspection may require disassembly and reassembly processes.

Traditionally, the detected/estimated value a_{mid} would be established based on a simple characterisation of the inspection equipment using a Probability of Detection (POD) experiment to associate a_{mid} with $a_{90/95}$ (a crack size with a 90% POD at a statistical confidence level of 95%). Using the crack growth curve with triangular symbols, the second inspection is set at T_2, which is the midlife of the crack growing from the detected/estimated a_{mid-1} to failure at $a_{critical}$=1.12″. The illustration shown in Figure 39 indicates that the second detected/estimated crack size a_{mid-2} (0.38″) is less than the expected crack size at the second midlife (a_{exp-2}=0.49″). Using the red crack growth curve with square symbols, the third inspection is set at the third midlife T_3 of the crack growing from the second estimated a_{mid-2} to $a_{critical}$=1.12″. So, the midlife inspection time is a function of the initial crack size. This function can be evaluated by repeating the application of the loads of the typical mission mix to the component with different initial crack sizes until failure.

A.3.6 Effect of Mission Mix Types

Reference [65] investigated the effects of mission severities and concluded that the relationship between the initial crack size and the expected midlife crack size is independent of the severity of the flown missions as illustrated in this section, Figure 40.

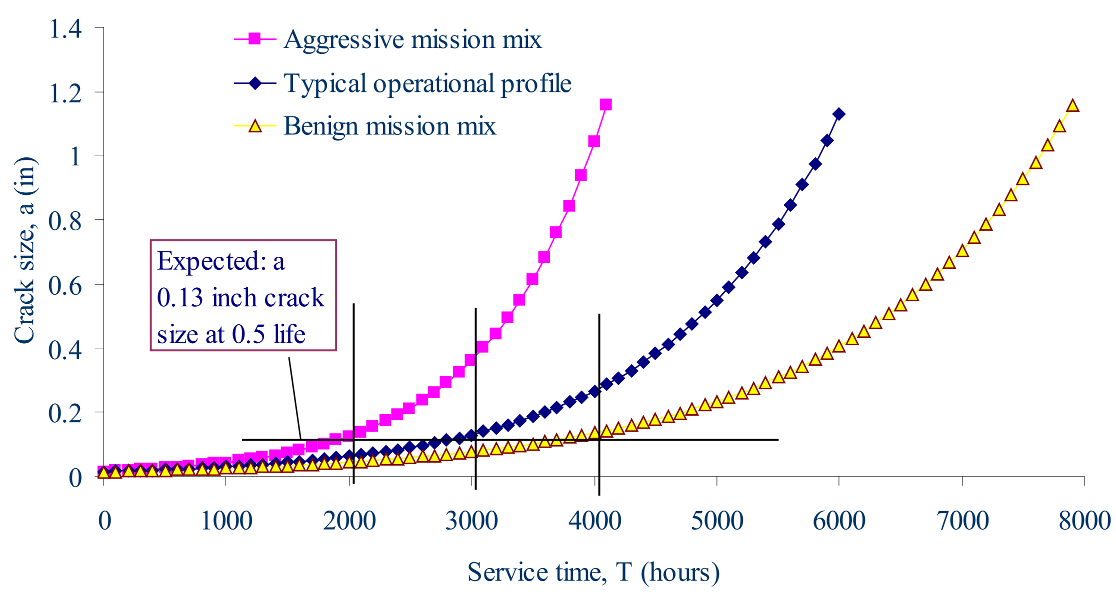


Figure 40: Crack growth curves of three mission mix types

The blue diamond curve of Figure 40 is the crack growth curve under a typical mission mix. According to the USAF approach, the first inspection is set at half the 6000-hour crack growth lifetime. If the individual aircraft is flown more aggressively than what was initially anticipated, the aircraft operations will contain more missions of high loading conditions such as air-to-air combats; the pink crack growth curve with square symbols is derived for aggressive mission mix with the first inspection set at half the 4000-hour crack growth lifetime. In peace time, it is possible that the mission mix will induce loadings less than those of the typical mission mix; the yellow crack growth curve with triangular symbols is derived for a benign mission mix with the first inspection set at half the 8000-hour crack growth lifetime. The inspection intervals under the aggressive mission mix are shorter than those of the typical mission mix; the inspection intervals under the benign mission mix are longer; the first midlife inspections are 2000, 3000, and 4000 hours for the three types of mission mix. Reference [65] concluded that although the midlife inspection times are very sensitive to the type of flown missions, the expected crack sizes at the midlife times are much less sensitive; at the first midlife, the cracks expected lengths grew from 0.015″ to values close to 0.13″ for the three types of mission mix over midlife hours of 2000, 3000, and 4000.For the three crack growth curves, the crack size at failure was assumed to be the same, which might seem to contradict the fact that failure occurs when the stress intensity factor reaches a critical value; the stress intensity factor increases with both the crack length and the stress value. The justification for the assumption is as follows: the aircraft stress time histories are not of constant amplitudes; the histories contain a variety of different stress cycles; stress histories across different aircraft/missions contain similar stress cycles due to similar operations such as take-off and landing; just below the crack size at failure, it is highly probable that the critical stress causing failure at this crack size will be encountered during a very small number of missions; similarly, non-linear and plastic effects can differ between stress cycles but the integral effects across a few missions can be similar.

The x-axis of Figure 41 shows the initial crack size; the y-axis shows the expected crack size at the midlife of the crack growing from the initial crack size to failure. As shown in the figure, it was found that the cracks grew from an arbitrary initial size to about the same expected midlife size regardless the length of the midlife period or the severity of the missions flown. Hence, the inspection intervals should be based on expected midlife crack sizes. They should not be based on flying hours; the midlife flying hours is very sensitive to the mission flown; any error in estimating the severity of these missions will result in large errors in the inspection times.

Data of legacy aircraft covering 15 years of military operations were used to validate these conclusions: by excluding some of the sorties and repeating others, it was possible to create three sets of sorties simulating aggressive, typical, and benign mission mix scenarios; the flight data of each series of sorties were converted to stresses and a crack growth model was used to compute the crack size to failure. The analysis produced an invariant relationship between the initial crack size and the midlife crack size similar to the relationship shown in Figure 41.

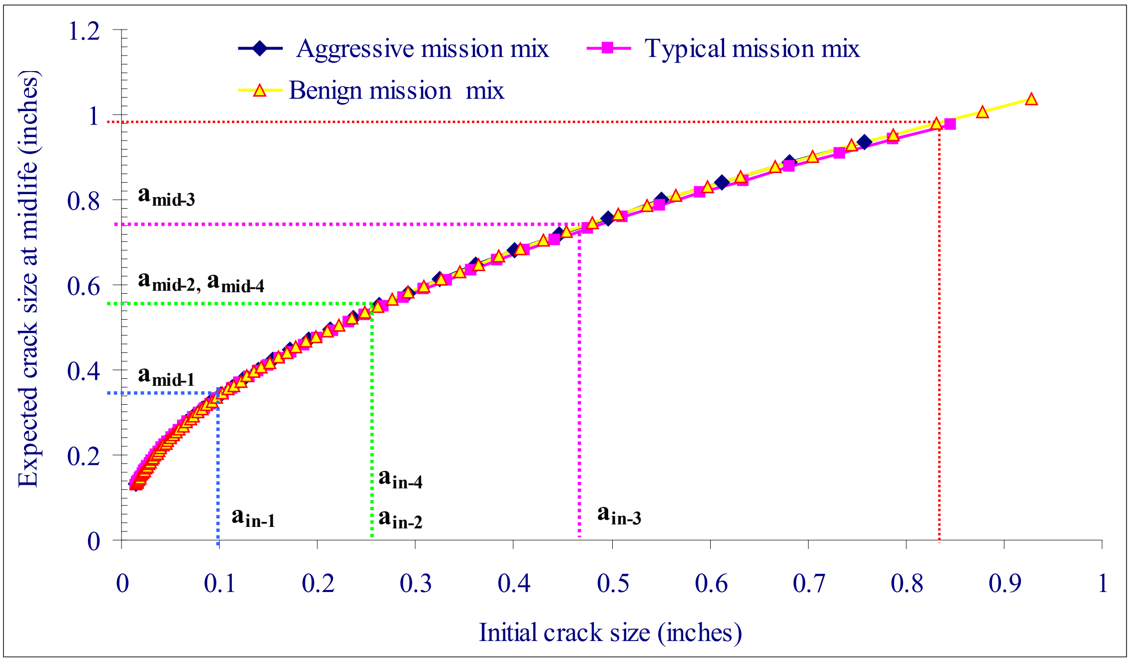


Figure 41: Inspection intervals based on midlife crack sizes

The above conclusions, if implemented, would lead to a flexile inspection approach that could be optimally integrated within a proactive maintenance system capable of taking advantage of arising opportunities to minimize costs and maximize readiness. The main steps of this flexible approach are as follows: for a selected component, derive the invariant relationship between the initial crack size and the expected midlife crack size, Figure 41; set a counter i to zero; then, repeat the following steps:

- Set i = i + 1
- Detect/estimate an initial crack size, $a_{in-i.}$
- Use the invariant relationship to readout the expected midlife crack size, $a_{mid-i.}$
- Use the service data acquired since the detection/estimation of a_{in-i}, as frequent as possible, to compute the corresponding service loads. Use the computed loads and fracture mechanics to compute the expected crack length to date, $a_{current}$.
- Plan an inspection when the value of $a_{current}$ approaches the value of a_{mid-i}.
- Plan the required repair if the inspection triggers the need for repairing a large crack. After restoring the strength of the component, repeat the above steps; start by estimating a next (i=i+1) initial crack size for the repaired structure, which should be small but can be larger than a manufacturing defect (a_{in-1}).
- Else, if the inspection did not trigger the need for repairs, repeat the above steps; start by using the inspection results (crack detection and/or estimation) to compute a next initial crack size.

A.4 Target Reliability

The target reliability referred to in the previous sections is the reliability associated with a low probability (risk) of failure condition that would have adverse consequence on safety, airworthiness, economy, environment, or performance through the equation.

“Reliability = 1 - Probability of such a Failure”.

The word risk is widely used to encompass both the probability of undesirable event (failure) and the consequence of the event if it happens; the consequence is estimated based on the possible failure mode and can be categorized into, for example, safety, performance, economical, or environmental consequences. Thus, the interpretation of target reliability is consistent with the common interpretation of risk.

For civil aircraft, it must be demonstrated that the occurrence of any failure condition that would prevent the continued safe flight and landing is extremely improbable, References [3] & [19]. If the probability of such an extremely improbable failure is one failure every 10,000,000 flying hours, the target reliability would be 0.9999999; this seven-nine target reliability must be demonstrated and maintained through the designs, inspections, and repairs of the damage tolerance philosophy, or through the designs and maintenance approaches of the safe life philosophies.

For military aircraft in the UK, and according to the Regulatory Article 5202, Appendix C: “Declaration of Compliance Criteria”, the MOD requires demonstrating by safety analysis that “the design meets the required level of safety for the proposed flight(s)” and “an overall system cumulative probability of catastrophic technical failure must be provided”. A number of MOD regulations require that measures should be taken to counter sources of threats to safety, airworthiness, and aircraft integrity, reducing the risks and through-life costs to “As Low As Reasonably Practicable” (ALARP).

According to Reference [30], The ALARP principle derives from Sections 2 and 3 of the Health and Safety at Work Act (HSWA) 1974. A risk can be said to be reduced to a level that is ALARP when the cost of further reduction is "grossly disproportionate" to the benefits of risk reduction. This cost may include more than financial cost and must consider the time and trouble involved in taking measures to avoid risk. Therefore, an ALARP argument should balance the "sacrifice" (in time, money, or trouble) of possible further risk reduction measures with their expected safety benefit (incremental reduction in risk exposure). The balance should be weighted in favour of safety, with a greater "disproportion factor" for higher levels of risk exposure. ALARP is essentially the "stopping condition" for risk reduction, so justifying and recording how this is reached is an important and vital step in safety management. A Duty Holder is required to make an argument that risks have been made ALARP; however, the validity of this argument can only be decided definitively by courts, should an accident occur, References [30] & [32] .

