The Impact of Composites to Aircraft Structural Integrity Management

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Publication Date:
2013

DOI:
https://doi.org/10.26190/unsworks/16566

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The Impact of Composites to Aircraft Structural Integrity Management

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A thesis in fulfillment of the requirements for the degree of

Masters by Research

July 2013

School of Engineering & Information Technology

Faculty of UNSW Canberra at ADFA

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Nomenclature and Terms Used

ε	Strain
σ	Stress or Applied Load
σ_r	Residual strength
σ_{fs}	Static strength
Φ	Relative humidity (in %)
AASA	Aging Aircraft Safety Act
AC	Advisory Circular
ACC	Airworthiness Certification Criteria
ACEE	Advanced Composites Energy Efficiency
ACI	Analytical Condition Inspection
AD	Accidental Damage
ADF	Australian Defence Force
AFHR	Airframe Hour
AFHRS	Airframe Hours
ASIAC	Aerospace Structures Information and Analysis Center
ASIP	Aircraft Structural Integrity Program
ASIM	Aircraft Structural Integrity Management
ASNT	American Society for Nondestructive TEsting
ATA	Air Transport Association
ATSB	Australian Transportation Safety Bureau
BBA	Building Block Approach
BVID	Barely Visible Impact Damage
CACRC	Composite Aircraft Composite Repair Committee
CPAB	Corrosion Prevention Advisory Board
DDP	Detail Design Point
DoD	Department of Defence
DSL	Design Service Life
E	Young's Modulus
ED	Environmental Damage
EDS	Evolving Discontinuity State
ELOS	Equivalent Level of Safety
FAA	Federal Aviation Administration
FAR	Federal Aviation Regulations
FD	Fatigue Damage
FOD	Foreign Object Damage

FoS	Factor of Safety
G_{IIC}	Dynamic mode II interlaminar fracture toughness
GAO	Government Accountability Office
GFRP	Glass Fibre Reinforced Polymer
GLARE	Glass Laminate Aluminium Reinforced Epoxy
H(2L)	Exceedance Intensity
HOLSIP	Holistic Structural Integrity Process
IAT	Individual Aircraft Tracking
ICA	Instructions for Continued Airworthiness
IDS	Initial Discontinuity State
II	Inspection Interval
JAL	Japan Air Lines
JAMS	Joint Advanced Materials and Structures
JSSG	Joint Service Specification Guide
L	Total Life of a Structure
L1	Nucleation
L2	Small Crack Growth
L3	Stress Dominated Crack Growth
L4	Failure (Fracture)
L/ESS	Loads/Environment Spectra Survey
LOV	Limit of Validity
M(t)	Moisture content
M_i	Initial moisture content
M_m	Maximum moisture content
MCDASP	Material Characterisation and Design Allowables Substantiation Plan
MDS	Modified Discontinuity State
MED	Multiple Element Damage
MMC	Metal Matrix Composite
MSD	Multiple Site Damage
n	Current number of cycles
N	Number of cycles to failure for the applied load σ.
NACA	National Advisory Committee on Aeronautics
NASA	National Aeronautics and Space Administration
NDI	Non-Destructive Inspection
NTSB	National Transport Safety Board
OEM	Original Equipment Manufacturer
p(100)	Probability of occurrence of a 100 ft-lb or higher energy
P(D|R)	Probability of safe design given safe regulation
P(DIMOR)	Probability of safe design and safe

	manufacture and safe inspection and safe operation and safe regulation
P(I|MRD)	Probability of safe inspection given safe manufacture, regulation and design
P(M|RD)	Probability of safe manufacture given safe regulation and design
P(O|MIRD)	Probability of safe operation given safe manufacture, inspection, regulation and design
P(R)	Probability of Safe Regulation
P(S)	Probability of Safe Structure
P_{xx}	Maximum average internal member load (without clipping) that will occur once in M times the interval inspection
PDF	Probability Density Function
PoD	Probability of Detection
PSE	Principle Structural Element
PSRA	Probabilistic Structural Reliability Assessment
RATO	Rocket Assisted Take-Off
RRS	Reduction in Residual Strength
RS	Residual Strength
S_D	Probability of Safe Design
S_I	Probability of Safe Maintenance
S_M	Probability of Safe Manufacturing
S_O	Probability of Safe Operation
S_R	Probability of Safe Regulation
S_T	Probability of Safe Structure
S-N	Stress-Life
SC	Special Condition
SIFCM	Structural Integrity Failure Causation Model
SIFCMsim	SIFCM simulation
SoI	Structure of Interest
SPC	Statistical Process Control
SSI	Structurally Significant Item
SSOR	Strength Summary and Operational Restrictions
t	Time
T	Temperature
T_{creep}	Temperature at which creep occurs
T_g	Glass transition temperature
T_m	Melting temperature
T_S	Time Step
TCDS	Type Certification Data Sheet
TTF	Time To Failure
USAF	United States Air Force

USN	United States Navy
UV	Ultraviolet
VGH	Velocity, normal acceleration and height
WFD	Widespread Fatigue Damage
WRSLJ	Wing Root Step-Lap Joint
Xm	Representative Modal Impact Energy Level
z	Moisture proportionality variable

Acknowledgements

I would like to thank my wife, Lisa, for tolerating my need to work on my research most weekends and holidays.

Publications arising from the development of this Thesis

The following publications were written during the course of development of this Thesis:

Warren, A., Heslehurst, R. & Wilson, E. (2011). *Evolution of Aircraft Structures and Integrity Management*. Paper presented at the meeting of 2011 Aircraft Airworthiness & Sustainment Conference, 26-28 July, Brisbane, Australia.

Warren, A. (2012). *Structural Integrity Regulation and Models*. Paper presented at the meeting of the 2012 Composites Australia and CRC for Advanced Composite Structures Conference - 'Diversity in Composites'.

Warren, A., Heslehurst, R. & Wilson, E. (2012). *Simulation of Structural Integrity*. Paper presented at SAMPE Tech 2012, 22-25 October, Charleston, SC.

Warren, A., Heslehurst, R. & Wilson, E. (2013). *Composites and MIL-STD-1530C*. Paper accepted for publication in International Journal of Structural Integrity.

1 - INTRODUCTION

Aircraft structural integrity management programs are aimed at ensuring that aircraft can operate safety and economically throughout their life. As typified by MIL-STD-1530, this is achieved by:

a. Establishing and validating the structural integrity of aircraft structures;

b. Using operational data to update the status of the structural integrity;

c. Providing quantitative data to support decisions related to conduct of inspections and priority for modification; and

d. Providing lesson-learnt to be applied to the next generation of aircraft design/modifications.

The current aircraft structural integrity management programs are primarily related to metallic structures. This is illustrated by Wanhill (2002), which identifies four aircraft incidents, which are considered milestones in aircraft structure integrity, all of which involve metallic structures. The incidents and the associated impact on aircraft structural integrity were:

a. 1954 - de Havilland Comet, loss of two aircraft due to fuselage failure. Impact: awareness that finite fatigue life is important.

b. 1969 - General Dynamics F-111 wing failure due to undetected material flaw, which initiated the development of the damage tolerance concept.

c. 1977 - Dan Air Boeing 707 loss of tailplane due to fatigue failure of spar, resulting in the realisation that old aircraft become more fatigue critical.

d. 1988 - Aloha Airlines Boeing 737 loss of large part of fuselage due to multiple fatigue cracks, which highlighted the issue of Multiple Site fatigue Damage (MSD) in aging aircraft.

The use of composites in aircraft construction has increased steadily since being introduced in the 1960s and 70s. Modern combat aircraft, such as the F-22 and F-35 have approximately thirty percent by weight of aircraft structure constructed from composites. This percentage is even higher for large commercial aircraft, such as the Boeing 787 Dreamliner, which has a structure, comprised of approximately fifty percent composites by weight. This increase in structural usage combined with the decrease in design safety factors resulting from the desire to optimise aircraft structure to minimise weight, raises the question of whether current aircraft structural integrity methodologies are appropriate for aircraft structures built largely from composites.

1.1 - What is a composite?

Composites (or composite system) is a combination of two or more different materials distinct from each other at the micro and macroscopic level, with fibres of one material embedded within a matrix of another material. Composites can be composed of a combination of the three material classes (metals, polymers and ceramic/glass). Example combinations include:

a. Polymer/Polymer: Carbon fibre/epoxy composite systems, with the carbon fibres being manufactured by a process involving the carbonisation of polymer precursor.
b. Polymer/Glass: Glass fibre/epoxy composite systems.
c. Metal/Polymer: Metal Matrix Composites (MMC) such as aluminium matrix and boron fibre composites.
d. More complicated composite systems, which include more than two different materials, exist. One such composite is GLARE (Glass Laminate Aluminium Reinforced Epoxy) as used on the Airbus A380 aircraft, which includes glass fibre composite, aluminium and epoxy within the one composite panel.

Composite systems used in aircraft structures have utilised fibres composed of glass, boron, aramid and carbon fibre within a polymer (epoxy) matrix. Dependent on the balance between material and manufacturing costs, operational environment and loading other composite systems could be used, such as, thermoplastic or metal matrix.

1.2 - Aim of Research

The aim of this research is to identify any changes required to be made to Aircraft Structural Integrity Programs to account for the increasing use of composites for aircraft structures.

1.3 - Scope of Research

The scope of the research conducted includes a literature review of the evolution of aircraft structural integrity, followed by discussion regarding the degradation models, which occur for composite materials and concluding with the development of a methodology for simulating structural integrity of composite structures. This methodology (titled SIFCM) was used to identify any changes required to be made to Aircraft Structural Integrity Programs.

2 - STRUCTURAL INTEGRITY MANAGEMENT

This chapter provides an overview of the evolution of Aircraft Structural Integrity with the intent of identifying underlying assumptions impacting the application of the military and civil Aircraft Structural Integrity requirements described at the end of this chapter.

2.1 - Evolution of Aircraft Structural Integrity

This section details a number of milestones in the evolution of Aircraft Structural Integrity (ASI) expanding on the milestones identified by Wanhill (2002):

a. de Havilland Comet accidents;

b. General Dynamics F-111 accident;

c. DAN Air Boeing 707 accident, and

d. Aloha Airlines Boeing 737 accident.

2.1.1 - In the Beginning

The Wright B Flyer was designed to provide pilot training and perform reconnaissance for the United States Army Signal Corps. The acquisition contract for the Flyer, dated 8th February, 1908 (United States Army Signal Corps, 1908), stated that the aircraft was to be designed to comply with Signal Corps Specification No. 486 (United States Army Signal Corps, 1907). This specification detailed the performance and design requirements for the aircraft and did not contain any structural integrity requirements beyond the aircraft being required to complete three speed test flights and three endurance (one hour duration) test flights prior to aircraft acceptance.

The philosophy used for structural integrity was the use of over-sized aircraft structures or more technically, the application of a Factor of Safety (FoS). As described by

Aerospace Structures Information and Analysis Center (ASIAC) (1980), the FoS accounted for:

a. Uncertainties in loads;

b. Inaccuracies in structural analysis;

c. Variations in strength properties of materials;

d. Deterioration during service life, and

e. Variations in build standard (quality).

This approach (Structural Strength philosophy) was typified by the United States Civil Air Regulations (1937), which included regulations associated with ultimate and yield FoS and requirements for structural proof testing, with no reference to structural fatigue.

During this period the average age of aircraft was quite short, thus limiting aircraft structure exposure to fatigue and environmental damage. For example, during the period from 1916 to 1920, the average design age[1] of aircraft operated by the United States air force was only 1.7 years (Ramey and Keating, 2009).

2.1.2 - The Post-War Years

After World War 2, the average age of aircraft increased steadily, as described by Ramey and Keating (2009), the lowest average design age of aircraft operated by the United States Air Force (USAF) post-World War 2 was during 1945 with an average age of approximately 3.9 years. Thus, the exposure of aircraft structures to fatigue and environmental damage has increased accordingly.

As this exposure increased, research into the fatigue behavior of aircraft structures intensified as a result of numerous aircraft accidents. Molent (2005) describes fatigue testing performed at Fisherman's Bend in Australia during the post-War period, which

[1] Design age is the timeframe between the first example of an aircraft design entering service and the last example leaving service.

included the development of the first stress-life (S-N) diagram for a full scale fabricated structure (CA-12 Boomerang wings). The presentation of a paper by Wills (1949) describing a methodology for estimating the life of aircraft structures, identified by Molent (2005) as being the basis of current lifing methods.

Thus, the transition to the Safe Life philosophy had begun, though it would take some time to complete. Safe-Life was introduced into the United States Civil Air Regulations (predecessor to the Federal Aviation Regulations) via Amendment 4b-3 (1956).

2.1.3 – 1954 – de Havilland Comet Accidents

The design for the de Havilland Comet commenced in 1946, with the first flight of the first production aircraft conducted on the 9th of January, 1951 (Cacutt, 1989).

On the 10th January 1954, whilst en-route from Rome to London, BOAC Comet G-ALYP broke up in-flight. At the time of the accident, the aircraft was 3 years old and had accumulated 3,681 Airframe Hours (AFHRS) and approximately 1,200 flight cycles (Job, 1994). After eleven weeks, BOAC recommenced Comet services as the investigation had not revealed any definitive explanation. However, on 8th April 1954, BOAC Comet G-ALYY went missing whilst en-route from Rome to Cairo.

So began one of the most intense and costly accident investigations in the history of aviation (Job 1994), with the outcome that both aircraft had broken up in flight due to fatigue cracks originating from cut-outs in the fuselage. During the investigation, a full-scale fatigue test was performed which identified that fatigue failure occurred after only 9,000 equivalent AFHRS even though during development fatigue testing at twice the cabin differential pressure demonstrated a fatigue life of at least 18,000 flight cycles (or approximately 55,000AFHR).[2]

[2] This value of AHFRS was determined by using the ratio of AFHRS to flight cycles identified for G-ALYP.

The Comet accidents highlighted the importance of finite fatigue life and problems with the Safe Life methodology, as typified by fatigue cracking that can occur earlier than expected due to errors in fatigue analysis (Wanhill, 2002). Thus, the civil aviation industry transitioned to the Fail-Safe philosophy. Fail-Safe was introduced into the United States Civil Air Regulations via Amendment 4b-3 (1956).

2.1.4 – 1958 - Boeing B-47 Accidents

As described by ASIAC (1980), the Boeing B-47 was introduced into USAF service in 1951 as a part of the United States nuclear bomber fleet. To ensure structural integrity, the aircraft was designed with a FoS of 1.5 and was subjected to static test and a limited load survey as part of its certification. Any structural integrity related problems encountered whilst in-service, were solved via expedited investigation and retrofit programs. The B-47 was, therefore, designed and certified using the Structural Strength methodology as the lessons from the Comet accidents had not been incorporated into its design or implemented in management practices.

The structural integrity management for the B-47 appeared to be adequate, even though the aircraft was:

a. Being operated in different missions and mission mix than original conceived during design;
b. Had undergone increases in engine thrust, and
c. Regularly performing Rocket Assisted Take-Off (RATO), which were not included as part of the initial design.

However, in 1958, this assumption was to be proven wrong as a result of a string of B-47 accidents:

13 March 1958 - B-47B disintegrated at 15,000ft with 2,077AFHRS

TB-47B broke up at 23,000ft with 2,418AFHRS

21 March 1958 - B-47E disintegrated with 1,129AFHRS

10 April 1958 - B-47E disintegrated with 1,265AFHRS

15 April 1958 - B-47E disintegrated with 1,419AFHRS

The USAF response to these accidents was two-fold:

a. Due to the criticality of the B-47 to the United States Cold War deterrent (over two thousand aircraft manufactured by three manufacturers), it was a high priority to ensure that the aircraft could continue limited operations as quickly as possible.

b. An aircraft structural integrity program was initiated by General Curtis LeMay to extend life of the B-47 and avoid future accidents incidents as that encountered during 1958. As part of this program, fatigue testing of three airframes was performed by Douglas, Boeing and the National Advisory Committee on Aeronautics (NACA).

As a result of these accidents, WCLS-TM-58-4 'Detail requirements for structural fatigue certification program' was issued by the USAF in 1958, which eventually evolved into MIL-STD-1530 and thus the concept of the Aircraft Structural Integrity Program (ASIP) was formalized within the USAF.

During this period the average age of aircraft had increased considerably, with the average design age of aircraft operated by the USAF during 1956 to 1960 had risen to 8.7 years (Ramey and Keating, 2009).

2.1.5 – 1969 - General Dynamics F-111 Accident

As stated by Lincoln (2000), the General Dynamics F-111 aircraft structure was certified in accordance with ASD TR-66-57 ‘Air Force Structural Integrity Program Requirements’, utilising the Safe-Life philosophy which required qualification of structure to a safe life via fatigue testing.

However, on 22nd December 1969, F-111A #94 (Serial Number 67-049) was lost due to premature failure of the D6AC steel wing pivot fitting caused by an undetected manufacture material flaw after only 107 AFHRS.

As a result of the investigation into the accident, the USAF adopted the Damage Tolerance philosophy in 1974 via the issue of MIL-A-83444. MIL-STD-1530 (issued in 1972) was used during the F-111 recovery program which occurred in the aftermath of this accident (ASIAC 1980) to re-establish the structural integrity of the F-111 fleet.

The FAA introduction of the concept of damage tolerance into the Federal Aviation Regulations occurred over a period of 15 years:

a. Transport Category Airplanes via FAR 25.571 Amendment 25-45 (1978);

b. Transport Category Rotorcraft via FAR 29.571 Amendment 29-28 (1989), and

c. Normal, Utility, Acrobatic and Commuter airplanes via FAR 23.571 Amendment 23-45 (1993).

The average age of aircraft continued to increase with the average design age of aircraft operated by the USAF from 1966 to 1970 increasing to 10.9 years (Ramey and Keating, 2009).

2.1.6 – 1977 - Dan Air Boeing 707 Accident

On the 14th May 1977, whilst on approach to Lukasa International Airport, Zambia from London, a Dan Air Boeing 707-321C impacted the ground short of the runway. At the time of the accident the aircraft was 14 years old, had accrued 47,621 AFHRS and performed 16,723 flight cycles.

The Boeing 707-321 type certification basis was ‘CAR 4b dated December 1953, Amendments 4b-1, 4b-2 and 4b-3[3] thereto; the Special Conditions and the provisions

[3] This amendment added fail-safe into the regulations.

amendments listed in Attachment A of CAA letter to Boeing dated October 30, 1957; and the provisions of Item 2 of Special Civil Air Regulation No. SR-422B.' (FAA, 1984).

As stated by Accident Investigation Branch (1978), the accident was the result of the failure of the right side horizontal stabilizer, which was the result of "long term fatigue damage, of the rear spar top chord and secondly, the inability of the redundant failsafe structure to carry the flight loads for a period long enough to enable the fatigue crack to be detected during routine inspection using the then current inspection procedures." (p.22).

Wanhill (2002) identified that the lessons learnt from this accident were that:

a. For a design to be considered fail-safe, the inspectability of the structure is as equally important as the structural design concept, and

b. Highlighted the inadequacy of older aircraft inspection methods and schedules to eliminate the threat of fatigue failure.

2.1.7 – 1978 – Issue of Advisory Circular 20-107

The Australian Transportation Safety Bureau (ATSB) (2007) stated that composites, in the form of glass fibre reinforced polymer (GFRP), have been used in aircraft since 1957, with usage increasing over time. The usage of composites in commercial aircraft (for example, certified in accordance with FAR 25) has been increasing since the 1978 introduction of the McDonnell Douglas MD-80, as shown in *Figure **1***.

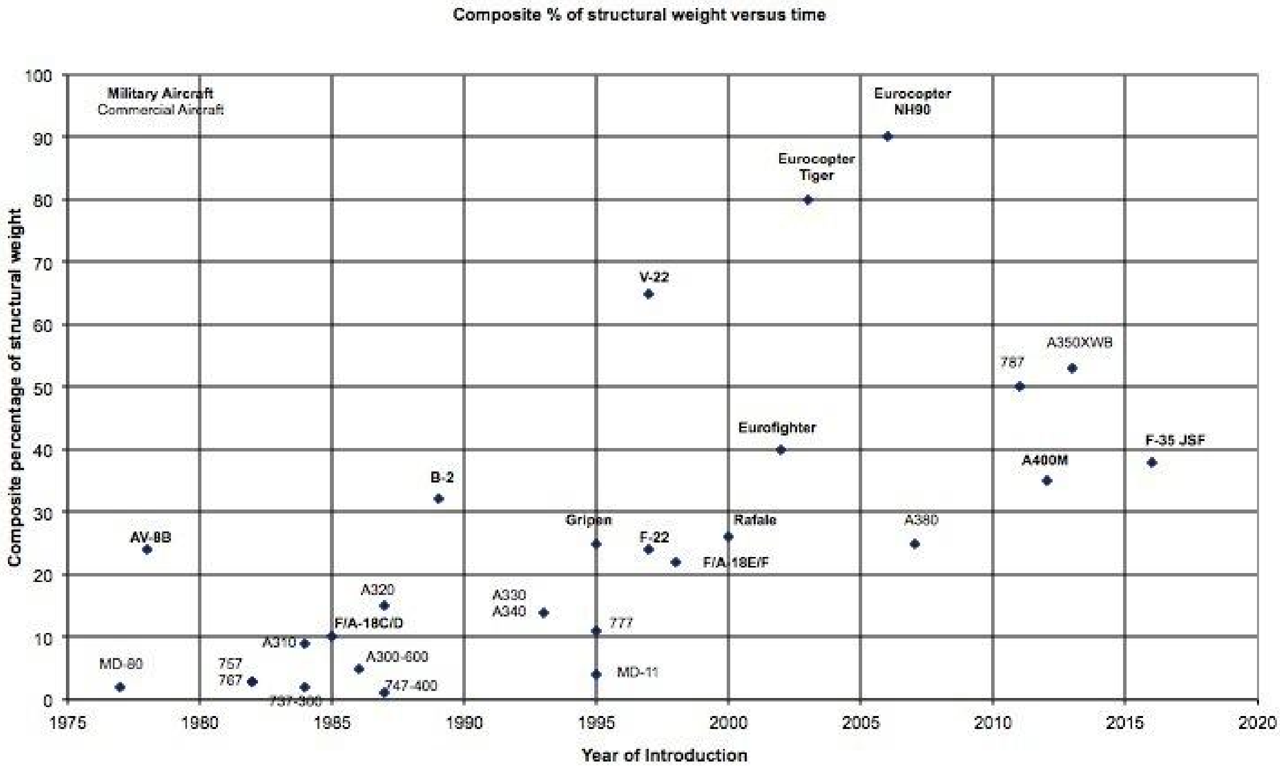

Figure 1: Growth of composite structure on major aircraft programs (1975-2010) as a percentage of weight (adapted from ATSB, 2007)

The FAA issued Advisory Circular (AC) 20-107 'Composite Aircraft Structure' on the 10th July 1978 which was applicable to aircraft certified under FAR 23 and 25 and rotorcraft under FAR 27 and 29. This AC provided guidance for demonstrating composite aircraft structural compliance with the airworthiness type certification requirements. This document was the first to provide guidance for composite aircraft compliance with the FAR and therefore is a foundation document for ASI for civil composite aircraft. The timing of the release of the AC coincides with the introduction of composites structures into commercial aircraft, as illustrated in *Figure 1*.

Since its introduction, AC 20-107 has undergone a number of revisions:

a. AC 20-107A issued on the 24th April 1984 covered proof of structure compliance (static, fatigue/damage tolerance and flutter) and additional considerations such as lightning protection. Whilst identified by Callus (2003) as representing the state of the art for the composite structures certification, Ilcewicz (2007) identified the need to update the document in the following areas:

   i) Removal of obsolete guidance;
   ii) Harmonisation with other regulations;
   iii) Update based on service and/or certification experience, and
   iv) Inclusion of new technology, including materials, engineering methods and maintenance procedures.

b. AC 20-107B issued on the 8th September 2009, incorporated a number of the changes identified by Ilcewicz (2007) and greatly expanded the information documented within the AC.

c. AC 20-107B (Change 1) issued on 24th August 2010, which corrected minor errors.

2.1.8 – 1988 - Aloha Airlines Boeing 737 Accident

On the 28th April 1988, after reaching a cruise altitude of 24,000 feet, an Aloha Airlines Boeing 737-297 suffered cabin decompression, when approximately 5.5m of the upper fuselage separated from the aircraft during flight. The aircraft was able to safety land, with only one fatality. At the time of the accident, the aircraft was 19 years old and had accumulated 35,496 AFHRS and 89,860 flight cycles (Job, 1996).

The Boeing 737-297 type certification basis was '14 CFR §25, Amendments 25-1 through 25-3, 25-7, 25-8, 25-15, 14 CFR §21, 14 CFR §1: and special conditions attached to FAA letter to Boeing dated October 15, 1965, and modified in letters dated December

23, 1966 and February 14, 1967, and Special Condition No. 25-89-NW-5 attached to FAA letter to Boeing dated April 10, 1979.' (FAA, 2010).[4]

This accident highlighted the issues associated with Widespread Fatigue Damage (WFD) and in particular, Multiple Site Damage (MSD) which is the presence of multiple fatigue cracks in same structural element. The other form of WFD, Multiple Element Damage (MED), is the presence of fatigue cracks in adjacent structural elements. The major concern with WFD is that it can negate the Fail-Safe philosophy, as illustrated in the Aloha accident.

This accident was a primary trigger for the Aging Aircraft Safety Act (AASA), which was passed into United States law in 1991. This Act required the FAA to:

a. Prescribe regulations that ensure the continuing airworthiness of aging aircraft. This was achieved by the issue of the Aging Airplane Safety Interim Final Rule (Federal Register Vol. 67 page 72726, 2002) and subsequently the issue of the Aging Airplane Safety Final Rule (Federal Register Vol. 69 page 5518, 2005).
b. Conduct inspections and review the maintenance and other records of each aircraft an air carrier uses to provide air transportation, which was achieved by the issue of Advisory Circular 120-84 Aging Aircraft Inspections and Record Reviews.

2.1.9 – 2010 – Introduction of Limit of Validity

On the 13th July 2009, a Southwest Airlines Boeing 737-300 on a flight from Baltimore to Nashville had to divert to Charleston, West Virginia after the aircraft's cabin depressurized when a 1 foot-by-2-foot hole appeared in its upper fuselage near its vertical stabilizer, with no serious injuries. At the time of the incident the aircraft had accumulated

[4] Thus the aircraft design did not include the Damage Tolerance philosophy, which was introduced at Amendment 25-45.

42,500 flight cycles and 50,500 AFHRS and was 15 years old (Federal Register Vol. 75 page 69746).

On the 26th October 2010, an American Airlines Boeing 757 flying from Miami to Boston had to return to Miami when it depressurized after a 1 foot-by-1-foot hole opened in the upper part of the fuselage near a cabin door toward the front of the plane, with no serious injuries. At the time of the incident the aircraft had accumulated some 22,000 flight cycles and was 20 years old (Federal Register Vol. 75 page 69746).

These incidents raised concerns about the presence of WFD, particularly as the age of commercial aircraft continues to increase. With military aircraft displaying a similar trend of increasing age, for example during the period from 1991 to 1995, the average design age of aircraft operated by the USAF had increased to 20 years (Ramey and Keating, 2009).

As a response on the 15th November 2010, the Federal Aviation Administration (FAA) formally issued a new rule (Federal Register Vol. 75 page 69746) which sought to prevent widespread fatigue damage (WFD) by requiring aircraft manufacturers and other certification applicants to establish the Limit of Validity (LOV) for each aircraft design, which is the number of flight cycles or AFHRS below which the aircraft will be free from WFD. Manufacturers had between 18 and 60 months to comply, depending on the particular aircraft type. This rule was applicable to all new transport aircraft (yet to be certified) and for existing aircraft over 75,000 lb (34,000 kg), operated under FAR Parts 121 and 129 with a type certificate dated after 1 January 1958. The rule specifically states that it was not applicable to composite structures, as these structures are covered by AC20-107B which details damage tolerance assessment of composite structures.

2.1.10 - Summary

The temporal relationship between the identified milestones is illustrated in *Figure* **2**.

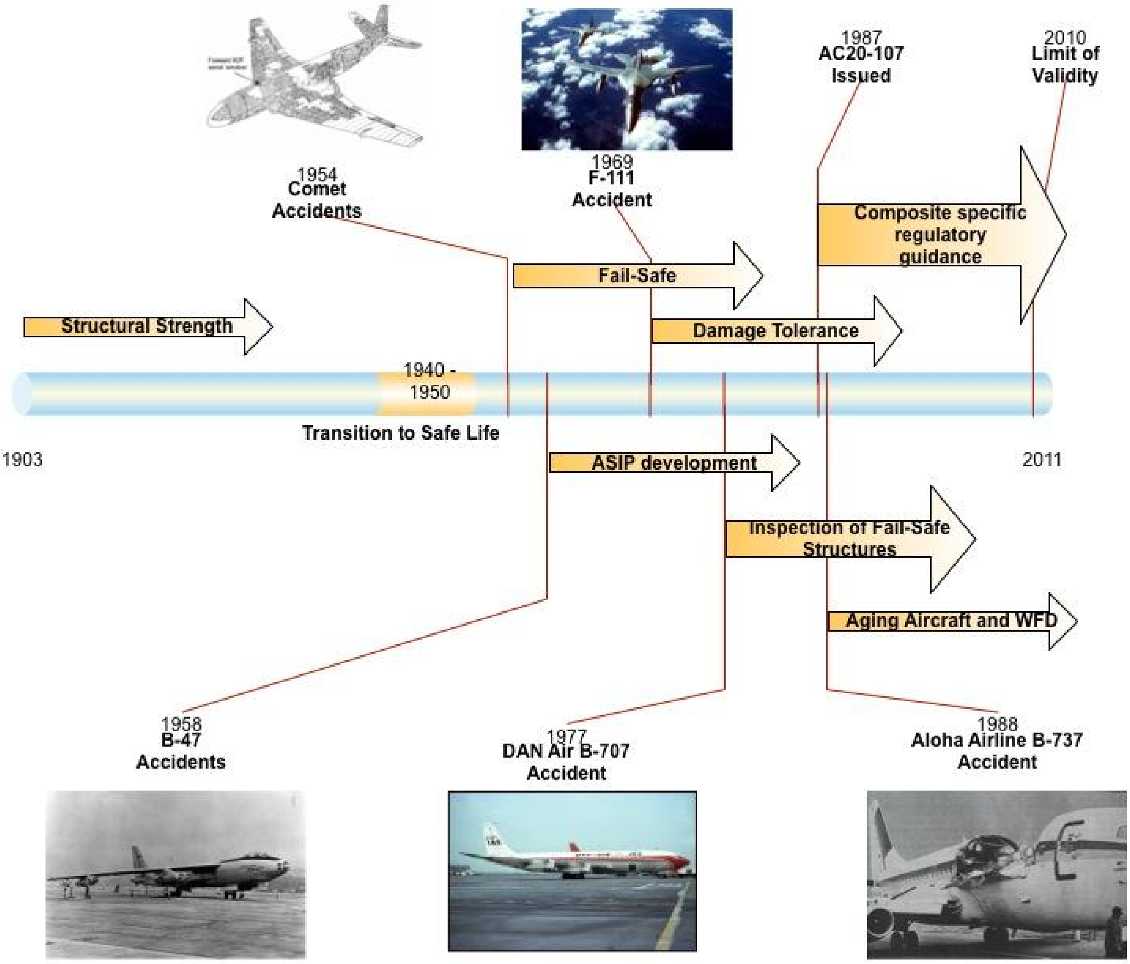

Figure 2: Milestones in Aircraft Structural Integrity.

2.2 - Review of Structural Integrity Regulations

2.2.1 - Military Aircraft Structural Integrity Regulations

Military Structural Certification

MIL-HDBK-516B (Change 1) establishes the Airworthiness Certification Criteria (ACC) to be used to assess the airworthiness of fixed and rotary-wing, manned and unmanned aerial systems and is approved for use by all departments and agencies of the

United States Department of Defence (DoD). The criteria provided in the handbook is to be tailored to meet the specific needs of the air vehicle being certified, with the tailoring being performed in accordance with the guidance provided within the document (paragraph 1.2.1).

*Figure **3*** illustrates the linkage between the ACC identified in MIL-HDBK-516 and referenced documents, such as MIL-STD-1530C and Joint Service Specification Guide (JSSG) 2006, which provide more detailed guidance for the establishment of structural and verification requirements for airframe.