The following sections list some of the MOD regulations that point to ALARP requirements.

A.4.1 MOD Regulations that Point to ALARP

The following paragraphs quote regulations, AMC and/or guidance materials that have called for ALARP. The regulations are distinguished by the executive verb “shall” to indicate “prescribed rule or authoritative direction” with the only choice is to comply with them. The quoted sentences that don’t contain “shall” are either AMC or guidance requirements.

- Regulation 5312(1) - In-Service Design Changes: “The PT should take responsibility for: Ensuring that safety, airworthiness and engineering risks and through-life costs introduced by design changes are identified and reduced to As Low As Reasonably Practicable (ALARP)".

- Regulation 5720(1) - Structural Integrity Management: "Structural Integrity should be regarded as vital to airworthiness, but only by undertaking the activities described in this RA and its associated RAs throughout the life of the aircraft will the risk of SI failure be kept at an acceptably low level. Moreover, the relative ease of incorporating improvements to aircraft systems means that the operational life of an aircraft should usually be determined by Structural Integrity considerations rather than by equipment obsolescence. Accordingly, measures should be taken to counter sources of threats to Structural Integrity, reducing the risks to 'As Low As Reasonably Practicable' (ALARP) levels".

- Regulation 5720(3) - Sustaining Structural Integrity: "Structural Integrity shall be sustained and managed appropriately in order continuously to monitor, measure and counter the threats to Structural Integrity so that the risk to structural airworthiness is reduced to As Low As Reasonably Practicable (ALARP)".

- Regulation 5720(4) - Validating Structural Integrity: "Structural integrity shall be periodically assessed to ensure that Structural Airworthiness assumptions remain valid and the Structural Airworthiness risk is reduced to As Low As Reasonably Practicable (ALARP)".

- Regulation 5720(5) - Recovering Structural Integrity: "Reductions of fatigue and damage-tolerance clearances: Validation activities such as SOIU reviews and OLM/ODR (RA 5720(4)) may introduce changes to fatigue formulae and/or fatigue and damage-tolerance clearances that may reduce available safe life or render damage-tolerance examinations overdue. In the short term, such reductions may indicate that the risk of fatigue failure is higher than originally thought and, in extreme cases, that safe lives have been over-flown. Action is then required to recover Structural Integrity to restore risks to ALARP levels".

- Regulation 5721(1) - System Integrity Management: "Measures should be taken to counter threats to System Integrity, thereby reducing the airworthiness risks to Tolerable and ALARP".

- Regulation 5721(5) - Recovering System Integrity: "More than one of the threats to System Integrity can be in effect at any one time. The recovery of System Integrity may require a number of measures to be applied in concert. Recovery action will be fully effective once the root cause(s) have been identified and any associated risk recovered to ALARP".

- Regulation 5722(1) - Propulsion Integrity Management: "Where threats to PI are identified, airworthiness risks should be reduced to tolerable and ALARP".

- Regulation 5722(2) - Establishing Propulsion Integrity: "Within an Air System Safety Case and through the introduction of an Air Safety Management System (ASMS), it is necessary to assess the hazards that a Propulsion System can present to a platform and to ensure the risks posed by these hazards are Tolerable and ALARP (see RA1220 Project Team Airworthiness and Safety). These risks will be reported up to the platform level hazard analysis. For further information on Engine Hazardous effects see Def Stan 970 Part 11"

- Regulation 5722(5) - Recovering Propulsion Integrity: "Recovery measures may include repairs, modifications, additional examinations or testing, changes to component lives, or the imposition of operating restrictions. Additionally, the results of validation activities and in-service experience may bring fleet-wide PI into question, which may reinforce any imposed operating restrictions. Any requirement to impose operating restrictions would need to be agreed with the relevant Aircraft Operating Authority. Following a loss of PI it may be necessary to quantify risk and generate recovery options. Component modifications that alter the performance of the Propulsion System (e.g. increasing shaft speed) may affect other components in the system and the impact on those components affected will need to be assessed. Although critical component fatigue lives are usually considered to be finite, in exceptional circumstances and in order to manage a platform back to ALARP, the TAA may, in consultation with the DO, authorize critical part life extensions. This extreme action is to be carefully considered by the TAA if it is to be included as part of a recovery plan as it can, in itself, trigger a further reduction in confidence in PI".

- Regulation 5723(1) - Ageing Aircraft Audit: "All UK military registered aircraft types shall be subjected to an Ageing Aircraft Audit (AAA), to give confidence that airworthiness risks are at least tolerable and As Low As Reasonably Practicable (ALARP), as the fleet ages and regulatory requirements evolve."
- Regulation 5725(1): Development and Implementation of an Out of Service Date Extension Programme: "Without an Out of Service Date Extension Programme (OSDEP) this increased risk may not be detected and the level of risk at which the fleet is operating, may no longer be Tolerable and As Low As Reasonable Practical (ALARP)".

A.5 The UK Military Perspectives on Safe Life and Damage Tolerance

A.5.1 A Perspective from UK Manned Aircraft Standards [39]

By examining Def Stan 00-970 Part 1 Section 3 Clause 3.2, it became apparent that the SL approach is the standard approach in the UK; the evident advantage of this approach is to minimize the need for in-service inspections; see Section A.5.1.1 for more details. Then, a "clear by inspection" approach may be adopted to enable life extension beyond the safe life, and is used to overcome the threats of AD, which may be induced by events such as impacts, reported overloads, or reported break of corrosion protection systems; see Section A.5.1.2 for more details.

Furthermore, these military standards require the provision of means for determining the fatigue life consumption of each individual aircraft during service. By examining the standards that are briefly discussed in Section A.5.1.3, and by replacing the word Structural Monitoring by SHM, it was concluded that the means for determining the fatigue life could include:

- SHM systems that compute loads, strains, and fatigue using data acquired from strain gauges, fibre optic sensors, or any other emerging sensors regardless their types as long as they reliably perform their intended functions,
- SHM systems that use fatigue meter and fatigue formulae along with information such as SPC to estimate the fatigue life consumptions,
- Non-adaptive SHM systems that estimate loads and fatigue from flight data such as speed and acceleration,
- SHM systems that directly detect fatigue damage using dedicated sensors capable of detecting cracks well before they grow and become critical; e.g. cracks having a mid-life crack size, see Figure 35.

Def Stan 00-970 Part 1 Section 3 does not directly state the latter systems, but if combined with crack size assessment capabilities, such systems would satisfy the fatigue determination requirement for each individual aircraft. In other words, Section 3 does not explicitly include any references to systems that directly detect and assess damages such as cracks, delamination, or corrosion; also, Section 3 does not explicitly exclude such systems. Section 3 refers to specific technologies for measuring or estimating strains and fatigue life consumptions, namely: strain gauges, fatigue meters, and non-adaptive prediction methods. Section 3 does not include or explicitly exclude technologies that can perform the function of strain gauges, e.g., fibre optic technologies.

A.5.1.1 Safe Life

According to Def Stan 00-970 Part 1 Section 3 Clause 3.2, the UK military standards require the structures and mechanical components to have acceptable safe lives and acceptable tolerances to defects and damage. The SL approach is the standard approach to always be adopted for components where fatigue damage cannot be identified readily. The safe life of a component must be substantiated under design spectrum, normally by tests supported by calculations. The SL substantiation activities include adequate testing of pre-production representative subassemblies. Pre-production testing must be planned so that any consequent design changes can be timely and optimally introduced into production. For stabilised production standard structures, the SL substantiation activities include testing under a load spectrum representative of the service spectrum. The service spectrum should be determined using data from development flight tests and available OLM programmes. Following the completion of testing, a satisfactory residual strength must be demonstrated by analysis, supported, where appropriate, by evidence from additional testing and teardown exercises. A teardown exercise involves carefully dismantling and inspecting all components subjected to major load paths to reveal any significant fatigue damage. Usually, the service safe life of a component is a fraction of a test life spanning 90% of the course of the test during which the component did not fail (i.e. did not contain any detectable damage). The SL substantiation activities must account for the variability of the manufacturing processes, product forms, and material fatigue properties.

Each powered flying control system must have a safe life determined as described above and demonstrated by a fatigue test.

To ensure that the structures and mechanical components have acceptable safe lives and acceptable tolerances to defects and damage, Clause 3.2 specifies the criteria for assemblies and material selection, which include:

- Good fatigue performance,
- Good resistance and tolerance to damage and crack growth,
- Good resistance to, or protection from, environmental degradation, and
- Tolerance to accidental damage.

A.5.1.2 Clearing by Inspection

Def Stan 00-970 Part 1 Section 3 Clause 3.2 indicates that for a component exposed to impact damage or cleared by inspection to remain in service beyond its safe life, an inspection-based substantiation must be performed. Normally, the SL substantiation of a component cleared by inspection requires demonstrating the safe life to be at least half of the life under the design spectrum (the specified life). The inspection periodicity is substantiated by calculations, supported by evidence from relevant crack growth testing, teardown inspections, and in-service inspections. Hence, the compliance with these UK "clearing by inspection" standards requires inspection when a component is exposed to reported damage or can be used to support a life extension; the compliance with this approach suggests that a component may remain in service if:

- The presence of fatigue cracks can be identified with acceptable confidence.
- Any crack that remains undetected after an inspection will not grow, under the service spectrum, to an unacceptable size before the next inspection or before scheduled replacement or retirement.
- The inspection penalty is acceptable on operational and economic grounds.
- In-service incident reports, if any, are taken into account.

A.5.1.3 Requirements for Service Monitoring

As mentioned previously, Def Stan 00-970 Part 1 Section 3 Clause 3.2 requires the provision of means for determining the fatigue life consumption of each individual aircraft during service. Therefore, every aircraft in the fleet must be provided with instrumentation, for the purpose of estimating the fatigue damage accumulation. If a fleet-wide instrumentation is insufficient to monitor all fatigue-critical components, then, a representative sample of aircraft must be fitted with instrumentation that is more extensive for in-service loads assessment. OLM for fixed wing aircraft or ODR for helicopters must be used to demonstrate the effectiveness of the monitoring system and underpin any simple lifing metrics adopted, e.g. hours and number of landings. Structures monitored by such simple lifing metrics are referred to as “unmonitored” structures. Structures are considered “monitored”, if all critical loading histories are rigorously assessed for life usage over the entire period of significant fatigue loading via direct monitoring, e.g. by strain gauges, or by inference from monitored flight parameters. The events and magnitude of design limit exceedance must be identified before the next flight and made available for post flight activities. The monitoring system should be active for the entire period of significant fatigue loading. Clause 3.2 also provides requirements for data integrity, accuracy, and reliability of the monitoring system.

Def Stan 00-970 Part 1 Section 3 Clause 3.2 describes the installation requirements of a monitoring system called “fatigue meter” and indicates that this system may be used for determining the fatigue life consumption of each individual aircraft during service. The fatigue meter measures the acceleration near the aircraft centre of gravity and counts the numbers of events when pre-set levels of vertical acceleration are exceeded. Fatigue formulae use the fatigue meter information along with information such as SPC and number of Air to Air fuel probe contacts to estimate the fatigue lives of components.

Def Stan 00-970 Part 1 Section 3 Clause 3.2 also describes in detail the standards required for monitoring systems that use non-adaptive prediction methods to estimate loads and fatigue. A prediction method is defined as a set of coefficients/weights and a set of transformation equations that operate on a set of inputs (such as flight parameters e.g. normal acceleration, roll rate, etc.) to produce outputs that approximate a target value (e.g. stress, strain, load, or fatigue damage). Prediction techniques use a range of mathematical or statistical methods that may include neural networks, model-based analysis, linear or non-linear regression, clustering algorithms, etc. Non-adaptive prediction methods use a fixed set of weights, which are evaluated through calibration/training using target data that contains examples of inputs and target outputs. After training, the coefficients of non-adaptive methods are fixed until the commencement of any further training. Reference [71] describes and reports some of the results of the first non-adaptive method introduced worldwide that has provided reasonable fatigue and load information. This method motivated the inclusion of non-adaptive methods within the UK Def Stan 00-970 and has been considered for the JSF prognostic SHM system.

A.5.2 A Perspective from Remotely Piloted Air Systems' Standards [40]

A Remotely Piloted Air Systems (RPAS) consists of several elements that are critical to engineering and flight safety including not only the flying Remotely Piloted Air Vehicle (RPAV) and all its associated flight safety-critical elements, but also elements such as the ground-based control unit and the ground-launch system. Def Stan 00-970 Part 9, Reference [40], presents the UK design and airworthiness requirements for RPAS; Part 9 requires making reference to the other parts of Def Stan 00-970 and to Def Stan 07-85 which, although relating to the design requirements for manned aircraft and guided weapons respectively, can also be applicable to RPAS. Def Stan 00-970 Part 9 explicitly quotes the term "damage tolerance" only in one clause within the section on "Fatigue Evaluation". Part 9 does not explicitly use the words "safe life"; it uses the words "fatigue life" only in Clause 609d of the section on "UAV Design and Construction"; Clause 609d requires that all major assemblies are uniquely identified by the manufacturer; Clause 609d also indicates that for the purpose of establishing fatigue life, a record should be kept of all exchangeable and replaceable items subjected to fatigue wear. In addition to the mandatory requirements of Def Stan 00-970 Part 9, a group of NATO airworthiness standards with minor UK national reservations are also mandated including those NATO standards set out in STANAG 4671 "Unmanned Aerial Vehicles System Airworthiness Requirements (USAR)", Reference [60]. STANAG 4671 quotes the words "safe life" and "damage tolerance" in two sections. In the "Fatigue Evaluation" section, Clause USAR.570, supported by the Acceptable Mean of Compliance AMC.570, mandates performing the following: (a) fatigue and damage tolerance analysis for UAV systems having metallic or composite structures, excluding those UAV for which certification is requested for very short life time, (b) adequate demonstration to predicted life time using USAR.572 to USAR.575, and (c) consideration of fatigue monitoring, in agreement with the Certifying Authority; USAR.572 and AMC.572 set out requirements and means of compliance for "Metallic fuselage, wing, empennage and associated structures"; USAR.573, USAR.575, AMC.573, and AMC.575 set out the requirements and means of compliance for "Damage tolerance and fatigue evaluation of composite and metallic airframe structure" and for "Inspections and other procedures". SL and DT requirements are also set out in the section on "UAV Design and Construction".

Def Stan 00-970 Part 9 also quotes the clauses of the "Indicators and Warning" section of STANAG 4671 including the following clauses:

- USAR.U1785 that requires colour coding the warning, caution or advisory information displayed by the UAV Control Station (UCS) red, amber, or green.
- USAR.U1787 that requires including an automatic diagnostic and monitoring capability in the UCS for the status of the UAV system and providing appropriate warnings to the UAV crew with the guidance for corrective actions provided either automatically or in the UAV system flight manual. In addition, the UK Def Stan 00-970 Part 9 requires making the health monitoring data available to be electronically entered.
- USAR.U1788 that requires a UCS configuration capable of informing the UAV crew of any abnormal or emergency mode, including cases in which there is an automatic switching to an alternate mode of operation.
- USAR.U1789 to USAR.U1829 that require providing the following: low speed warning, UAV mode of control indicator, wing flaps position indicator, landing gear position indicator/warning, pressurised compartment indicator, fuel pumps warning, air induction indicator, battery discharge warning, indicators for power-assisted valves in the power plant, UAV electrical systems warning/indicator, de-icer boot system indicator, hydraulic systems indicator, fire protection warning, pitot heat indicator if a pitot heating system is installed, UCS power distribution indicator, flight control system lock warning, flight-path deviation warning, and UAV safety status indication.

A.5.3 MOD Regulations that Point to SL and DT [36]

A.5.3.1 Regulation 5309(1) - Fatigue Type Record for Aircraft

This regulation mandates that "The PTL shall decide the requirement for a Fatigue Type Record (FTR), part of a FTR, or revision to a part of a FTR." As AMC with Regulation 5209(1), the FTR should consist of four parts:

- Part 1: A historical record of the fatigue substantiation for the aircraft,
- Part 2: A reassessment of fatigue life and damage tolerance in the light of service usage and fatigue test results,
- Part 3: A reassessment of inspection requirements shown to be necessary by the analysis of Part 2, and
- Part 4: A Life Extension document, based upon information in Parts 1 through 3, OLM and the most recent full-scale fatigue test.

In summary, the four parts should comprise, but not limited to the following: (a) the principles used to underpin the platform SL and its tolerance to unforeseen sources of damage based upon a survey of all existing fatigue and damage tolerance analyses and tests used in the original design; (b) reanalyses of the fatigue lives of critical components based upon the service user spectrum and/or any fatigue data taking account of test or operational failures; the service user spectrum is defined by the platform's Statement of Operating Intent and Usage (SOIU); (c) reassessments of inspection requirements including NDI methods used, post inspection flaw assumptions, and inspection intervals; (d) reviews of original calculations to extend the aircraft life; the reviews should contain: any required structure rework, revised operational data (revised mission profiles, utilisation, etc.), new inspection procedures, etc.

A.5.3.2 Regulation 5720 - Structural Integrity Management

The following section, Section A.6, is dedicated to discussions covering this regulation.

A.6 Structural Integrity

Structural Integrity (SI) is defined by MOD as "The ability of an aircraft structure to withstand without collapse or unacceptable deformation the loads imposed throughout the aircraft's service life by operation of the aircraft within the limitations of the Military Aircraft Release and to the usage described in the Statement of Operating Intent and Usage". Safety is defined as "The freedom from unacceptable risks of personal harm". Airworthiness is defined as "The ability of an aircraft or other airborne equipment or system to be operated in flight or on the ground without significant hazard to aircrew, ground-crew, passengers, or to third parties; it is a technical attribute of material throughout its lifecycle". The third parties include the general public and friendly military personnel over which airborne systems are flown. These definitions imply that maintaining airworthiness would involve maintaining the main aspects of SI and safety, which is achieved by identifying potential risks and taking appropriate preventive and corrective actions to mitigate the risks and their consequences on SI and safety. This section is dedicated to SI; Section A.7 briefly discusses the main causes of failures that can lead to airworthiness risks. Sections A.8, A.9, and A.10 briefly review the maintenance approaches implemented to mitigate these risks.

A.6.1 References to SI within the UK Defence Standards

Only three clauses in Def Stan 00-970 Part 1 Section 3 cite "structural integrity":

- According to Clause 3.2.22, "Every aircraft in the fleet shall be provided with instrumentation, for the purpose of estimating the fatigue damage accumulation for the maintenance of structural integrity. Provision shall be made for any required instrumentation during production".
- According to Clause 3.10.55, "In the assessment of structural integrity consideration shall be given to all Active Control System (ACS) modes, including those degradations and failures from which the aeroplane can reasonably be expected to recover". For example, a loading assessment must include those loading conditions which would exist following the occurrence of reasonable combinations of system degradation and structural damage (e.g. a bird-strike or minor battle damage) from which recovery is expected.
- According to Clause 3.10.100 in performing Manoeuvre Load Alleviation (MLA) function "the system shall not compromise overall structural integrity".

Def Stan 00-970 Part 1 Section 3 contains the military requirements for designing and qualifying certifiable structures that have acceptable safe lives and acceptable tolerances to defects and damage. Section 3 explicitly requires service monitoring having instrumentation for the purpose of estimating fatigue. Section 3 also requires NDI equipment to enable Inspection-Based Substantiation. However, Section 3 does not explicitly state any requirements for other systems that can directly detect or assist in detecting damages threatening SI such as cracks, corrosion, and delamination.

The following leaflets of Def Stan 00-970 Part 1 Section 3 cite "structural integrity":

- Leaflets 1 "Static Strength and Deformation, Underlying Principles" outlines the basic principles underlying strength and deformation requirements.
- Leaflet 2 "Static Strength and Deformation, Static Test Philosophies" describes a qualification test route via a 'pyramid' approach starting with testing many coupons, leading to testing elements, details, and sub-components, and finally, testing components and/or airframe.
- Leaflet 28 "Active Control Systems, Structural Implication of ACS" indicates that the procedures used for static and fatigue design of aeroplanes incorporating ACS are similar to those for conventional aeroplanes; however, critical design cases may be more difficult to determine, which necessitates the integration of the procedures used for structural, aerodynamic, and ACS designs.
- Leaflet 44 "Impact Damage Resistance of Composite Material Structures" gives guidance on the certification and qualification route for composite structures, with particular reference to impact damage resistance.
- Leaflet 36 "Fatigue Inspection-Based Substantiation" describes acceptable procedures for demonstrating compliance with the requirements of Clause 3.2. The leaflet covers the following: (a) conditions governing the use of inspection-based substantiation, (b) detection of cracks, (c) time to first inspection, (d) conditions governing the determination of crack-growth curves, (e) allowances for uncertainties in estimates of inspectable life, (f) derivation of inspection intervals, and (g) check for the sensitivity of inspection intervals to increases in loading severity.

The first four leaflets provide guidance on the design and qualification requirements of certifiable structures. The last leaflet contains requirements for inspection to be carried out by trained operators, i.e. using NDI equipment. SHM systems performing similar inspection tasks can be developed to meet these requirements.

A.6.2 Regulatory Article 5720 - Structural Integrity Management

It is worth noting that the only explicit reference to SHM is cited in the guidance materials of RA 5720(2); all other MAA publications and UK defence standards do not explicitly refer to SHM. Also, SHM is not formally defined by MAA or within the UK defence standards. Nevertheless, the MAA articles 5720(1) to 5720(6), which are presented in the following sections, mandate SI regulations; the AMC with these regulations include the following requirements that can be fulfilled by SHM systems:

- "OLM/ODR and maximum capture of usage data using serviceable monitoring systems": these requirements can be directly addressed by SHM systems that monitor usage and loads/strains.
- "Monitoring fatigue consumption": this requirement can be addressed directly or indirectly by the same SHM systems with acceptable fatigue computations added to convert usage or strains to fatigue consumption.
- "Determining and controlling mass, Centre of Gravity (CG), and mass distribution": it is possible to develop a SHM system using few sensors and aircraft data that can determine these three parameters on the ground and on the air, and can assist in controlling the determined parameters.
- "Structure Examination Program (SEP) to monitor AD, ED, and FD": currently this requirement relies on NDI equipment and can be addressed by SHM systems having sensors that directly detect these damages and assess their impact on SI.
- "Reviewing and amending the SOIU": to assist achieving these requirements, SHM systems can be developed to monitor the usage using Flight Condition Recognition (FCR) algorithms and to identify events such as landings and Air-to-Air refuelling.

The AMC with the MAA regulations also include key requirements that can be addressed by a single powerful Structural Health Management System; examples are:

- compiling and maintaining a list of Structurally Significant Items (SSIs),
- managing fatigue qualification evidence and maintaining computer model to generate qualification evidence,
- publishing, validating, and updating SOI/SOIU,
- downloading any aircraft electronic data and entering the data into electronic systems,
- establishing a system to record the structural configuration and condition of each aircraft and interchangeable components in the fleet,
- using systems for: usage data capture, usage computation, managing overall fleet usage, and monitoring individual aircraft usage,
- reporting damage and repair events and recording them on a suitable database,
- establishing procedures for Individual Aircraft Tracking (IAT) fill-in rates for unmonitored flights, and
- communicating information, reviews, and reports with relevant stakeholders.

A.6.2.1 Regulation 5720(1) - Structural Integrity Management

"All aircraft operated within the Military Air Environment (MAE) shall be managed to ensure an acceptable and demonstrable level of Structural Integrity".