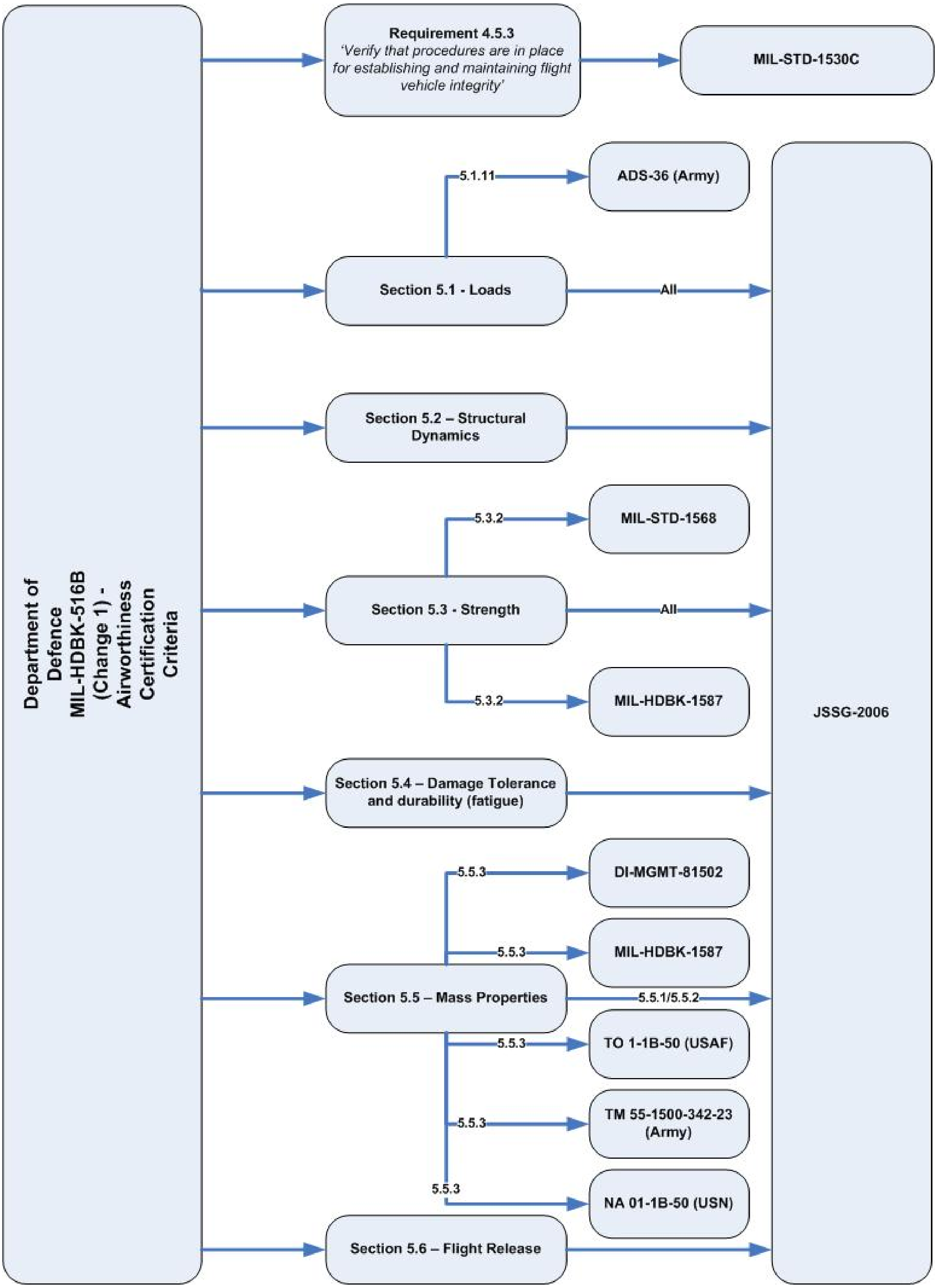

Figure 3: US Military Aircraft Structural Integrity related certification requirements.

The documents referenced by Section 5.5 are related to the verification of mass properties, centre of gravity margins and the documentation of weight and balance processes and therefore will not be further discussed. ADS-36 is for use by the United States Army only and will not be further discussed. Thus, the primary documents out referenced by the ACC related to structural integrity management are:

a. MIL-STD-1530 – The Aircraft Structural Integrity Program (ASIP);

b. Joint Service Specification Guide 2006 – Aircraft Structures;

c. MIL-STD-1568 – Materials and Processes for Corrosion Prevention and Control in Aerospace Weapons Systems, and

d. MIL-HDBK-1587 – Materials and Process Requirements for Air Force Weapons Systems.

MIL-STD-1530 - The Aircraft Structural Integrity Program (ASIP)

The Evolution of ASIP

The B-47 accidents led the USAF to develop and issue WCLS-TM-58-4 in 1958, which describes the detailed requirements for the structural fatigue certification program. This document evolved into MIL-STD-1530 which describes aircraft structural integrity program requirements. The evolution of MIL-STD-1530 is described below:

1958 - Issue of Technical Memorandum WCLS-TM-58-4 'Detail requirements for structural fatigue certification program'.

1959 - Release of 'ARDC-AMC Program Requirements for the Structural Integrity Program for High Performance Aircraft' which delineated the breakout of structural integrity program activities into eleven sub-program areas (http://www.afgrow.net/applications/DTDHandbook/sections/page1_1.aspx):

a. Static test;

b. Flight load summary;

c. Fatigue test;

d. Low-altitude gust environment;

e. Mission profile data;

f. Interim service load;

g. VGH (velocity, normal acceleration and height) life history recording;

h. Eight-channel service load recording;

i. Sonic fatigue;

j. High-temperature structure, and

k. Design criteria.

1960 - Release of a number of Military Specification 8800 series documents to support the implementation of the Aircraft Structural Integrity Program (ASIP) described in the 1959 document, for example MIL-A-8860 'Airplane Strength and Rigidity General Specification for' and MIL-A-8861 'Airplane Strength and Rigidity Flight Loads'. (http://www.afgrow.net/applications/DTDHandbook/sections/page1_1.aspx)

1968 - Release of ASD-TR-66-57 'Air Force Aircraft Structural Integrity Program Airplane Requirements', where the Safe-Life philosophy for fatigue requirements was stated, with structure qualification via fatigue testing as identified by Lincoln (2000).

1969 - Release of Air Force Regulation 80-13 was the definitive ASIP establishment document. The document added Phase VI inspections and assigned responsibilities to the various USAF organisations for the implementation of the program. (http://www.afgrow.net/applications/DTDHandbook/sections/page1_1.aspx)

1970 - Update of ASD-TR-66-57.

1972 - Issue of MIL-STD-1530 'Aircraft Structural Integrity Program, Airplane Requirements'. This document added the requirement for a ASIP force structural maintenance plan which identifies inspection and modification requirements and economic life estimates. On 22nd December 1969, F-111A #94 (Serial Number 67-049) was lost due to premature failure of the D6AC steel wing pivot fitting caused by an undetected manufacture material flaw after only 107 AFHRS. As stated by Lincoln (2000), the General Dynamics F-111 aircraft structure was certified in accordance with ASD TR-66-57 'Air Force Structural Integrity Program Requirements'. MIL-STD-1530 was used during the F-111 recovery program, which occurred in the aftermath of this accident (ASIAC 1980) to re-establish the structural integrity of the F-111 fleet.

1975 - Issue of MIL-STD-1530A. Introduced the damage tolerance philosophy.

1988 - Issue of MIL-STD-1530A Notice 1, which states that MIL-STD-1530A (dated 1975) is valid for use in acquisition.

1996 - Issue of MIL-STD-1530A Notice 2, MIL-STD-1530A (1988) cancelled and replaced by MIL-HDBK-1530 'Aircraft Structural Integrity Program, General Guidelines for'.

1996 - Issue of MIL-HDBK-1530, which was identical to MIL-STD-1530A except for the cover page, which identifies that the document is to be used for guidance only.

2002 - Issue of MIL-HDBK-1530A

2002 - Issue of MIL-HDBK-1530B, changes include revision of definitions and introduction of climatic testing to Task III.

2004 - Issue of MIL-HDBK-1530B Notice 1, MIL-HDBK-1530B (2002) cancelled and replaced by MIL-STD-1530B.

2004 - Issue of MIL-STD-1530B 'Aircraft Structural Integrity Program (ASIP)', which supersedes both MIL-HDBK-1530B and MIL-STD-1530A.

2005 - Issue of MIL-STD-1530C

2010 - Issue of MIL-STD-1530C Notice 1, which states that MIL-STD-1530C (dated 2005) is valid for use in acquisition.

ASIP Overview

The United States Department of Defence (DoD) requirements for aircraft structural management (termed ASI Program (ASIP)) are documented within MIL-STD-1530C. This standard states that the purpose of ASIP is to ensure that aircraft can operate safely and economically throughout its life from a structural integrity viewpoint. This is achieved by using the following four pillars:

a. Establishing and validating the structural integrity of aircraft structures;

b. Using operational data to update the status of the structural integrity;

c. Providing quantitative data to support decisions related to conduct of inspections and priority for modification; and

d. Providing lessons-learnt to be applied to the next generation of aircraft design/modifications.

The tasks (and associated activities) to implement these pillars are identified in Table 1 and discussed in the following text.

Task I Design Information	Task II Design Analyses & Development Testing	Task III Full-Scale Testing	Task IV Certification & Force Management Development	Task V Force Management Execution
5.1.1 – ASIP Master Plan	5.2.1 – Material and Joint Allowables Testing	5.3.1 – Static Tests	5.4.1 – Certification Analyses	5.5.1 – Individual Aircraft Tracking (IAT) Program
5.1.2 – Design Service Life & Design Usage	5.2.2 – Loads Analysis	5.3.2 – First Flight Verification Ground Tests	5.4.2 – Strength Summary & Operating Restrictions (SSOR)	5.5.2 – Rotorcraft Dynamic Component Tracking (RDCT) Program
5.1.3 – Structural Design Criteria	5.2.3 – Design Service Loads Spectra	5.3.3 – Flight Tests	5.4.3 – Force Structural Management Plan (FSMP)	5.5.3 – Loads/Environment Spectra Survey (L/ESS)
5.1.4 – Durability and Damage Tolerance Control Program	5.2.4 – Design Chemical/Thermal Environment Spectra	5.3.4 – Durability Tests	5.4.4 – Loads/Environment Spectra Survey (L/ESS) Development	5.5.4 - ASIP Manual
5.1.5 – Corrosion Prevention & Control Program (CPCP)	5.2.5 – Stress Analysis	5.3.5 – Damage Tolerance Tests	5.4.5 – Individual Aircraft Tracking (IAT) Program Development	5.5.5 - Aircraft Structural Records
5.1.6 – NonDestructive Inspection (NDI) Program	5.2.6 – Damage Tolerance Analysis	5.3.6 – Climatic Tests	5.4.6 – Rotorcraft Dynamic Component Tracking (RDCT) Program Development	5.5.6 – Force Management Updates
5.1.7 – Selection of Materials, Processes, Joining Methods & Structural Concepts	5.2.7 – Durability Analysis	5.3.7 – Interpretation and Evaluation of Test Results		5.5.7 - Recertification
	5.2.8 – Corrosion Assessment			
	5.2.9 – Sonic Fatigue Analysis			
	5.2.10 – Vibration Analysis			
	5.2.11 – Aeroelastic and Aeroservoelastic Analysis			
	5.2.12 – Mass Properties Analysis			
	5.2.13 – Survivability Analysis			
	5.2.14 – Design Development Tests			
	5.2.15 – Production NDI Capability Assessment			
	5.2.16 – Initial Risk Assessment			

Table 1: MIL-STD-1530C Tasks and activities

Task 1 - Design Information

Firstly, the structural integrity criteria to be applied during design are established and involve:

a. Development of various management plans, including those associated with ASI and corrosion;

b. Definition of the aircraft Design Service Life (DSL), mission mix and mission profiles;

c. Selection of materials, and

d. Identification of structural design requirements. MIL-HDBK-5165 provides guidance regarding the airworthiness certification criteria applicable for all US DoD air vehicle systems. This handbook identifies the US DoD Joint Service Specification Guide 2006 (JSSG-2006)[6] for the certification of aircraft structures. JSSG-2006 includes both the USAF preferred damage tolerance and the United States Navy (USN) preferred safe life philosophies.

Task 2 - Design Analysis and DT&E

After Task 1 has been completed, the characterization of the environment in which the aircraft must operate, the initial testing of materials, components and assemblies and the analysis of the aircraft design are performed. The analysis performed during this task includes survivability, mass properties, aeroelasticity, vibration, sonic fatigue, stress, durability, corrosion and damage tolerance. Assessments are performed regarding production Non-Destructive Inspection (NDI) capability and a risk assessment is performed to demonstrate that risks to structural integrity have been adequately mitigated. The design

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[5] Section 5

[6] Annex A, Section A.3.12.1 contains details specific to composites.

service spectra identifying the frequency, distribution and sequencing of anticipated loads is developed. Additionally, the climatic/thermal environmental spectra anticipated during the aircraft's service life are defined. Finally, testing and evaluation as required to support the design of the aircraft (including material/joint allowable testing) is performed.

Task 3 – Full Scale Testing

As defined in MIL-STD-1530C, full scale testing "consists of flight and laboratory testing of the aircraft structure to assist in the determination of the adequacy of the analysis and design of the aircraft" (p.8). The purpose of this task is to verify the outcomes from Task 2 and includes the conduct of:

a. Static tests;
b. First flight verification ground tests;
c. Flight tests;
d. Durability tests;
e. Damage tolerance tests, and
f. Climatic tests.

Task 4 – Certification and Force Management Development

As defined in MIL-STD-1530C, certification and force management development "consists of the analysis that leads to the certification of the aircraft structure as well as the development of the processes and procedures that will be used to manage force operations (inspections, maintenance, modifications, damage assessments, risk analysis, etc.) when the aircraft enters the inventory" (p.8).

As part of this task, the following activities are performed:

a. Certification analysis to determine compliance with certification requirements;

b. Compliance with other regulatory requirements, for example, requirements for a Loads/Environment Spectra Survey (L/ESS) and Individual Aircraft Tracking (IAT) programs;

c. Documentation of Strength Summary and Operational Restrictions (SSOR) providing descriptions of aircraft structures, critical design conditions, minimum margins and structural limitations, and

d. Development of a plan for the structural management of the aircraft once in service (known as a Force Structural Management Plan in MIL-STD-1530C). This document contains details of:

   i) Inspection and structural maintenance programs, including the methods and inspection intervals.

   ii) Structural maintenance database, and

   iii) Structural surveillance program, for example fleet leader and aircraft teardown requirements.

Task 5 – Force Management Execution

As defined in MIL-STD-1530C, force management execution "executes the processes and procedures developed under Task IV to ensure the structural integrity throughout the life of each individual aircraft. This task may involve revisiting elements of earlier tasks, particularly if the service life requirement is extended or if the aircraft is modified" (p.8). Example processes conducted includes L/ESS and IAT programs, maintenance of structural maintenance records and are conducted by the aircraft sustainment organisations, such as squadron personnel and support contractors.

Joint Service Specification Guide (JSSG) - 2006 - Aircraft Structures

JSSG-2006 "establishes the joint structural performance and verification requirements for the airframe" (p. 1). The guide requires the addition of supplemental structural performance information (identified as blanks in the document) before it can be used for contractual purposes. Appendix A to the guide provides rationale, guidance and lessons learnt for the performance and verification requirement to assist in the provision of the supplemental information. The guide is applicable to both metallic and non-metallic (composite) structures (paragraph 1.2.3). The structural integrity related guidance provided within JSSG-2006 has been grouped into the following topics:

Materials and processes

Considerations for the use of composites include: temperature, moisture, insensitivity to low cycle fatigue, extreme loading, battle damage and residual strength with insignificant damage growth of flaws. The static strength allowable for composites is to be based on the temperature appropriate for the associated flight condition combined with the most critical range of moisture conditions. The 'B' basis allowables for composites must include the effects related to lay up, geometry and type of loading[7]. The process for establishing these allowables can be quite expensive and therefore the determination of 'B' basis allowables from coupon data representative of the desired lay up and loading is desirable. Composite material properties should be obtained from or developed using the guidance provided in MIL-HDBK-17[8], including the use of the Building Block Approach[9] (BBA) for the characterisation of materials.

[7] JSSG-2006, p.358
[8] Note that MIL-HDBK-17 has been replaced by CMH-17.
[9] JSSG-2006, p. 351

Building block approach

MIL-HDBK-17-3F states, that "To accommodate the unique features of composites, a method for determining relevant design properties has been devised. This is the "building block approach." This method provides a systematic way of treating composite materials to obtain design information" (p. 4-24). This incremental approach is generally referred to as the Building Block Approach (BBA), refer to *Figure* ***4***.

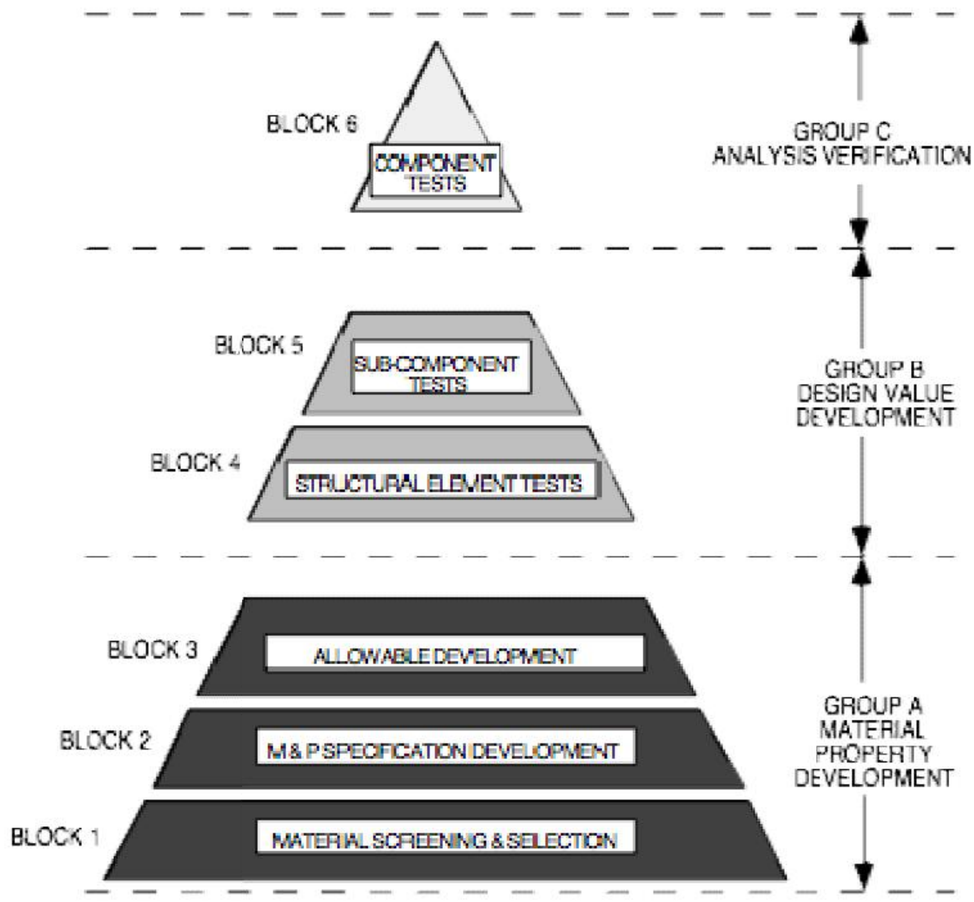

Figure 4: Building Block Approach (from MIL-HDBK-17-3F)

Whilst the building block approach has been used for metallic structures, it is more critical for composite structures due to the environmental sensitivity, multiple failure modes and sensitivity to out of plane loads of composites. This complexity results in an inability to adequately determine structural performance using analytical methods alone.

Statistical process control

To counter the significant variability possible in the manufacture of composite components the use of Statistical Process Control (SPC) is recommended. SPC is used to determine if a process is proceeding correctly, via analysis of process variations, determining which variations are natural and those which need to be corrected. As such, SPC is a decision making aid to assist in determining when and what to change in a process to minimise variations in output.

Foreign Object Damage

JSSG-2006 Appendix A contains lengthy discussion regarding Foreign Object Damage (FOD), however there is no detailed discussion of the increased susceptibility of composites to impact damage. The following maximum acceptable probabilities related to FOD are identified:

a. Loss of aircraft due to FOD = 1×10^{-7} per flight, and

b. FOD impact causing unacceptable damage (for metallic aircraft) = 1×10^{-5} per flight.

Whilst, the probability of FOD impact causing unacceptable damage for metallic aircraft is provided. No value is provided for composite aircraft or discussion provided as to the reason for any difference between the structures in this regard. The details of low (tool impact) and high (hail and runway debris) energy impacts are provided in Tables VII and VIII of JSSG-2006.

Composite Aircraft In-Service Experience

JSSG-2006 documents specific composite structures in-service experience associated with various structural performance requirements, for example, delaminations in a channel in the empennage due to out of plane bending of during fastener removal was not considered during design.

Design Analysis

JSSG-2006 states “In laminated composites, the stresses and ply orientation are to be compatible and residual stresses of manufacturing are to be accounted for, particularly if the stacking sequence is not symmetrical” (p. 353). There is a large discussion regarding in-service experience regarding static strength and raises the point that as more aircraft are designed from composites it will be interesting to see what happens in regard to static strength failures.

Durability

JSSG-2006 provides guidance related to the durability of composites. It states that durability is difficult to assess due to the large scatter in fatigue test results and the fatigue growth due to large spectrum loads. However, in general the durability of composites is excellent when the structure is sized to meet its strength requirements.

Damage Tolerance

JSSG-2006 provides specific guidance related to the damage tolerance of composite structures, which in addition to the threats identified for metallic structures must be designed to meet additional damage due to manufacture and battle damage. The residual strength requirements are the same as for metallic structures except that *Pxx*[10] is not limited to 1.2 times the maximum load in one lifetime. For high confidence, it is necessary for either

[10] *Pxx* - Maximum average internal member load (without clipping) that will occur once in M times the inspection interval.

damage growth rate to be insignificant or for damage to not grow to critical size in two lifetimes.

MIL-STD-1568 Materials and Processes for Corrosion Prevention and Control in Aerospace Weapons Systems

This document outlines the requirements to establish and implement a Corrosion Prevention Advisory Board (CPAB) and provides guidance for the avoidance of various forms of corrosion for metals. The document contains no specific guidance in regard to composite structures.

MIL-HDBK-1587 - Materials and Process Requirements for Air Force Weapons Systems

This document includes discussion regarding the selection of materials and process for metals (aluminium, titanium, steel and beryllium) and non-metallics (including composites, elastomers and transparencies). The following composite specific considerations are identified: development of Material Characterisation and Design Allowables Substantiation Plan (MCDASP), development of material acquisition, material qualification, process and product fabrication specifications, validation of fastening methods, thermal expansion mismatch between joined materials, avoidance of galvanic corrosion between certain metals (such as aluminium) and carbon composites, repairability/supportability and electromagnetic/electrical behaviour.

Military Structural Integrity Summary

The US DoD airworthiness requirements (as defined by MIL-HDBK-516B (Change 1)) provides the requirements for Aircraft Structural Integrity and includes reference to a number of sub-ordinate standards, handbooks and guides. MIL-STD-1530C describes the requirements for Aircraft Structural Integrity Management and could be considered material-neutral. Material (composite and metal) specific guidance is provided in JSSG-2006, MIL-HDBK-17 (now CHM-17) and MIL-HDBK-1587.

2.2.2 - Civil Aircraft Structural Integrity Regulations

As one of the two major civil aviation airworthiness authorities, the civil structural certification requirements as defined by the FAA will be summarised.

Federal Aviation Regulation 25 - Transport Category Airplanes

The civil equivalent to MIL-HDBK-516B Change 1 for transport category aircraft, such as the Boeing 787 is FAR 25. The following sections discuss the FAR 25 regulations associated with structural integrity.

Load Definition

FARs 25.301, 25.303, 25.305 and 25.307

Limit Loads are defined as the maximum loads expected in service, with the structure not suffering any permanent deformation and any temporary deformation up to limit loads shall not interfering with safe operation of the aircraft.

Ultimate Loads are defined as the limits loads multiplied by a Factor of Safety (FoS) of 1.5 (unless otherwise specified) with the structure sustaining Ultimate Loads for 3 seconds prior to failing.

Compliance with the strength and deformation requirements must be demonstrated by test unless structural analysis method has been demonstrated by experience to be reliable.[11]

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[11] This could drive increased testing for composite structures.

Loading Conditions

FARs 25.321, 25.331-351, 25.361-373, 25.391-459, 25.471-519, 25.521-537, 25.561-563

There are seven groupings of loading conditions identified, including: flight loads, flight manoeuvre and gust conditions, supplementary conditions (for example, engine torque and gyroscopic loads), control surface and system loads, ground loads, water loads and emergency landing conditions.

Damage Tolerance and Fatigue Evaluation of Structures

FAR 25.571

This requirement states that an evaluation of the strength, detail design and fabrication must demonstrate that catastrophic failure of the structure due to fatigue, corrosion, manufacturing defects or accidental damage will be avoided throughout its operational life. With the introduction of the Limit of Validity (LOV) in November 2010, the operational life of the aircraft became limited to LOV, which is defined as the time period (in either AFHRS, flight cycles or both) within which it has been demonstrated that Widespread Fatigue Damage (WFD) will not occur. The evaluation includes consideration of expected loading spectra, temperatures and humidities; the identification of Principal Structural Elements (PSE) and Detail Design Points (DDP), which if failed, would cause catastrophic failure. The service history of aircraft with a similar structural design (accounting for differences in operating conditions and procedures) may be used as part of the evaluation, however the establishment of the LOV must be supported by evidence from full-scale testing. As an outcome of the evaluation, any inspections and other procedures (structural replacement) required to achieve operational life are developed and documented within the Instructions for Continued Airworthiness (ICA), as required by FAR 25.1529. Inspection intervals must be based on crack growth rates assuming that the structure contains the maximum initial flaw size expected as a result of manufacture or in-service damage. Whilst the introduction of LOV was driven by the threat of WFD, Advisory Circular (AC) 25.571-1D states that all maintenance actions to address fatigue, corrosion and accidental damage up to the LOV are to be included in the structural maintenance program.

Two structural integrity philosophies are discussed within the regulation: Damage Tolerance as the preferred philosophy, with Safe-Life only to be used when it is demonstrated to be impractical for a specific structure to be assessed using the Damage Tolerance philosophy. The regulation also includes the consideration of the effects of sonic fatigue. Relevant structures must be able to withstand the damage and remain intact when exposed to the static loads (considered as ultimate) that are reasonably expected during flight, including: impact with 4lb bird, uncontained fan blade impact, uncontained engine failure and high energy rotating machinery failure.

Material Properties

FARs 25.603 and 25.613

Material strength properties and design values are to be established within specific probabilities and confidences, depending on the type of structure which it will be used within and take into account environmental effects (such as temperature and humidity):

a. Redundant structure - 90% probability with 95% confidence, and

b. PSE - 99% probability with 95% confidence, and

c. Materials selected for safety critical structures shall have properties conforming to approved standard (such as a Military Specification), be established on test or previous service experience and shall take into consideration the effects of the in-service environment.

Specific Design Considerations

FARs 25.581, 25.619, 25.629 and 25.631

The regulations in coverage of special factors for castings, bearing and fitting, aeroelasticity, birdstrike and lighting protection.

Federal Aviation Regulation 26 - Continued Airworthiness and Safety Improvements for Transport Category Airplanes

This regulation provides additional requirements for the support of the continued airworthiness of and safety improvement of transport category airplanes, and covers a number of topics including electrical wiring, fuel tank flammability and aging aircraft safety. The aging aircraft safety topic is directly related to aircraft structural integrity as it covers:

a. Widespread Fatigue Damage (WFD), definition and compliance requirements for the LOV and the mechanism for extending the LOV of a specific aircraft, and

b. Damage Tolerance Data for repairs and alterations.

Federal Aviation Regulation 121 - Operating Requirements: Domestic, Flag, and Supplemental Operations

This regulation requires that any person holding an air carrier or operating certificate under FAR 119, ensure the continued airworthiness of the aircraft being operated. This includes the revision of maintenance programs, the incorporation of design changes and the update of ICA. Certificate holders must also demonstrate that the maintenance of age-sensitive parts and components has been sufficiently timely to ensure the highest degree of safety. As part of the regulation, certificate holders are required to present aircraft and associated service and maintenance data to the FAA to perform aging aircraft inspections. Finally, the maintenance program for aircraft must include damage-tolerance based inspections for fatigue critical structures. Therefore, it appears that the requirements within this regulation have similar intent to MIL-STD-1530C Task 5 'Force Management Execution'.

AC 20-107B Change 1 - Composite Material Structure

This composite specific AC provides additional guidance covering FARs 23, 25, 27 and 29 with topics including material and fabrication development, proof of structures (static, fatigue/damage tolerance and flutter), continued airworthiness, crashworthiness, lightning protection and fire protection/flammability and thermal issues. It is important to note that there is no specific mention of the LOV within the AC. As with military certification, civil composite structural certification also utilises the building block approach.

It is identified that there are very few industry standards sufficiently detailed to establish design criteria or test/analysis processes for complete damage tolerance analysis and that impact damage is a concern for composite structures. The AC provides details regarding five different categories of composite damage and discussion regarding no-growth, slow-growth, arrested growth damage tolerance methodologies.

Civil Structural Integrity Summary

The FAA regulations have a distinct metallic flavour, for example the emphasis on widespread fatigue damage (FAR 26.21). There is no mention of 'composite' in any of the design regulations (FAR 23, 25, 27 and 29) with the exception for FAR 23.573 which is concerned with damage tolerance and fatigue evaluation of structures for Normal, Utility, Acrobatic and Commuter category aircraft. However, Advisory Circular (AC 20-107B Change 1) provides additional guidance regarding demonstrating regulation compliance for composite structures.

3 - DRIFTING INTO STRUCTURAL FAILURE

3.1 - What is our experience with composite aircraft structures?

Composites have been used in aircraft structures for over forty years, with the percentage of the composite usage in structure steadily increasing over time. This section provides a brief overview of the current in-service experience with composite aircraft structures.

3.1.1 - United States Navy Boeing F/A-18

Seneviratne, Tomblin, Kittur and Rahman (2011) investigated the aging effects on F/A-18 wing root step-lap joints (WRSLJ) which is one of the key examples of bonded primary structure certified and deployed on an air vehicle in the United States. The WRSLJ is the transition from the carbon/epoxy upper and lower wing skins to a titanium fitting for attachment to the fuselage of the aircraft. Retired F/A-18 aircraft were used as the source of coupons for subsequent static and fatigue testing, with the expended fatigue cycles of the various WRSLJ ranging from 1/2 to 1 lifetime of service. Prior to the removal of the coupons, an inspection of each WRSLJ for potential service induced defects was performed, with no defects detected. The static testing indicated that the service experience (including associated environmental exposure) did not degrade the integrity of the WRSLJ; fatigue testing indicated that the remaining life of the joints was significant and that the additional fatigue cycles induced via testing did not affect the residual strength of the WRSLJ.

3.1.2 - Beechcraft Starship

The Beechcraft Starship was developed as the successor to the highly successful King Air aircraft series. The aircraft was design using a significant amount of composites (approximately 70% of airframe weight) for increased durability and higher strength to weight ratio. The first of fifty-three production aircraft flew on the 25th April, 1989 with the last aircraft being manufactured in 1995.

Tomblin, Salah and Davies (2010) investigated the aging effects on a main wing of the Beechcraft Starship which had been in-service for 12 years and 1800 AFHRS. Testing revealed that there was no major change in structure stiffness/compliance and response, no obvious visual signs of degradation and no degradation of thermal properties. Non-Destructive Inspection (NDI) of the wing revealed no major defects and the results of full-scale testing of the wing correlated well with the results obtained from the certification article.

3.1.3 - Sailplane Experience

Composites have been used in primary structure of sailplanes (gliders) for a large number of years, with the first flight of the first composite sailplane Akaflieg Stuttgart FS-24 'Phoenix' occurring was in 1957. The composite in this instance was balsa wood/glass fibre sandwich.

Since then the usage of composites in sailplanes as continued with the Slingsby T.65 (fibreglass construction) and Scheicher ASW-20 (glass reinforced plastic) first flights in 1977 and the LAK-19 manufactured from kevlar, carbon and glass fibre which conducted it's first flight in 2001.

With the use of composites in high-performance sailplanes and the associated quest for increased structural performance it would be expected that structural integrity issues associated with composites would be readily apparent. A preliminary review of the National Transport Safety Board (NTSB) aviation accident database (performed during March 2012) did not reveal any composite structural integrity related trends. For example during the period from 1965 to 1981 there were eight structural integrity related accidents (ignoring those associated with overload failures - exceedence of flight limitations).

A preliminary review of the NTSB aviation accident database identified a large number of accidents associated with airframe structural overload due to exceeding design limitations. A contributing factor is the quest for sailplanes with higher performance and better handling qualities, which as identified by Tarode (1996) results in the removal of 'soft' limits as design conservatism is reduced. Thus, introducing the condition where just exceeding the design limits can result in structural overload failure.

3.1.4 - NASA Research into Composite Aging

National Aeronautics and Space Administration (NASA) has performed numerous research programs related to the in-service performance of composite aircraft structures. These programs include:

NASA Contractor Report 4733, Advanced technology composite fuselage - repair and damage assessment supporting maintenance dated April 1997

The report states that for the airlines consulted, experience with composites, in general, ranged from poor to very successful. The poor service record of some components was attributed to fragility or inclusion of nondurable design features. The fragility, which is an issue for thin-gage components (secondary structure), should be less of an issue for thicker-gage components.

This report documents the performance of a number of composites parts, which were substituted for metal components on a number of large commercial aircraft (Boeing 727 and 737) and was part of the NASA-sponsored Advanced Composites Energy Efficiency (ACEE) program. The composite components introduced as part of the program and their associated service life are documented in Table 2.

Component	Quantity In-service	Total accrued AFHRS	Total accrued cycles
Boeing 727 elevators	5 ship-sets	>331,000	>189,000
Boeing 737 spoilers	108	>2,888,000	>3,781,000
Boeing 737 horizontal stabilisers	5 ship-sets	>133,500	>130,000

Table 2: ACEE Composite Component Experience

Review of the performance of the components indicated no durability or corrosion problems, though minor corrosion pitting was discovered in the fastener holes of the 737 stabiliser aluminium fittings due to obsolete sealing practises. Whilst in-service, numerous repairs where successfully completed on the 727 elevators and 737 horizontal stabilisers.

NASA Technical Memorandum 89067, Long-term environmental effects and flight service evaluation of composite materials, January 1987

This report documented the long-term durability of advanced composites in a number of different environments:

a. 10 years of world-wide ground based exposure on graphite and kevlar reinforced composite systems;

b. In-service exposure on various components on large fixed-wing aircraft (as documented in Table 3), and

c. In-service exposure on various components on helicopters (as documented in Table 4).

Component	Composite System	Oldest Individual Component Age	Notes
Lockheed L-1011 fairing	Kevlar-49/Epoxy	10 years	Performed similar to production fibreglass/epoxy fairings.
Lockheed L-1011 ailerons (Four ship-sets)	Graphite/epoxy	4 years	No damage incidents or major maintenance actions required.
Boeing 737 spoilers	Graphite/epoxy	13 years	Fewer maintenance problems than production aluminium spoilers. Hail, birdstrike and other impact damage was detected on a number of spoilers. Minor damage was repaired, whilst for major damage the spoilers where removed from service due to repair expense.
Boeing 737 horizontal stabilisers (Five ship-sets)	Graphite/epoxy	~3 years	No damage incidents or maintenance actions required.
Lockheed C-130 centre-wing boxes (reinforced with composite system) (Installed on two aircraft)	Boron/epoxy	12 years	No damage or defects reported on the two modified C-130 aircraft. Based on ground test article, superior fatigue endurance was anticipated.
Douglas DC-10 aft pylon skin (Installed on three aircraft)	Boron/aluminium	7 years	Have not performed as well as production titanium skins. Issues associated with corrosion were considered to be due to the use of inadequate corrosion protection methods.
Douglas DC-10 upper aft rudder (Quantity 15)	Graphite/epoxy	5.7 years 22,265 AFHRS	No degradation was evident in Air New Zealand aircraft rudder (service life identified in the In-Service Experience column). Lightning damage on other rudders have been successfully repaired.
Boeing 727 elevators (Quantity 10)	Graphite/epoxy	Unknown	Damage as a result of ground handling and lightning strike, all of which where repaired by airline maintenance personnel.

Table 3: Large Fixed Wing Aircraft Composite Component Experience

Component	Composite System	Oldest Individual Component Age	Notes
Bell 206L forward fairing, litter and baggage doors	Kelvar-49/epoxy	<34 months <3387 AFHRS	Performed better than equivalent metallic components.
Bell 206L vertical fin	Graphite/epoxy	<34 months <3387 AFHRS	Performed better than equivalent metallic components.
Sikorsky S-76 tail rotors	Graphite/epoxy	<51 months	Excellent in-service performance.
Sikorsky S-76 horizontal stabilisers	Hybrid kelvar-49/epoxy	<4051 AFHRS <66 months	Excellent in-service performance.
Sikorsky CH-53D cargo ramp skin	Kelvar-49/epoxy	~4 years	No damage or service related problems reported.

Table 4: Helicopter Composite Component Experience

The strength and modulus degradation due to exposure to moisture and UV radiation was around 25% for the unpainted ground-exposure test articles. This degradation is expected to be an upper limit due to the flight exposure articles (components) demonstrating less moisture absorption and usually have a painted surface. It was identified that kelvar/epoxy composite systems are more affected by various environments than the graphite/epoxy systems.

During the period of monitoring (for some components up to 13 years) of the various composite components, the monitored components displayed excellent performance. In general, the composite aircraft components demonstrated better ‘durability’ than the comparable metallic components.

3.1.5 - Boeing 787 Certification

During FAA compliance planning for the Boeing 787 twelve special conditions (as stated in TCDS T00021SE) were identified, with five concerned with the novel features of the composite wings and fuselage as identified in GAO-11-849:

a. 25-348-SC - Composite Wing and Fuel Tank Structure—Fire Protection Requirements;
b. 25-360-SC - Composite Fuselage In- Flight Fire/Flammability Resistance;
c. 25-362-SC - Crashworthiness;
d. 25-363-SC - Tire Debris Penetration of Fuel Tank Structure, and
e. 25-414-SC - Lightning Protection of Fuel Tank Structure To Prevent Fuel Tank Vapor Ignition.

These special conditions do not identify any specific structural integrity issues associated with the certification of the composite fuselage and wing of the aircraft.

3.1.6 - In-Service Summary

In summary, there do not appear to be any significant structural integrity issues associated with the use of composites in aircraft structures identified to date.

3.1.7 - Future Direction of Composite Structures

Warren, Heslehurst and Wilson (2012) provides a summary of in-service experience with composites in aircraft structures and identifies that there do not appear to be any significant structural integrity issues based on experience to date. However, Warren et al. (2012) also identifies that future good performance in regard to composite structural integrity cannot be guaranteed by good previous performance due to:

a. Contemporary structural integrity methodologies have evolved based on experience with metal structures;
b. Trend towards increased structural efficiency, and
c. Increasing percentage of composites used within aircraft structures.

3.2 - Accident Theory

3.2.1 - Introduction to Resilience

EUROCONTROL (2009) defines resilience as "The ability of a system to succeed under varying and adverse conditions" (p.2) and Woods (2006) identifies that resilience refers to the art of managing the unexpected. Taking the viewpoint that ASI Management (ASIM) is the 'system' and 'success' is considered to be no structural integrity related aircraft failures, resilience appears to be a viable metric for ASIM performance.

Hollnagel (2011) furthermore identifies the four cornerstones of Resilience as:

a. *Responding (actual):* The system must be able to provide a response to the actual event. This corresponds to the MIL-STD-1530C pillar of providing quantitative data to support decisions related to conduct of inspections and priority for modification. However, personnel within the ASI System are required to implement the ASIM and to provide the response.

b. *Monitoring (critical):* The purpose of monitoring is to identify potential or actual threats to the systems, identification of the critical. The scope of monitoring includes both the external environment and the internal behaviour of the system. This corresponds to MIL-STD-1530C pillar of using operational data to update the status of the structural integrity.

c. *Anticipating (potential):* The purpose of anticipation is to identify potential future threats and opportunities. This is achieved via the use of a model of the system and the environment within which it interacts.

d. *Learning (factual):* The system must be able to learn from experience, determine the facts and therefore update its ability to respond, monitor and anticipate. This corresponds to providing lessons-learnt to be applied to the next generation of aircraft design/modifications in the ASIM.

3.2.2 - Structural Resilience

The capability of a structure to withstand unforeseen damage, is similar to the concept of resilience, as defined by Hollnagel (2011) resilience is “The intrinsic ability of a system to adjust its function prior to, during, or following changes and disturbances, so that it can sustain required operations under both expected and unexpected situations.” (Hollnagel (2011), page xxxvi). Therefore, this capability can be termed structural resilience.

Structural resilience can be provided within a structure via a number of different mechanisms:

a. Large factor of safety, as provided by the structural strength methodology. This can result in heavy structures however, and

b. Redundant structure, as provided by the fail-safe methodology.

Since the first flight at Kitty Hawk, there has been a trend towards developing more efficient and lighter structures. This efficiency is achieved by removal of excess material, based on structural analysis knowledge at the time of design.

In the early years, aircraft structures were generally significantly over designed due to the use of the Structural Strength philosophy used. As knowledge increased (reducing the amount of Factors of Safety to consider, compare CAR 4B dated 1938 to the latest FAR will identify a decrease in the number of Factors of Safety to consider) and ASI evolved other philosophies to account for newly encountered degradation mechanisms (e.g. fatigue), increased analysis capability and newly identified degradation modes/influencers. With the increase in structural understanding and the pursuit of highly optimised structures, there is a risk that the capability of the structure to withstand unforeseen damage has been degraded, which could also be stated as reducing the resilience of the structure (structural resilience). This is similar to the loss of ‘soft’ limits discussed previously in relation to sailplanes.

3.2.3 - Total Structural Residual Strength

Total Structural Residual Strength is the difference between Predicted Residual Strength and Actual Residual Strength, where:

a. Predicted Residual Strength = Predicted Strength - Limit Load, examples include:

   i) Structural Strength: The strength of the structure was considered static since manufacture, with the Predicted Strength remaining unchanged overtime.

   ii) Safe-Life: With the introduction of the Safe-Life methodology, strength was considered to decrease over time due to the effects of fatigue.

   iii) Damage Tolerance: The damage tolerance methodology in addition to fatigue also considered to effects of manufacturing/material defects.

   iv) Limit of Validity: The introduction of the Limit of Validity by the FAA in 2010 represented an attempt to ensure that the effects of Widespread Fatigue Damage (WFD) is considered.

b. Actual Residual Strength = Actual Strength - Limit Load;

c. Predicted Strength is the strength of the structure predicted by structural analysis and consideration of unknown degradation modes (as expected to behave in-service), and

d. Actual Strength is the strength of the structure taking into account all influencers on strength accounted for (as actually behaves in-service).

The relationship between these Predicted Strength examples (Structural Strength, Safe-Life, Damage Tolerance and Limit of Validity) and the Actual Strength and Limit Load (in this case represented by Applied Load) is illustrated in *Figure **5***. The green dashed lines in *Figure **5*** represents the Actual Strength restarting at the start of each Predicted Strength examples.

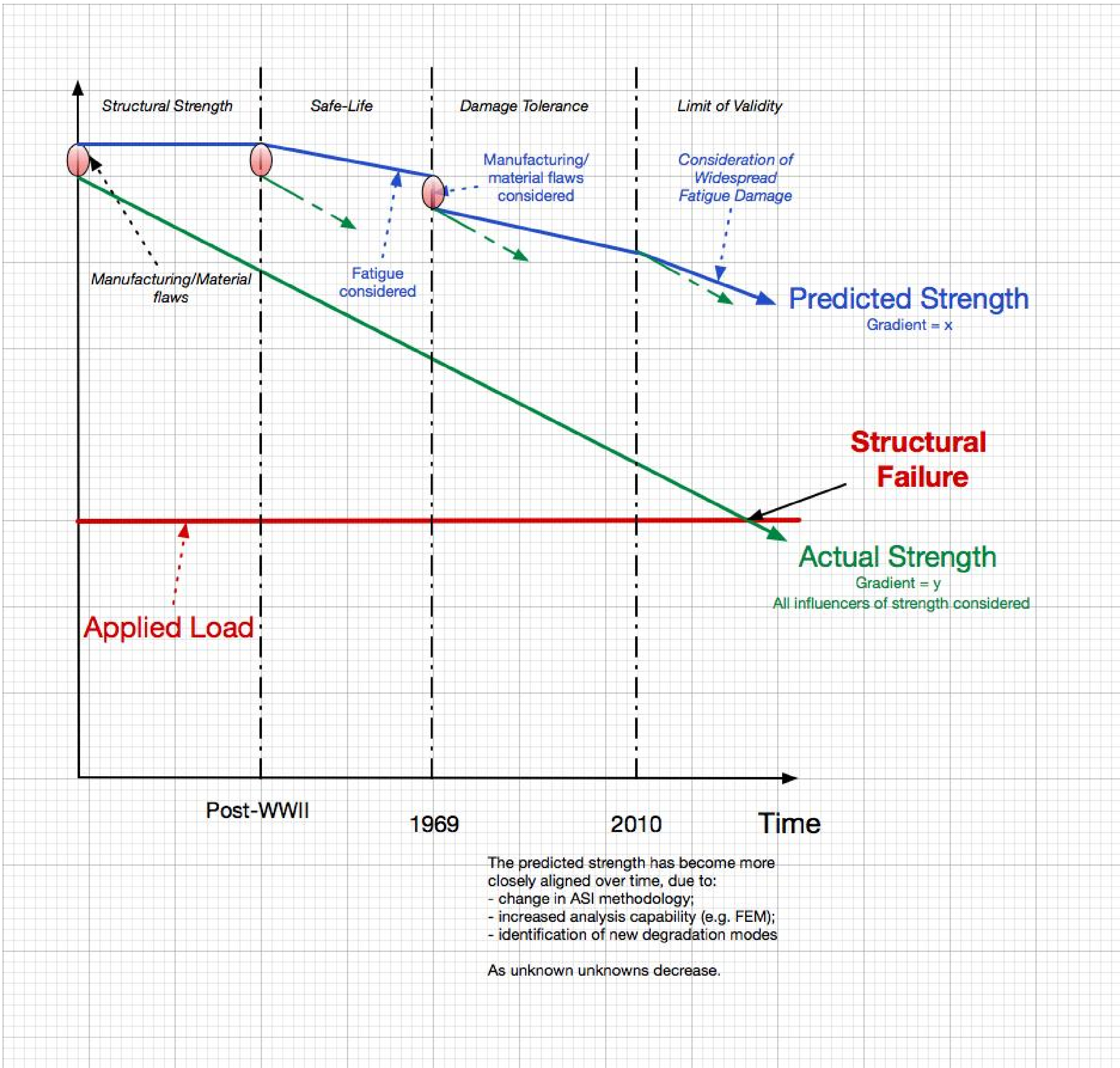

Figure 5: Total Structural Residual Strength

As illustrated, the gap between the predicted and actual residual strength of a structure has reduced over a number of decades of experience with metal structures (structural optimization). However, there are different degradation modes between metal and composite structures (as discussed in Section 5), with the response of the common degradation modes also being different.

3.2.4 - Unruly Technology

Dekker, p.42 (2011) discusses the term 'unruly technology' which is used "to capture the gap between our image of tidiness and control over technology through design, certification, regulation, procedures, and maintenance on the one hand and the messy, not-so governable interior of that technology as it behaves when released into a field of practice."

An example of unruly technology identified by Dekker (2011) resulted in the loss of Alaska Flight 261. On January 31, 2000, the MD-80 aircraft servicing this flight crashed as a result of a screw-jack failure from lack of lubrication. The maintenance extensions to the screw-jack were authorised by the Federal Aviation Administration and represents the gap between the development of a system and its implementation in operation. The aircraft manufacturer had initially (1967) established a 30,000 AFHR replacement period for the screw-jack with no wear check within that period. However, after only one year, the wear detected was significantly higher than that predicted (technology was not behaving as predicted). In response, the manufacturer introduced a wear check every 3,600 AFHR (or every C-Check). Over the intervening years, the C-Check interval was extended, resulting in the wear check being extended to every 15 months, which equated to an interval of about 9,550 AFHR. At the last wear check (in 1997), the accident aircraft was on the acceptable limit and was released for flight.

Unruly technology is similar to Total Structural Residual Strength, which is the difference between the Predicted Residual Strength (to be established and maintained via design, certification, regulation, procedures and maintenance) and the Actual Residual Strength (as behaves in-service).

*3.3 - Resilience of the Aircraft St*ructural Integrity 'System'

3.3.1 - Aircraft Structural Integrity System

The aircraft structural integrity system is composed of the organisations, which influence the decisions associated with structural integrity management. The classification of these organisations and examples are provided at Table 5.

Category	**Civil**	**Military**
Regulator	Federal Aviation Administration Civil Aviation Authority	Military (e.g. United States Air Force (USAF) or Australian Defence Force (ADF))
Customer	Flying Public	Military (e.g. USAF, ADF)
Manufacturer (OEM)	e.g. Boeing/Airbus	e.g. Boeing/Lockheed Martin
Operator	Airlines	Military (e.g. USAF, ADF)
Accident Investigation	National Transportation Safety Board (NTSB) or Australian Safety Transport Bureau (ASTB)	Military (e.g. USAF, ADF)

Table 5: ASI System Organisations

It can be observed from Table 5, that the number of independent organisations involved in the ASI System is significantly less in the military arena than for civil aviation. The civil aircraft structural integrity system is illustrated in *Figure **6***, and will be used as the basis for discuss during the remainder of this section.

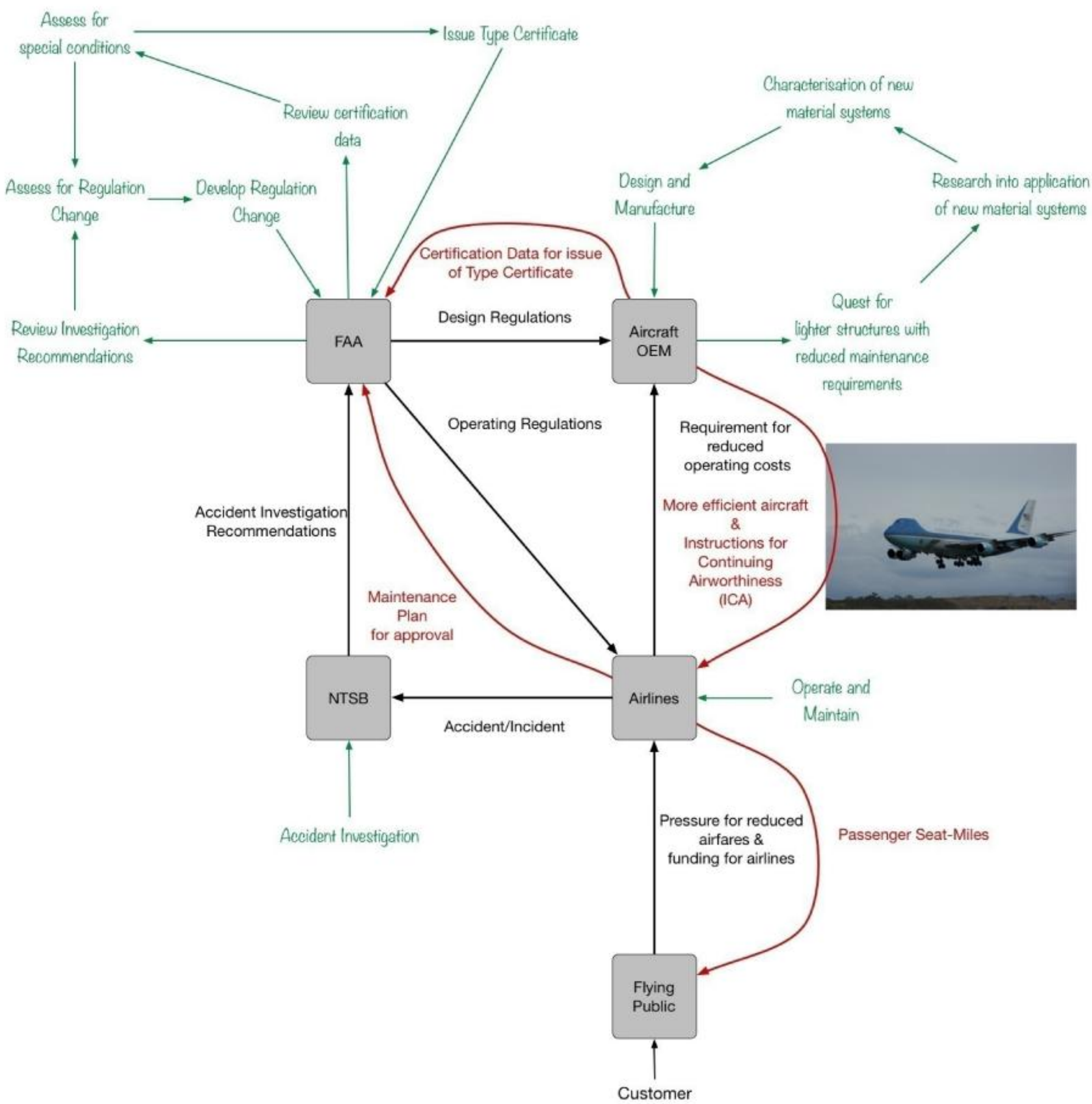

Figure 6: Civil Aircraft Structural Integrity System

The flying public is the customer, purchasing seat-miles (i.e. flights) from the airlines. The flying public is after value for money, which results in downward pressure on the cost of airfares. This combined with external factors such as the increasing cost of fuel and tighter emission and noise regulations being imposed by numerous countries results in the airlines placing requirements on the aircraft Original Equipment Manufacturers (OEM) for more efficient aircraft. In response to these pressures, Boeing has introduced a number of new or updated aircraft: Boeing 747-8 which is 15% more fuel efficient, 15% less CO_2 emissions and 30% smaller noise footprint than its predecessor (Spaeth 2012); Boeing 787 which reduces structural weight by using approximately fifty percent composites by weight, more efficient General Electric or Rolls-Royce engines and the increased aerodynamic efficiency (using smooth wing technology and active fly-by-wire flight controls); Boeing 737 Max which via the use of new powerplant (the CFM International LEAP-1B) and winglets will be 11 to 13.5% more fuel efficient than the Boeing 737NG (Broadbent 2012). The aircraft OEM use a number of different means to increase the efficiency of aircraft, such as:

a. Aerodynamic improvements;

b. Increased engine efficiency, and

c. Reduce aircraft empty weight.

From a structural integrity perspective, reducing aircraft empty weight is the most important, as this drives the OEM to increase structural efficiency and investigate new material systems. This has led to the increased use of composites (by weight) in aircraft structures (refer to *Figure **1***).

The FAA issues a Type Certificate for new aircraft design using the process outlined in *Figure **7***.

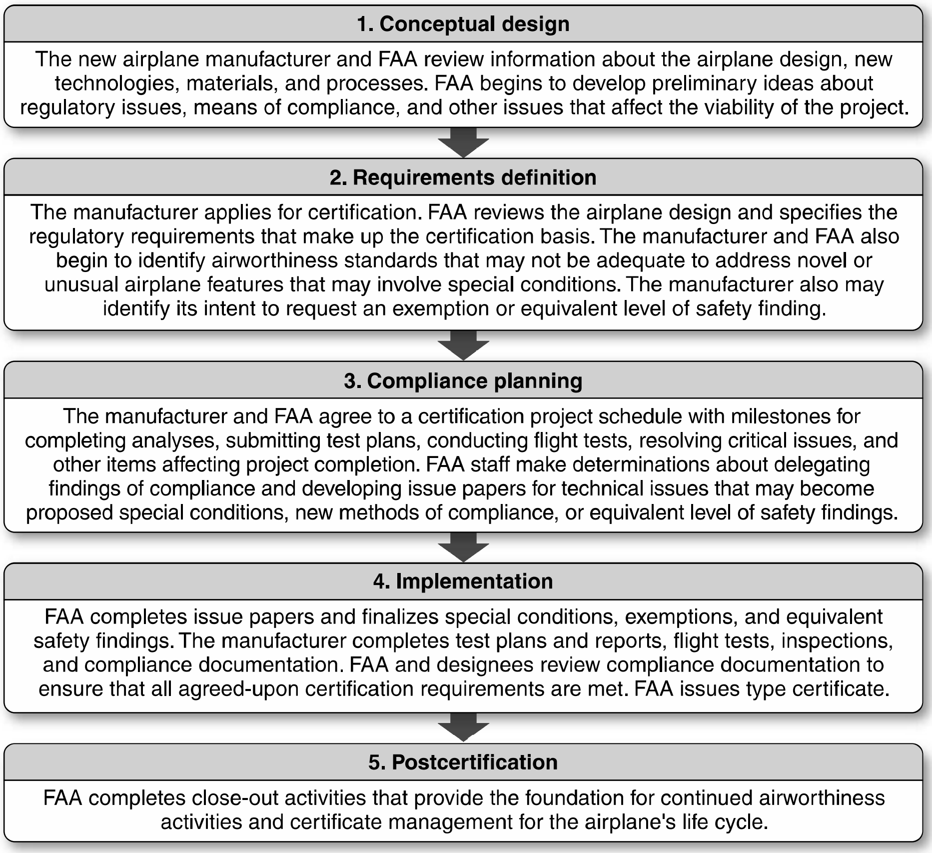

Figure 7: FAA Type Certification Process for New Aircraft Design (from GAO-11-849).

When the FAA determined that a regulatory standard may not be adequate for an aircraft design due to novel features, a Special Condition[12] is developed to address the deficiency. Additionally, the FAA may issue an Equivalent Level of Safety (ELOS), which is used when literal compliance with a certification regulation cannot be shown, and compensating factors exist which can be shown to provide an equivalent level of safety. Experience with Special Conditions may drive the FAA to update regulation or provide additional guidance detailing acceptable means of compliance. An example of compliance guidance was the release of Advisory Circular AC20-107, in response to the increasing use of composites in aircraft structures. The FAA may also introduce regulatory changes as a result of accident investigations performed by the National Transportation Safety Board (NTSB). An example was the FAA prescribing regulations to manage aging aircraft as a

[12] A Special Condition is a rulemaking action that is specific to an aircraft make and often concerns the use of new technology that the Code of Federal Regulations do not yet address.

result of the crash of an Aloha Airlines Boeing 737-297 in 1988. The FAA defines the aviation regulations and authorise the maintenance programs used by the airlines, usually based on OEM guidance.

3.3.2 - Resilience Classification of ASI System

Amalberti (2006) identifies four classes of system resilience: Ultra performing, egoistic, collective expansion and ultra-safe (described in Table 6).

	Ultra Performing	**Egoistic**	**Collective Expansion**	**Ultra-Safe**
Safety Objectives	$< 10^{-3}$	$< 10^{-5}$	$<10^{-6}$	$>10^{-6}$
Examples	Mountaineering, Extreme Sports, Transplant Surgery	Drivers on public roads, patients selecting a doctor	Food industry, banks and services.	High risk-complex systems, transportation and energy industries.
Model for success	Outstanding performance, constant search and expression for maximum performance	Individual satisfaction. The choice of service fully dependent on customer decision.	Collective satisfaction. Customers are not making a direct choice of these services.	No accidents. One accident anywhere means the end of business.
Model of failure	Low competency	Poor team work. Unstable quality and delivery. Fatalities here and there. Individual victims sue individual workers.	Poor top organisation. Accidents possible. Victims form a group and sue local social entities and local politicians.	Complacency. Large accident is the rule. Victims form a group and may sue the system as a whole.
Criteria for Resilience	Training competitiveness	Quality and control procedures	Transparency, HRO, management regulations.	Show compliance, accept supervision, ready for the 'big' one.
Who is in charge of organising resilience?	Everyone and no one	Quality managers. Business risk control department	Safety managers	Safety managers scrutinised by International and Government agencies.

Table 6: System Resilience Classifications (adapted from Amalberti (2006))

The ASI System is an integral part of the aviation transportation system, with very low accident rates. For the year 2010, the National Transportation Safety Board identified that for United States Air Carriers operating under 14 CFR 121, the accident rate was 0.157 per 100,000 flight hours or an accident rate of 1.57×10^{-6}. Aviation accidents can result in either airlines or aircraft OEMs either going out of business or being adversely impacted, for example:

a. Lockheed Electra: As a result of a number of aircraft accidents (American Airlines 320 (1959), Braniff Airways 542 (1959) and Northwestern Airlines 710 (1960) resulting from unstable whirl mode of the propellers, Lockheed received no further new orders for the aircraft.

b. De Havilland Comet: The loss of two BOAC Comet 1 aircraft during 1955, resulted in de Havilland missing the opportunity to become a market leader in the jet era.

c. ValueJet 592 crashed on the 11th May 1996 as the result of a cargo compartment fire due to errors by both ValueJet and SabreTech (maintenance organisation): ValueJet suspended all revenue flight operations and returned its operating certificate to the FAA for a period of a year and then subsequently merged with AirTran Airways and assumed its identity.

d. COMAIR 5191 crash on take-off from Lexington, Kentucky in August 27, 2006 primarily as a result of pilot error. The families of the victims sued COMAIR, the FAA and the departure airport seeking compensation.

Therefore, based on *Table **6***, the ASI System is categorised as an Ultra-Safe in terms of resilience.

3.3.3 - Resilience of ASI System

What is the resilience of the ultra-safe system, as such the ASI System (as illustrated in *Figure **6***)? In particular, how resilient is the ASI System to the introduction of new material systems in aircraft structures?

Threats to ASI System

GAO-11-849 identified the following safety-related concerns associated with the introduction of composites into civil aircraft structures:

a. Limited information regarding the behaviour of composite structures over time. This results in the potential inaccuracy in the models used by the OEMs to predict the damage growth in composites.

b. Technical concerns related to the unique properties of composite structures. This includes:

   i) The lack of non-destructive inspection procedures to confirm the strength of a bonded composite repair;

   ii) Higher susceptibility of composite repairs to human error, and

   iii) Difficulty in detecting impact damage in composites.

c. Limited standardisation of composite materials and repair techniques. GAO-11-849 states that there are approximately a dozen metal alloys used in the construction of aircraft structures as opposed to over 60 unique materials used for various composite repairs.

d. Level of training and awareness of composite materials. Personnel that interact with the aircraft in service (maintenance, airline and airport personnel) may not be fully aware of the differences between metallic and composite structures. For example, the potential for significant damage to occur to a composite structure post-impact, without any visible surface damage.

These concerns (limited service life information, unique technical properties, limited standardisation and level of training and awareness) would be applicable to any new material system being introduced and are considered to be the threats to the ASI System which are manifest when a new material system is introduced.

Response of the ASI System

Response of Airlines

With the introduction of composite materials, airlines have commenced the conduct of composite training.

Response of OEMs

The question can be asked how does an aircraft OEM introduce a new material in aircraft structure? To provide insight and assuming that the behaviour of The Boeing Company can be considered industry standard practice, a review of Boeing experience with composites was considered beneficial.

Over a number of decades Boeing has been increasing the percentage of composites used in aircraft structures using a number of different composite systems, as illustrated below:

a. 1969 - The Boeing 747 structure contains ~1% composites (fibreglass honeycomb), used for control surfaces, fairings and trailing edge panels.

b. 1982 - The Boeing 757/767 structures contains ~3% composites (carbon, aramid and hybrid composite systems) used in trailing edge panels, control surfaces and gear doors.

c. 1995 - The Boeing 777 structure contains ~11% composites (carbon, glass and hybrid composite systems) used in control surfaces, fairings, floor beams and stabiliser/fin torque boxes.

d. 2011 - The Boeing 787 structure contains ~49% composites (carbon and fibreglass composite systems) used in fuselage, fairings, control surfaces and wing/fin/stabiliser torque boxes.

It can be seen that Boeing initially introduced composites to secondary structures (such as fairings, control surfaces) and then expanding the use of known composite systems to new structural application (such as floor beams and torque boxes). Based on Boeing's experience it can be seen that the introduction of a new material can take a number of decades (four decades from 747 to 787), until it is used for significant percentage of primary aircraft structure. This period of introduction allows for the regulations to be developed in harmony with the structures and reduces the potential for a major deficiency regarding the regulation of the structures developed from the new material.

Response of Regulator

The response of the FAA has been to:

a. Conduct research into the behaviour of composites, this includes the establishment of the National Aging Aircraft Research Program (which includes composite as well as metallic structures) and the Joint Advanced Materials and Structures (JAMS) Centre of Excellence (established 2002) which investigates safety and standardised certification of existing and emerging structural applications of composite materials and advanced materials.

b. Conduct composite material training courses for FAA safety inspectors and designees.

c. Collaborate with industry (aircraft OEMs and airlines), such as the Composite Aircraft Composite Repair Committee (CACRC). The CACRC charter is to develop and improve maintenance, inspection, and repair of commercial airplane composite structures and components. As such, has developed numerous guidance documents related to composites, such as: AE-27 Design of Durable, Repairable, and Maintainable Aircraft Composites (1997) and ARP5089 Composite Repair NDT/NDI Handbook (2011). The FAA is part of the organization, which develops and maintains the Composite Materials Handbook CMH-17, which superseded MIL-HDBK-17.

d. Release guidance regarding how to achieve regulatory compliance for composite aircraft structures, such as:
   i) AC 20-107B Composite Aircraft Structures (first issued in 1978);
   ii) AC 21-26A Quality Systems for Manufacture of Composite Structures (first issued in 1989);
   iii) AC 23-20 Acceptance Guidance on Material Procurement and Process Specifications for Polymer Matrix Composite Systems (issued 2003), and
   iv) AC 65-33 Development of Training/Qualification Programs for Composite Maintenance Technicians (issued 2011).

How resilient is the ASI System?

So, how resilient is the ASI System, based on the discussion in this section? The migration of the use of composites from secondary aircraft structures (minimal percentage of structure) to primary structure (large percentage of structure) has taken a number of decades. This gradual introduction has allowed structural integrity related regulations (and guidance) to be development in-step with the composite usage. This development in regulation has been supported by research and collaboration between the regulators and industry (aircraft OEMs and airlines). During this period of introduction no apparent significant structural integrity issues associated with the use of composites in aircraft structures have been identified (refer to Section 3.1). It therefore appears that the ASI System has high resilience to the introduction of composite materials.

4 - SIMULATING STRUCTURAL INTEGRITY

4.1 - Structural Integrity Models

4.1.1 - Probabilistic

Backman (2008) presents a model for structural integrity, which is probability based, where "the probability of a safe structure (S_T) is equal to the joint probability of safe design (S_D), safe manufacturing (S_M), safe maintenance (S_I), safe operation (S_O) and safe regulation (S_R)."(p. 4). This, as stated by Backman (2005) can be expressed as:

$$P(S) = P(DIMOR) = P(O|MIRD).P(I|MRD).P(M|RD).P(D|R).P(R)$$

Where:

P(S) is probability of safe structure;

P(DIMOR) is probability of safe design and safe manufacturing and safe maintenance and safe operation and safe regulation;

P(O|MIRD) is probability of safe operation given safe manufacture, inspection, regulation and design;

P(I|MRD) is probability of safe inspection given safe manufacture, regulation and design;

P(M|RD) is probability of safe manufacture given safe regulation and design;

P(D|R) is probability of safe design given safe regulation, and

P(R) is probability of safe regulation.

This model is presented to support the safe use of composites in aircraft structures, however it is just as applicable to metallic structures. Backman (2005) uses this foundation as a means of providing contrast between metallic and composite structures in a number of structural integrity areas.