The RA specifies the activities required to achieve and maintain SI for any military aircraft type from its project inception through to its eventual disposal. The MOD SI management framework consists of 5 groups of activities: Establish, Sustain, Validate, Recover, and Exploit (ESVRE); the costs of the key elements of ESVRE should be included in a project Through Life Management Plan (TLMP). The ESVRE framework should be applied to the acquisition cycle of new types and should be retrospectively applied to legacy types. SI should achieve airworthiness, but only by undertaking the activities described in RA 5720 and associated RAs, which require measures to be taken to counter sources of threats to SI, reducing the risks to ALARP.

A.6.2.2 Regulation 5720(2) - Establishing Structural Integrity

"Structural Integrity shall be established to demonstrate that the aircraft structure is airworthy to operate under agreed conditions".

As AMC with Regulation 5720(2), the Project Team (PT), for each aircraft type under their control, should ensure:

- establishment of a SI strategy describing the intended approach to implementing the required through-life SI management activities,
- availability of adequate funding to support the SI strategy throughout the anticipated service life of the aircraft,
- compiling and maintaining a complete list of SSIs for use in management of SI,
- providing and updating static and the fatigue qualification evidence determined by the Design Organization (DO) throughout the life of the aircraft,
- seeking the MAA guidance necessary for ensuring the existence of an initial Statement of Operating Intend as early as possible in the acquisition cycle but no later than the aircraft's Introduction-to-Service (ITS),
- publishing the SOI for each aircraft type and reviewing the SOI with the DO whenever the intended use of the aircraft type is changed,
- supporting and maintaining computer models used to generate qualification evidence in an accessible and usable state for the life of the aircraft type, and
- forwarding any key and cross-platform SI issues to the Combat Air Airworthiness Management Group (CAAMG), Air Support Airworthiness Management Group (ASAMG), and Helicopter Airworthiness Management Group (HAMG).

Characterizing SSIs as SL or DT items is essential to the derivation of preventive maintenance programmes using MSG-3/RCM logic. According to the guidance materials of RA 5720(2), "the SL items are those items designed to have a fatigue life at least as long as the in-service life of the aircraft, or those where application of a DT approach is not possible"; the interpretation of the quoted definition should be extended to clearly include SL items having limited SL less than the in-service life of the aircraft and, in the same time, cannot be considered as DT items because, for example, (a) they can suffer rapid crack growth under dynamic loads within a period less than any reasonable service period, or (b) their replacement is less expensive than their inspection programmes. The vulnerability of SL items to AD and ED is analysed during the development of the preventive maintenance programme to determine whether the threats of AD and ED must be detected by an appropriate inspection regime. The airworthiness of the DT items is assured by a specified inspection regime. The DT items do not only include SSIs designed and demonstrated to be DT items, but also include SSIs designed to be SL items and subsequently re-categorized as DT items (a) because of failure on test or in service, or (b) by life extension beyond the designed SL limit and where SI can be assured using an inspection-based regime.

The SI strategy should address topics such as: (a) defining the requirement for continued airworthiness, (b) engaging SI stakeholders including independent advisors, (c) production, publication, and evaluation of SOI/SOIU, (d) determination and implementation of IAT methodologies, e.g. Fatigue Meter Formula (FMF)/Structural Health Monitoring (SHM)/ HUMS, RA 5720(3), and (e) determination and implementation of usage validation requirements: OLM, ODR, or Manual Data Recording Exercise (MDRE); see RA 5720(4).

Aircraft accepted into UK military service may be designed to satisfy one of a number of different design requirements or standards such as Def Stan 00-970, international military standards, or civil standards. Notwithstanding the wealth of evidence required for certification and qualification of the aircraft structure, the minimum evidence required to sustain the management of the aircraft throughout its service life is summarised in the static and fatigue evidence document set.

A.6.2.3 Regulation 5720(3) - Sustaining Structural Integrity

"Structural Integrity shall be sustained and managed appropriately in order continuously to monitor, measure, and counter the threats to Structural Integrity so that the risk to structural airworthiness is reduced to As Low As Reasonably Practicable (ALARP)".

As AMC with Regulation 5720(3), Aircraft Operating Authorities (AOAs) should ensure:

- maximizing the capture of usage data by maintaining serviceable monitoring systems,
- promptly downloading any electronic data and entering the data into electronic systems,
- implementing fatigue budgeting measures, and

As AMC with Regulation 5720(3), the Project Team (PT) should ensure:

- planning SI assurance measures in their aircraft TLMP,
- periodically reviewing, updating, and marinating the SI strategy and its associated document and plan, broken down into individual programme, recurring and one-off activities, key milestones, and decision points, as appropriate,
- monitoring fatigue consumption, investigating fatigue modification or refurbishment programmes, and appropriately authorizing, in exceptional circumstances, any temporary extension beyond the cleared SL or DT examination interval,
- sponsoring, producing, and implementing a SEP identifying SSIs and characterizing them as either SL or DT, and collating/reviewing SEP results including: (a) AD/ED examination results for both SL and DT items, and (b) FD examination results for DT items,
- establishing a system to record the structural configuration and condition of each aircraft and interchangeable components in the fleet,
- the existence of systems for: usage data capture, usage computation, managing overall fleet usage, and monitoring individual aircraft usage, and for sponsoring changes to these systems as and when required,
- the existence of systems to determine and control individual aircraft mass, CG, and mass distribution, and to make any consequent overall changes to the fatigue and usage monitoring systems, SOIU, FTR, and/or Static Type Record (STR) containing static qualification evidence,
- the existence of a Obsolescence Management Plan (OMP) as part of the aircraft TLMP.

As AMC with Regulation 5720(3), The DO should:

- Specify for each SSI the usage parameter to be used and either the SL or DT threshold and interval, in terms of the appropriate usage metric.
- Provide the usage monitoring system and associated FMF along with any necessary support for the implementation and prototyping of new or revised FMF.

The guidance materials of 5720(3) include but not limited to the following mandated measures for SI assurance:

Table 1: Structural integrity assurance measures

Structural Integrity Assurance Measure	Mandated Regulation	Notes	References
Fatigue and Usage Monitoring and HUMS Assess Fleet Leader status and need for budgeting.	Continuous	Vital for planning inspections and other maintenance activities and for ensuring test clearances are adequate	Def Stan 00-970 Pt 1 Sect 3 Lft 38 RA 5720(3) Sustaining Structural Integrity
Conduct a continuous or periodic OLM/ODR programme throughout aircraft life.	Initially within 2 years of entry into service 5 year maximum between subsequent OLM/ODR programmes Requirement to be reviewed 3-yearly, on change of role, or prior to life extension programme	OLM for fixed wing Temporary ODR for helicopters	Def Stan 00-970 Pt 1 Sect 3 Lft 38 RA 5720(4) Validating Structural Integrity
Conduct Ageing Aircraft Structural Audit.	Initially after 15 years service. Repeated every 10 years	The insidious nature of degradation and the interaction of apparently unrelated ageing processes is often only found by rigorous periodic audit of trend data, procedures and, if necessary, the aircrafts physical condition.	RA 5723(1) Ageing Aircraft Audit
Conduct structural sampling and tear- down.	Scheduled plan required once aircraft declared ageing	Use to be made of Cat 3, 4, and 5 aircraft. Review fleet impact of results.	RA 5723(1) Ageing Aircraft Audit
Life extension measures	Continuously review life extension measures.	Most military aircraft retire after original planned life.	Def Stan 00-970 Pt 1 Sect 3 Lft 39 RA 5723(1) Ageing Aircraft Audit
Structural Inspection Programme	For aircraft following Damage Tolerance principles including USAF ASIP programme	Supports Damage Tolerant designs and although planned, can be burdensome.	Def Stan 00-970 Pt 1 Sect 3 Lft 36

SEP is one of the key requirements for sustaining SI. Whilst regular examinations to detect cracks before they become critical is required for DT structures, inspecting and repairing detected damages is the only way to overcome the threats of AD/ED for both SL and DT structures. In addition, safe lives, DT examination thresholds, and DT inspection intervals are based on analysis and/or testing of structure that has not been exposed to AD/ED. Furthermore, as aircraft age, the likelihood of interaction between the different threats to SI increases. Therefore, a SEP supported by ‘structural sampling’ is required to (a) monitor

AD/ED in all structure types, (b) monitor FD in DT structures, and (c) validate design, usage, test, and maintenance assumptions.

SEP requires the characterization of each SSI as a 'SL structure' having a substantiated safe life or a 'DT structure' having an examination threshold (time to first inspection) and examination interval (time between subsequent inspections). Also, each SSI must be assessed for its vulnerability to AD/ED as part of a maintenance schedule development process (e.g. a RCM process), which commonly deal with AD or ED mechanisms acting independently; hence, the vulnerability of each SSI to the interaction between the threats to SI should also be assessed. The result of the assessment of vulnerability is to designate each item as either At Risk (AR) or Not At Risk (NAR) of AD/ED. The AR items must be included in the Master Maintenance Schedule (MMS) to ensure they are examined at a suitable frequency. The exclusion of NAR SSIs from MMS may occur because they are SL items or DT items with a long examination threshold. The SI of the excluded items can be maintained by carrying out checks on sample of them before they reach the end of their SL or first DT inspection. Therefore, the SSI list must be cross-referred to the MMS list. SSIs that are not included in the MMS must be subject to structural sampling to confirm that they do not suffer from AD, ED, or FD earlier than expected.

A.6.2.4 Regulation 5720(4) - Validating Structural Integrity

"Structural integrity shall be periodically assessed to ensure that Structural Airworthiness assumptions remain valid and the Structural Airworthiness risk is reduced to As Low As Reasonably Practicable (ALARP)".

The AMC with Regulation 5720(4) includes:

- reviewing, approving, and amending the SOIU when necessary to confirm that the fleet is being operated within the Release To Service (RTS) limitations, and conducting MDREs if required,
- identifying, sourcing, planning, and carrying out OLM/ODR programmes as summarised in Table 1, and
- conducting sampling plan development, maintenance schedule reviews, and structural qualification evidence reviews.

A.6.2.5 Regulation 5720(5) - Recovering Structural Integrity

"Structural Integrity shall be recovered to restore confidence in Airworthiness if there is a loss of, or potential compromise to, Structural Integrity".

The AMC with Regulation 5720(5) includes, but not limited, to the following:

- As far as possible, report damage and repair events; and record them on a suitable database for structural configuration control that includes structural concessions, repairs, modifications, accidental damage, and environmental damage for all aircraft.
- Evaluate any arisings of structural damage; the evaluation and recovery actions are monitored by the Structural Integrity Working Group (SIWG).
- Whenever possible, carry out investigations to recover lost usage data.
- Introduce and apply procedures for IAT fill-in rates for unmonitored flights as required.
- Assess the impact of any changes in aircraft/component lifing resulting from validation activities; approve/accept the associated recovery action/mitigation.
- Review component lifing, particularly where components that do not have individual lifing

records may be moved between aircraft and may exceed their original cleared life.

A.6.2.6 Regulation 5720(6) - Exploiting Structural Integrity

"Structural Integrity shall be exploited to make best use of the inherent capabilities of the aircraft structure".

The AMC with Regulation 5720(6) includes, but is not limited to, the following:

- Ensure that static and fatigue clearances are developed, by test and analysis, and are adequate to meet the requirements of the Release to Service Authority (RTSA) and AOA.
- Ensure that cleared fatigue safe lives are maintained ahead of the fleet leader.
- Consider the need for life extension: plan the life extension decision point; engage the DO and Subject Matter Experts (SMEs) to perform the required analysis and work.
- Consider the use of the Principal Structural Element (PSE)/Structural Control Point (SCP) method if the need arises to move from a safe life to a damage-tolerance philosophy.
- Consider whether pre-emptive reinforcement may be preferable to repair or modification.

The PSE/SCP method is a DT approach in which inspection requirements are minimized: a PSE is a feature within an airframe structure such as wing carry-through structures and undercarriage back-up structures; each PSE contains a number of SSIs that are located in the same region and subjected to similar stress fields; the SCP of a PSE represents the lead fatigue feature, i.e. it is the SSI that is expected to fatigue before the other SSIs within the PSE; by examining the SCP only, the condition of all SSIs within the PSE can be inferred.