Tuegel (2011) discusses the concept of using Probabilistic Structural Reliability Assessments (PSRA) to support structural integrity decision making. The concept of PSRA is based on the probability that for a given flight, the flight loads will exceed the available residual strength (structural failure) or not (survival), in this case calculated using Monte-Carlo simulation. The probability distribution for residual strength is determined by a combination of material fracture toughness and uncertainty on crack size. Whilst the probability distribution for maximum load is determined by the load exceedance curve for a single flight. Tuegel (2011) identifies three challenges to performing PSRA: lack of data to determine probability distribution, uncertainty about the crack size and uncertainty about future loads on the aircraft.

The probabilistic analysis detailed in FAA (1999) is similar to the PSRA in concept, with *Figure 8* illustrating the probabilistic analysis concept, where failure occurs at the union of the probability distributions of 'applied stress' and 'component strength'.

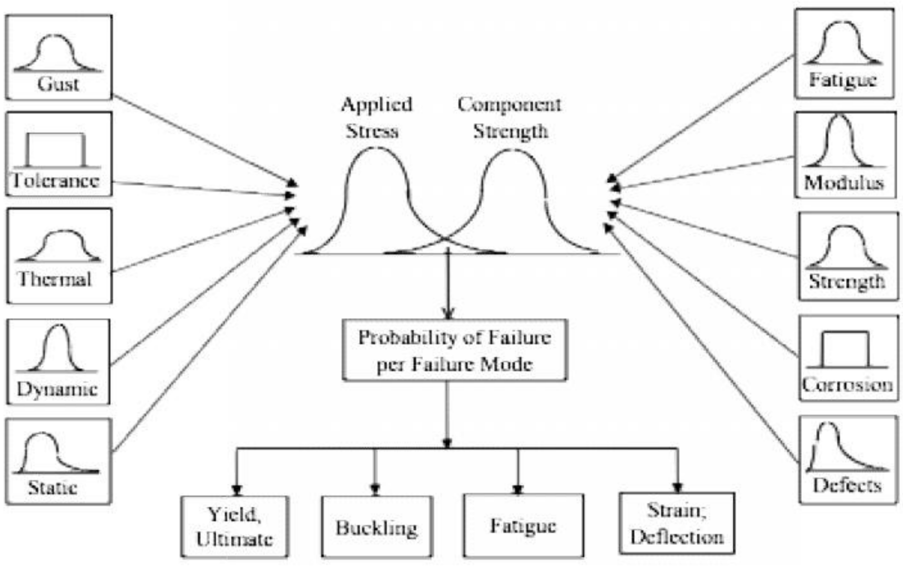

Figure 8: Probabilistic Analysis Concept (from FAA, 1999)

4.1.2 - The Diamond

Eastin & Swift (2005) presented the 'Diamond' concept for damage tolerance, focusing on crack related damage. The diamond concept is composed of five elements:

a. Site: Where could cracks start?

b. Scenario: How will cracks grow?

c. Detectability: What is the smallest crack size, which can be reliably detected?

d. Dangerous: At what crack size is the residual strength of the structure compromised (i.e. the structure becomes dangerous)?

e. Duration: How long does a crack take to grow from detectable to dangerous?

Swift (2009) identifies that the diamond can be used for the determination of the threshold (duration between manufacture and first in-service inspection) and interval (duration between in-service inspections). Swift (2007) extended the application of the diamond concept from damage tolerance to corrosion and concluded that the damage tolerance principles should be applied to corrosion and not just constrained to fatigue. Additionally, Swift (2009) identifies that the initial flaw method of damage tolerance is not always appropriate as the damage may not behave like a crack. Thus, rendering fracture mechanics analysis inappropriate to determine damage growth rates.

4.1.3 - The Three-Legged Stool

Grandt (2004) identifies that structural integrity is akin to a three-legged stool, with the legs comprised of:

a. Residual Strength;

b. Inspection, and

c. Crack growth.

An example of the implementation of elements of this model was applied to the F-111 following the in-flight failure of a wing pivot fitting in 1969, as detailed in Richey (2005):

Inspection: Low-temperature proof testing was used to reduce the fracture toughness of the components and allow the loading to be limited to 100% (+ve) and 90% (-ve) design limit load. Prior to the low-temperature proof test, a non-destructive inspection of the structure was performed to ensure that the structure did not undergo major failure as a result of a detectable pre-existing defect.

Crack Growth/Residual Strength: If the aircraft did not fail during the proof testing then any defects present were considered to be below critical crack size for that temperature. The time for a crack to grow from the low-temperature critical crack size to the critical crack size at operational temperature and environmental conditions was determined and used to determine the low-temperature proof-testing interval.

4.1.4 - Holistic Structural Integrity Process

NRC-CNRC (2005) states "HOLSIP is a new safe-life paradigm that will allow you to more accurately determine when you need to inspect different components'". As defined by Hoeppner (n.d.) the fundamental tenet of Holistic Structural Integrity Process (HOLSIP) is that all failure modes or mechanisms are interconnected. HOLSIP accounts for the lack of corrosion considerations, which are lacking in safe-life and damage tolerance methodologies used for fatigue life critical components. HOLSIP considers three states of a structure: Initial Discontinuity State (IDS), which represents the initial flaw in the structure, which evolves due to external influences including environment and structural loading. The evolution is termed the Evolving Discontinuity State (EDS) and when the evolution stops there is the Modified Discontinuity State (MDS). Hoeppner (2010) identifies the phases of the holistic design methodology as:

a. Nucleation (*L1*): This phase considers the material failure mechanisms, the presence of discontinuities and the possibility of corrosion, fretting, fatigue, creep and mechanical damage.

b. Small Crack Growth (*L2*): This phase considers structure dominated crack growth and the rate of onset of stress dominated crack growth considering the effects of stress ratio, stress state, environment (temperature, chemical and time) and loading spectrum.

c. Stress Dominated Crack Growth (*L3*).

d. Failure (Fracture) (*L4*).

Komorowski, Forsyth, Bellinger and Hoeppner (2001) identifies that the total life of a structure (*L*) is equal to $L1 + L2 + L3 + L4$.

4.2 - Structure to be Modeled

4.2.1 - Evolution of Aircraft Structures

As identified by Fielding (2004), aircraft structures are designed to optimise a number of conflicting requirements, such as:

a. Minimise weight;

b. Minimise material and manufacturing costs;

c. Adequate strength with safety factor;

d. Minimise distortion, and

e. Minimise structural degradation.

With these conflicting requirements, aircraft structures have evolved significantly since the flight of the Wright Flyer in 1903. The techniques used in structural design started with a space frame where the frame is load bearing and skin provides the aerodynamic covering, such as that used in the Hawker Hurricane fuselage (refer to *Figure **9***), with materials used transitioning from wood to metal to composites.

Figure 9: Hawker Hurricane Cutaway (from Fiddlersgreen (2013))

The next step in structural design was the use of semi-monocoque structures, where both the skin and support structure (stringer, longerons and frames) attached to the skin are all load-bearing. An example of a semi-monocoque structure is presented in *Figure **10***, which illustrates the structure of the Boeing 747-400.

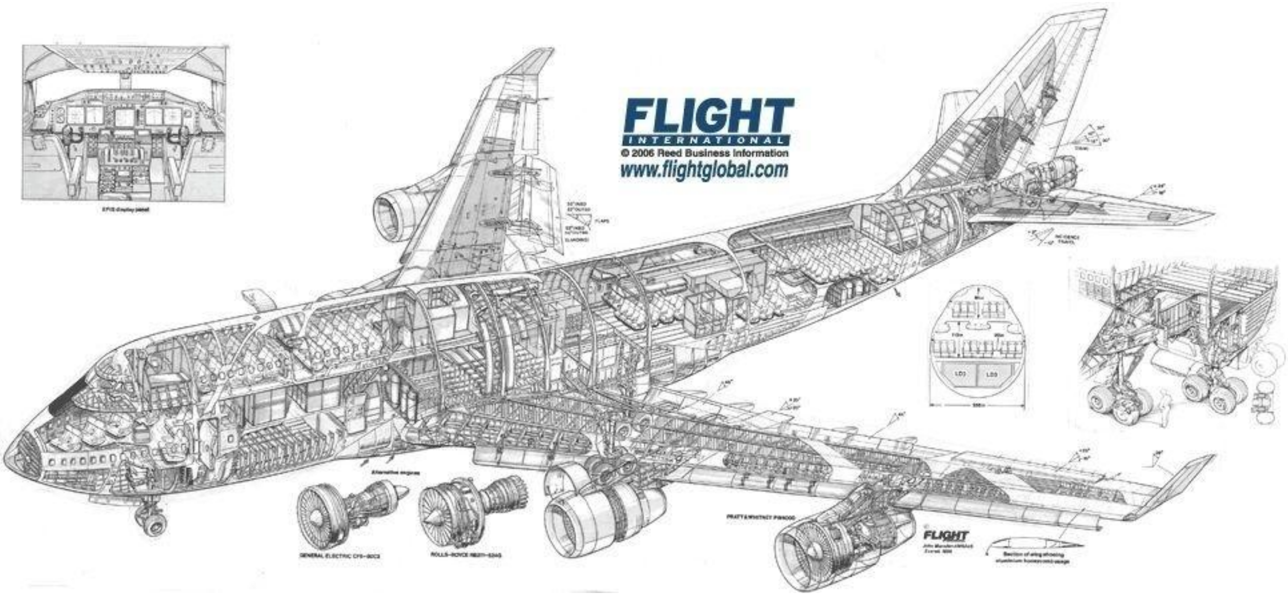

Figure 10: Boeing 747-400 Cutaway Diagram (retrieved from www.flightglobal.com)

4.2.2 - Principal Structural Elements (PSE)

AC 25.571-1D defines a Principal Structural Element (PSE) as an ‘element that contributes significantly to the carrying of flight, ground, or pressurisation loads, and whose integrity is essential in maintaining the overall structural integrity of the airplane.’ PSE includes all structural elements susceptible to fatigue degradation. Examples of PSE include:

a. Engine mounts;

b. Landing gear and attachments;

c. Window frames;

d. Door skins, frames, latches;

e. Pressure bulkheads;

f. Control surfaces, and

g. Spar caps.

4.2.3 - Significant Structural Items (SSI)

ATA MSG-3 defines structural elements as either: Significant Structural Items (SSIs) or as Other Structure. ATA MSG-3 defines a Significant Structural Item (SSI) as “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 safety of the aircraft.” (p.89). Where assemblies are identified as SSI, those elements of the assembly that comply with the definition of SSI must also be included. Therefore, the set of SSI for a given structure will include both PSE and DDP.

4.2.4 - Structure for Simulation

A generic structure shall be used as the Structure of Interest (SoI) for simulation. The SoI will consist of a number of PSE, each of which is represented by a number of DDP. The PSE of the SoI will represent the basic elements of an aircraft such as wing skins (upper and lower), wing spars, vertical tail and fuselage. *Figure **11*** provides a generic representation of the construction of the SoI.

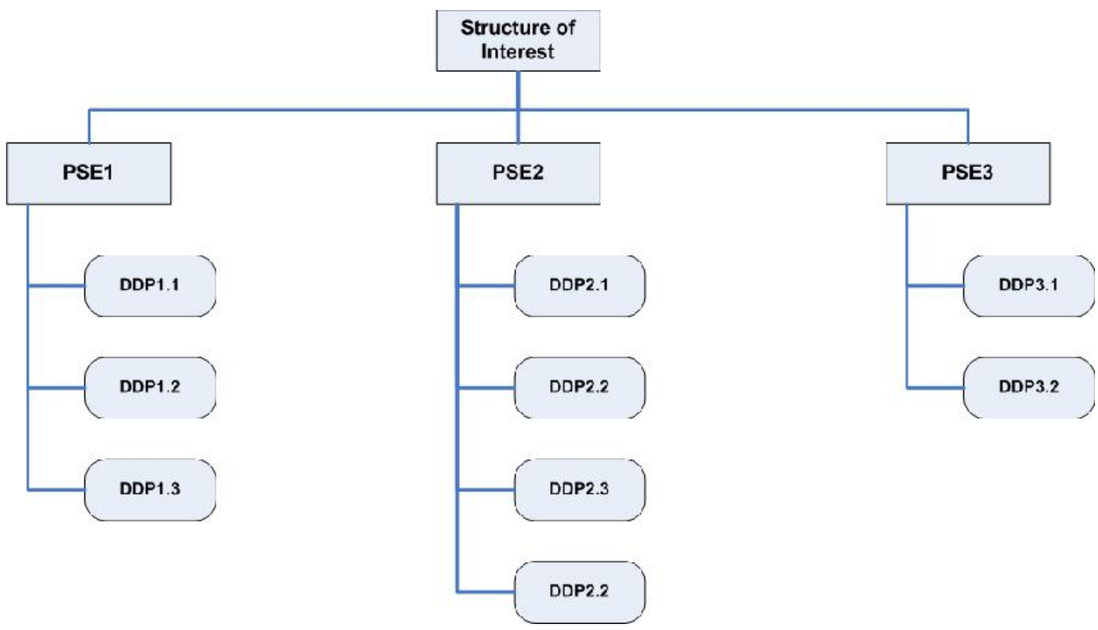

Figure 11: Generic SoI for simulation.

The selection of aircraft structure to be represented in the SoI is performed in a manner to ensure that all representative damage sources and associated probabilities (for initiation, growth and detection) are represented. For example, impact damage from maintenance tools has a higher probability of initiation on the upper as opposed to lower wing surface structure. However, impact damage from runway debris has a higher probability of initiation on the lower as opposed to upper wing surface. To ensure that the probability of structural failure is representative, it is desirable that the number of PSE and DDP be the same order of magnitude as that defined for current aircraft. The details of the SoI used for the simulation will be further discussed in Section 7.

4.2.5 - When does the Structure of Interest fail?

When the damage at a DDP reaches critical size, the available residual strength is less than the applied load and the PSE fails at the DDP. Once a single PSE fails, for a non-redundant structure the entire structure is considered to have failed from a structural integrity perspective.

The use of redundant structure is the foundation of the Fail-Safe ASI methodology and as such, the method for inclusion of redundant structures within the SoI needs to be defined. The following rules shall be used for the implementation of redundant structures within structural integrity simulation (considering that PSE1 and PSE2 are redundant PSE):

a. PSE1 fails, PSE2 intact and SoI remains intact, though the loads on PSE2 increase, and
b. PSE1 fails, PSE2 fails and SoI fails.

4.3 - Modeling Structural Integrity (the SIFCM)

To provide a framework for the benchmarking of current ASI concepts as applicable to composites and to provide contrast between the current state of the art in ASI management for metallic and composite airframes, the concept of the Structural Integrity Failure Causation

Model (SIFCM) was developed. Whereas the model presented by Bachman (2008) is an aircraft lifecycle-centric view of structural safety, the SIFCM presents an alternative viewpoint of damage progression. Where the probability of safe structure (S_T) is related to the probability of damage being initiated, probability of the damage growing to critical size before detection and the probability of the damage being successfully repaired. The relationship between these elements is illustrated in *Figure **12***, which represents an overview of the SIFCM (as applied to each DDP).

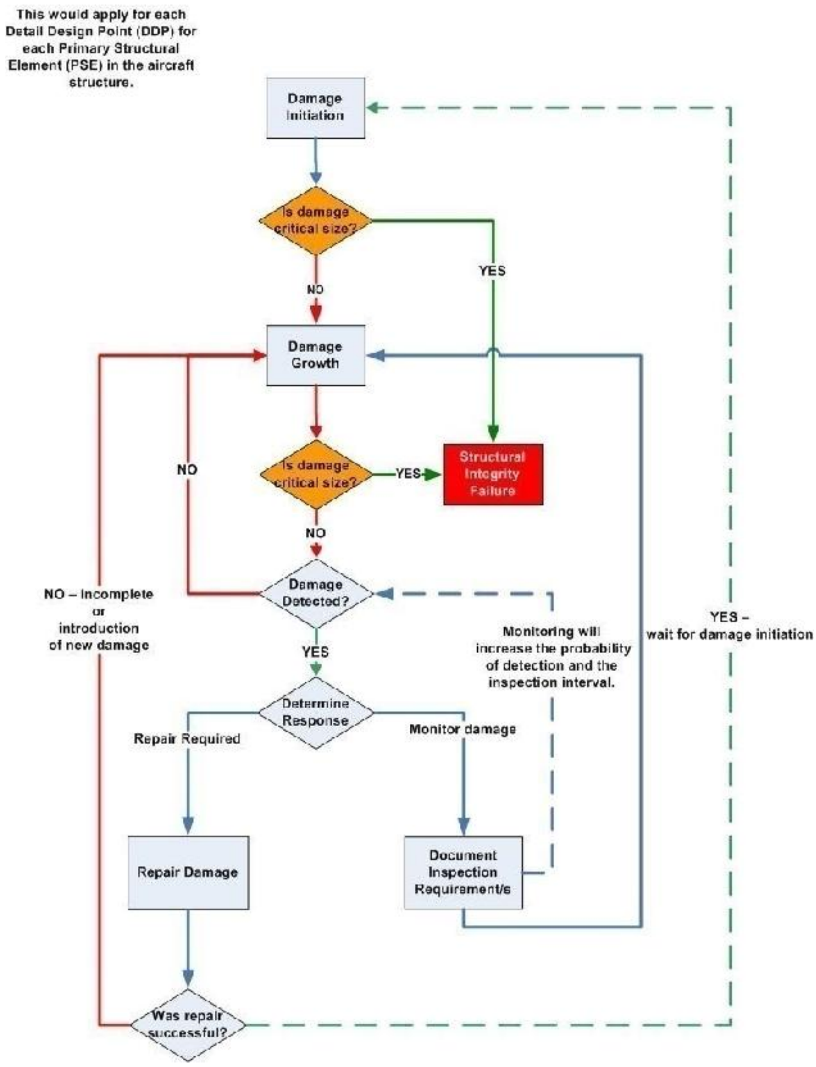

Figure 12: Structural Integrity Failure Causation Model (SIFCM) (from Warren, 2012).

The SIFCM has been developed to be applicable to both metallic and composite aircraft structures, allowing for applicability of ASI methodologies for the two types of structures to be compared.

The SIFCM as illustrated in *Figure **12***, would apply to each DDP of each Principle PSE within the SoI. For a given DDP, the SICFM begins with Damage Initiation, which includes both damage initiated during manufacture (for example, inadequate bonding) and in-service damage (for example, hail impact or corrosion). Instances of inadequate design or regulation would be represented by damage being initiated unexpectedly. For example, the introduction of regulations associated with LOV was to address the unexpected occurrence of widespread fatigue damage in a number of aircraft. Once the damage is initiated (for example hail impact), what is its size? If the damage is greater than or equal to critical damage size, the PSE fails. Otherwise, the damage will continue to grow in size until either the damage is detected or the damage reaches critical damage size at which point residual strength < load at the DDP under investigation, resulting in the parent PSE failing. There could be multiple inspections of the DDP until detection, with the probability of detection increasing as the damage grows in size. Section 5 discusses the various forms of structural degradation and methodology for implementation within the simulation.

Once detected, the damage is either repaired or documented within maintenance documentation (resulting in 100% probability of detection at the next inspection). If the damage is repaired, there is the possibility of the repair being performed incorrectly, as was the case for Japan Air Lines (JAL) Flight 123. JAL Flight 123 impacted with Mt Osutaka in 1985. The primary cause of the accident was an incorrect repair to the rear pressure bulkhead performed seven years earlier. In the case of incorrect repair, the damage continues to grow (though its growth characteristics may have be altered) and its size may have significantly reduced (though not zero) by the repair. If the repair is successful, then the damage is eliminated and the DDP is returned to pristine condition awaiting damage initiation once again. Section 6 discusses damage inspection and repair and methodology for implementation within the simulation.

4.4 - SIFCM Simulation

The next step is to simulate the SIFCM using the Monte-Carlo technique; this simulation is termed SIFCM simulation (SIFCMsim). The aim of the simulation is to determine the probability of structural integrity failure for given criteria and to identify aspects of structural integrity management of composite structures which require further research.

The SIFCM, which is based on a single DDP, is placed within a framework to iterate the SIFCM for each DDP within a SoI. Initially, the structural loading is determined and each DDP is simulated using the SIFCM. Once all DDP have been simulated a check is conducted to determine if the SoI has failed, if the SoI remains intact, then the simulation moves to the next time step (*Ts*) and the process repeats with the determination of the structural loading at the new time step. This process is illustrated in ***Figure 13***, with the application of SIFCM for each DDP in the center of the figure (refer to ***Figure 12***).

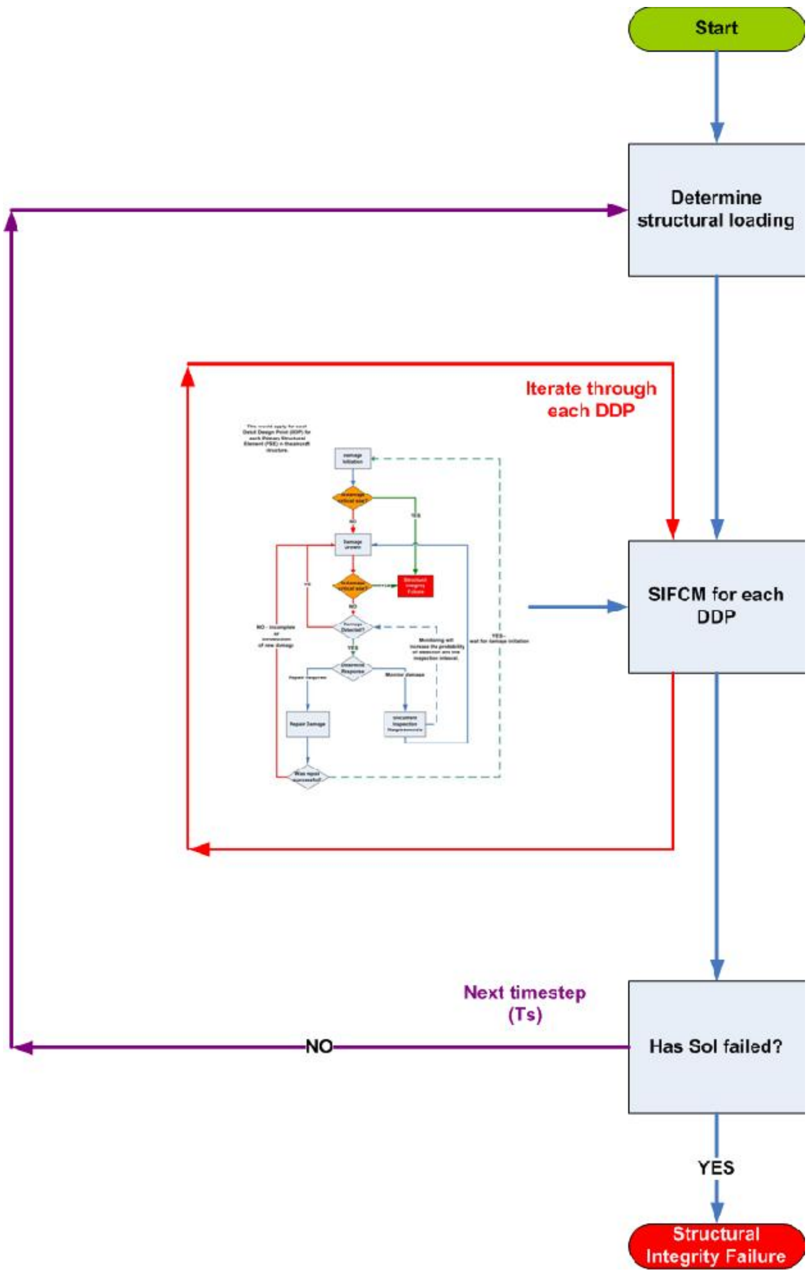

Figure 13: SIFCMsim process.

A phase approach was taken to the implementation of the SIFCMsim concept, with the phases defined by the scope of the SoI being simulated:

a. SIFCM_PSE representing the implementation of SIFCM for a single PSE (with multiple DDP), and

b. SIFCMsim representing the implementation for a generic aircraft structure (multiple PSE).

This approach allowed for the scope of the simulation to be increased gradually, assisting the simulation development process. The further discussion regarding the implementation of these phases is provided at Section 7.

5 - STRUCTURAL DEGRADATION

The term Degradation encompasses both the damage initiation and damage growth blocks in the SIFCM, as highlighted in the *Figure 14*. This section discussed the various forms of structural degradation and the implementation within SIFCM.

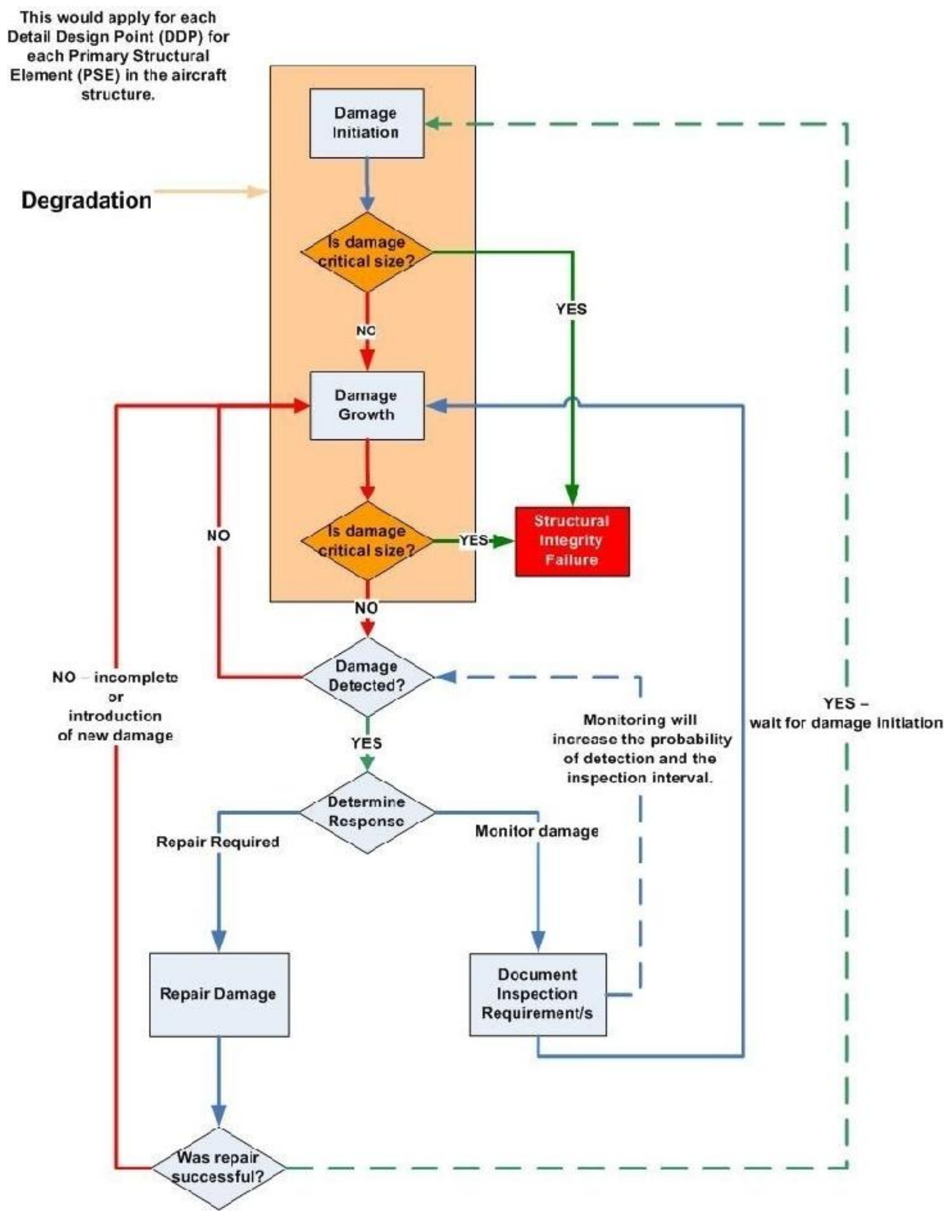

Figure 14: Degradation with SIFCMsim.

The initiation of degradation within aircraft structure can occur both during manufacture and once in service, during the operational phase (e.g. flight loads, exposure to adverse weather conditions, foreign object damage etc) and during maintenance (incorrect repair/maintenance, tool impact etc). In addition, incorrect or inadequate design can result in degradation being initiated when not expected. As such, sources of degradation can be classified as either: Design, Manufacture or Service induced. Table 7 identifies the sources of degradation from a number of accidents associated with the previously identified ASI evolution milestones (Section 1).

Air Transport Association (ATA) MSG-3 states that the sources for structural deterioration are:

a. Accidental Damage (AD), the occurrence of a random discrete event which reduces the residual strength, including human error during the manufacture, operation and maintenance of the aircraft.

b. Environmental Damage (ED), degradation of the structure as a result of the interaction between the structure and chemicals within the climate or environment.

c. Fatigue Damage (FD), cracking (and associated crack growth) as a result of cyclic loading of the structure.

*Figure **15*** illustrates the hierarchy of degradation modes, which will be discussed further in this section.

ASI Milestone	Degradation Source
1954 - de Havilland Comet	Design induced - over estimation of fatigue life due to inappropriate testing. Service induced - flight loads (fatigue).
1958 - Boeing B-47	Design induced - inadequate definition of mission mix and mission profiles. Service induced - flight loads (fatigue).
1969 - General Dynamics F-111	Manufacture induced - flaw in D6AC steel forging. Service induced - flight loads (fatigue).
1977 - Dan Air Boeing 707	Service induced - flight loads (fatigue) Design induced - inadequate damage detection methodology.
1988 - Aloha Airlines Boeing 737	Manufacture induced - tear-strap disbond, resulting in low environmental durability or lack of bonding (NTSB 1989, Finding 9). Service induced - flight loads (fatigue). Design induced - inadequate damage detection methodology.

Table 7: Degradation Sources for ASI Milestones

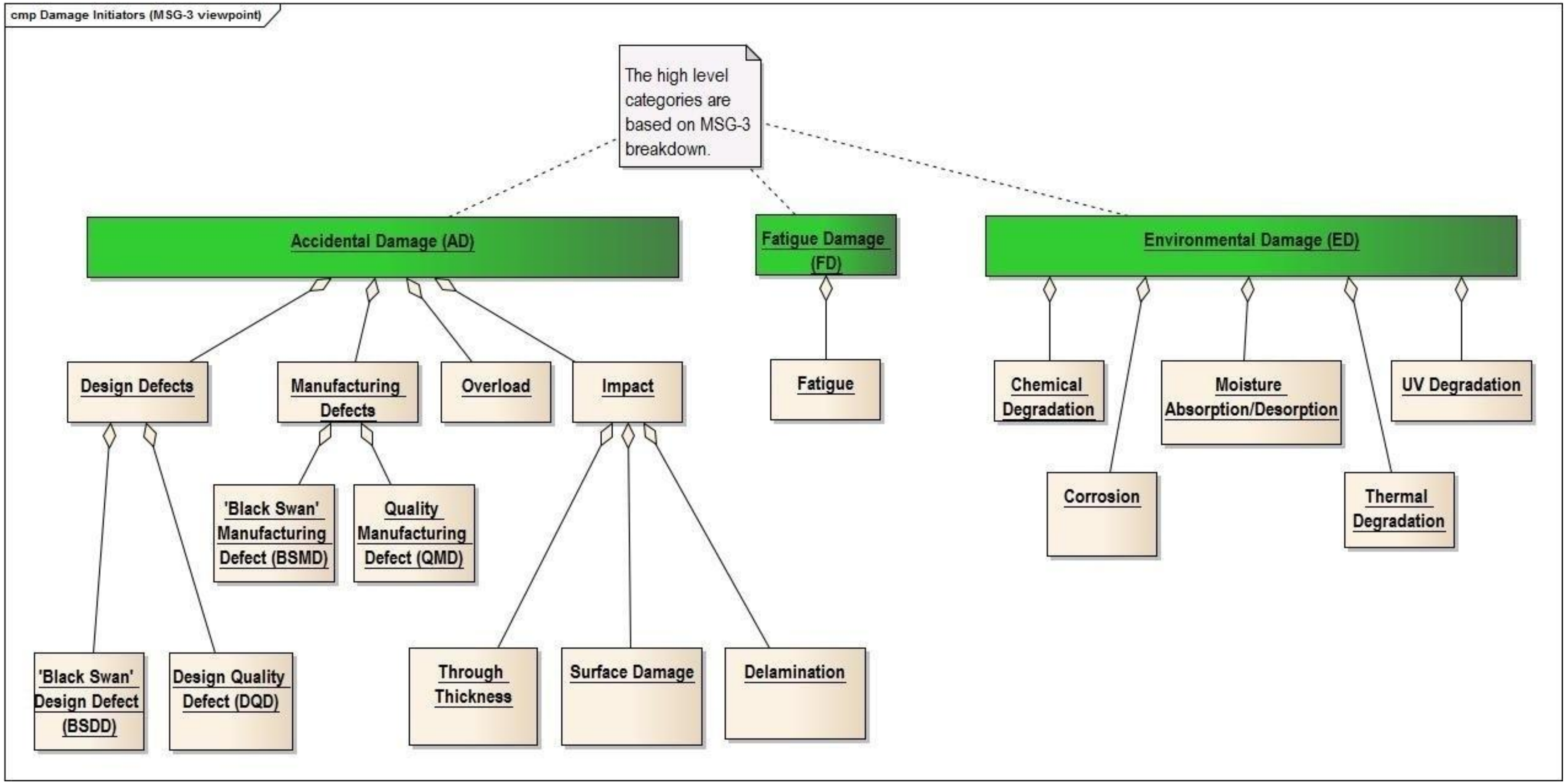

Figure 15: Hierarchy of structural degradation modes

5.1 - Accidental Damage (AD)

5.1.1 - Design Induced Damage

From a structural integrity perspective, the following activities (derived from the MIL-STD-1530C tasks) are included as part of the design process:

a. Design Information;

b. Design Analysis;

c. Full Scale Testing, and

d. Force Management Development.

Therefore, when these tasks are performed inadequately the structure design will not be able to resist the damage induced during manufacture and in-service. This could result in structural integrity failure even if the structure has been manufactured, maintained and operated in accordance with the design.

Design Information

During this activity, the structural integrity criteria to be applied during the design of the structure are established. To a large extent, the outputs from the Design Information task are independent of the structural material with the exception of the selection of materials, processes, joining methods, and structural concepts, which includes the conduct of material screening and selection (Block 1 - BBA). MIL-STD-1530C states that "Prior to a commitment to new materials, processes, joining methods, and/or structural concepts (i.e. those not previously used in the military and/or commercial aviation industry), an evaluation based on their stability, producability, inspectability, supportability, and mechanical and physical properties shall be performed. The risk associated with the selection of the new materials, processes, joining methods and/or structural concepts shall be estimated and risk mitigation actions defined." (p.16).

Therefore, the risks associated with a new material are identified at the beginning of the development process, which allows for numerous approaches to be undertaken mitigating the identified risks. During the 1970s and 80s, composite materials as a whole were relatively new to aerospace structures and risk assessments would have been performed. However, over time a significant repository of composite material property data (for example, MIL-HDBK-17-2F) has been developed which will reduce the perceived need for this risk assessment. This is a concern when composites used for secondary or small primary structures (for example, Boeing 777 vertical stabiliser) are used for large primary structures (for example, Boeing 787 fuselage), without reassessment of the possible risks.

An example of the type of error which can occur during this task applicable to both metallic and composite structures involves the series of accidents involving the Boeing B-47 during 1958. One of the contributing factors was that the aircraft was designed to a different mission mix and mission than was actually being flown.

Design Analysis

During this activity, the operational environment is defined, design analysis and testing of the structure is performed and the structure is sized to achieve strength, durability, damage tolerance and rigidity requirements. As part of this task, the material qualification of new materials is performed via BBA Blocks 2 and 3; material and process specification development and allowables development respectively. FAR 25.613 requirements for determination of material design values, includes reference to environmental conditions (with additional clarification provided by AC 25.613-1 for composites). Anticipated areas of risk within the conduct of this task include:

a. Material Allowables Characterisation, and

b. Design Analysis: During the analysis, the structure is sized to ensure that the stress levels remain within the selected material design values and to analytically determine the fatigue life and response to other in-service damage (which will be discussed in a subsequent section). Inadequate design analysis can have significant impacts on the structural integrity of a structure. For example, the De Havilland Comet fuselage contained a number of square cornered cut-outs which resulted in stresses far higher than initially anticipated during design (Cohen, Farren, Duncan & Wheller, 1955). The structural analysis of metallic structures is relatively mature and inadequate or incorrect analysis is expected to be unlikely. However, the high likelihood of incorrect/inadequate structural analysis of composite structures is one of the reasons the building block approach to testing is so critical for development of composite structures. Baker, Kelly & Dutton (2004) states that the reliability of techniques to predict matrix failure and the growth of delaminations in composites needs to be improved.

Full-Scale Testing

During this activity, the results of the analysis performed during Task 2 are verified. There is a requirement to ensure that the correct individual testing is performed and that the testing is performed in the right sequence when multiple tests are being performed on the same structural component. An illustration of the latter being performed incorrectly is the fatigue testing performed during the development of the De Havilland Comet. As part of the development of the Comet, fatigue testing was performed to demonstrate that the fuselage had an adequate Safe Life, in this case a life of greater than 18,000 flight cycles (or approximately 55,000 AFHRS) was demonstrated. Subsequent fatigue testing during the investigation into the loss of two Comets in-flight resulted in fatigue failure of the fuselage after only 9,000 equivalent AFHRS, significantly less than that previously determined during development.

The cause of this inconsistence was the conduct of a proof test prior to the conduct of the fatigue test on the same fuselage structure. The proof test being performed on an airframe which had undergone proof load testing which Cohen et al. (1955) identified could have increased its fatigue life above that to be expected from a standard production aircraft. This view is supported by Dawicke, Poe, Newman and Harris (1990), which demonstrated that proof testing of metals (in this case Aluminium 2024-T3) can increase its fatigue life due to increased crack closure.

During the development of the General Dynamics F-111, the D6AC test specimens used during the structural certification program were of a different temper to that used in during aircraft manufacture. The temper used for manufacture resulted in a lower fracture toughness (approximately half) that determined during the structural certification program and used during design (Richey, 2005).

In-Service Management

As part of the design activity, any structural integrity management related processes and procedures are developed with the aim of maintaining structural integrity once the aircraft enters service. Examples of the processes and procedures include:

a. Structural Inspection and Maintenance program to determine maintenance inspection intervals and inspection methods.

b. Individual Aircraft Tracking (IAT) program to adjust the maintenance intervals of individual aircraft using damage growth rates predicted from recorded aircraft parameters during flight.

c. Structural Surveillance programs are utilised to improve damage repair estimates and can include such programs as Analytical Condition Inspection (ACI) and Structural Teardown.

d. Loads/Environment Spectra Survey (L/ESS) to record the actual usage spectrum to confirm or update the design spectrum. A limited number of the aircraft fleet (for example 20 percent) will be fitted with sensors to record time-history data such as: strain, acceleration, load factors, pitch and yaw rates and ground loads. Additionally, the thermal and chemical environment environments within the aircraft and basing locations shall be recorded.

5.1.2 - Manufacture Induced Damage

Manufacturing errors during processing, machining and structural assembly of structural components can result in the introduction of adverse residual stress and reduced mechanical properties (for example, fatigue life, yield strength etc). It is for this very reason that the Damage Tolerance philosophy for structural integrity was introduced. This manufacturing induced damage not only occurs during the initial manufacture of the structure but also during repair and modification of the structure. Structures have been manufactured from metals for over 100 years and as such, there is significant industry experience regarding the manufacture of metal structures.

Material Composition

Errors can be introduced during the manufacture of the base material to be used in the construction of the structure, for example during the manufacture of pre-impregnated (pre-preg) composite laminate sheets or the incorrect alloy composition, heat treatment or flaw in a billet for metals. This can result in the selected material not achieve the required material allowables as used during design. Examples of the impact of material composition errors include:

a. United Airlines McDonnell Douglas DC-10 (19th July 1989): during the manufacture of the titanium No. 2 engine fan disk, a cavity formed within a heat-related hard alpha defect from which a fatigue crack propagated resulting in an uncontained engine failure (Job, 1996).
b. General Dynamics F-111A (22nd December 1969): during the forging of D6AC wing pivot fitting a defect was introduced which resulted in premature fatigue failure of the fitting.
c. During the manufacture of the Lockheed SR-71, it was found that certain metal parts were failing due to the presence of chlorine in water being used to quench the components modifying the crystalline structure of the metal (Roskam, 2007).

Structural Element Manufacture

During the manufacture of individual structural elements errors can be introduced which adversely reduce the structural integrity of the overall structure. For example during the manufacture of composite laminates or the forging, machining or casting metals element. Examples of structural element manufacture errors include:

a. American Airlines Boeing 757 on 26th October 2010: chemical milling of a pocket of a fuselage skin panel exhibited a channeling defect, which resulted in reduced skin thickness significantly reducing the fatigue life of the panel (NTSB, 2011).
b. Zhang & Mason (1999) identify that liquid contaminates encountered during manufacture may affect the structure of carbon fibre composite systems by: increasing void content, increasing fibre content and inhibiting matrix (epoxy) bonding during B-stage cure (the partial cure used to manufacture pre-preg laminate sheets).

Assembly Mishaps

During the assembly of structural elements to form the structure mishaps can occur, for example:

a. Japan Air Lines Boeing 747 (12th of August 1985): a structural repair to the rear pressure bulkhead was performed incorrectly using two separate doublers instead of a single doubler plate, resulting in only one as opposed to two rows of fasteners (as required by the repair design) joining the splice repair. The reduced number of rivet rows, reduced the fatigue life of the pressure bulkhead by approximately 70% (Job, 1996).

b. Failure of adhesive bonding used on Boeing 737 aircraft manufactured prior to April 1972 (up to production number 291) for fuselage skin lap joints. The failure of the adhesive increased the stresses on the joint rivets, reducing fatigue life (Job, 1996).

c. During production acceptance test flight of Cessna Corvalis, approximately seven feet of wing skin disbonded from the wing upper spar (AD 2010-26-53). This disbonding was identified by the FAA to be the result of manufacturing errors (FAA, 2011).

Accidental Damage From Dropped Tools/Other Equipment

Accidental damage from dropped tools/other equipment is predominately the result of impact with the structure, which can result in dents, scoring and penetrations. Therefore, details of this damage type are discussed in the 'In-Service' section. An example of this type of damage initiation is that of American Airlines Douglas DC-6 in August 1950, where scoring of the interior surface of the No.3 engine propeller blade during manufacture was the origin of fatigue cracking which resulted in part of the propeller separating during flight (Roskam, 2007).

Manufacture Induced Damage Summary

The damage can be induced during manufacture of composite structures via a number of mechanisms, such as those identified by Backman (2005): laminate processing mishaps, co-bonding/co-curing mishaps, errors during installation, assembly mishaps and damage from dropped tools and other equipment. AC 20-107B Change 1 states "One of the unique features of composite construction is the degree of care needed in the procurement and processing of composite materials. The final mechanical behaviour of a given composite material may vary greatly depending on the processing methods employed to fabricate production parts." (p.2). Structures have been manufactured from metals for over 100 years and as such, there is significant industry experience regarding the manufacture of metal structures. Therefore, whilst errors during manufacture are applicable to both metal and composite structures, it is considered more probable that significant errors will occur during the manufacture of composite structures.

5.1.3 - Structural Overload

When a structure is exposed to loads higher than ultimate design loads, the structure will fail as a result of overload. This type of damage initiation is common to both composite and metallic structures. Examples of this type of failure include:

a. American Airlines Flight 587 Airbus A300 (12th November 2001): overload of the composite vertical stabiliser in response to wake turbulence correction resulted in separation of the vertical stabiliser from the aircraft (NTSB, 2004).
b. Braniff International Airways BAC-111 (6th August 1966): encounter with an extreme wind gust resulted in the aircraft tail-plane failing whilst passing through a storm front (Job, 1994).

5.1.4 - Impact

During the service life on an aircraft, the aircraft structure will endure impacts from a variety of sources, including: foreign object damage (for example, runway debris), bird strikes, hail, stones (for example, thrown up by undercarriage during take-off/landing) and maintenance impacts (for example, dropped tools).

The Federal Aviation Administration (2002) (DOT/FAA/AR-01/55) acknowledges that metals can absorb impacts via plastic deformation. Avery (1981) identifies that impact can cause the following damage in metal structures depending on the structure thickness, projectile energy and angle of impact with the structure: dents, gouging, cracks, holes, petals, and spallation. These forms of damage can be classified as either surface damage or through-thickness damage.

Unlike metals, composites fail as a brittle material and therefore as stated by Baker et al. (2004) composite structures are extremely sensitive to impact damage, even when the impact damage is barely visible, termed Barely Visible Impact Damage (BVID). DOT/FAA/AR-01/55 (2002) identifies that low (impact velocity < 6-8 m/s) and mid-speed (< 30 – 200 m/s) impacts can cause:

a. Surface damage, for example scratches/gouges, fracture notch (*Figure **16***(a)).

b. Delamination, which can be split into the following two sub-categories:

   i) BVID, internal delaminations that are visually undetectable from the surface of the structure (*Figure **16***(b)). This is followed by matrix cracking on the side opposite the initial impact (*Figure **16***(c)).

   ii) Surface Delaminations, which appear at the impact surface due to cracking and fragmentation and are therefore visually detectable on the impact surface (*Figure **16***(d)).

   iii) For both BVID and Surface Delaminations, the actual damage size in the interior of the composite may be significantly greater than the damage size visible on the surface of the structure.

c. Through thickness damage, for example cracks and punctures. The damaged area is through the entire thickness of the composite structure consists of either clean holes (*Figure **16***(e)) or punctures which includes delaminations and sharp cracks in the damage area (*Figure **16***(f)).

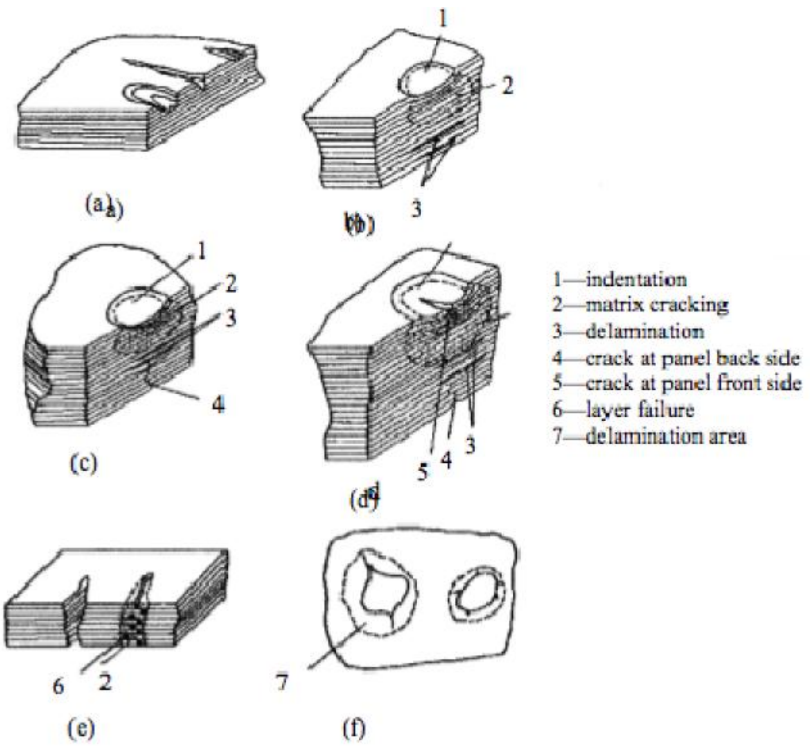

Figure 16: Types of In-Service Damage (from DOT/FAA/AR/01-55, 2002).

The Federal Aviation Administration (1997) (DOT/FAA/R-96/111) reports on research conducted related to the certification of composite materials in aircraft structures. As part of this work, the sources of in-service impact damage and the characterisation of effect of structural strength and probability of occurrence were investigated. This work classified impact damage into three threat levels: high, medium and low, with each threat level having a unique Probability Density Function (PDF) and representative modal impact energy level (Xm), as illustrated in *Figure **17***. The probability of occurrence of a 100 ft-lb or higher energy (p(100)) is also identified in *Figure **17***. *Figure **18*** provides the cumulative probability of occurrence for the three threat levels against impact energy.

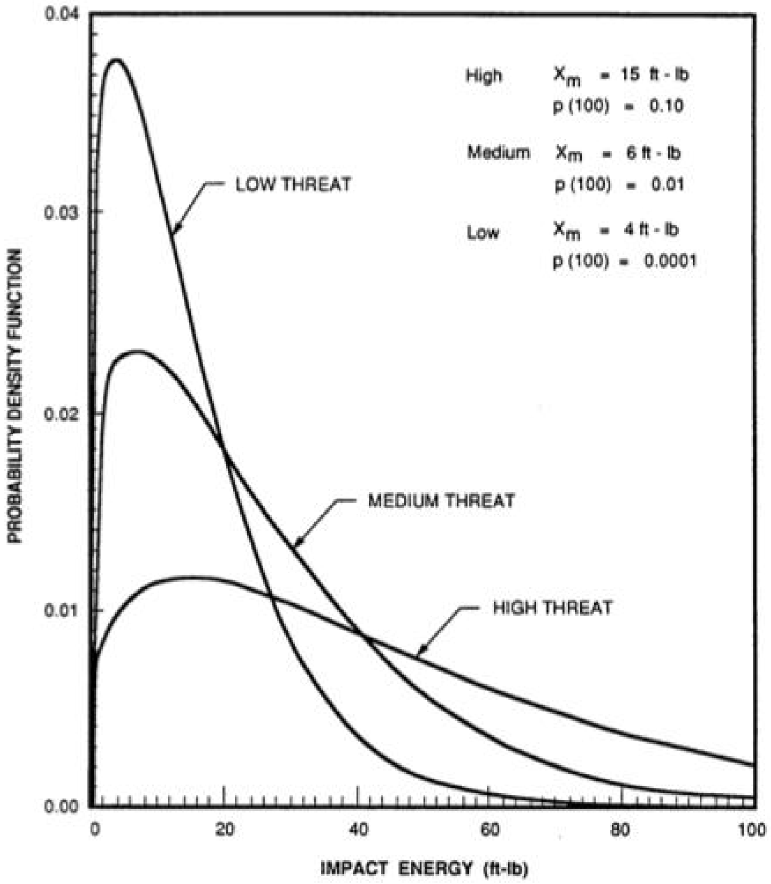

Figure 17: Probability density function for various impact threats (from DOT/FAA/AR-96/111, 1997).

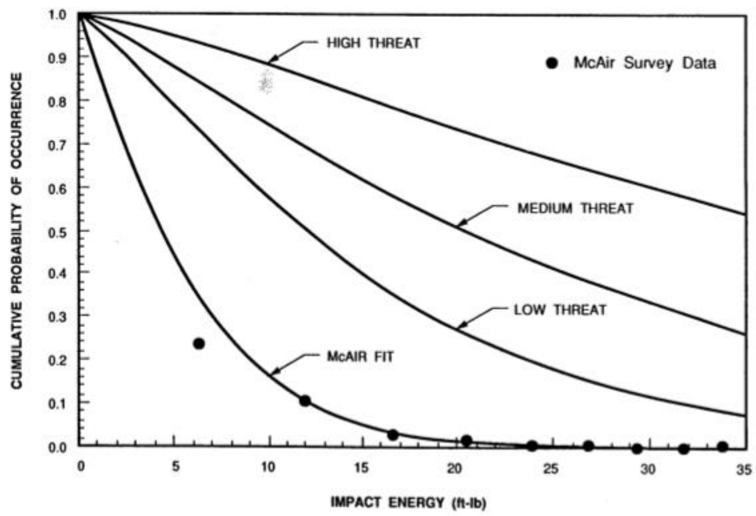

Figure 18: Cumulative probability of occurrence versus impact energy for various impact threats (from DOT/FAA/AR-96/111, 1997).

FAA (2002) (DOT/FAA/AR-01/55) reports on the efforts to develop a methodology for establishing the reliability of composite aircraft structures and included statistical analysis of historical impact damage on a number of different Russian aircraft types, including SU-29, TU-204 and Lear Fan-2100. The analysis of historical impact damage identified three groups of impact damage (I, II and III) and provided an approximation equation to describe the damage occurrence rates for:

a. delaminations;
b. crack + hole, and
c. all damage (surface scratches + delaminations + crack + hole).

The approximation equation for the exceedance intensity is $H(2L) = H_0.e^{(-2L/b)}$, where H_0 and b are in units 1000 AFHRS per sq. m and identified in the *Table 8* and $2L$ (mm) is the length of the damage.

The (Crack + Hole) damage type maps to the previously identified through thickness damage, with the delamination types equivalent and the previously identified surface damage being equivalent to (All Types of Damage) minus Delamination minus (Crack + Hole).

5.2 - Environment Damage (ED)

5.2.1 - Chemical Degradation

For metal structures, the primary chemical degradation is that of corrosion which is further discussed in the corrosion section.

Group	Unit	H_0			b		
		All Types of Damage	Delaminations	Crack + Hole	All Types of Damage	Delaminations	Crack + Hole
I—High damage rate	Maneuverable airplanes: • wing leading edges • engine fairings • vertical stabilizer actuator fairings • nose landing gear fairings, • nose landing gear doors	3.6	1.6	0.97	38	64	58
II—Intermediate damage rate	Maneuverable airplanes • air inlet channels • lower vault panels	1.6	0.43	0.81	28	22	27
III—Low damage rate	Maneuverable airplanes • vertical stabilizer skins • wing skins Low-maneuverable airplanes • wing/fuselage fairings • landing gear doors • cargo bay doors	0.33	0.27	0.13	56	69	55

Table 8: Recommended values for H_0 and b (from DOT/FAA/AR-01/55, 2002).

The chemical degradation (which may include material loss or thinning as discussed for corrosion) of composite structures can occur as a result of a number of chemical reactions. Allara (1975) identifies a number of chemical reactions which can result in the degradation of a polymer over time, such as thermal and photo-oxidation, hydrolysis (attack of water on esters/amides and acetals in the polymer) and reaction with sulfur and nitrogen oxides.

There is a significant body of evidence to suggest that graphite/epoxy composite systems are relatively immune from chemical attacks as illustrated by:

a. Thompson, White and Snide (1985) state that graphite/epoxy composite systems are resistant to chemical attack.

b. Rider & Yeo (2005) state that the effect of fuel on adhesive joints is difficult to determine, with the possibility of joint durability decreasing when exposed fuel environments for thousands of hours.

c. Broughton, Lodeiro & Mulligan (2000) states that carbon fibers exhibit excellent chemical resistance and can be expected to remain unaffected by exposure to water, weak acids and weak alkalis at elevated temperatures.

For any new composite system planned for use, MIL-HDBK-17-3F identifies the requirement for performing fluid screening of any new polymer material (matrix) and states that epoxy are fairly resistant to chemical degradation with the exception of methylene chloride. Thus, in general, it is considered unlikely that current graphite/epoxy composite systems used in aircraft structures will suffer from significant chemical degradation.

5.2.2 - Corrosion Initiated Damage

Grandt Jr (2004) defines corrosion as “material degradation due to chemical attack.” (p.15) with Craig and Pohlman (1987) identifying the following forms of corrosion with further description provided by Grandt Jr (2004):

a. General corrosion: results in wide area degradation (or thinning) of a structural component. Causes of this form of corrosion include oxidation and galvanic corrosion.

b. Localised corrosion: results from high rates of chemical attack in small localised areas which results in corresponding localised areas of damage, for example pitting corrosion.

c. Metallurgical influenced corrosion occurs as a result of metal alloying composition and heat treatment, for example intergranular and exfoliation.

d. Mechanically assisted degradation, involves the wear or fatigue of structural component combined with chemical attack, for example erosion, cavitation and corrosion fatigue.
e. Environmentally induced cracking, for example stress corrosion cracking.

Cole, Clark and Sharp (1997) identify the following forms of corrosion as affecting structural integrity of an metallic aircraft structure pitting corrosion, intergranular corrosion, exfoliation, stress corrosion cracking, corrosion fatigue and uniform corrosion. Cole et al. (1997) states that these forms of corrosion are able to:

a. Reduce the load carrying capacity (residual strength) of a structure via the reduction of structural material cross-section;
b. Become initiation sites for fatigue crack growth, and
c. Increase the crack growth rates for existing fatigue cracks (corrosion fatigue).

It can be seen that the impact of corrosion on the structural integrity of a metallic aircraft structure is multifaceted, which results in a complex failure mechanism, which is independent of aircraft usage as identified by Grandt Jr. (2004). Using the corrosion definition provided by Grandt Jr (2004) and the taxonomy for corrosion provided by Craig and Pohlman (1987) composites are susceptible to:

a. General corrosion as a result of contact with hostile environments;
b. Localised corrosion as a result of corrosive material pooling in matrix surface defects;
c. Mechanically assisted degradation such as erosion, and
d. Environmentally induced cracking, such as stress corrosion cracking at elevated temperatures in glass-fibre (E-glass) composite systems, as identified by Kajorncheappunngam (1999).

Examples of corrosion related structural failure include, British Airways BAe/Aerospatiale Concorde 102 (12th April 1989): moisture ingress into the upper rudder section resulted in corrosion between the rudder skin and honeycomb, deterioration in bond strength ultimately resulting in delamination between the skin and honeycomb and separation of a portion of the upper rudder from the aircraft (AAIB, 1989).

5.2.3 - Absorption

Absorption is the incorporation of a substance in one state (for example, liquid) into a substance of another state (for example, solid). Moisture absorption does not occur in metals.

Baker et al. (2004) states that thermoplastics absorb a relatively small amount of moisture (approximately 1% by weight) whereas epoxy resins absorb over 4% moisture by weight. It is usually assumed that the fibres don't absorb moisture, whilst this is true for carbon, boron and glass, it is not valid for polymeric fibres such as aramid. Moisture absorption affects the material properties of the matrix, including glass transition temperature (T_g) and Young's Modulus (E). As an example, for AS4/3501-6 carbon/epoxy composite system, T_g varies from over 160 degrees C (<20% relative humidity) to 140 degrees C (>60% relative humidity) as presented by Baker et al. (2004). As expected, the matrix dominated composite properties are influenced by the absorption of moisture.

Todo, Nakamura & Takahashi (2000) investigated the effects of moisture absorption on the dynamic mode II interlaminar fracture toughness (G_{IIC}) of T800H/2500 and T800H/3631 carbon composite systems:

a. For T800H/2500, depending on the wet conditions (80°C-wet or 80°C-90% relative humidity), G_{IIC} decreased as a result of fibre/matrix interfacial degradation or increased due to plasticisation of the matrix which dominated the interfacial degradation, respectively.
b. For T800H/3631, both wet conditions resulted in a decrease in G_{IIC}, which was considered to be the result of fibre/matrix interfacial degradation.

Todo et al. (2000) observed that Langmuir-type two phased model predicted the moisture absorption behaviour for both composite systems and identified that the moisture content of a composite system may be represented by the following equation:

$$M(t) = G(t)(M_m - M_i) + M_i$$

Where M_m is the maximum moisture content of the composite system,

M_i is the initial moisture content, and

$G(t) = 1 - \frac{\alpha}{\alpha+\beta} . exp^{(-\beta.t)}$

Baker et al. (2004) states that for many composite systems, the maximum moisture uptake (absorbed) is related to humidity by the following equation:

$$\text{Maximum moisture update} = k.\Phi^n$$

Where Φ is the relative humidity (in %), k is a constant and n is close to 1 for many aerospace composite systems. It should be noted that it can take considerable time (hundreds of weeks) for moisture absorption to reach equilibrium.

Baker et al. (2004) states that glass fibre composite systems subjected to moisture are prone to weakening of the fibre/matrix interface as a result of chemical attack. Aramid fibre composite systems also suffer from reduction in material properties, for example, 5% decrease in tensile strength and stiffness at 4% moisture absorption.

Moisture absorption can result in swelling of the composite material which can introduce additional stresses into the structure, though these may be offset to some extent by residual stress in the composite as a result of manufacture, Baker et al. (2004).

It can be seen from the previous discussion that the effect of moisture absorption on the properties of a composite system can vary depending on the type of composite system exposed. This emphasises the importance of performing material design property characterisation as outlined in FAA Advisory Circular AC 20-107B Change 1 (para 6.d).

5.2.4 - Desorption

Moisture absorption by composite systems at initial glance does not appear to be damage mechanism per se. Baker el al. (2004) states that composite properties appear to return to normal values when absorbed moisture is desorbed. However, there appears to be little research regarding the repeated absorption and desorption of water or other contaminates. Specifically, do contaminates remain within the matrix and degrade the matrix properties as a build up occurs?

5.2.5 - Thermal Effects

The effects of temperature on materials are outlined in the following list:

a. Creep, as described by Ashby & Jones (1980) is a slow continuous deformation with time where the strain is dependent on temperature in addition to stress:

$\varepsilon = f(\sigma,t,T)$ creeping solid or 'high' temperature behaviour

Where ε is the strain, σ is stress, t is time and T is temperature. The room temperature behaviour of most metals and ceramics, which is described by:

$\varepsilon = f(\sigma)$ elastic/plastic solid or 'low' temperature behaviour

b. Oxidation, as described by Ashby el al. (1980) is the process when a material reacts with oxygen to produce an oxide, for example the formation of iron oxide (blackening) on the surface of iron when sufficiently heated or the ignition of polymers if burnt.

c. Material property degradation, as the temperature approaches the melting (metals) or glass transition (polymers) temperature the Young's Modulus (E) of the material reduces as the material begins to transition from a solid to liquid phase.

Thermal effects on metal

Creep

Ashby et al. (1980) indicate that as a general rule, if a metal can be limited to being heated to temperatures less than 0.3 to 0.4 of its melting (T_M) the effects of creep can be avoided.

$$T_{creep} = 0.3 \text{ to } 0.4\ T_M$$

For example, using a factor of 0.3 (to be conservative):

a. Tungsten - T_M = 3680K, T_{creep} = 1104K;

b. Zinc - T_M = 692K, T_{creep} = 208K, and

c. Mercury - T_M = 235K, T_{creep} = 71K.

Oxidation

The degradation of material caused by Oxidation is considered under corrosion.

Material property degradation

In general, as temperature increases the strength and stiffness of metals decreases, as indicated in MIL-HDBK-5H. For example:

a. Magnesium alloy (AZ31B-O sheet), at 260 deg C, Young's modulus (E) decreases to 64% room temperature value.

b. Aluminium alloy (2014-T6), at 260 deg C (after 0.5hrs exposure), ultimate tensile strength (F_{tu}) decreases to 32% room temperature value.

c. Titanium alloy (Ti-6Al-4V), at 260 deg C (after 0.5hrs exposure), F_{tu} decreases to 72% room temperature value.

Thermal effects on composites

Glass transition temperature (T_g) for polymers is the temperature where the polymer begins to transition from a solid to liquid phase. In general, as the temperature increases the strength begins to decrease. Whilst, increased temperature can adversely effect the strength of fibres, this generally occurs at a much higher temperature than the glass transition temperature for the matrix.

Creep

Ashby el al. (1986) describes the process by which polymers transition from a glassy state at low temperature to a visco-elastic state as T_g is approached and exceeded. With a subsequent 'rubbery' phase, viscous phase[13] (> $1.4T_g$) and finally decomposition where polymerisation breakdown occurs. Polymers are used as matrix material (for example, epoxy) and as a fibre (for example, Kevlar). Thus, creep can occur with composites, with Baker et al. (2004) stating that for unidirectional composites, creep is a fibre-dominated property as strain in the longitudinal direction is a fibre-dominated property.

Oxidation

The degradation of material caused by Oxidation is considered under corrosion.

Material property degradation

In general, as the temperature increases the strength begins to decrease, as indicated in MIL-HDBK-17-2F. For example:

a. T-500 12k/976 unidirectional tape, F_1^{tu} [14] (at 24 deg C) decreases 93% at 121 deg C.
b. T-500 12k/976 unidirectional tape, F_2^{tu} [15] (at 24 deg C) decreases 77% at 121 deg C.

[13] Thermo-plastics are moulded at this temperature.
[14] Ultimate tensile strength parallel to fibres, fibre dominated property.

MIL-HDBK-17-3F provides guidance regarding the required margin, usually 10 deg C between wet T_g of the composite system and the use temperature of the structure. The use of wet T_g is considered the worst case due to the time required for the composite to become fully saturated. The decrease in T_g with moisture absorption is considered to be reversible.

5.2.6 - Ultraviolet Degradation

Metals are not affected by ultraviolet (UV) radiation.

The matrix resins of composites are damaged by UV radiation breaking down the chemical bonds of the matrix, as stated by Baker et al. (2004). Kumar, Singh and Nakamura (2002) state that the UV photons are absorbed by the polymers, as the photons have similar energy to the dissociation energies of the polymer covalent bonds. Resulting in photo-oxidative reactions which can cause:

a. Molecular chain scission, which lowers molecular weight, reducing strength and heat resistance;
b. Chain crosslinking, which increases brittleness and can lead to microcracking, and
c. Discolouration in the polymer due to the production of chromophoric chemical species.

Epoxy resins are more susceptible than polyester based resins. This damage can be avoided by painting the exposed surfaces of the composite structure to minimise exposure to UV radiation or by the addition of UV absorbers as additives to the matrix resin. However, as described by Shokieh & Bayat (2007), possible long term UV degradation of the additive remains an unknown. Shokieh el al. (2007) performed testing on glass/polyester composite system specimens, which indicated that there is degradation due to UV exposure in resin tensile and shear mechanical properties. Kumar el al. (2002)

[15] Ultimate tensile strength transverse to fibres, matrix dominated property.

investigated the influence of UV radiation and/or condensation on carbon fibre composite system. The investigation revealed that the transverse (matrix dominated) properties were degraded after UV radiation exposure. For example, after 500 hours of exposure, transverse tensile strength was decreased by 9%.

5.3 - Fatigue Damage (FD)

Fatigue cracking occurs as either the continued growth of a pre-existing crack (which may be the result of manufacturing defects or in-service damage) or as the result of very small cracks forming in the material as a result of local yielding at the microscopic level, eventually coalescing to form a single larger crack.

It is generally considered that fatigue is not as much of a concern for composites as it is for metallic structures, this is particularly true when the load is taken by the fibres as opposed to the matrix. As described by Baker el al. (2004), the mechanism for fatigue crack growth within a composite is composed of the following damage mechanisms:

a. Matrix cracking;
b. Crack coupling and delamination, and
c. Fibre breakage.

The fatigue behaviour for composites is affected by the direction of the loading, as discussed by Baker et al. (2004):

a. During tensile cyclic loading, the matrix in 90 degree plies crack, with off-angle plies becoming ineffective until the 0 degree fibres fail due to overload.
b. During compression cyclic loading, micro-buckling of the fibres can occur, leading to crack growth. However, micro-buckling should be suppressed under compressive cycling doe to the presence of the matrix, reducing the crack growth in undamaged structure.

c. During tension/compression cyclic loading, the damage caused to the matrix and matrix/fibre interface during tension reduces the ability of the matrix to reduce micro-buckling of the fibres during compression.