In-service cracking problems may be addressed by repairs or modifications. Pending repair or modification, SI would normally be maintained by a DT approach. Repairs and modifications have to be qualified to ensure that the necessary static strength and fatigue endurance will be achieved; the qualification activities can be expensive and time-consuming. Pre-emptive reinforcements would not require the same qualification rigour and are intended to reduce the risk of damage by reducing stresses and stress concentrations in known sensitive regions of structure. Pre-emptive reinforcement is quicker and simpler to approve than full modification. An example of a pre-emptive reinforcement is a bonded composite reinforcement applied to a component suffered from damage to reduce the risk of damage growing beyond repair limits.

A.6.3 The Use of HUMS data for SI Tasks

According to MAA, the term HUM encompasses a variety of techniques including operational load monitoring, vibration analysis, visual inspections, oil & wear debris analysis. MAA has mandated the application of HUMS through the two regulatory articles. Furthermore, several guidance materials of regulatory articles encourage the use of HUMS for structural integrity and life extension tasks.

A.6.3.1 RA 4500(1) - Application of Health and Usage Monitoring Systems (HUMS)

"HUMS shall be included on all new aircraft platforms and retrofitted to existing aircraft fleets where justified by airworthiness and/or cost considerations"

A.6.3.2 RA 4500(2) - Exploitation of HUMS requires

"HUMS data shall be exploited to preserve and enhance flight safety and realize maintenance benefits".

A.6.3.3 Guidance Materials of Regulatory Articles Pointing to HUMS

The guidance materials of RA 5720(2), RA 5729(3), RA 5720(6), and RA 5724 have highlighted the following:

- HUMS can be used for Individual Aircraft Tracking (IAT).
- HUMS can be used for Structural Integrity Assurance.
- Operational Data Recording (ODR) or HUMS can be used for fatigue conservation purposes.
- HUMS data, as available, can be used to identify the most significant contributors to structural life consumption.
- Using HUMS data to support Life Extension Programme (LEP) could provide invaluable information to understand in-service usage for all types of aircraft.

A.7 The Main Failure Causes of Structural and Mechanical Components

The SL and DT approaches are adopted to deliver airworthy components and maintain their target reliabilities throughout their in-service lives. The target reliability is achieved by minimizing the risks of failures to very low acceptable levels. Through careful identification of all potential failure modes, integrity management programmes are established and associated maintenance activities undertaken to overcome the threats of functional and component failures, and hence, to ensure that the target reliabilities are maintained. A failure mode is the root cause of a functional failure. Each potential failure mode should be adequately identified so that the most appropriate failure management action can be established. Root causes of failure, e.g. failure modes such as fatigue, lack of lubrication, corrosion, and pilot fatigue, are far more useful than the failure itself to identify effective failure preventive tasks. The following sections present brief discussions covering the main causes of failures.

A.7.1 General

The failure of a component occurs as a result of usage, hazardous-operations, exposure to environments, and/or component interactions with other components or objects. Damages caused by repeated stresses, thermal cycles, or overloads are examples of the usage and hazardous-operation damage. The exposure to salty water and sandstorms can cause corrosion/erosion leading to loss of material strength and damage caused by environments. Accidental damage, rubbing, foreign object strikes, battle damage, and damaging effects of vibration induced by highly unbalanced rotating components on neighbouring components are examples of interaction damage. The causes and probability of component failure can change with component age and can be influenced by how the component is designed, manufactured, and maintained. The ‘bath tub’ failure probability, curve introduced by Carhart in 1953 provides a general model that encompasses the three causes of failure and describes how the probability of failure can change with component age, Figure 42. For aircraft components, significant departures from the ‘bath tub’ failure curve have been observed. For most aircraft components, the middle and final age regions are characterized by age-independent probabilities (almost-constant probabilities). These age-independent probabilities of failures are attributed to rigorous airworthiness requirements and associated maintenance programmes that eliminate the increasing trend of probability at the final-age region; in other words, aircraft components are maintained, repaired, or retired and replaced before any possible failures under expected usage spectra. For well designed/manufactured components, the high probabilities of failure at the early age region have not been encountered.

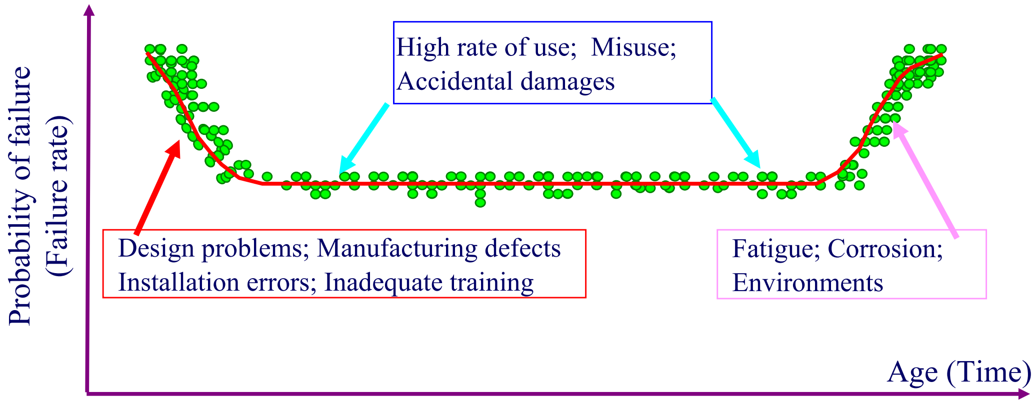


Figure 42: Changes in dominant failure causes with age

Generally, the life of a component is consumed as a result of the component being stressed mechanically, thermally, and/or chemically. For example, varying stresses can cause fatigue damage and growth of microscopic defects to damaging cracks. The progression of fatigue cracks can be split into three stages: initiation, crack growth, and final fracture. The well-known S-N curve gives the number of cycles (N) that cause final fracture under repeated stress cycles (S); the S-N concept is an attempt to capture the three progression stages from initiation to final fracture. Crack initiation depends on factors such as surface conditions (scratches, corrosion, and residual tensile stresses). Crack growth depends on the average stress in the surrounding region and depends less on stress concentration factors but can be aggravated by highly corrosive environments. Often, final fracture occurs in a brittle manner independent of the average stress but dependent on the crack tip condition. Damage tolerant approaches address the last two stages of crack progression. Aircraft critical components are therefore designed with safety factors to eliminate the probability of the fatigue being accumulated (or the cracks being grown) to high values leading to failure during promulgated lives. The aircraft manufacturers use the safety factors to compensate for factors such as: aircraft usage outside the usage spectra, operations under stress levels higher than the expected levels, and scatter in material data. Under the simultaneous action of corrosion and repeated stresses, the fatigue strength of most metals is reduced.

Corrosion can occur in components exposed to salty water. Equally, metallic surfaces rubbing together that release sufficient energy for chemical formation can cause corrosive (oxidative) wear. The fatigue fretting associated with surface rubbing can be aggravated by humidity. Centrifugal loads and gas loads induce varying stresses and fatigue in engine components. Temperature gradients between hot and cold parts of components induce significant stresses even under steady state conditions. Life consumption of hot engine components does not only depend on varying stresses but also depends on the time spent at constant amplitude loads. Such components can develop considerable strains over a time and are said to creep.

Life consumption can also depend on factors such as stress concentrations, oxidation, and microstructure transformation at high temperatures. Furthermore, the damage mechanisms of two engine components can be at a great variance. For example, the term "Thermal Mechanical Fatigue (TMF)" has been used to refer to the damage induced by interacting thermal, mechanical, creep, oxidation, and microstructure transformation effects. Cooled high-pressure turbine blades can experience TMF. The term "Fatigue" or "Thermal Transient Fatigue (TTF)" refers to the damage induced by interacting thermal and mechanical stresses. TTF models can be applied to components such as high-pressure turbine discs. Isothermal fatigue models (non-transient) can be applied to components such as low-pressure compressor components.

Erosion can be induced by sand and dust. The spalling of a bearing track starts with a fatigue crack below the surface caused by high stresses. Cavitation bubbles that implode on a component surface induce high local impact forces which may cause damage to the surface. Repeated implosion of cavities causes cyclic stresses, which result in surface fatigue wear.

A.7.2 Low Cycle Fatigue (LCF)

LCF is mainly induced by stress variations between aircraft/engine operational conditions. For example, LCF is induced by manoeuvres, gusts, ground-air-ground cycles, cabin pressurizations, landing gear movements, releasing stores, catapult launching, firing weapons, taxiing, rotor start-up and shut-down, changes in engine power, thermal changes, hydraulic and fuel system pressurizations, and arrester hook use. If the stress variations between conditions are high, LCF can consume the component life after a relatively small number of stress cycles. For example, LCF caused by variations in rotational speed stresses can cause engine disc failures after a number of cycles ranging from 1000 to 50000 cycles depending on material and stress variation levels. Generally, engine thermal stresses reduce the effect of centrifugal stresses. The centrifugal stresses can change significantly if re-burst takes place. Re-burst occurs as a result of temperature gradients that cause high accumulative stresses several seconds after major throttle movement, for example, during take-off where the disc bore is relatively cold and the rim is very hot.

A.7.3 High Cycle Fatigue (HCF)

HCF is mainly induced by high frequency stress variations (vibration) within quasi-steady operational condition variations. In other words, HCF is caused by high frequency cycles, which are superimposed on the quasi-steady stresses that induce LCF. HCF includes vibration caused by acoustic loading, flutter, buffeting, and rotating components in aircraft dynamic systems. HCF of helicopters is mainly induced by rotor vibratory loads at blade passing frequencies and hence, can be influenced by the usage described by times spent in flight conditions. In contrast, HCF of engines is hardly influenced by usage and, hence, should be controlled by good designs. Often, such designs cannot completely eliminate HCF because of the effects of wear, manufacturing irregularities and mishandling, which can induce loads that excite, for example, blades or structures at natural frequencies.

A.7.4 Creep

Damage of hot mechanical components does not only depend on varying stresses but also depends on the time spent at constant amplitude loads. Such components can develop considerable strains over time and are said to creep. Creep is a function of load and the time spent at high temperature. Creep failure starts in the grain boundaries rather than within the grains leading to distortions. Turbine blade materials are currently designed such that creep is avoided by providing many load-bearing paths along the blade radius without crossing grain boundaries (uni-axially solidified and single crystal turbine blades).

A.7.5 Corrosion

Corrosion can be defined as the deterioration of materials due to electrochemical reactions with environments. Corrosion can attack materials including metals, polymers such as plastics and rubbers, ceramics, composites, and mixtures of two or more materials with different properties. Corrosion occurs in many forms. General corrosion occurs due to direct exposure to corrosive fluids such as acids and salty water and can attack structures almost uniformly causing slow weight losses and strength reductions. Crevice corrosion occurs when corrosive fluids are trapped between two crevice surfaces such as flanges, fasteners, and lap joints. A crevice susceptible to corrosion would have sufficient width to permit entry of corrosive fluids but narrow enough to trap stagnant fluids. Crevice corrosion is a very similar mechanism to pitting corrosion. Pitting corrosion causes a localised material loss, and if aggravated by stresses can lead to fatigue failure. Galvanic corrosion occurs when two dissimilar conducting materials are electrically connected in the presence of an electrolyte; it is caused by an electrochemical reaction in the presence of an electrolyte and an electron conductive path. Inter-granular corrosion occurs along multiple grain boundaries, causing for example, exfoliation in aluminium alloys. Stress corrosion cracking is caused by the combined effects of tensile stresses and corrosive environments. Stress corrosion cracking develops rapidly in the grain boundaries and cracks grow under the simultaneous effects of the tensile stresses and the corrosive environments. Corrosion fatigue is caused by cyclic stresses applied to corroded metals. Fretting corrosion occurs at the interface between two contacting surfaces due to a relative motion in a corrosive medium causing oxidation of wear debris and can take the form of accelerated atmospheric oxidation. Unlike fretting and fretting wear, fretting corrosion is associated with an additional electrochemical reaction.