Smith and Wilson (1985) investigated the effects of various types of damage on the growth of fatigue damage in a AS6/2220-3 composite panel. Five types of damage were considered: delaminations from a 31.63 N.m (280 in-lb) and 56.5 N.m (500 in-lb) impacts, open hole with delamination damage, delaminations resulting from 56.5 N.m (500 in-lb) impact in through-stitched laminate and multiple simulated impact damage (using nine Teflon discs). Little or no damage growth was observed in specimens subjected to spectrum fatigue (truncated Boeing 767 cyclic spectrum) even when the maximum cyclic load was 80% of ultimate load. Of the types of damage considered, only those specimens containing either simulated or actual delaminations around an open hole displayed damage growth as a result of constant amplitude fatigue. *Figure **19*** provides an illustration of the fatigue damage growth observed for a 31.63 N.m (280 in-lb) impacted specimen subjected to constant amplitude (R=10.0) cycling.

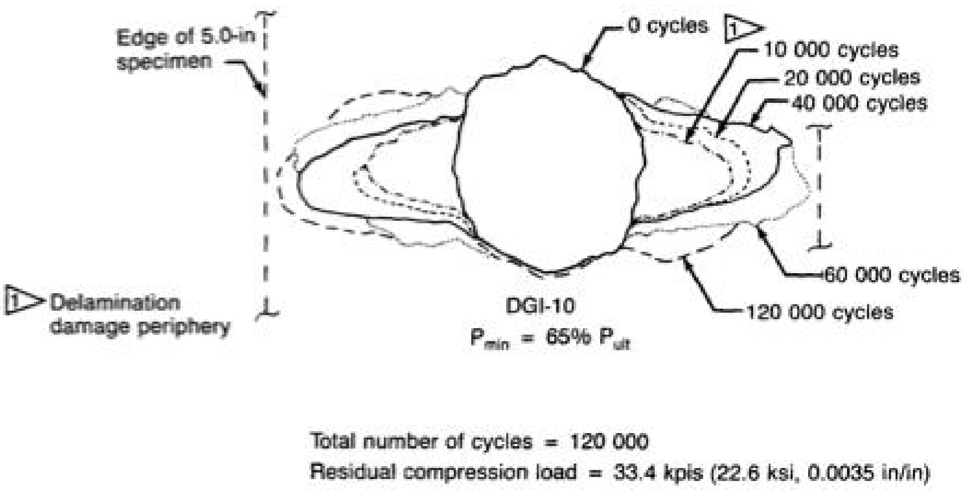

Figure 19: Damage growth for 31.63 N.m (280 in-lb) impacted specimen (from Smith and Wilson, 1985).

Kassapoglou (2009) identified the following expression for calculating the residual strength as a function of cycles (with the endurance limit set to zero):

$$\sigma_r = \sigma_{fs}\left(\frac{\sigma}{\sigma_{fs}}\right)\left(\frac{n}{N-1}\right)$$

Where σ_r is residual strength

σ_{fs} is static strength

σ is the applied load

n is the current number of cycles, and

N is the number of cycles to failure for the applied load σ.

5.4 - Simulation of Degradation

This section outlines the mechanism used to simulate the various forms of structural degradation within the SIFCM.

5.4.1 - Accidental Degradation Simulation

Model of Design Induced Degradation

Design induced damage is considered to be composed of damage of two parts. One is the result of breakdown in the quality assurance processes (design quality) used during design. Examples of design quality defects include: not following best practise, failure to comply with design regulation and error during design process (including incorrect assumptions). The second form of design induced damage is the result of an unanticipated/unknown design considerations. Examples includes: incorrect or inadequate regulation or other unknown design considerations and damage which is the result of an unanticipated/unknown design effect. These two aspects can be represented as follows:

a. ***Design Quality Defect:*** This defect is represented by a probability of occurrence (likelihood) and Reduction in Residual Strength (RRS):

RRS = Base RRS * random number (between 0 and 1), and
Likelihood = For upper and lower 10% of probability distribution function (i.e. when random number is < 0.1 and > 0.9).

Where Base RRS is a variable with an initial value of 0.02, based on engineering judgment of the author.

b. ***Black-Swan Design Defect***: Presence of defect is considered low likelihood (probability). The value of this probability of defect is based to a certain extent with the experience and knowledge regarding the material system and design tools. This probability for metallic structures would be significantly less than that for composite systems. Likewise it would be anticipated that the effect of the defect on the residual strength of the structure at manufacture and through life, would be less for metallic than composite structures.

This defect is represented by a probability of occurrence (likelihood) and RRS:

RRS = Base RRS * random number (between 0 and 1), and
Likelihood = For upper and lower 1% of probability distribution function (i.e. when random number is < 0.01 and > 0.99).

Where Base RRS is a variable with an initial value of 0.1, based on engineering judgment of the author.

Model of Manufacture Induced Degradation

Manufacture Induced Damage is considered to be composed of damage as the result of breakdown in the quality assurance processes used during manufacture and damage and is the result of an unanticipated/unknown manufacture effect. Examples include the defect in the wing pivot fitting which resulted in the loss of an F-111 in December 1969. These two aspects can be represented as follows:

a. ***Manufacture Quality Defect:*** This defect is represented by a probability of occurrence (likelihood) and RRS:

RRS = Base RRS * random number (between 0 and 1), and
Likelihood = For upper and lower 10% of probability distribution function (i.e. when random number is < 0.1 and > 0.9).

Where Base RRS is a variable with an initial value of 0.01, based on engineering judgment of the author.

b. ***Black-Swan Manufacture Defect:*** Presence of defect is considered low likelihood (probability). The value of this probability of defect is based to a certain extent with the experience and knowledge regarding the material system and design tools. This probability for metallic structures would be significantly less than that for composite systems. Likewise it would be anticipated that the effect of the defect on the residual strength of the structure at manufacture and through life, would be less for metallic than composite structures.

This defect is represented by a probability of occurrence (likelihood) and RRS:

RRS = Base RRS * random number (between 0 and 1), and
Likelihood = For upper and lower 1% of probability distribution function (i.e. when random number is < 0.01 and > 0.99).

Where Base RRS is a variable with an initial value of 0.2, based on engineering judgment of the author.

Model of Overload Failure

Overload is not considered further within the SIFCM, as once this form of damage is initiated, failure occurs as the damage reaches critical size bypassing the opportunity for damage detection prior to failure.

Model of Impact Damage

Three types of impact damage are modeled for the various PSE/DDP, which compose the SoI: Surface Damage, Laminate Damage and Through Thickness Damage. The exceedence rates (used to determine probability of impact) are calculated using the parameters identified in Table 8, dependent on the type of structure (Group I, II or III) with the following corresponding reduction in residual strength:

a. Surface Damage - (All Types of Damage) minus Delamination minus (Crack + Hole): RRS = 0;

b. Laminate Damage - Delaminations: RRS = 0.05, and

c. Through Thickness Damage - Crack + Hole: RRS = 0.2.

These values of RRS were selected based on engineering judgment of the author.

5.4.2 - Environmental Degradation Simulation

Chemical Degradation

As discussed in Section 5.2.1, graphite/epoxy composite systems are highly resistant to chemical attack and MIL-HDBK-17-3F directs that fluid screening be performed for any new polymer (matrix) material to determine sensitivity to chemical degradation. Due to these factors, chemical degradation is not modeled within the SIFCM.

Corrosion

The effects galvanic corrosion between composite and metal structures can be eliminated via good design practice. The simulation currently only considers all-composite structure, therefore the effect of corrosion is not modeled within the SIFCM.

Moisture Absorption/Desorption

Absorption

The percentage of moisture absorbed is calculated by using the following equation (Todo et al. (2000)):

$$M(t) = G(t)(M_m - M_i) + M_i$$

Where $M_m = 1.27\%$ (values for T800H/2500 composite system from Todo et al. (2000), table 1).

$M_i = 0\%$, and

$$G(t) = \left(1 - \frac{\alpha}{\alpha - \beta}\right) e^{(-\beta t)}$$

$\alpha = 3.00 \times 10^{-3}$ /hr (values for T800H/2500 composite system from Todo et al. (2000), table 1).

$\beta = 2.91 \times 10^{-3}$ /hr (values for T800H/2500 composite system from Todo et al. (2000), table 1).

Reduction in residual strength of a composite structure is proportional to the percentage of moisture absorbed by the structure, that is: Reduction in Residual Strength (RRS) = $-z * M(t)$, where z is the proportionality variable. Initially value is 10^{-6}.

Absorption/Desorption cycles

Due to the lack of research in this area, the effect of absorption/desorption cycles will not be specifically modeled. However, it will be accounted for implicitly as part of the In-Service Black Swan (unforeseen) degradation mode within the SIFCM.

Thermal Degradation

MIL-HDBK-17-3F recommends that a temperature margin between T_g and anticipated maximum temperature that the composite will be exposed to. Following this recommendation should ensure that thermal degradation of composites due to high temperature should be avoided. Therefore, thermal degradation is not modeled within the SIFCM.

UV Degradation

It is standard practice to paint the external surfaces of aircraft structure; the paint will eliminate the exposure of composite materials to UV exposure. Internal surfaces may not be painted, but are not exposed to UV. Therefore, UV degradation is not modeled within the SIFCM.

5.4.3 - Fatigue Degradation Simulation

The reduction in residual strength is calculated by using the following equation

Kassapoglou (2009):

$$\sigma_r = \sigma_{fs}\left(\frac{\sigma}{\sigma_{fs}}\right)\left(\frac{n}{N-1}\right)$$

Which can be rearranged such that:

$$\frac{\sigma_r}{\sigma_{fs}} = \left(\frac{\sigma}{\sigma_{fs}}\right)\left(\frac{n}{N-1}\right)$$

$RRS = \left(\frac{1}{RS}\right)\left(\frac{n}{N}-1\right)$, which is modeled within SIFCM.

Where σ_r is residual strength

σ_{fs} is static strength

σ is the applied load

RS is the current Residual Strength

n is the current number of cycles, which is calculated using 10 cycles per AFHR, and

N is the number of cycles to failure for the applied load σ, which is a variable set to 100000 (selected as an initial setting).

5.4.4 - Black Swan In-Service Damage Simulation

This damage represents in-service degradation, which is unanticipated/unknown. As such, it is represented by a probability of occurrence (likelihood) and *RRS*:

RRS = Base *RRS* * random number (between 0 and 1), and

Likelihood = For upper and lower 1% of probability distribution function (i.e. when random number is < 0.01 and > 0.99).

Where Base *RRS* is a variable with an initial value of 0.1, based on engineering judgment of the author.

5.5 - Degradation Summary

*Table **9*** compares the response of composite and metallic structures to the various degradation modes, which an aircraft structure could be exposed to. Table 9 also identifies when and how the degradation mode is implemented within the Structural Integrity Failure Causation Model (SIFCM).

Degradation Mode	**Material System affected**	**Implementation in SIFCM**
Accidental Damage (AD)		
Design-Induced	Metal and Composite	Yes Quality and Black Swan
Manufacture-Defect	Metal and Composite	Yes Quality and Black Swan
Overload	Metal and Composite	No Overload would result in structural failure regardless of ASI methodology implemented
Impact	Metal and Composite	Yes
Environmental Damage (ED)		
Corrosion	Metal	No Simulation will consider composite-only structure.
Chemical Degradation	Composite	No Graphite/epoxy composite system used as baseline within the simulation.
Moisture Absorption	Composite	Yes
Moisture Absorption/Desorption cycle	Composite	No Considered part of Black Swan In-Service.
Thermal Degradation	Metal and Composite	No Design guidance identifies a margin between Tg and operational temperature.
UV Degradation	Composite	No External surfaces are painted and internal structures are not exposed to UV.
Fatigue Degradation (FD)		
Fatigue	Metal and Composite	Yes
Black Swan In-Service	Metal and Composite	Yes

Table 9: Degradation Summary

6 – MAINTENANCE

The term Maintenance encompasses both the detection and repair of structural damage, as highlighted in *Figure **20***. This section discussed the various forms of maintenance applicable to structural integrity and the implementation within SIFCM.

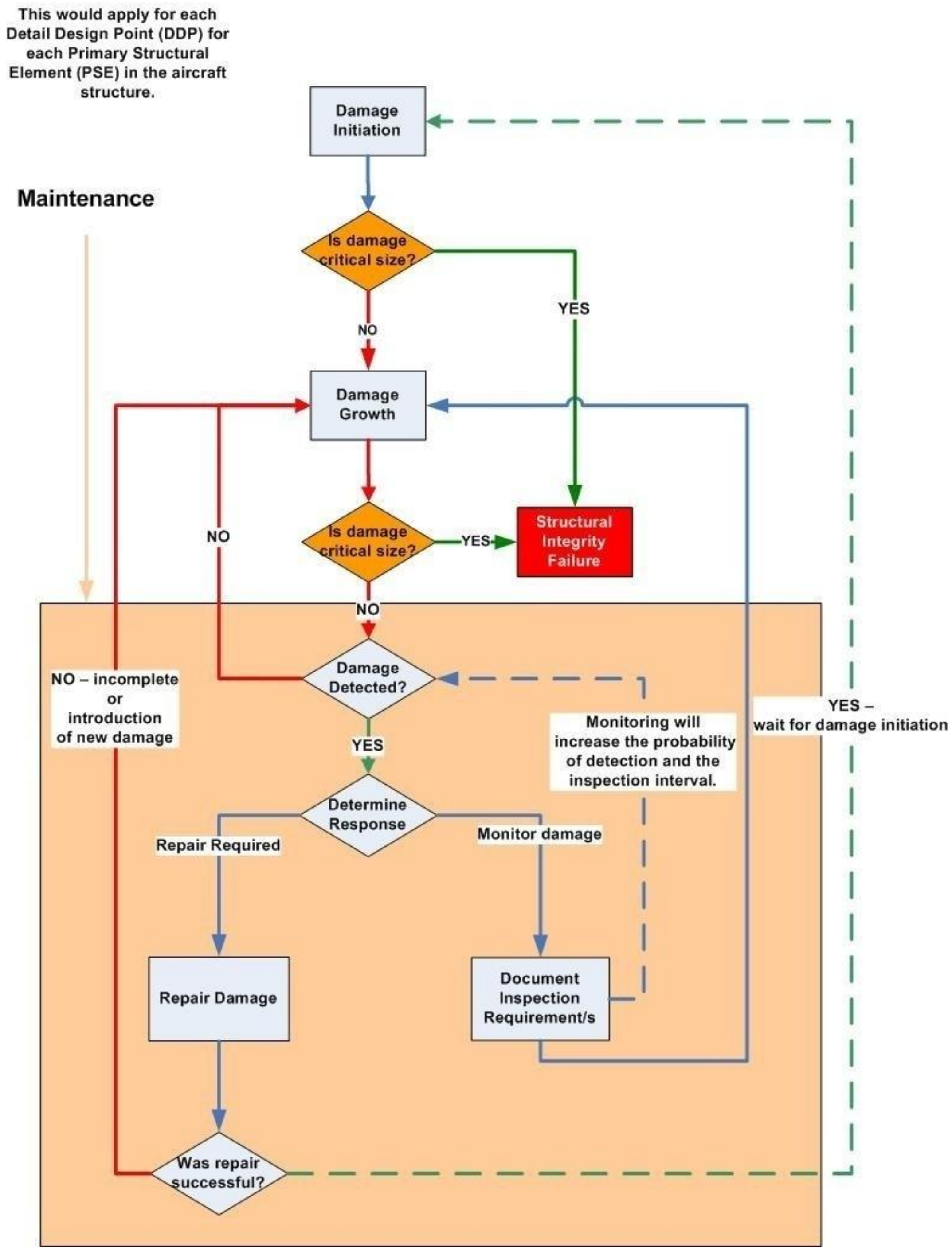

Figure 20: Maintenance with SIFCMsim.

6.1 - Development of Scheduled Maintenance

The Maintenance Review Board (MRB) process as described in AC 121-22C (as illustrated in *Figure* ***21***) is utilised by the FAA as the means to determine the initial scheduled maintenance requirements for a new aircraft type.

The analysis performed as part of the Maintenance Type Board Process is usually performed using the guidance provided in Air Transport Association (ATA) Maintenance Steering Group 3 (MSG-3). ATA MSG-3 states that it "presents a means for developing the scheduled maintenance tasks and intervals which will be acceptable to regulatory authorities, the operators, and the manufacturers." (p.20). As such, ATA MSG-3 covers:

a. Systems/Powerplant, which includes auxiliary power units,

b. Aircraft Structures, including both metallic and non-metallic considerations.

c. Zonal Inspections, visual inspections performed in a specific zone/area to confirm the security and good condition, and

d. Lightning/High Intensity Radiated Fields.

The development of scheduled maintenance tasks related to aircraft structures includes consideration of the forms of degradation (environmental, accidental and fatigue), the susceptibility of the structure to the forms of degradation, the consequence to airworthiness if the structure deteriorates, the effectiveness of prevention, detection and control techniques and finally, any structural health monitoring systems planned for use by the aircraft manufacturer.

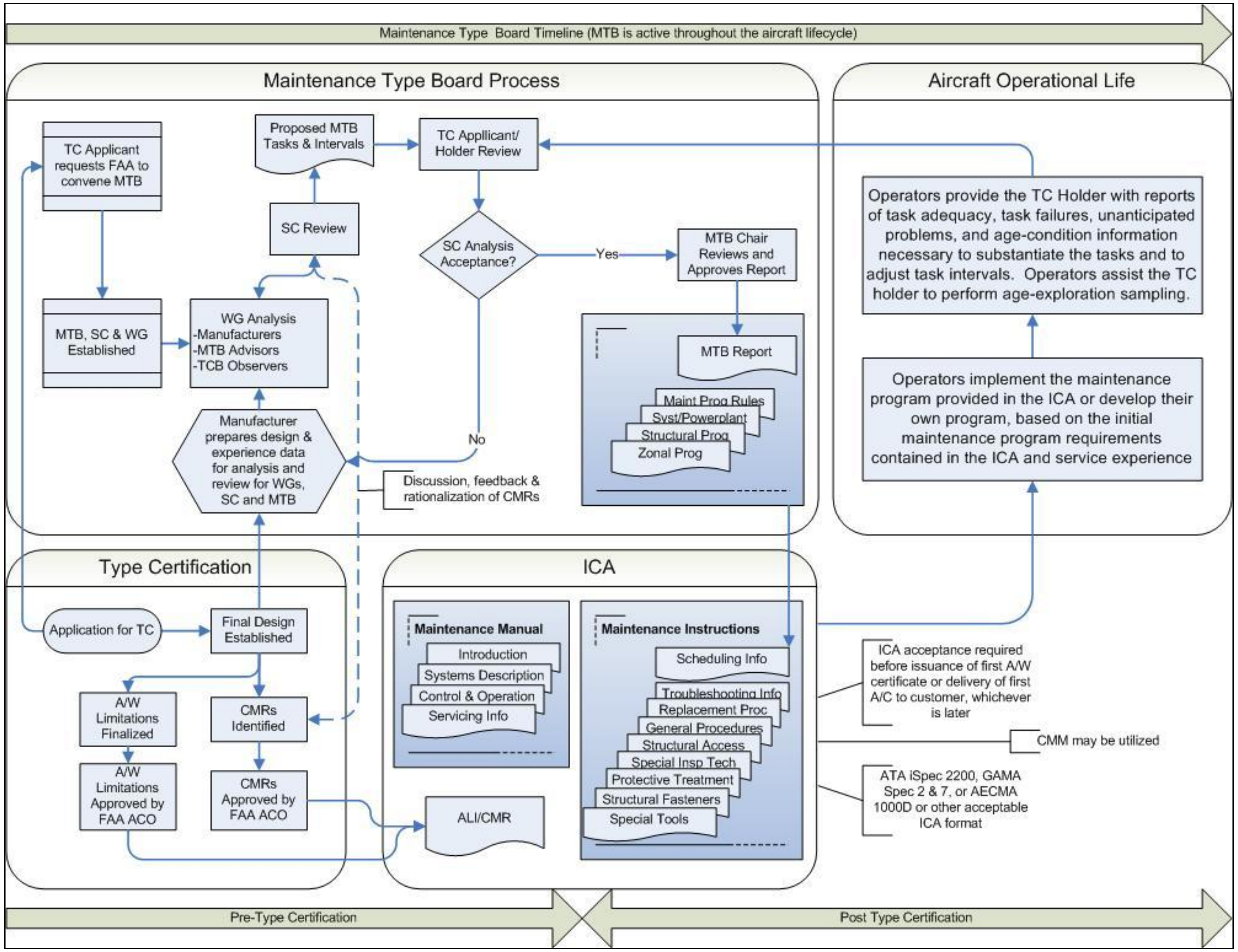

Figure 21: Maintenance Review Board Process Flowchart (from AC 121-22C).

The damage tolerance approach used during the design of the structure will impact the development of the scheduled maintenance requirements for an aircraft structure. For example, ATA MSG-3 states "Inspections related to FD (fatigue damage) detection in non-metals should not be required if their design is based on a 'no- damage growth' design philosophy, and substantiated by testing." (p.56).

Scheduled maintenance includes the conduct of airframe inspections at pre-determined intervals and the repair of any damage detected. The various inspection methods and repair techniques will be discussed in Section 6.2 and 6.3 respectively.

6.2 - Degradation Detection

6.2.1 - Detection Methods

As identified by The American Society for Nondestructive Testing (ASNT), the primary non-destructive testing (inspection) methods used within the Aerospace community are:

a. Visual Testing: This method involves the use of unaided and aided (such as, microscopes, borescopes and background lighting) human sight to determine the damage.

b. Liquid Penetrant Testing: This method involves the application of a low-viscosity fluid (penetrant) to the surface of the structure. The penetrant is allowed to soak (dwell) for a period of time to allow for it to enter any surface defects. After the dwell period, the surface is wiped clear of penetrant and a developer is applied to the surface, which reacts with the penetrant remaining within surface defects to highlight surface defects. Penetrant compatibility with composite structures must be considered prior to application.

c. Magnetic Particle Testing: Using this method, a ferro-magnetic structure is magnetised with magnetic flux leakage fields caused by defects/damage being detected. As composites are not ferro-magnetic, this method is not suitable for use with composite structures.

d. Electromagnetic Testing: This method includes the use of eddy current inspection, which entails generating small electrical currents within the structure via the use of varying electro-magnetic field. The eddy currents generate a reverse magnetic field, which is detected by a detection coil. Due to composites being non-conductive, this method is not suitable for use with composite structures.

e. Ultrasonic Testing: Defects (such as cracks, voids, regions of high porosity) alter the way an acoustic pulse is reflected/scattered.

f. Radiographic Testing: The absorption of X-rays passing through the structure varies depending on the material density. This method can be used to detect through-thickness cracks, voids, foreign objects and crushed cores within composite structures.

g. Infrared and Thermal Testing: This class of inspection methods involves changing the temperature of the structure and looking for changes in the surface temperature of the structure for evidence of damage/inconsistencies (such as delaminations, large voids and foreign objects within composites).

h. Shearography and Holography: Surface strains of a loaded structure are measured as fringe patterns. Damage/defects (such as delaminations and crushed cores) within the structure are detected as variations in the surface strain continuity.

i. Acoustic Emission Testing: This method involves loading the structure and sensing any resulting acoustic emission and correlating the emission with the source of damage. This technique can be used for composite structures, detecting damage such as, fibre breakage and delaminations.

6.2.2 - Probability of Detection

The reliability of an inspection method of detecting damage/flaws of a given size under the specified inspection conditions and procedures, is represented by the Probability of Detection (PoD). The current use of PoD can be traced back to a program initiated by NASA in 1969 to determine, for a number of different non-destructive inspection methods used during Space Shuttle design and production, the largest flaw size that could be missed. The PoD methodology was then implemented by the US Air Force, US commercial aircraft industry and spread to other industries in the subsequent decades.

As described by Georgiou (2006), the recommended practice for production of PoD curves includes:

a. Manufacture/acquire flaw specimens for testing. The number of specimens required for testing is dependent on the desired PoD and lower confidence limit. For example, to develop PoD curves for a combination of 90% PoD and a lower confidence limit of 95% over 6 width intervals would require a minimum of 174 flaw specimens.
b. Inspect flaw specimens with required inspection method;
c. Record the inspection results as a function of flaw size, and
d. Plot PoD curve as a function of flaw size, as example PoD curve for three inspection methods is provided at *Figure **22***.

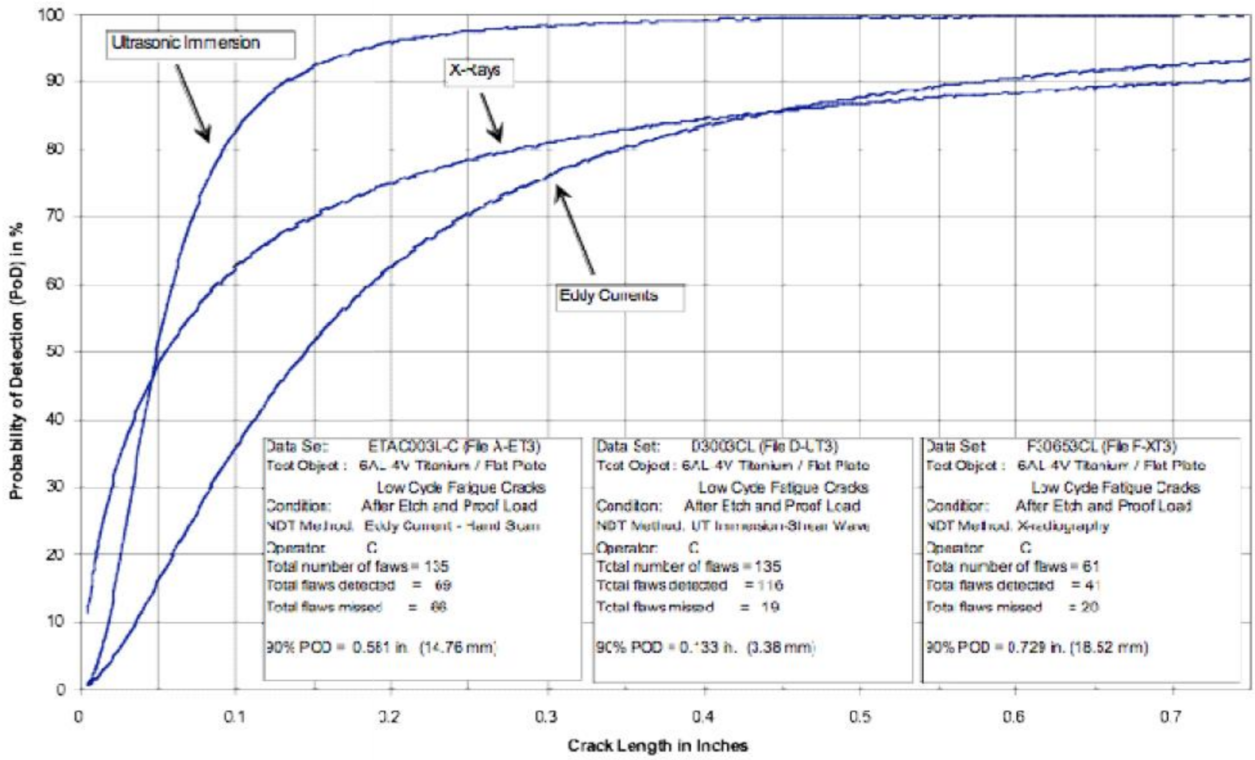

Figure 22: PoD curve for different inspection methods applied to the same flaw specimen (from Georgiou, 2006).

Not only can the PoD vary between inspection methods for the same flaw, but the PoD of a specific inspection method can vary due to a number of reasons, including:

a. Material being inspected, as illustrated by Georgiou (2006) where using eddy current inspection to detect a 2.54mm (0.1in) crack the PoD varied from 70% in Aluminium to less than 20% in 4340 steel.

b. Flaw/damage type (degradation), as illustrated by Georgiou (2006) where using fluorescent penetrant inspection to detect a longitudinal 0.1in crack (>90% PoD) whereas for a transverse 2.54mm (0.1in) crack (30% PoD).

c. Operator/inspector, as illustrated by Georgiou (2006) where using ultrasound inspection on the same flaw (0.2in (5.08mm) crack), different operators had PoD which varied from 85 to 100%.

d. Surface finish, as described by Civil Aviation Authority (CAA) (2013), where the PoD for a 40.45mm wide and 0.25mm deep surface flaw was less than 70% on all surface colours/finishes, only 6% for gloss white surfaces and 0% on matt blue surfaces.

6.3 - Repair Methods

This section discusses the various aircraft structural repair techniques.

6.3.1 - Replacement

Repair of the structure is not viable, and therefore replacement is required. This technique can be used for both composite and metallic structures.

6.3.2 - Non-Patch Repairs

Resin Injection (composite only)

This repair technique is used for composite structures to repair delaminations and disbonds in composite structures. This technique involves the injection of resin under pressure into the area of delamination or disbond. Baker et al. (2004) identifies that this type of repair is limited to non-critical structures due to incomplete penetration of delaminations (and associated low strength recovery) due to the recommended resin systems.

Surface Coating

This repair technique is used to repair protective and conductive surface layers of the structure and can be applied to either composite or metal structures.

Potting (filling) repair (composite only)

This repair technique is used to repair defective region with resin and can be used to repair dents to composites and elongated fastener holes (for low loading applications). In the case of fastener holes, a machine-able potting compound is used to fill the elongated hole and then re-machined as required.

6.3.3 - Patch Repairs

Bolted Patch

This repair technique is used for both composite and metallic structures and entails placing a patch of either composite or metallic construction over the damaged section of structure and bolting the patch into place. This repair technique is suitable for the repair of thick section elements (low strain), it is easily implemented and as such can be readily performed in the field.

External Bonded Patch

This repair technique is used for both composite and metallic structures and entails bonding a patch of either composite or metallic (e.g. titanium) over the damaged section of structure. This repair technique is suitable for the repair of thin skin with no aerodynamic or low-observable (stealth) limitations. It provides a more effective means to restore the mechanical properties of a structure, than using a bolted patch repair. This repair is unobtrusive as there is not requirement to drill holes for fasteners, as required by the bolted patch technique.

Airbus will be concentrating on bolted repairs for the A350WXB, as bonded repairs for primary load bearing structures are not supported by regulations (Gubisch (2012)). The regulators main concern with bonded repairs is the difficulty with confirming the strength of the bond.

Scarf Patch

This repair technique is used for composites and entails removing the damage composite structure and replacing it with composite structure, as illustrated in *Figure **23***. This repair is used to repair thick skin and/or structure with aerodynamic or low-observables limitations.

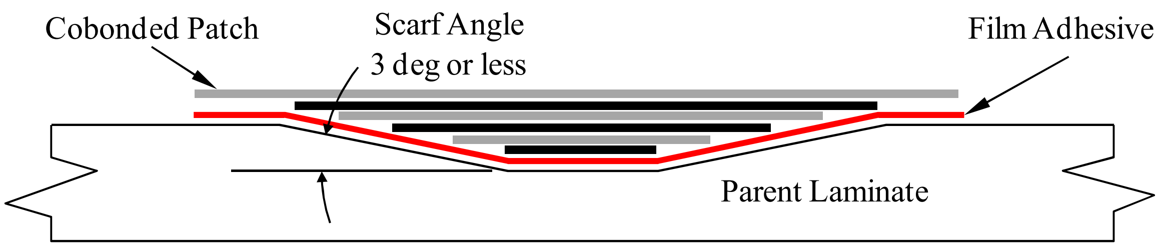

Figure 23: Example of a scarf repair (from Heslehurst (2012)).

6.4 – Simulation of Maintenance

The simulation of maintenance is illustrated in *Figure **24*** and discussed in the following sections.