Whilst aircraft structures are subject to corrosion control programs, the thermodynamic tendency of materials to return to their stable state eventually leads to corrosion incidents.

A.7.6 Cavitation

Damage caused by cavitation can occur in control valves, pumps, propellers, impellers, and bends where a sudden change in the direction of fluid occurs. Cavitation is the formation and then implosion of bubbles (cavities) in a liquid. Cavitation usually occurs when the liquid is subjected to rapid reductions in pressure, which cause the formation of cavities; then the cavities are carried downstream until they reach an area of higher pressure where they collapse or implode. In addition to vaporization at low pressure, cavitation can be formed as a result of air ingestion, flow turbulence, and internal re-circulation.

Cavitation bubbles that implode on a component surface induce high local impact forces which may cause damage to the surface. Repeated implosion of cavities causes cyclic stresses, which result in surface fatigue wear. If cavitation occurs in the presence of contamination (e.g. erosive/corrosive chemicals in dirty lubricants) the surfaces may become eroded or pitted. The presence of bubbles can also cause the obstruction of flow passages and the formation of eddies giving rise to vibration and leading to loss of performance and efficiency.

The cavitation process is classified as vaporous cavitation or gaseous cavitation if the void is filled by primarily water vapour or gases respectively. The following sentences shed light on the difference between boiling and cavitation. Water boils at a temperature that reduces as the local pressure reduces. Since the pressure reduces at high altitudes, boiling near the top of the mountain Everest occurs at about 70^{o}C; boiling at higher altitudes can occur at a 20^{o}C room temperature. The change of water into vapour first occurs as localized bubbles (cavities) when the vapour pressure reaches the value of the local pressure. The subtle difference between boiling and cavitation is as follows: (a) boiling is the state transformation from liquid to vapour by changing temperature while holding the local pressure constant; (b) cavitation is the state transformation from liquid to vapour by changing the local pressure while holding the temperature constant. For example, "gaseous cavitation" can occurs when opening a bottle containing a carbonated liquid.

A.8 Reliability Centred Maintenance

Disciplined experts and well-trained engineers use unambiguous maintenance procedures to maintain airworthiness, safety, structural integrity, and adequate performance. These procedures stem from well-established maintenance philosophies. The procedures are usually updated when new matured technologies emerge. In the presence of several philosophies, the RCM logic does not necessarily rule out one philosophy or another but incorporates the best of all philosophies. For example, aircraft maintenance procedures can be compiled from preventive and corrective maintenance tasks.

RCM provides a structured framework for analyzing the functions and potential failures for a physical asset such as an aircraft with a focus on preserving system functions. RCM is used to develop scheduled maintenance plans that provide an acceptable level of operability, with an acceptable level of risk, in an efficient and cost-effective manner. RCM involves logic that answers the following questions:

- What are the functions and associated desired standards of performance of the asset in its present operating context (functions)?
- In what ways can it fail to fulfil its functions (functional failures)?
- What causes each functional failure (failure modes)?
- What happens when each failure occurs (failure effects)?
- In what way does each failure matter (failure consequences)?
- What should be done to predict or prevent each failure (proactive tasks or preventive maintenance tasks and task intervals)?
- What should be done if a suitable proactive task cannot be found (default actions)?

Thus, RCM is a method for planning maintenance and defining maintenance requirements based on the consequences (effects) of failures; the maintenance requirements include the type of maintenance required. For example, RCM can identify preventive maintenance tasks, schedule them in an optimal way, and assign corrective maintenance tasks to items that are found to be non-critical or redundant. The selection from available preventive and corrective maintenance technologies and tasks is driven by requirements for reducing costs, maintaining functionality, avoiding unacceptable loss of operational capability, restoring the target reliability, avoiding expensive repairs, and reducing lengthy downtimes. By assessing the safety and operational consequences of failure, the RCM logic can evaluate the effects of available maintenance technologies along with the effects of rework, replacement, and any other tasks undertaken to reduce risk.

A.9 Preventive Maintenance [33]

Preventive maintenance is systematic and prescribed work undertaken at predetermined intervals to reduce the probability of failure, to restore the inherent level of equipment reliability, and to ensure that performance is not degraded by time or usage. There are three types of preventive maintenance: servicing, scheduled, and condition-based maintenance.

A.9.1 Servicing

Servicing is the maintenance required to determine the condition of an aircraft or other item of equipment after a period of use and to prepare for its next period of use. It comprises the checking and replenishment of consumables and may include such minor maintenance as the replacement of bulbs and the identification of obvious signs of un-serviceability. The user or operator may carry out servicing.

A.9.2 Scheduled Maintenance

Scheduled maintenance is that preventive maintenance undertaken at regular, predetermined intervals to keep an aircraft or other item of equipment in a sound-overall condition and to minimize the amount of corrective maintenance and other day-to-day attention it requires. For aircraft, the requirements for scheduled maintenance are derived using a version of Maintenance Steering Group logic and Reliability Centred Maintenance; similar logic techniques may be applied to other equipment.

Scheduled maintenance tasks may be retained as individual tasks, allowing completion to coincide with corrective maintenance or other downtime as operations dictate. Alternatively, the individual tasks may be grouped by periodicity and allocated to the appropriate level of maintenance. This grouping defines the scheduled maintenance system, which may be based either on units of usage or calendar time and may, if appropriate, have an upper limit specified in the other unit of measure. A grouping may be divided into smaller groups or packages and equalized over the relevant part of the maintenance cycle so that the whole requirement is satisfied within the specified period.

There are a number of tried and proven maintenance systems; the principles behind these systems are summarised in the following sub-paragraphs.

A.9.2.1 Flexible Maintenance System

The individual tasks identified by the RCM analysis may be carried out individually at the identified periodicities. These tasks may also be combined into small work packages, carried out within a specified period to coincide with corrective maintenance or other downtime. Each task or work package should have sufficient latitude in its application requirements to permit flexibility in its satisfaction. Flexible maintenance activities may be satisfied and re-forecast individually when carried out on an opportunity basis, or re-calculated upon component replacement when conducting corrective maintenance. This system will tend to generate aircraft quickly, but will involve more frequent requirements for scheduled maintenance.

A.9.2.2 Grouped Maintenance System

The individual tasks identified by the RCM analysis are formally grouped into packages to be carried out at set intervals in a maintenance cycle; generically termed Low, Medium, and High frequency maintenance tasks. Each group of scheduled maintenance forms packages of work content that should broadly equate to the capabilities of the organization (Forward/Depth) responsible for carrying it out. Low frequency task groups provide, through extended downtime, an opportunity to embody time-consuming modifications or upgrades. Additional groups of scheduled maintenance may be introduced, to provide intermediate frequency groupings (legacy: RAF Primary) to meet maintenance requirements.

A.9.2.3 Equalized Maintenance System

The equalized system employs the high/medium/low frequency groupings identified at paragraph A.9.2.2 , but the maintenance is carried out progressively throughout the maintenance cycle by completing, for example, ¼ of the low frequency work group in conjunction with each of 4 medium frequency work groups. This system exchanges the protracted down-time normally associated with low frequency scheduled maintenance for slightly extended down-times during the more frequent groupings, allowing better use of resources, although reducing the opportunity to embody time-consuming modifications.

A.9.2.4 Use of Maintenance Backstops

For aircraft and airborne equipment, the periodicities of the grouped maintenance tasks described at paragraph A.9.2.2 are normally expressed in flying hours. However, within a particular maintenance schedule, there may be activities that, should the aircraft or equipment have low utilization, warrant inspection based on calendar time. For example, this may be an inspection or series of inspections for corrosion that are embedded in a flying hour-based group of activities. In such cases, a calendar upper limit may be advisable.

A.9.2.5 Out-of-Phase Maintenance

If, when using a grouped or equalized maintenance system, maintenance activities still do not align to groupings or work blocks, they may be forecast and carried out as individual activities. They are known as out-of-phase operations. For example, when the safe lives of critical components expire, out of phase replacement will be required if the maintainer is unable to defer the replacement to coincide with other maintenance activities.

A.9.3 Condition Based Maintenance (CBM)

Condition Based Maintenance is that preventive maintenance initiated as a result of knowledge of the condition of an item gained from routine or continuous monitoring. Where adequate and realistic condition monitoring techniques are available for a particular item to detect incipient failure, condition-based maintenance is applied to the item in preference to routine repair or replacement as part of scheduled or out-of-phase maintenance. Where possible, the requirement for condition monitoring of the item should be included in the relevant servicing or maintenance schedule; otherwise it is incorporated as out-of-phase maintenance. Additionally, the accuracy of the condition-monitoring technique is normally sufficiently refined to permit anticipation of the need to repair or replace the item concurrent with scheduled or other maintenance opportunities.

A.10 Corrective Maintenance

A.10.1 Corrective Maintenance of Materiel

Corrective maintenance embraces those maintenance activities carried out after a fault has occurred in order to restore an item to a serviceable state. Maintenance activities undertaken when a fault is indicated, but in the event not confirmed, are also corrective maintenance. A fault may be identified from the application of condition-monitoring techniques as part of preventive maintenance.

A.10.2 Inspect and Repair As Necessary (IRAN)

IRAN is a methodology intended to ensure that the most cost-effective corrective maintenance activities are undertaken to return an item to a condition to meet operational commitments. It is rarely necessary to return a faulty component to 'as new' condition; consideration should always be given to repairing an identified fault, rather than reconditioning whole equipments.

B Appendix B: Definitions of Technology Readiness Levels

These levels are used to assess the maturity of evolving aerospace technologies and, systematically, incorporate them into aerospace systems when they reach a high TRL. Usually, Levels 1 to 4 relate to creative, innovative technologies before or during mission assessment phase; Levels 5 to 9 relate to existing technologies and to missions in definition phase.

B.1 The ESA TRL Definitions

Table 2: The European Space Agency (ESA) definitions

Level	Description
TRL 1.	Basic principles observed and reported
TRL 2.	Technology concept and/or application formulated
TRL 3.	Analytical & experimental critical function and/or characteristic proof-of-concept
TRL 4.	Component and/or breadboard validation in laboratory environment
TRL 5.	Component and/or breadboard validation in relevant environment
TRL 6.	System/subsystem model or prototype demonstration in a relevant environment (ground or space)
TRL 7.	System prototype demonstration in a space environment
TRL 8.	Actual system completed and "Flight qualified" through test and demonstration (ground or space)
TRL 9.	Actual system "Flight proven" through successful mission operations

B.2 The NASA TRL Definitions

Table 3: The National Aeronautics and Space Administration (NASA) definitions

Level	Description
1. Basic principles observed and reported	This is the lowest "level" of technology maturation. At this level, scientific research begins to be translated into applied research and development.
2. Technology concept and/or application formulated	Once basic physical principles are observed, then at the next level of maturation, practical applications of those characteristics can be 'invented' or identified. At this level, the application is still speculative: there is not experimental proof or detailed analysis to support the conjecture.
3. Analytical and experimental critical function and/or characteristic proof of concept	At this step in the maturation process, active research and development (R&D) is initiated. This must include both analytical studies to set the technology into an appropriate context and laboratory-based studies to physically validate that the analytical predictions are correct. These studies and experiments should constitute "proof-of-concept" validation of the applications/concepts formulated at TRL 2.
4. Component and/or breadboard validation in laboratory environment	Following successful "proof-of-concept" work, basic technological elements must be integrated to establish that the "pieces" will work together to achieve concept-enabling levels of performance for a component and/or breadboard. This validation must be devised to support the concept that was formulated earlier, and should also be consistent with the requirements of potential system applications. The validation is "low-fidelity" compared to the eventual system: it could be composed of ad hoc discrete components in a laboratory.
5. Component and/or breadboard validation in relevant environment	At this level, the fidelity of the component and/or breadboard being tested has to increase significantly. The basic technological elements must be integrated with reasonably realistic supporting elements so that the total applications (component-level, sub-system level, or system-level) can be tested in a 'simulated' or somewhat realistic environment.
6. System/subsystem model or prototype demonstration in a relevant environment (ground or space)	A major step in the level of fidelity of the technology demonstration follows the completion of TRL 5. At TRL 6, a representative model or prototype system or system - which would go well beyond ad hoc, 'patch-cord' or discrete component level bread boarding - would be tested in a relevant environment. At this level, if the only 'relevant environment' is the environment of space, then, the model/prototype must be demonstrated in space.