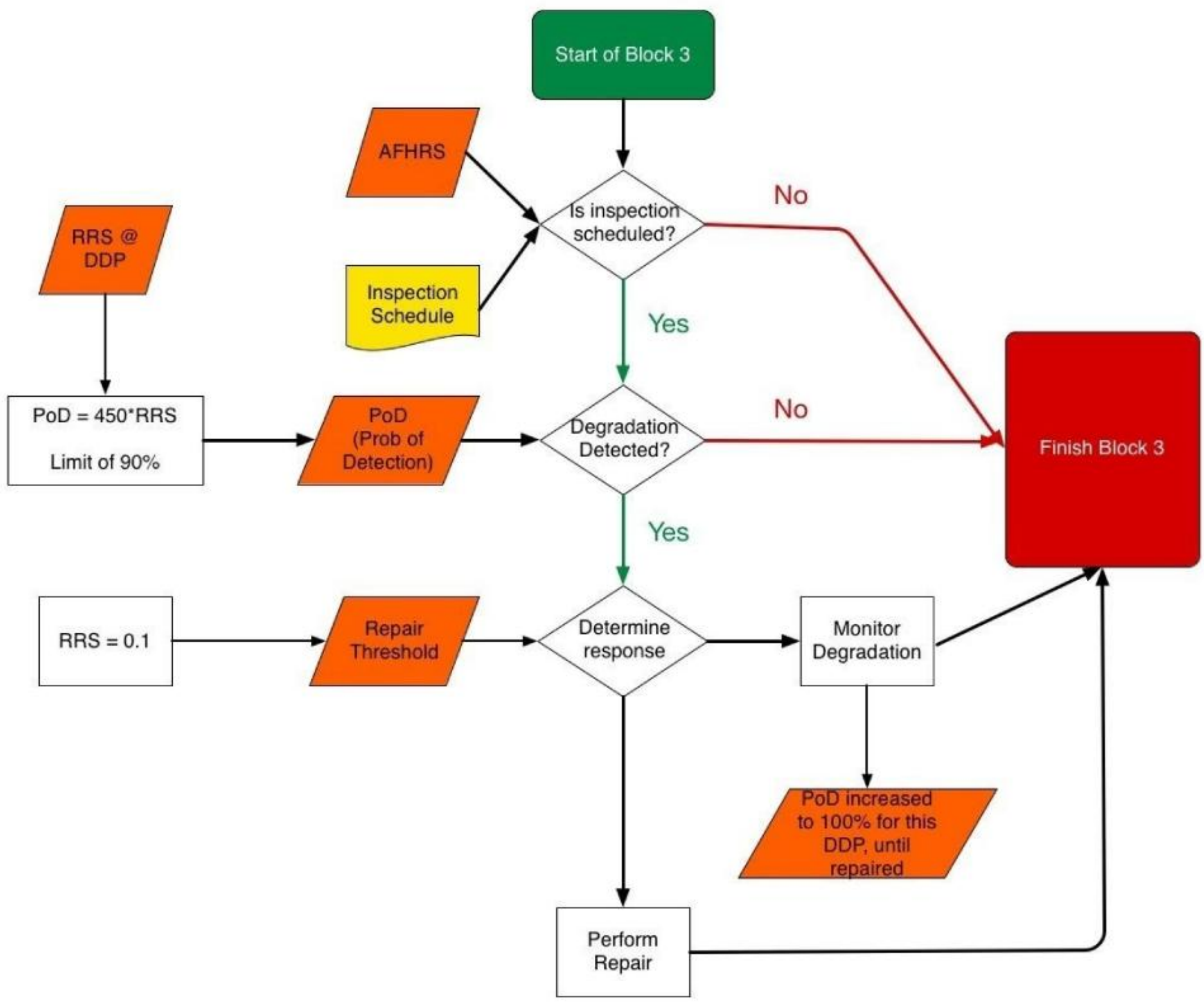

Figure 24: Simulation of Maintenance block diagram.

6.4.1 - Damage Detected?

Due to variability of PoD as described in Section 6.1.2 and the lack of detailed degradation information tracked within the SIFCM, modeling of specific inspection methods will not be performed within the SIFCM. Instead, a generic inspection method will be modeled with PoD varying with the Reduction in Residual Strength present at the DDP compared with the 'as-manufactured' RS, as described by the following equation:

PoD = 450*RRS up to a maximum PoD value of 90 (%)[16]

Inspections will be performed for the PSE (and associated DDP) at predefined intervals, as defined within an Inspection Schedule, which will be further discussed in Section 7.

6.4.2 - Determine Response

Once detected, if the RRS from 'as-manufactured' RS is greater than 0.1, a repair will be conducted (refer to Section 6.4.3), if less than 0.1 the degradation is documented for monitoring (refer to Section 6.4.4).

6.4.3 - Repair Damage

The modeling of individual repair techniques will not be undertaken due to the similarity of the techniques used to that used during manufacture and additional complexity required to map damage within the simulation against specific repair techniques. The repair when conducted, will restore the RS at the DDP to 'as-manufactured' levels taking into consideration Design and Manufacture Induced Degradation as discussed in 'Was the repair successful?'

[16] The value of 450 was selected to ensure that through-thickness damage (RRS = 0.2) would have the maximum PoD of 90%.

Was the repair successful?

The repair of damaged structure can be broken down into two phases: design and manufacture (installation) of the repair. As such, the mechanism to determine the accidental damage during design and manufacture will be applicable to the repair:

a. Repair Design Induced Degradation: As per the Design Induced Degradation discussed in Section 5.4.1, and

b. Repair Manufacture Induced Degradation: As per the Manufacture Induced Degradation discussed in Section 5.4.1.

6.4.4 - Document Inspection Requirements

When it is determined to document an inspection requirement to monitor damage growth, it will increase the PoD (at that DDP) at the next inspection interval to 100%, until the degradation has been repaired. Once repaired the PoD will revert back to the value calculated as per Section 6.4.1.

6.5 – Maintenance Summary

This section discussed the various inspection methods and repair techniques and applicability to composite or metallic structures, which is summarized in Tables 10 and 11.

Inspection Method	Material System affected	Implementation in SIFCM
Visual	Metal and Composite	Generic Inspection Method is implemented.
Liquid Penetrant	Metal and Composite	
Magnetic Particle	Metal	
Electromagnetic	Metal	
Ultrasonic	Metal and Composite	
Radiographic	Metal and Composite	
Infrared and Thermal	Metal and Composite	
Shearography and Holography	Metal and Composite	
Acoustic Emission Testing	Metal and Composite	

Table 10: Inspection Summary

Repair Technique	Material System affected	Implementation in SIFCM
Replacement	Metal and Composite	Generic Repair Method is implemented.
Non-Patch Repairs		
Resin Injection	Composite (Not used for primary structure)	
Surface Finishing	Metal and Composite (Not used to restore strength)	
Potting	Composite	
Patch Repairs		
Bolted Patch (external or flush)	Metal and Composite	
Externally Bonded Patch	Metal and Composite (Regulations do no allow the use for primary structures)	
Scarf Repair	Composite	

Table 11: Repair Summary

7 - SICFM MODELS

7.1 - SIFCM_PSE

The SIFCM_PSE model was developed to simulate a single PSE consisting of four DDP. SIFCM_PSE was used as a prototype to support the development of the more detailed model (SIFCMsim), comprised of a representative aircraft structure. SIFCM_PSE was developed using *MATLAB* and consists of the following files (refer to Appendix A for details):

a. SIFCM_PSE.m, and

b. PSE_initialisation.m.

The script file 'SIFCM_PSE.m' consists of number of software blocks as illustrated by Figure 25 and further described in the subsequent text.

7.1.1 - Block 1 - Establish 'as-manufactured' Residual Strength for each DDP

Block 1 runs the file 'PSE_initialisation.m' which establishes the RS at each DDP within the PSE (includes Design-Induced and Manufacture-Induced degradation), using the methodology identified in Section 5.4.1.

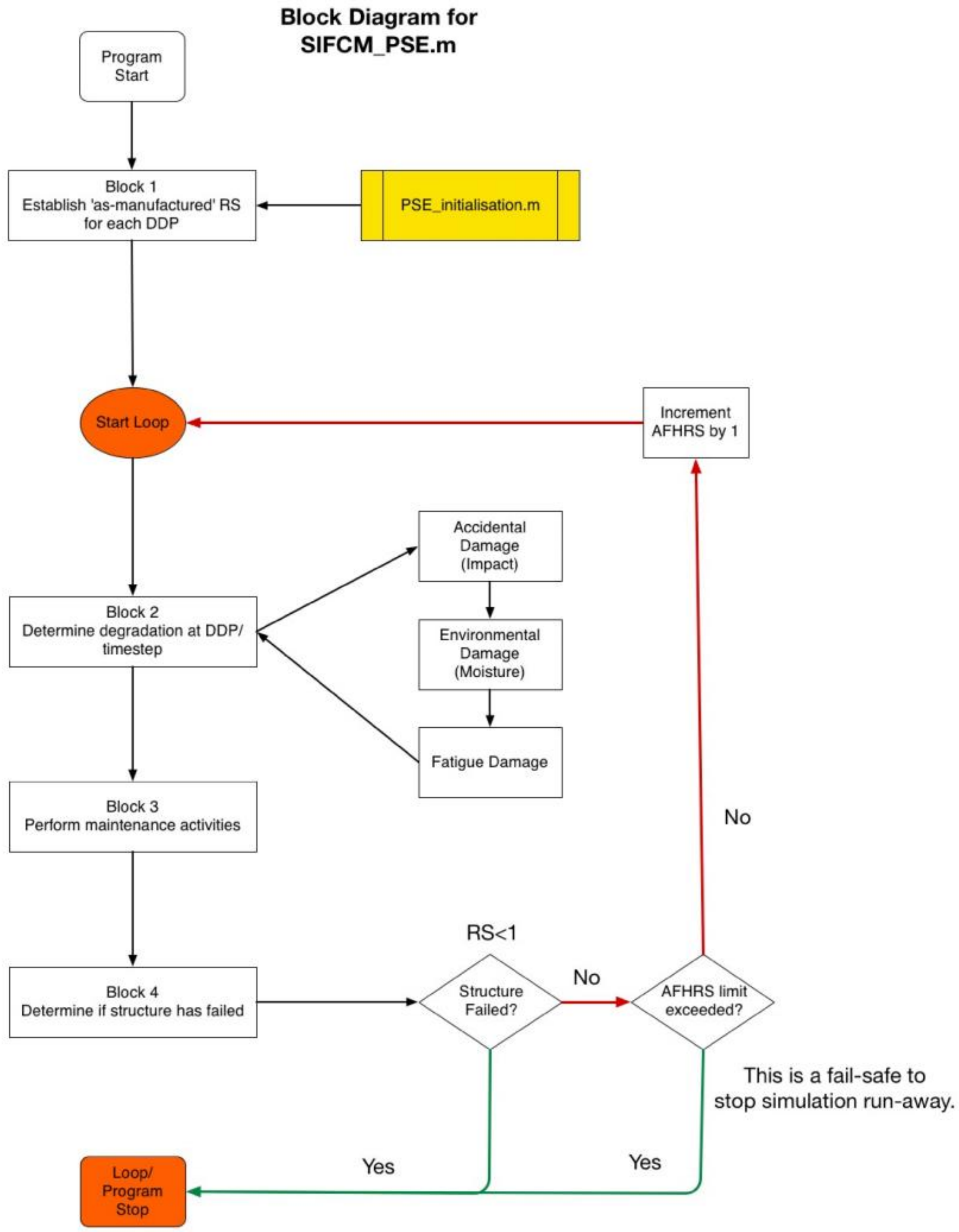

Figure 25: SIFCM_PSE Block Diagram

7.1.2 - Block 2 - Determine the degradation at each DDP per time step

Block 2 determines the Reduction in Residual Strength (RRS) at each DDP as a result of the following degradation modes:

a. Impact, as per Section 5.4.1 with the following probabilities:

10% probability per AFHR of Surface Damage (RRS = 0);

5% probability per AFHR of Laminate Damage (RRS = 0.05);

2.5% probability per AFHR of Through Thickness Damage (RRS = 0.2), and

82.5% probability per AFHR, no damage RRS = 0.

b. Moisture Absorption, as per Section 5.4.2; and

c. Fatigue, as per Section 5.4.3.

7.1.3 - Block 3 - Perform Maintenance Activities

Block 3 operates as described in Section 6.4 with the inspection schedule based on AFHRS, with the time between inspections (for each DDP) being defined within variable 'Inspection_Interval'. The default value is 10 AFHRS for each DDP.

The detection of damage is based on RRS from design value (prior to any design-induced or manufacture-induced degradation), which is 1.5. Once detected, the damage is repaired by restoring the residual strength, at the DDP to its 'as-manufactured' value using 'PSE_initisiation.m'.

7.1.4 - Block 4 - Determine if the structure has failed

Block 4 determines when the structure has failed (when Residual Strength at any DDP is reduced below 1) and contains a cycle-limit to stop the program from running indefinitely. The cycle-limit stops the program at a pre-defined number of AFHRS (for example, 6,000 AFHRS).

7.2 - SIFCMsim

The SIFCMsim model was developed to simulate a complete structure consisting of multiple PSE and DDP and includes the ability to model fail-safe structure. SIFCMsim was developed using *MATLAB* and consists of the following files (refer to Appendix B for details):

a. SIFCM_PSE.m;

b. SICFMsim_Input.txt, and

c. PSE_initialisation.m (modified to support different output format).

The script file 'SIFCMsim.m' consists of number of software blocks as illustrated by Figure 26 and further described in the subsequent text.

7.2.1 - Block 0 - Establish the Structure on Interest (SoI) matrix

To allow for the management of a large number of DDP with different parameters, a matrix, 'SoI(DDPnumber, attribute index)', is used to capture the various parameters related to each DDP, as identified in Table 12 and described below:

a. Column 1: Identifies the PSE which the DDP is part of;

b. Column 2: The current residual strength of the DDP;

c. Column 3: The impact type of the structure, high, intermediate or low damage rate as described in DOT/FAA/AR-01/55, Table 1-11 (2002). A fourth type, Internal, was added to cover internal structure which has zero probability of impact damage.

d. Column 4: Identifies if the DDP is part of a fail-safe structure, which could be composed of up to three PSE.

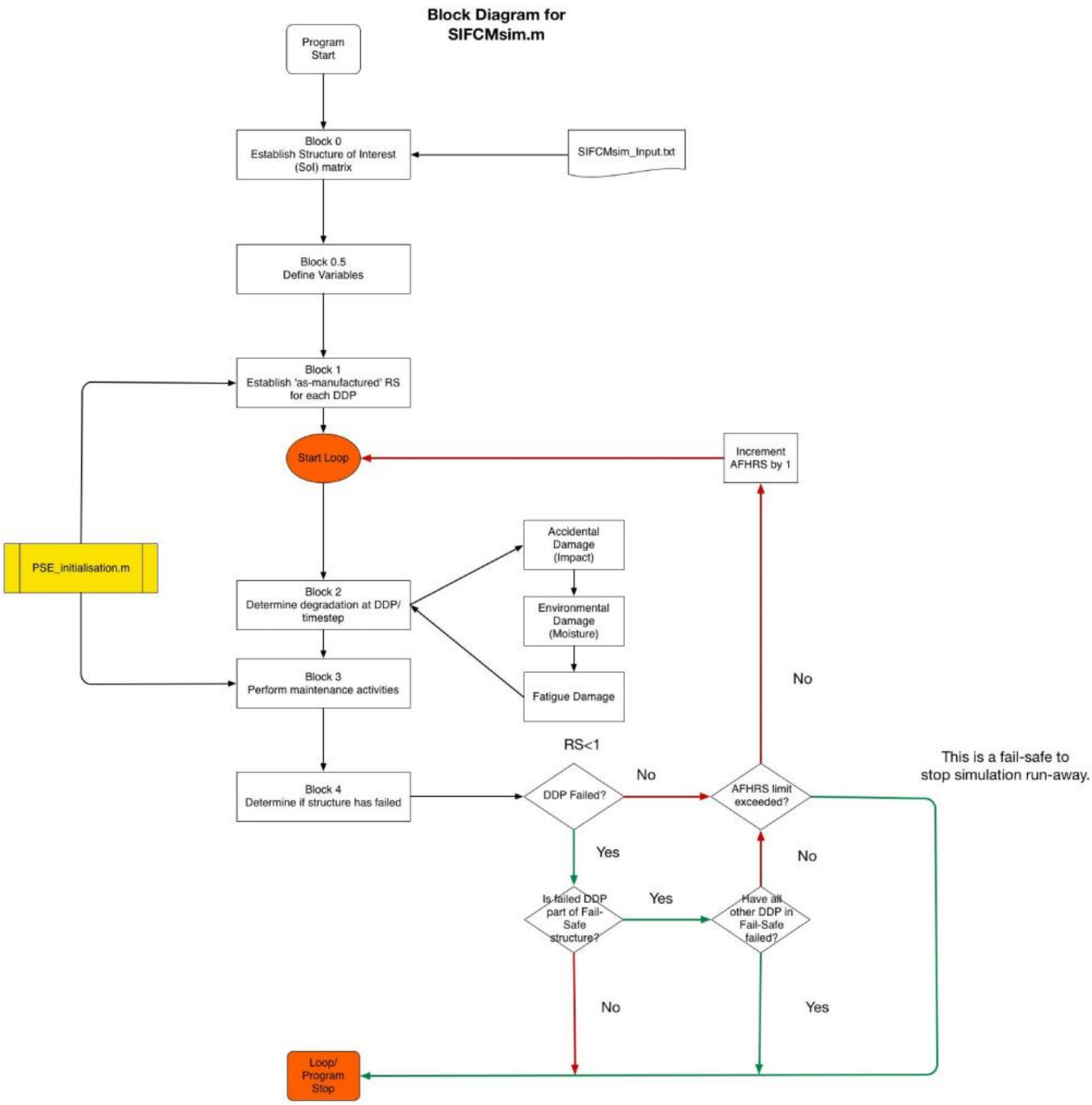

Figure 26: SIFCMsim Block Diagram

e. Column 5-7: Identifies the PSE which comprise the fail-safe structure which the DDP is part of. The model allows for a maximum of three PSE to form part of a fail-safe structure.

f. Column 8: This captures when the DDP has failed and is used to determine if the entire fail-safe structure has failed.

g. Column 9: This captures when (in AFHRS) the DDP failed (i.e. Residual Strength less than 1).

h. Column 10: Identifies the inspection interval for the DDP.

i. Column 11: Used to determine when the DDP is required to be inspected.

j. Column 12: The current Probability of Detection value for the DDP.

The 'SoI' matrix is populated using the text file, SICFMsim_Input.txt, which has the following format:

a. Column 1: Identification number for the DDP (not used by the model).

b. Column 2: Identifies the PSE which the DDP is part of.

c. Column 3: The impact type of the structure.

d. Column 4: Identifies if the DDP is part of a fail-safe structure.

e. Columns 5-7: Identifies the PSE which comprise the fail-safe structure which the DDP is part of.

f. Column 8: The number of cycles to fatigue failure for the DDP. This populates a separate array titled 'NtoFailure'.

g. Column 9: Inspection interval for the DDP.

7.2.2 - Block 0.5 - Define Variables

Block 0.5 is use to define variables for use during the simulation

PSE ID	***Residual Strength***	***Impact Damage Structure Type***	***Part of Fail-Safe PSE***	***Partner Fail-Safe Structure ID (1)***	***Partner Fail-Safe Structure ID (2)***	***Partner Fail-Safe Structure ID (3)***	***Fail-Safe DDP has failed***	***Fail-Safe DDP Failure time (AFHRS)***	***Inspection Interval***	***Interval Counter***	***PoD***	***Name***
Integer	Floating	Integer	Integer	Integer	Integer	Integer	Integer	Integer	Integer	Integer	Float	Variable Type
1	2	3	4	5	6	7	8	9	10	11	12	Attribute Index
		1 - High 2 - Intermediate 3 - Low 4 - Internal	0 - No 1 - Yes				0 - No 1 - Yes					

Table 12: Detail Design Point Matrix Definition

.

7.2.3 - Block 1 - Establish 'as-manufacture' Residual Strength for each DDP

Block 1 runs the file 'PSE_initialisation.m' which establishes the RS at each DDP within the Structure of Interest (Detail Design Point) as defined in the matrix 'SoI' and includes Design-Induced and Manufacture-Induced degradation (using the methodology identified in Section 5.4.1).

7.2.4 - Block 2 - Determine the degradation at each DDP per time step

Block 2 determines the Reduction in Residual Strength (RRS) at each DDP as a result of the following degradation modes:

a. Impact, as per Section 5.4.1 with probabilities (Table 13) were derived from Table 8 with 1 sqm, 2L = 1 mm.
b. Moisture Absorption, as per Section 5.4.2; and
c. Fatigue, as per Section 5.4.3.

7.2.5 - Block 3 - Perform Maintenance Activities

Block 3 operates as described in Section 6.4 with the inspection schedule based on AFHRS, with the time between inspections (for each DDP) being defined within variable 'Inspection_Interval'.

The detection of damage is based on Reduction in Residual strength from design value (prior to any design-induced or manufacture-induced degradation), which in is 1.5. Once detected, the damage is repaired by restoring the residual strength at the DDP to its 'as-manufactured' value using 'PSE_initisiation.m'. In addition, for fail-safe DDP (identified at matrix *SoI(i,4)*), when repaired, the failure flag (identified at matrix *SoI(i,8)*) is reset to zero

Damage Type	Through-Thickness Damage (RRS = 0.2)	Laminate Damage (RRS = 0.05)	Surface Damage (RRS = 0)
Structure Type	Probability of Occurrence (per AFHRS)	Probability of Occurrence (per AFHRS)	Probability of Occurrence (per AFHRS)
High	0.001	0.00165	0.00114
Intermediate	0.00087	0.000471	0.000375
Low	0.000135	0.000278	0
Internal	0	0	0

Table 13: Impact Probabilities for various Impact Type Structure

.

7.2.6 - Block 4 - Determine if the structure has failed

Block 4 determines when the structure has failed and contains a mechanism to stop the program from running indefinitely (cycle-limit). The cycle-limit stops the program at a pre-defined number of AFHRS (for example, 6,000 AFHRS).

The logic used to determine if the Detail Design Point has failed is:

a. Determine if any DDP have failed (residual strength < 1);

b. If the failed DDP is not fail-safe (identified by 0 in *SoI(i,4)*), then the Detail Design Point has failed.

c. If the failed DDP is fail-safe (identified by 1 in *SoI(i,4)*) then:

   i) Residual strength of all DDP, which form part of the failed DDP fail-safe structure, is reduced by 0.002.

   ii) Cycles to fatigue failure of all DDP, which form part of the failed DDP fail-safe structure, is reduced by 2%.

   iii) If all other DDP, which forms part of the fail-safe structure, have failed, then the Detail Design Point has failed.

8. DISCUSSION

8.1 - Composites and MIL-STD-1530C[17]

The concept of the Aircraft Structural Integrity Program (as defined in MIL-STD-1530C) has evolved over a number of decades, primarily based on the experience gained from metal aircraft structures (for example, Boeing B-47 and General Dynamics F-111). However, with the increased use of composite materials in aircraft structures since the introduction of glass fibre reinforced polymer (GFRP) in 1957 (ATSB 2007). This section discusses the aspects of the ASIP, which may need to be modified to adequately take into account the unique properties and failure mechanisms of composite structures as presented in the previous section.

8.1.1 - Task I - Design Information

Impact Damage Criteria

MIL-STD-1530C, Section 5.1.3.5 identifies the need to define damage tolerance criteria, with impact damage being identified within the text as one aspect to consider. Due to the vulnerability of composite materials to impact damage, the establishment of impact damage criteria and associated impact environment is a critical consideration. Therefore, it is recommended that impact damage criteria be a distinct task within Section 5.1.3.5.

Damage Tolerance Criteria

MIL-STD-1530C, Section 5.1.3.5.1 identifies the following approaches to damage tolerance: fail-safe (safety by design) and slow growth (safe life). This section also specifically states that for composite structures, no significant damage growth shall originate

[17] This section (with minor changes) is from Warren, A., Heslehurst, R. & Wilson, E. (2013). *Composites and MIL-STD-1530C,* accepted for publication in International Journal of Structural Integrity.

from manufacturing defects or from damage due to high-energy impact. This equates to the 'no growth' damage tolerance approach discussed in Advisory Circular 20-107B Change 1 (2010).

8.1.2 - Task II - Design Analyses and Development Testing

Material Properties and Joint Allowables

MIL-STD-1530C Section 5.2.1 identifies a number of sources for material and joint allowables to be used during design. These sources include CINDAS Aerospace Structural Metals Handbook, CINDAS Structural Alloys Handbook and Damage Tolerant Design Handbook. These sources provide metal specific information, no composite specific references, such as Composite Materials Handbook (CMH-17), are identified in MIL-STD-1530C. Recommend that composite specific material properties references be provided.

Degradation Mechanisms

The principle degradation mechanisms discussed within MIL-STD-1530C are:

a. Fatigue, with the determination of onset of WFD as a specific analysis task to be performed as part of the durability analysis discussed in MIL-STD-1530C Section 5.2.7, and WFD is used to determine the service life of the aircraft during durability testing as discussed in MIL-STD-1530C, Section 5.3.4.
b. Corrosion, as there is a requirement for the establishment of a corrosion prevention and control plan (MIL-STD-1530C, Section 5.1.5).

Impact is as significant for composite materials as corrosion is for metals, therefore it is recommended that the susceptibility of composites to impact damage as a degradation mechanism be identified and discussed. Impact damage becomes sources for compression related fatigue with delamination propagation.

Design Analysis

The design of composite aircraft structures is governed by airworthiness requirements, as captured by JSSG-2006 for United States military aircraft. Each aircraft manufacturer will have design guidelines 'design best practice' to achieve compliance with the required airworthiness requirements. For example:

a. JSSG-2006 Section 3.10.5 specifies that as part of static strength analysis delaminations in composite structures should not occur at or below 115% of limit loads.

b. The current design practice in composite structures, and in particular shear structures, is to not allow the occurrence of shear buckling (tension field). A shear resistant composite structure design will ensure that secondary bending is significantly reduced at the attachment points of the shear structure. Thus riveted joints will not experience excessive fastener pull-though loads and bonded structure will significantly reduce induced peeling loads. Both connection types have the potential to induce delaminations into the composite structure via interlaminar stresses at the local load transfer point if buckling of the structure is allowed.

The quest for higher efficiency aircraft structures combined with an increasing understanding of the performance of composite structures, will drive a reduction in conservatism in both regulation and design best practice.

8.1.3 - Task III - Full-Scale Testing

Climatic Tests

MIL-STD-1530C, Section 5.3.6 states that full scale climatic testing is to be performed to identify potential corrosion issues. In addition to corrosion, composite structures experience a number of additional climatic related degradation modes:

a. Moisture absorption,

b. Moisture absorption/desorption cycles,

c. UV exposure, and

d. Chemical degradation.

Therefore, the scope of full-scale climatic testing should be increased to confirm the exposure of the composite structures to these degradation modes.

8.1.4 - Task IV - Certification and Force Management Development

Certification

Analysis is performed during this task as part of the certification activity performed by the government. The certification is to be performed using the guidance provided by MIL-HDBK-516B 'Airworthiness Certification Criteria', which for structures references JSSG-2006. Warren (2012) provides a high-level review of JSSG-2006, which indicates that composite specific structural considerations are discussed.

Development of structural integrity related surveillance programs

During Task IV surveillance programs to support the management of structural integrity of the aircraft fleet are established for implementation as part of Task V - Force Management Execution. The need to modify the programs to account for composite structures is discussed below:

Analytical Condition Inspection (ACI) Program

MIL-STD-1530C, Section 5.4.3.3.1 identifies the requirement for an Analytical Condition Inspection (ACI) program to be performed as required by Air Force Material Command Inspection (AFMCI) 21-102, where AFMCI 21-102 defines an ACI as "the systematic disassembly and inspection of a representative sample of aircraft to find hidden defects, deteriorating conditions, corrosion, fatigue, overstress, and other deficiencies in the aircraft structure or systems. ACIs are normally over and above those inspections specified in the technical order or PDM work specifications."

MIL-STD-1530C places additional emphasis on the determination of when and where corrosion occurs. Due to the sensitivity of composites to impact damage, it is recommended that emphasis be added for impact, with the purpose of determining where impact has occurred, estimation of the impact energy and extent of damage caused. Thus, allowing for assessment of impact energy versus damage data used during Task II to be confirmed/updated.

Structural Teardown

MIL-STD-1530C, Section 5.4.3.3.2 states that a structural teardown may be required to support the operation of an aircraft beyond the design service life. This is equally applicable to composites as to metal aircraft structures, with the teardown including confirmation that the composite system material properties have not significantly degraded whilst in-service.

Loads/Environment Spectra Survey (L/ESS)

MIL-STD-1530C, Section 5.4.4 identifies the requirement for loads and environment spectra survey to be performed to confirm the original (and update) design spectrum (as developed during Task II), with at least 20% of the aircraft fleet to be instrumented to gather data to support the survey. The L/ESS shall include the requirement to include the collection of impact damage data with the intent of confirming the probability distribution for impact likelihood (for example, probability per m^2) and impact energy used. FAA (1997) and FAA (2002) provide examples of this probabilistic information.

Thus, due to this critical degradation mode for composites being probabilistic, there is an increasing the importance of probability-based risk assessment for structural integrity management, such as that proposed by Backman (2008), Tuegel (2011) and FAA(1999). It is recommended that the risk assessment required by MIL-STD-1530C, Section 5.4.1.1 provide references to probabilistic structural design methodologies.

Individual Aircraft Tracking (IAT) program development

MIL-STD-1530C, Section 5.4.5 identifies the requirement for a individual aircraft tracking program to be developed to monitor the actual usage of an individual aircraft (i.e. by tail number) with the system used to record the data have sufficient reliability to ensure capture of at least 90% of all flight data during the life of the aircraft. Tracking using data derived from L/ESS. It is recommended that the IAT program include the capture of impact and moisture absorption data.

8.1.5 - Task V - Force Management Execution

As this task represents the execution of the programs and plans developed during previous tasks, there are no specific recommendations for Task V.

8.2 - SIFCM Results

8.2.1 - SIFCM_PSE Results

Prior to the development of SIFCMsim, it was considered important to perform a sensitivity analysis for a number of reasons: determine the sensitivity of a number of variables within the model and to determine if any of the value of the variables require modification for use within the SIFCMsim. The variables assessed for sensitivity regarding the number of AFHRS to failure of the PSE, were the following:

a. Laminate Damage RRS (defined in Section 5.4.1);

b. Through Thickness Damage RRS (defined in Section 5.4.1);

c. Inspection Interval (defined in Section 7.1.3);

d. Probability of Detection equation (defined in Section 6.4.1), and

e. Moisture proportionality variable (z) (discussed in Section 5.4.2).

To assess the variables, each variable of interest was varied (with at least baseline value, one above and one below) refer to Appendix C for the supporting data. Unless otherwise stated all values in the model will be as defined in the previous Sections.

Laminate Damage RRS

The variation in time to failure versus RRS due to laminate damage was investigated by setting the RRS value to 0.05, 0.075 and 0.1. After 40 runs[18] at each RRS value, the average time to failure was determined and plotted against Laminate Damage RRS in *Figure 27*.

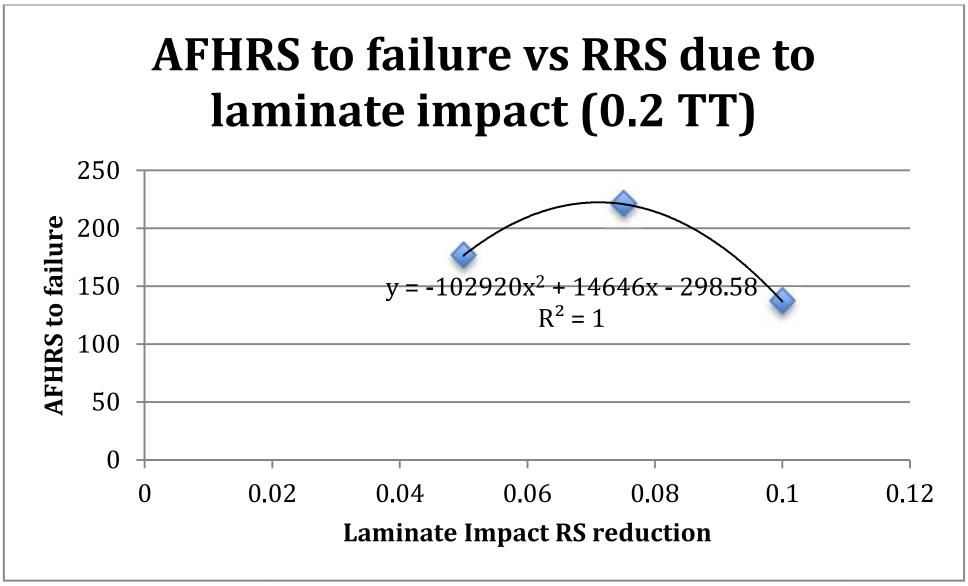

Figure 27: AFHRS to failure versus Laminate Damage RRS

(with Through-Thickness RRS = 0.2)

Through Thickness Damage RRS

The variation in time to failure versus RRS due to through-thickness damage was investigated by setting the RRS value to 0.1, 0.2 and 0.25. After 40 runs[18] at each RRS value, the average time to failure was determined and plotted against Through-Thickness RRS in *Figure 28*.

[18] This number was selected to provide a large sample size, but minimise the required simulation time.

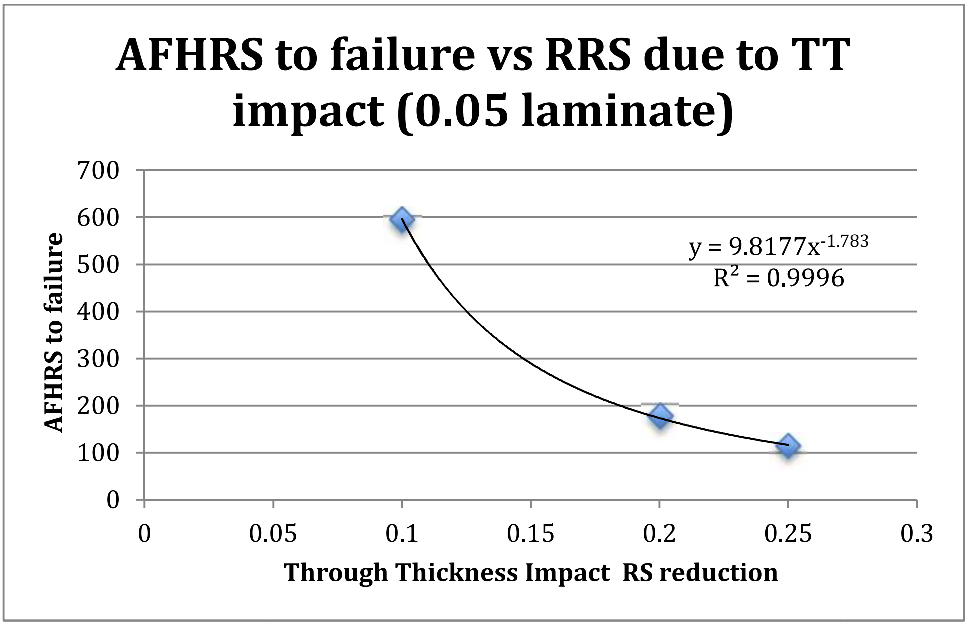

Figure 28: AFHRS to failure versus Through Thickness RRS

(with Laminate Damage RRS = 0.05)

Inspection Interval

The variation in time to failure versus Inspection Interval was investigated by setting the inspection interval for all the DDP to 2, 5, 10 and 20 AFHRS. A linear PoD equation as defined in 6.4.1, was used with all other variables set to the baseline values discussed in Sections 5 and 6. After 40 runs[19] at each inspection interval, the average time to failure was determined and plotted against Inspection Interval in Figure 29.