Level	Description
7. System prototype demonstration in a space environment	TRL 7 is a significant step beyond TRL 6, requiring an actual system prototype demonstration in a space environment. The prototype should be near or at the scale of the planned operational system and the demonstration must take place in space.
8. Actual system completed and 'flight qualified' through test and demonstration (ground or space)	In almost all cases, this level is the end of true 'system development' for most technology elements. This might include integration of new technology into an existing system.
9. Actual system 'flight proven' through successful mission operations	In almost all cases, the end of last 'bug fixing' aspects of true 'system development'. This might include integration of new technology into an existing system. This TRL does not include planned product improvement of ongoing or reusable systems.

B.3 The DOD TRL Definition

Table 4: The Department of Defense (DOD) definitions

Level	Description
1. Basic principles observed and reported	Lowest level of technology readiness. Scientific research begins to be translated into applied research and development. Examples might include paper studies of a technology's basic properties.
2. Technology concept and/or application formulated	Invention begins. Once basic principles are observed, practical applications can be invented. Applications are speculative, and there may be no proof or detailed analysis to support the assumptions. Examples are limited to analytic studies.
3. Analytical and experimental critical function and/or characteristic proof of concept	Active research and development is initiated. This includes analytical studies and laboratory studies to physically validate analytical predictions of separate elements of the technology. Examples include components that are not yet integrated or representative.
4. Component and/or breadboard validation in laboratory environment	Basic technological components are integrated to establish that the pieces will work together. This is "low fidelity" compared to the eventual system. Examples include integration of 'ad hoc' hardware in a laboratory.
5. Component and/or breadboard validation in relevant environment	Fidelity of breadboard technology increases significantly. The basic technological components are integrated with reasonably realistic supporting elements so that the technology can be tested in a simulated environment. Examples include 'high fidelity' laboratory integration of components.
6. System/subsystem model or prototype demonstration in a relevant environment	Representative model or prototype system, which is well beyond the breadboard tested for TRL 5, is tested in a relevant environment. Represents a major step up in a technology's demonstrated readiness. Examples include testing a prototype in a high fidelity laboratory environment or in simulated operational environment.
7. System prototype demonstration in an operational environment	Prototype near or at planned operational system. Represents a major step up from TRL 6, requiring the demonstration of an actual system prototype in an operational environment, such as in an aircraft, vehicle or space. Examples include testing the prototype in a test bed aircraft.
8. Actual system completed and 'flight qualified' through test and demonstration	Technology has been proven to work in its final form and under expected conditions. In almost all cases, this TRL represents the end of true system development. Examples include developmental test and evaluation of the system in its intended weapon system to determine if it meets design specifications.
9. Actual system 'flight proven' through successful mission operations	Actual application of the technology in its final form and under mission conditions, such as those encountered in operational test and evaluation. In almost all cases, this is the end of the last "bug fixing" aspects of true system development. Examples include using the system under operational mission conditions.

C Appendix C: Quality Characteristics of SHM Measurements & Functions

C.1 General

The measuring capabilities of the majority of SHM systems are unlike those of direct measuring devices that measure quantities such as temperature and acceleration. The SHM measuring capability can involve specialized algorithms operating on data acquired from more than one sensor. The SHM measuring capability should provide reliable results under all the foreseeable operational conditions of the SHM system. Therefore, the performance of the SHM sensors/algorithms should be evaluated in terms of quality attributes adequate for each associated intended elementary function: the values of these quality attributes should not be too high or too low; they should be values consistent with the intended elementary function, the function intended use and its assigned DAL level. For example, the measurements of an advisory non-critical system may not need to be as accurate as those of a safety critical system. For each SHM elementary function, the validation process should check the presence of quantitative values of quality attributes that adequately characterize the SHM sensor/algorithm measurements. Therefore, the following subsections present widely used measurement characteristics and discuss their interpretations or extensions for SHM measurements. Requirement allocation and validation activities should identify, from these characteristics, those characteristics that are applicable to the SHM application under consideration; not all the characteristics are necessarily required for each SHM application.

C.2 Accuracy

The accuracy is the degree of closeness of agreement between a measured quantity value and a true quantity value of a measurand. The true quantity value is obtained by a device that has been widely accepted as being accurate with high degree of confidence. The accuracy may be quantified by the differences between the true value and the two extremes of corresponding observed measured values (a maximum and a minimum values). In simple terms, the accuracy of a measured quantity value is the degree to which it is true and free from error; the accuracy can be measured as the difference between the measured quantity value and an accepted quantity value. The accuracy determination for SHM measurements should include sufficient quantitative measures that relate to the intended function. For example, these quantitative measures can indicate: (a) accuracy of a crack length, damage size, or computed load values (b) accuracy of an identified zone containing structural damage.

C.3 Reliability

The reliability is measured by the probability of repeatedly and successfully observing a desirable outcome from an entity under prescribed conditions. The entity can be any observable subject such as structure, system, sensor, mission, or event. In other words, the reliability is measured by the probability of the success, failure-free, desirable performance of the entity. A common example of a desirable outcome as sited in system engineering literature is the ability of a system or component to perform a required function under stated conditions for a specified period. Therefore, for SHM systems, reliability can involve evaluating the probability of successfully delivering an intended function under specified environmental conditions for a specified period (e.g. the maintenance free period of the system); for example, the reliability of a crack detection system can involve evaluating the probability of successfully detecting cracks having lengths greater than a specified minimum under specified environmental conditions for a specified period.

The reliability of an entity can be evaluated from the following equation: Reliability = Probability of Success = 1.0 - Probability of Failure. If the probability of failure of a structural component or a system is 1/100000 flying hours (one in a million chance of failure), the reliability of the structure/system will be 0.999999 (a six-nine reliability).

Reliability can indicate the quality of detecting the occurrence of structural events such as hard-landings or detecting the presence of faults by computing three probabilities: the probability of correct identification of faults/events, the probability of missing the faults/events (the probability of false negative), and the probability of indicating the presence of faults/events that did not exist (the probability of false positive). An increase in the probability of false negatives of a SHM system may lead to an increase in the failure risk of the structural components monitored by the SHM system and, hence, may lead to a potential reduction in the reliability of the structural components. An increase in the probability of false positives triggered by a SHM system may lead to a reduction in the failure risk of the structural components and, hence, may lead to a potential improvement in the reliability at unjustifiable additional costs associated with "no-fault-found" inspections.

For structural NDI applications, an estimate of the Probability of Detection (POD) is used to indicate the capability of the NDI applications: tests are performed to establish POD curves that indicate the probabilities and confidence levels associated with detecting different damage sizes; often, the capability of NDI is measured by the smallest damage size "a_s" found with 90% probability at a 95% confidence level (detecting "a_s" with "90/95" capability). Having established the capability of NDI applications, procedures are adopted to avoid the use of degraded capability and ensure measurement repeatability to assure reliability. The quality terms used to characterize NDI applications should be used for SHM systems performing the same tasks as the NDI applications. For more information refer to MIL-HDBK-1823.

C.4 Precision

The precision of a measurement system is the degree to which repeated measurements under specified conditions show the same results; precision is usually expressed numerically by a measure of imprecision such as the standard deviation of the repeated measurements. A precise device can be inaccurate if the repeated measurements of the same quantity are very close to each other but significantly differ from the accepted true value. A formal definition of precision is the closeness of agreement between indications or measured quantity values obtained by replicate measurements on the same or similar objects under specified conditions. Measurement precision is used to define measurement repeatability and is not the same as mathematical precision; the latter is defined as the number of digits used to perform a given computation; precise computation can be inaccurate.

The SHM measurements should be precise (repeatable) and reliable (high success rate/low failure rate) under all foreseeable measurement conditions. The degree of precision and reliability would depend on the DAL assignments, the intended functions, and how the intended functions is used.

C.5 Measurement Repeatability

The term repeatability indicates measurement precision under a set of repeatability conditions of measurement. A repeatability condition is a condition out of a set of conditions that include the same measurement procedure, same operators, same measuring system, same operating conditions, same location, and replicate measurements on the same or similar objects over a short period of time.

C.6 Resolution

The resolution is the smallest change in a quantity being measured that causes a perceptible change in the corresponding indication. The resolution of a displaying device is the smallest difference between displayed indications that can be meaningfully distinguished. The resolution of an ADC, which converts sensed analogue signals to digits, indicates the number of discrete values it can produce over the range of analogue values. An ADC with a resolution of 8 bits (bit is an abbreviation for binary digit) can encode an analogue input to one in 2^8 different levels; these levels can represent the input range from 0 to 255 (unsigned integers) or from -128 to 127 (signed integers).

C.7 Range, Dynamic Range, and Bandwidth

The range of a sensor input is the maximum and minimum values of an applied parameter that can be measured. The range of a sensor output is the maximum and minimum values of the sensor response to the applied parameter. The dynamic range is the difference between the range maximum and minimum values. The digital band width is the rate of data transfer, measured in values per second or, bit rate or throughput measured in bits per second.

C.8 Sensitivity

The Sensitivity is the quotient of the change in the indication of a measuring system and the corresponding change in the value of a quantity being measured. The sensitivity of the SHM system should be carefully chosen and should not be too high: consider a linear SHM system that measures crack lengths ranging from 0 to 20mm with indications (outputs) that cannot exceed 10 Volts; a system having high sensitivity of 10/10 Volts/mm will be saturated for all cracks having actual lengths between 10 and 20mm and will give a wrong indication of 10 Volts for all these lengths; a system with 10/40 Volts/mm will give correct indications from 0 to 5 Volts for crack lengths from 0 to 20mm, and can also indicate crack lengths up to 40mm.

Furthermore, the sensitivity of SHM can be susceptible to factors such as environmental condition; a quantity value measured by a SHM system may deviate from the true value not because the accuracy but because changes in environmental conditions. Therefore, it is important to evaluate and minimize the effect of factors such as (a) typical changes in environmental parameters such as temperature and humidity, (b) resolution and accuracy of input parameters to SHM algorithms, and (c) calibration and re-calibration methods/skills.

C.9 Durability

The durability determination should include sufficient quantitative measures covering for example: (a) Mean Time between Failure (MTBF) and (b) the fatigue life or the damage-tolerance service period of the SHM system and its items. MTBF is usually expressed in hours as the mean value of the lengths of time between consecutive failures under stated conditions for a stated period in the life of a functional unit; MTBF indicates the average time over which a device will function before failing under a set of expected conditions.

C.10 Maintainability

The qualitative measures for maintainability should include the information that enables the evaluation of maintenance intervals for the SHM system; they should also indicate the efforts and costs required for system maintenance, repair, and/or replacement.

REFERENCES

[1] FAA (2012) "Certification Procedures for Products and Parts", Electronic Code of Federal Regulation, Title 14, Part 21 as of August 2012.

[2] FAA (2012) "Airworthiness Standards: Normal, Utility, Acrobatic, and Commuter Category Airplanes", Electronic Code of Federal Regulation, Title 14, Part 23 as of August 2012.

[3] FAA (2012) "Airworthiness Standards: Transport Category Airplanes", Electronic Code of Federal Regulation, Title 14, Part 25 as of August 2012.

[4] FAA (2012) "Airworthiness Standards: Normal Category Rotorcraft", Electronic Code of Federal Regulation, Title 14, Part 27 as of August 2012.

[5] FAA (2012) "Airworthiness Standards: Transport Category Rotorcraft", Electronic Code of Federal Regulation, Title 14, Part 29 as of August 2012.

[6] FAA (2012) "Continued Airworthiness and Safety Improvements for Transport Category Airplanes", Electronic Code of Federal Regulation, Title 14, Part 26 as of August 2012.

[7] FAA (2012) "Airworthiness Directives" Electronic Code of Federal Regulation, Title 14, Part 39 as of August 2012.

[8] FAA (2012) "Maintenance, Preventive Maintenance, Rebuilding and Alteration", Electronic Code of Federal Regulation, Title 14, Part 43 as of August 2012

[9] FAA (2012) "General Operating and Flight Rules", Electronic Code of Federal Regulation, Title 14, Part 91 as of August 2012.

[10] FAA (2012) "Operating Requirements: Domestic, Flag, and Supplemental Operations", Electronic Code of Federal Regulation, Title 14, Part 121 as of August 2012.

[11] FAA (2004) "The FAA and Industry Guide to Product Certification", Second Edition, Prepared by AIA, GAMA and FAA Aircraft Certification Service, September 2004.

[12] FAA (2010) "Maintenance Review Board Report, Maintenance Type Board, and OEM/TCH Inspection Program Procedures", FAA Advisory Circular, AC No: 121-22B, 29th October 2010.

[13] FAA (2011) "Damage Tolerance and Fatigue Evaluation of Structure", FAA Advisory Circular, AC No: 25.571-1-D, 13 January 2011.

[14] FAA (2011) "Fatigue Management Programs for In-Service Issues", FAA Advisory Circular, AC No: 91-82A, 23 August 2011.

[15] FAA(2011) "Certification Maintenance Requirements", FAA Advisory Circular, AC No: 25-19, 3 August 2011.

[16] FAA (2007) "Instruction for Continued Airworthiness of Structural Repairs on Transport Airplanes", AC No: 25.1529-1A, 20 November 2007.

[17] EASA (2003) "Certification of Aircraft and Related Products, Parts and Appliances, and of Design and Production Organizations", EASA Regulation (EC) No 748/2012, Annex I, Part 21, 21 August 2012.

[18] EASA (2010) "Certification Specifications for Normal, Utility, Aerobatic, and Commuter Category Aeroplanes", European Aviation Safety Agency, CS-23, Amendment 2, as of 28 September 2010.

[19] EASA (2011) "Certification Specifications and Acceptable Means of Compliance for Large Aeroplanes", European Aviation Safety Agency, CS-25, Amendment 11, as of 4 July 2011.

[20] EASA (2008) "Certification Specifications for Small Rotorcraft", European Aviation Safety Agency, CS-27, Amendment 2, as of 17 November 2008.

[21] EASA (2008) "Certification Specifications for Large Rotorcraft", European Aviation Safety Agency, CS-29, Amendment 2, as of 17 November 2008.

[22] CARs (2012) "Canadian Aviation Regulations (CARs) 2012-1, Part I - General Provisions, Subpart 1 – Interpretation" (CARs 101.01).

[23] CARs (2012) "Canadian Aviation Regulations (CARs) 2012-1, Part V - Airworthiness, Subpart 21 - Approval of the Type Design or a Change to the Type Design of an Aeronautical Product, Division IV - Changes to a Type Design" (CARs 521.151 to 521.161).

[24] ATA (2009) "Operator/Manufacturer Scheduled Maintenance Development", ATA MSG-3, Revision 2009.1, Air Transport Association, 2009.

[25] SAE (1996) "Certification Considerations for Highly-Integrated or Complex Aircraft Systems", SAE ARP4754, Issuing Committee, S-18, Aircraft And Sys Dev And Safety Assessment Committee, November 1996.

[26] SAE (2010) "Guidelines for Development of Civil Aircraft and Systems", SAE ARP4754 Rev. A, December 2010. (Supersede ARP4754).

[27] SAE (1996) "Guidelines and Methods for Conducting the Safety Assessment Process on Civil Airborne Systems and Equipment", SAE ARP4761, December 1996.

[28] SAE (2003) "Safety Assessment of Transport Airplanes in Commercial Service", SAE ARP5150, November 2003.

[29] The UK MOD (2014) "Military Aviation Authority Regulatory Policy", Military Aviation Authority (MAA), MAA01, Issue 3, December 2014.

[30] The UK MOD (2014) "Military Aviation Authority Master Glossary", Military Aviation Authority (MAA), MAA02, Issue 5, December 2014.

[31] The UK MOD (2014) "Military Aviation Authority Regulatory Processes", Military Aviation Authority (MAA), MAA03, Issue 3, December 2014.

[32] The UK MOD (2014) "GEN 1000 Series Regulatory Articles", Military Aviation Authority (MAA), 9 December 2014.

[33] The UK MOD (2014) "Manual of Maintenance and Airworthiness Processes", Military Aviation Authority (MAA), CAE 4000 - MAP 01, Issue 6, 17 December 2014.

[34] The UK MOD (2014) "Manual of Maintenance and Airworthiness Processes Supplement - MOD Form 700 Series of Forms", Military Aviation Authority (MAA), CAE 4000 - MAP 02, Issue 6.3, 17 December 2014.

[35] The UK MOD (2014) "Continuing Airworthiness Engineering Regulations (CAE) 4000 Series Regulatory Articles", Military Aviation Authority (MAA), 9 December 2014.

[36] The UK MOD (2014) "Design and Modification Engineering (DME) 5000 Series Regulatory Articles", Military Aviation Authority (MAA), 1 Dec 2014.

[37] The UK MOD (2014) "RA 1500 – Certification of UK Military Registered Air Systems", "GEN 1000 Series Regulatory Articles", Military Aviation Authority (MAA), 9 December 2014.

[38] The UK MOD (2014) Def Stan 00-970 Part 0: "Design and Airworthiness Requirements for Service Aircraft, Part 0: Procedures for Use, Content and Definitions", the UK Ministry of Defence, Issue 14, 30 September 2014.

[39] The UK MOD (2014) Def Stan 00-970 Part 1: "Design and Airworthiness Requirements for Service Aircraft, Part 1: Fixed Wing, Section 3: Structure", the UK Ministry of Defence, Issue 13, July 2014.

[40] The UK MOD (2014) Def Stan 00-970 Part 7: "Design and Airworthiness Requirements for

Service Aircraft, Part 7: Rotorcraft, Section 2 Supplement C: Strength Requirements (Subpart C)", the UK Ministry of Defence, Issue 4, September 2014.

[41] The UK MOD (2014) Def Stan 00-970 Part 7: "Design and Airworthiness Requirements for Service Aircraft, Part 7: Rotorcraft, Section 2 Supplement D: Design and Construction (Subpart D)", the UK Ministry of Defence, Issue 4, September 2014.

[42] The UK MOD (2014) Def Stan 00-970 Part 9: "Design and Airworthiness Requirements for Service Aircraft, Part 9: Remotely Piloted Air Systems", the UK Ministry of Defence, Issue 11, July 2014

[43] The UK MOD (2014) Def Stan 00-56: "Safety Management Requirements for Defence Systems Part 1: Requirements and Guidance", Issue 5, 21 February 2014.

[44] The UK MOD (2014) Def Stan 00-45 Part 1: "Using Reliability Centred Maintenance to Manage Engineering Failures Part No: 1, Requirements for the application of Reliability Centred Maintenance" Issue 3, 28 November 2012.

[45] The UK MOD (2014) Def Stan 00-45 Part 2: "Using Reliability Centred Maintenance to Manage Engineering Failures Part No: 2: Developing an RCM Project Plan", Issue 3, 28 November 2012

[46] The UK MOD (2014) Def Stan 00-45 Part 3: "Using Reliability Centred Maintenance to Manage Engineering Failures Part No: 3: Guidance on the Application of Reliability Centred Maintenance", Issue 2 dated: 28 November 2012

[47] The UK MOD "Defence Environment and Safety Management", Joint Services Publication, JSP 815, February 2009.

[48] The UK MOD "Defence Logistics Support Chain Manual, Supportability Engineering, Integrated Logistic Support Management", Joint Services Publication, JSP 886, Volume 7, Part 2, June 2013

[49] The UK MOD "Defence Logistics Support Chain Manual, Supportability Engineering, Maintenance Planning", Joint Services Publication, JSP 886, Volume 7, Part 8.03A, August 2013.

[50] The UK MAA "Military Aviation Engineering, Preparation and Amendment of Maintenance Schedules", Joint Air Publication, JAP(D) 100C-20, August 2012.

[51] The UK MAA "Military Aviation Engineering, Guide to Developing and Sustaining Preventive Maintenance Programmes", Joint Air Publication, JAP(D) 100C-22, August 2012.

[52] The US DOD "Joint Service Specification Guide, Aircraft Structure", JSSG-2006, Department of Defense, 30 October 1998.

[53] USAF "Aircraft Structural Integrity Program," MIL-HDBK-1530A, US Air force (USAF).

[54] MIL-STD-810 "Environmental Engineering Considerations and Laboratory Tests".

[55] MIL-STD-461 "Requirements for the Control of Electromagnetic Interference Characteristics of Subsystems and Equipment".

[56] Deputy Under Secretary of Defense for Logistics and Materiel Readiness (2008) "Condition Based Maintenance Plus DoD Guidebook," May 2008.

[57] The US DOD "Nondestructive Evaluation System Reliability Assessment", MIL-HDBK-1823A, April 2009.

[58] The US Naval Air (NAVAIR) Systems Command "Guidelines for the Naval Aviation Reliability-Centered Maintenance Process", NAVAIR 00-25-403, July 2005

[59] The US Naval Sea (NAVSEA) Systems Command "Reliability-Centered Maintenance (RCM) Handbook", S9081-AB-GIB-010, Revision 1, April 2007.

[60] The NATO Standardization Agreement (STANAG) 4671, "Unmanned Aerial Vehicles System Airworthiness Requirements (USAR)", Edition 1, 3 September 2009.

[61] Peter Foote et al (2006) “Structural Health Monitoring Road Mapping Process”, Aerospace Vehicle Systems Institute (AVSI), August 29, 2006.

[62] ASTM International, “Standard Practices for Cycle Counting in Fatigue Analysis”, American Society for Testing and Materials, ASTM E1049, 1985.

[63] K B Amer (1988) “A new" philosophy of structural reliability, fail safe versus safe life”, 44th Annual National Forum of the American Helicopter Society, 1988.

[64] J P Gallagher, L M Butkus, J C Brausch, C A Babish, J C Malas and A P Berens (2007) “Demonstrating the Effectiveness of an Inspection System to Detect Cracks in Safety of Flight Structure,” Aging Aircraft Conference 2007, Palm Springs, CA.

[65] Hesham Azzam and Peter Knight (2008) “Rotor System Diagnostics and HUMS Extensions for Affordable CBM,” Presented at the American Helicopter Society 64th Annual Forum, Montréal, Canada, April 29 – May 1, 2008.

[66] Hesham Azzam (2011) “Development, Validation, Qualification and Certification of Structural Health Monitoring Systems,” HAHN Spring Report 1/B002, Jun-11

[67] Hesham Azzam and Jim McFeat (2011) “A Validation Methodology for Structural Health Management,” SAE International Paper 2011-01-2608, Presented in SAE 2011 AeroTech Congress and Exhibition, Toulouse, France, October 2011.

[68] Alan P. Berens (1988) “NDE Reliability Data Analysis”, Metals Handbook,”Volume 17, 9th Edition: Nondestructive Evaluation and Quality Control, 1988.

[69] Hesham Azzam (2012) “Development, Validation, Verification, Certification of Structural Health Monitoring Systems (SHM) For Military Aircraft, Progress Report 3,” HAHNS/R007/B003, 1st September 2012.

[70] Berens, Hovey, Donahue, and Craport (1988) “User’s Manual for Probability of Detection,” University of Dayton Research Institute, UDR-TR-88-12, January 1988.

[71] Hesham Azzam. (1997) “A Practical Approach for the Indirect Prediction of Structural Fatigue from Measured Flight Parameters,” Proceedings of the Institution of Mechanical Engineers, Part G: Journal of Aerospace Engineering, Vol. 211 No. G1, pp 29-38, 1997.