[19] This number was selected to provide a large sample size, but minimise the required simulation time.

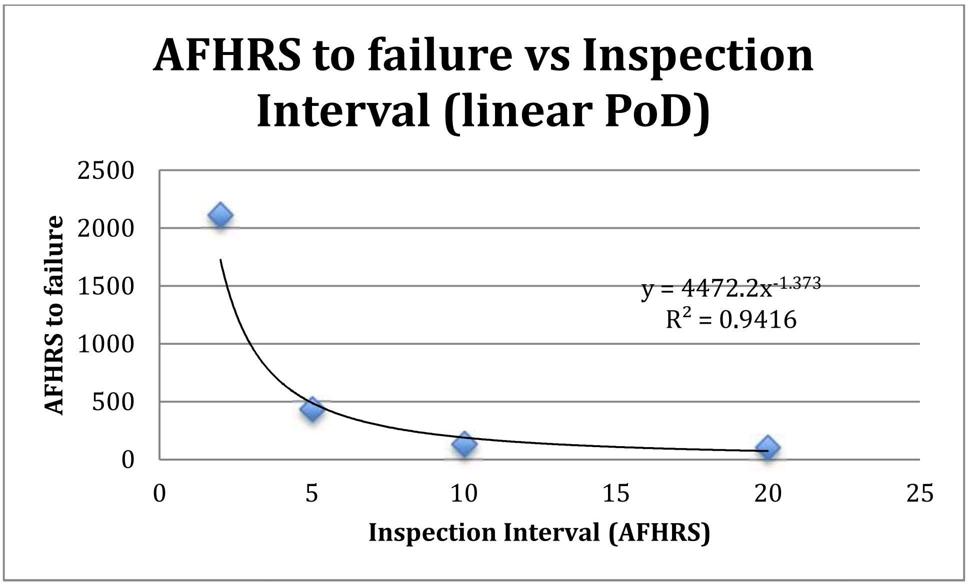

Figure 29: AFHRS to failure versus Inspection Interval (linear PoD)

Probability of Detection Equation

The linear equation defined as the baseline equation, was changed to a logarithmic equation (log PoD), which better represents actual PoD curves (as illustrated in *Figure **22***), was approximated as:

$$PoD = 7.0292*\log(RRS)+96.494$$

Figure 30 illustrates the difference between the linear (red) and logarithmic (green) curves.

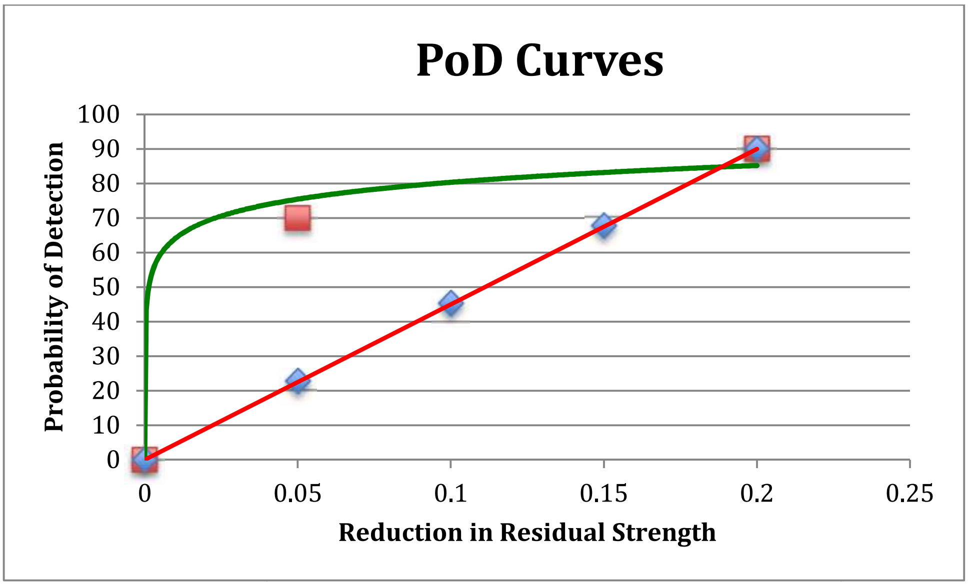

Figure 30: Linear and Logarithmic PoD curves

The variation in time to failure versus Inspection Interval with logarithmic PoD was investigated by setting the inspection interval for all the DDP to 2, 5 and 10 AFHRS. After 40 runs at each inspection interval, the average time to failure was determined and plotted against Inspection Interval in *Figure 31*.

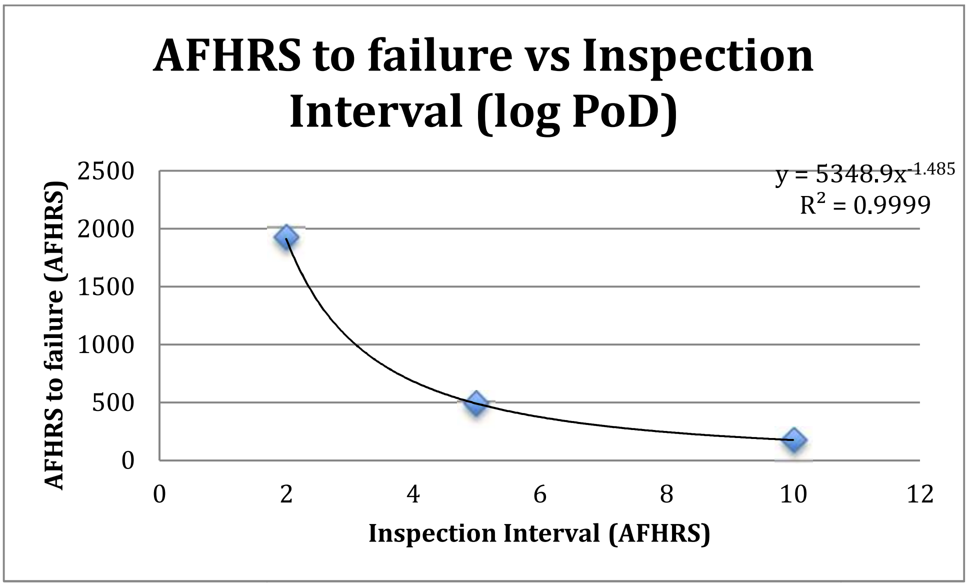

Figure 31: AFHRS to failure versus Inspection Interval (logarithmic PoD)

Moisture Proportionality variable (z)

The variation in time to failure versus Moisture Proportionality variable (z) was investigated by setting *z* to 10^{-5}, 10^{-6} and 10^{-7}. The logarithmic PoD equation was used and Inspection Interval set to 10 and after 40 runs[20] at each inspection interval, the average time to failure was determined and plotted against Inspection Interval in *Figure 32*. The process was then repeated with the Inspection Interval set to 2, *Figure 33*.

[20] This number was selected to provide a large sample size, but minimise the required simulation time.

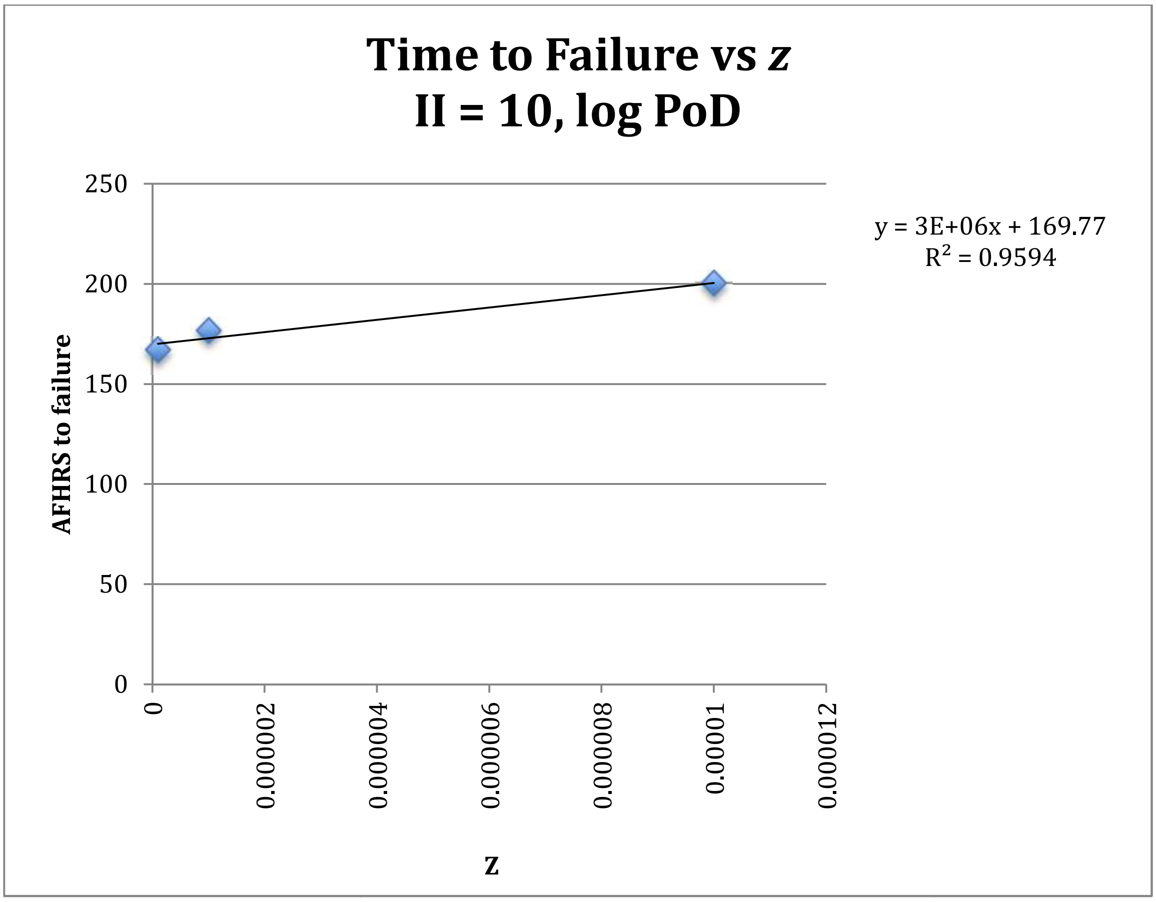

Figure 32: AFHRS to failure versus Moisture Proportionality Variable (z) with logarithmic PoD and Inspection Interval = 10

8.2.2 - SIFCMsim Results

The SoI used for the SIFCMsim was composed of 73 DDP spread over 15 PPE, with inspection intervals ranging from 50 to 500 AFHRS with N = 10,000,000 cycles to fatigue failure. Refer to Appendix D for details of SIFCMsim_Input.csv files representing the following SoI:

a. SoI 1: No fail-safe PSE.

b. SoI 2: Six paired fail-safe PSE and one triple fail-safe PSE.

c. SoI 3: Mixture of fail-safe and non-fail-safe PSE.

Over 40 simulations[21] the average time to failure for three structures SoI 1, 2 and 3 are presented in *Table 14*.

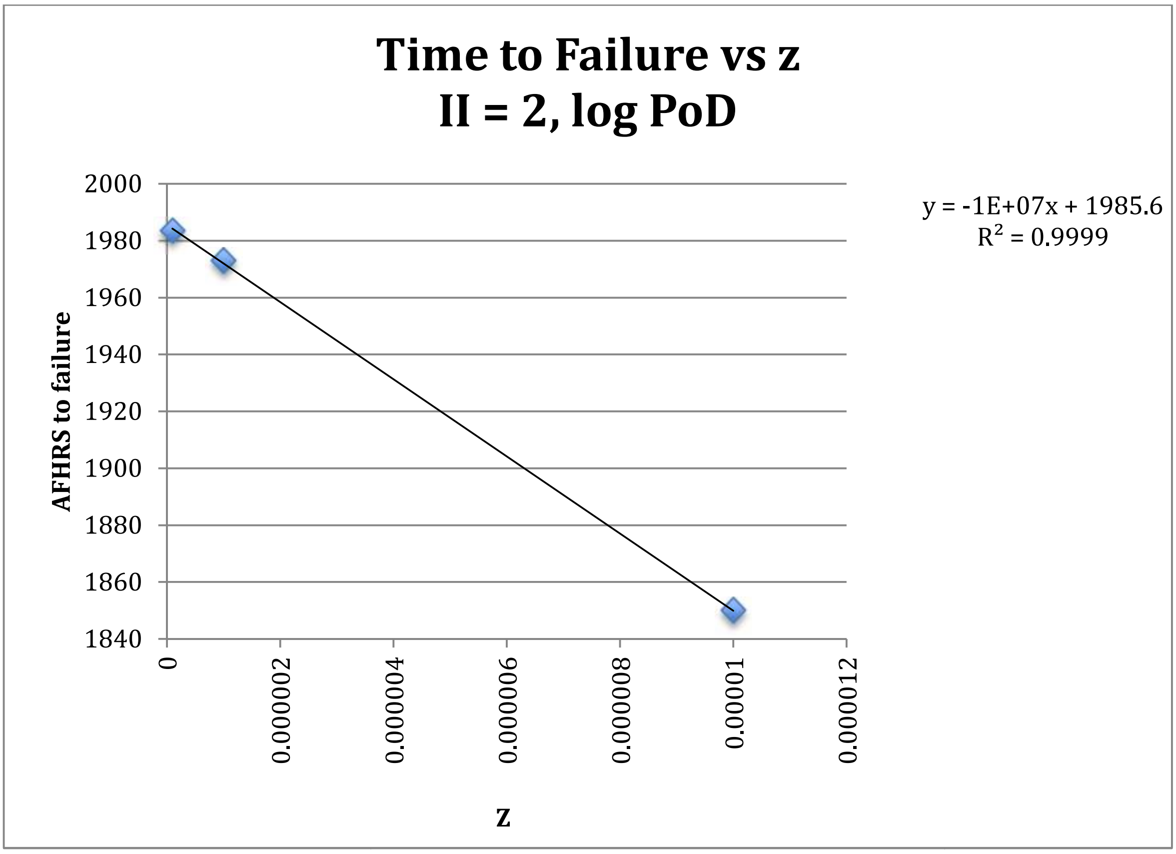

Figure 33: AFHRS to failure versus Moisture Proportionality Variable (z) with logarithmic PoD and Inspection Interval = 2

Time to Failure (AFHRS)	SoI 1	SoI 2	SoI 3
Average	1208	9239	3651
Standard Deviation (SD)	524	869	1639
SD (%)	43%	9.4%	45%

Table 14: Time to Failure Comparison

[21] This number was selected to provide a large sample size, but minimise the required simulation time.

The large standard deviations for SoI 1 and 3 are due to the inclusion of non-fail-safe PSE within the structure. The non-fail-safe PSE are more susceptible to the probabilistic nature of impact damage, whereas the fail-safe PSE in SoI 2 averages out the probabilistic nature of the impact damage, thus reducing the standard deviation.

8.3 - SIFCM Analysis

8.3.1 - Impact Damage

It was expected that as the RRS (magnitude of damage) for Through-Thickness and Laminate impact damage was increased, the time to structural failure would decrease. This expectation is supported by *Figure 28* (Through-Thickness RRS variations), however variations in Laminate RRS (*Figure 27*) did not produce a consistent trend.

The probabilistic nature of impact damage can result in structural failure even with a frequent inspection period. This is illustrated by the structure failing in most cases when the SIFCM_PSE results were generated (refer to *Figure 29*). When the effect of impact damage was removed (by reducing the Through-Thickness and Laminate damage RRS to 0), the structure exceeded a life of 100,000 AFHRS without failure (inspection interval of 5 AFHRS), well in excess of the 2106 AFHRS achieved with an inspection interval of 2 AFHRS with impact damage present (as illustrated in *Figure 29*). This could lead to the invalidation of the structural integrity methodologies:

a. Safe Life: considers fatigue only and therefore lifing would be invalidated by the probabilistic nature of impact damage.
b. Fail-Safe: provides additional protection from the probabilistic nature of impact damage, as illustrated using SIFCMsim, where there as a 760% increase in time to failure for a fail-safe (SoI 2) compared to a non-fail-safe structure (SoI 1) (Table 12). As expected, the mixture of fail-safe and non-fail-safe structure had a time to failure between the two extremes.

c. Damage Tolerance: The inspection intervals used as part of damage tolerance can be invalidated by the probabilistic nature of impact damage. For example, it is possible for two impacts to occur with sufficient severity to cause structure failure prior to the next inspection interval as the interval was selected based on the degradation from a single impact. To mitigate this risk,

   i) The use of redundant (fail-safe) structures should be used for composite structures expected to experience a severe impact environment, and

   ii) Increase the probability of detection of impact damage, including mechanisms to actively monitor structural condition.

8.3.2 – PoD Sensitivity Analysis

Equations of Time To Failure (*TTF*) vs Inspection Interval (*II*) for the linear and logarithmic PoD equations, as illustrated in *Figure 29* and *Figure 31* are compared in *Table 15*.

Linear (Lin) (from *Figure 29*): $TTF = 4472.2.II^{-1.373}$

Logarithmic (Log) (from *Figure 31*): $TTF = 5348.9.II^{-1.485}$

Inspection Interval (II)	Linear (TTF – AFHRS)	Logarithmic (TTF – AFHRS)	Difference (Log – Lin)
1	4472	5349	877
2	1727	1911	184
3	990	1047	57
4	667	683	16
5	491	490	-1
10	189	175	-14
15	109	96	-13
20	73	63	-10
25	54	45	-9
30	42	34	-8
35	34	27	-7

Table 15: PoD Equation Comparison

It can be observed that at shorter Inspection Intervals, the logarithmic equation provides longer structure life. However, as the Inspection Interval extends beyond 5 AFHRS, the linear equation provide for slightly longer life.

8.3.3 – Moisture Proportionality Variable Sensitivity Analysis

The relationship between *TTF* vs 'Moisture Proportionality Variable (z)' was determined for two Inspection Intervals, 2 AFHRS (*Figure 32*) and 10 AFHRS (*Figure 33*) using the logarithmic PoD curve:

$$II = 2: TTF = -1\text{x}10^7.z + 1985.6$$

$$II = 10: TTF = 3\text{x}10^6.z + 169.77$$

The different inspection intervals illustrate opposite trends, for $II = 2$ *TTF* increases with increasing z, whereas for $II = 10$, *TTF* decreases with increasing z. The result for $II = 10$ is more intuitive as increasing z increases the RRS due to moisture absorption.

9. CONCLUSION

The use of composites within aircraft structures has been increasing over a number of decades. This increase in structural usage combined with the quest for increased performance, decreased weight and more optimised aircraft structures.

Aircraft Structural Integrity Management has evolved, primarily based on metallic aircraft structures. This evolution has generally been reactionary based on aircraft accidents, such as de Havilland Comet (1954), Boeing B-47 (1958) and General Dynamics F-111 (1969). However, there has been preemptive evolution to support composite aircraft structures, such as the release of AC 20-107 (1978) to support composite structural certification.

A review of the ASI System (composed of regulators, airlines, aircraft OEMs, flying public and accident investigation organisations) indicates that it can be defined as ultra-safe system, which appears to be highly resilient to the introduction of a new material system, such as composites. The introduction of composites into the ASI System has occurred over a number of decades in small, incremental stages. A review of long-term testing of composite structures, primarily be NASA, indicated that if designed correctly, composite structures can perform as well, if not better, than contemporary metallic structures. This was support by a review of glider accidents in the NTSB accident databases, which did not reveal any adverse structural integrity trends for composite sailplanes. Due to the needs to minimise weight, sailplanes have highly optimised structures, with minimal Residual Strength.

After reviewing the difference in degradation modes between metallic and composite structures, it was identified that the most significant composite degradation mode, is that of impact. Impact damage by its nature, is probabilistic and difficult to detect visually in composite structures. A method of simulating Structural Integrity, the Structural Integrity Failure Causation Model (SIFCM) was developed. Using the SIFCM, it was identified that a combination of fail-safe and damage tolerant structures should be used for composite structures anticipated to experience a severe impact environment. This should be combined with:

a. The probability of detection for impact damage should be as high as possible. This could be achieved by developing structures, which visually indicate impact damage, development of improved NDI techniques and ultimately the use of active monitoring of the structure. Active monitoring of the structure would account weakness of scheduled inspection intervals to probabilistic damage.
b. ASI standards such as MIL-STD-1530C, should be modified to include more specific guidance regarding composite unique degradation modes, specifically impact. This would include the requirement to record aircraft impact damage (refer to following paragraph) and would include reference to authoritative composite material characterisation information, such as CHM-17.
c. Development of impact damage criteria and definition of impact environment to support design and certification activities. This would require a change to MIL-STD-1530C (and applicable civil aviation regulations) to add the requirement for operators to record instances of in-service impact damage for provision to the relevant airworthiness authority. This would allow an appropriate impact environment to be defined for specific aircraft categories. From this environment, impact damage criteria used to support design could then be developed.

10. RECOMMENDATIONS

It is recommended that future research be conducted in the following areas:

a. Development of impact damage criteria and definition of impact environment to support future aircraft designs.

b. Development of smart structures, which can be actively monitored for impact damage.

c. Refinement of SIFCMsim, including the values for RRS.

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APPENDIX A – SIFCM_PSE FILES

SIFCM_PSE.m

```
%This is SIFCM_PSE
% It has been moved from SIMULINK due to the introduction of a large number
% of matrices.
% This is the main program for the SIFCM_PSE model.
% Currently only using four Detail Design Points (DDP).
% ++++++++++++++++++++++++++++++++++
% STATUS 28/04/13
%++++++++++++++++++++++++++++++++++
%*****************************************
%Block 0 - Variable definition
PoD = [0,0,0,0];    %PoD value for each DDP...initially set to 0
I_Count = [0,0,0,0];
PoD_function = 4.5; %Linear relationship between RS and PoD
Fail_Safe_AFHRS = 100000; %maximum AFHRS which the simulation will run
DDP_Var = zeros(4,Fail_Safe_AFHRS)
%Block 1 - Establish initial residual strength of DDPs
disp('Establish RS at manufacture');

pse_manufacture = PSE_initialisation %m-script call

%Time Loop Start- Loop over time
pse_current = pse_manufacture;
%pse_current is the time-based variable for RS
SoI_intact = 1;
AFHRS = 0;
while SoI_intact > 0
% Establish timescale
% Time step = each loop cycle = 1 AFHR.

% ****************************************
% Block 2 - Impact of Degradation on RS

pse_degradation_timestep = degradation; %m-script call, ED and AD

% ++++++++++++Accidental Damage (AD)++++++++++
qdep=rand(1,4);           % establish probability for surface impact damage
P_Surf = 0.1;             % Probability of Surface Damage
AD1_RS = 0;               % RS degradation of Surface Damage
P_Laminate = 0.15;        % Probability of Laminate Damage
AD2_RS = 0;            % RS degradation of Laminate Damage
P_T_Thickness = 0.175;  % Probability of Through-Thickness Damage
AD3_RS = 0;              % RS degradation of Through-Thickness Damage
RS_Impact = [0 0 0 0];    % reset Moisture array
% Impact Damage/Degradation
for i = 1:length(qdep)
    if qdep(i)<P_Surf
        RS_Impact(i) = pse_current(i)-AD1_RS;
    end
    if qdep(i)<P_Laminate
        if qdep(i)>P_Surf
            RS_Impact(i) = pse_current(i)-AD2_RS;
        end
```

```
    end
    if qdep(i)<P_T_Thickness
        if qdep(i)>P_Laminate
            RS_Impact(i) = pse_current(i)-AD3_RS;
        end
    end
    if qdep(i)>P_T_Thickness
        RS_Impact(i) = pse_current(i);
    end
end
pse_current = RS_Impact;

% ++++++++++++Environmental Damage (ED)++++++++++
% ED1 - Moisture degradation
Max_Moisture_Content = 1.2729;    % Maximum moisture content (%)
Moisture_RS_factor = 0.000001;       % Relationship between moisture
content and residual strength

Moisture_Content = 1-(0.003/(0.003+0.00291))*exp(-0.003*AFHRS); %determine
current moisture level
if Moisture_Content < Max_Moisture_Content
else
    Moisture_Content = Max_Moisture_Content;
end

moisture_effect = Moisture_Content*Moisture_RS_factor;
RS_Moisture = [0 0 0 0];    % reset Moisture array
for i = 1:length(RS_Moisture)
    RS_Moisture(i) = pse_current(i)-moisture_effect;
end

pse_current = RS_Moisture;

% ++++++++++++Fatigue Damage (FD)++++++++++
% Definition of variables:
N = 10000000;                   % Cycles to failure at applied cyclic load
cycle_AFHR = 10;            % Cycles per AFHR
RS_Fatigue = [0 0 0 0];  % reset the Fatigue Damage array
for i = 1:length(RS_Fatigue)
    fatigue_effect = (1/pse_current(i))^((AFHRS*cycle_AFHR)/(N-1));
    RS_Fatigue(i) = fatigue_effect*pse_current(i);
end
pse_current = RS_Fatigue;

% ***************************************
% Block 3 - Perform maintenance activities
% use an input file (txt) to define the maintenance schedule for the
% structure - Maintenance_Schedule.
Inspection_Interval = [5,5,5,5]; %AFHRS between inspections per DDP
for i = 1:length(Inspection_Interval)
% count AFHRS until next inspection per DDP
    temp = I_Count(i);
    I_Count(i) = temp + 1;
% +++++++++++++++++++++++++
    if I_Count(i) == Inspection_Interval(i)
        %Block 3A - perform inspection
        %Determine PoD per DDP
        disp('++++++Performing Inspection+++++++++');
        pse_current(i);
        RRS = 1.5 - pse_current(i);
```

```
        temp1 = RRS*PoD_function; %linear PoD relationship
%        temp1 = (7.0292*log(RRS)+96.494)/100;
        if temp1 > 0.9
            PoD(i) = 0.9;
        else
            PoD(i) = temp1;
        end
        %++++++++++++++++++++++
        %Was damage detected?
        inspection_prob = rand; %probability of successful inspection
        if inspection_prob < PoD(i)
            % damage has been detected, determine response
            if RRS > 0.1
                % perform repair
                disp('++++++Perform Repair++++++++');
                PoD(i) = 0; % reset PoD value.
                pse_repair = PSE_initialisation; %m-script call
                pse_current(i) = pse_repair(i)
            else
                % monitor damage
                disp('*******Monitor Damage********');
                PoD(i) = 1;
            end
        else
            % damage not detected....no further action
        end
        %++++++++++++++++++++++
        I_Count(i) = 0; %reset the inspection interval counter
    else
        %no further action performed
    end
end

%*****************************************
%Block 4 - Determine loop end
% Add current value of RS to tracking matrix DDP_Var
if AFHRS==0
else
    for i = 1:length(pse_current)
        DDP_Var(i,AFHRS) = pse_current(i);
    end
end
% Time Loop Stop - when structure fails
for i = 1:length(pse_current)
    if pse_current(i) < 1
       SoI_intact = 0;
       disp('+++++ Structure has failed ++++++');
       x = 1:4;
       plot(x,pse_current,'r*')
       ylabel('Residual Strength')
       title('Residual Strength at failure')
    end
end
% This decision statement is used as a fail-safe
if AFHRS == Fail_Safe_AFHRS %AFHRS fail-safe
    SoI_intact = 0;
    disp('+++++Loop Fail-Safe Enacted+++++');
    x = 1:4;
    figure(1)
    plot(x,pse_current,'r*')
    ylabel('Residual Strength')
```

```
    title('Residual Strength at Fail-Safe')
    hold on
else
    AFHRS = AFHRS + 1;
    disp('^^^^^^^^^^^^^^^End of Cycle^^^^^^^^^^^^^^^^^^');
end
end
AFHRS
pse_current
% print out graphs of RS variations
AFHRS_seq=(1:AFHRS);
figure(2)
plot (AFHRS_seq,DDP_Var(1,AFHRS_seq))
title('DDP1 variation in RS')
hold on
figure(3)
plot (AFHRS_seq,DDP_Var(2,AFHRS_seq))
title('DDP2 variation in RS')
hold on
figure(4)
plot (AFHRS_seq,DDP_Var(3,AFHRS_seq))
title('DDP3 variation in RS')
hold on
figure(5)
plot (AFHRS_seq,DDP_Var(4,AFHRS_seq))
title('DDP4 variation in RS')
```

PSE_initialisation.m

```
function pse_manufacture = PSE_initialisation
% PSE_initialisation
% Purpose: Set the initial Residual Strength (RS) values of the
% PSE matrix.
% This function will include the impact of design and maufacturing defects.
% ++++++++++++++++++++++++++++++++++
% STATUS 18/10/12
%+++++++++++++++++++++++++++++++++++
% Establish initial RS values (4 Detail Desing Points - DDP)
% Define variables
RS_initial = 1.5;
pse_ideal = [RS_initial RS_initial RS_initial RS_initial];

design_base_defect = 0.02;
BSdesign_base_defect = 0.1;
manufacture_base_defect = 0.01;
BSmanufacture_base_defect = 0.2
pse_manufacture = pse_ideal;
% **********************************
% Determine effect of Design Defects
disp('Design-Quality*************');
qdep=rand(1,4)          % establish probability for the defect of interest
% Quality Design Defect
defect_RS = [0 0 0 0];  % reset the defect array
for i = 1:length(qdep)
    if qdep(i) < 0.1
       defect_RS(i) = rand*design_base_defect;
    end
```

```
    if qdep(i) > 0.9
       defect_RS(i) = rand*design_base_defect;
    end
end
defect_RS
for j = 1:length(pse_manufacture)
    pse_manufacture(j) = pse_ideal(j) - defect_RS(j);
end
% Black Swan Design Defect
disp('Design-Black Swan***********');
defect_RS = [0 0 0 0];  % reset the defect array
qdep=rand(1,4)          % establish probability for the defect of interest
for i = 1:length(qdep)
    if qdep(i) < 0.01
       defect_RS(i) = rand*BSdesign_base_defect;
    end
    if qdep(i) > 0.99
       defect_RS(i) = rand*BSdesign_base_defect;
    end
end
defect_RS
for j = 1:length(pse_manufacture)
    pse_manufacture(j) = pse_manufacture(j) - defect_RS(j);
end

% *************************************
% *************************************
% Determine effect of Manufacturing Defects
disp('Manufacture-Quality***********');
qdep=rand(1,4)             % establish probability for the defect of
interest
defect_RS = [0 0 0 0];     % reset the defect array
% Quality Manufacture Defect
for i = 1:length(qdep)
    if qdep(i) < 0.1
       defect_RS(i) = rand*manufacture_base_defect;
    end
    if qdep(i) > 0.9
       defect_RS(i) = rand*manufacture_base_defect;
    end
end
defect_RS
for j = 1:length(pse_manufacture)
    pse_manufacture(j) = pse_manufacture(j) - defect_RS(j);
end
% Black Swan Manufacture Defect
disp('Manufacture-Black Swan***********');
defect_RS = [0 0 0 0];  % reset the defect array
qdep=rand(1,4)          % establish probability for the defect of interest
for i = 1:length(qdep)
    if qdep(i) < 0.01
       defect_RS(i) = rand*BSmanufacture_base_defect;
    end
    if qdep(i) > 0.99
       defect_RS(i) = rand*BSmanufacture_base_defect;
    end
end
defect_RS
for j = 1:length(pse_manufacture)
    pse_manufacture(j) = pse_manufacture(j) - defect_RS(j);
end
```

```
end
```

APPENDIX B – SIFCMsim FILES

SIFCMsim.m

```
%This is SIFCMsim
% Date: 11/5/13
% This is the main program for the SIFCMsim model.
% Multiple PSE with multiple DDP
% ++++++++++++++++++++++++++++++++++
% STATUS 11/5/13
% Remaining actions:
% -
%++++++++++++++++++++++++++++++++++++
%****************************************
%30 is the number of DDP in the SoI
DDPnum = 73;
%Block 0 - Establish SoI matrix
load SIFCMsim_Input.csv;    %load maintenance data file
SIFCMsim_Input(2,1)
SoI = zeros(DDPnum,12);
NtoFailure = zeros(DDPnum);
for i=1:DDPnum
    SoI(i,1)=SIFCMsim_Input(i,2); %PSE ID
    SoI(i,3)=SIFCMsim_Input(i,3); %Impact Structure Type
    SoI(i,4)=SIFCMsim_Input(i,4); %Fail-Safe Flag
    SoI(i,5)=SIFCMsim_Input(i,5); %Fail-Safe PSE partner
    SoI(i,6)=SIFCMsim_Input(i,6); %Fail-Safe PSE partner
    SoI(i,7)=SIFCMsim_Input(i,7); %Fail-Safe PSE partner
    SoI(i,10)=SIFCMsim_Input(i,9); %Inspection Interval
    NtoFailure(i)=SIFCMsim_Input(i,8); %cycles to fatigue failure
end
%Block 0.5 - Variable definition
SoI
PoD = zeros(1,DDPnum);    %PoD value for each DDP...initially set to 0
I_Count = zeros(1,DDPnum);
Failed_DDP_RRS = 0.002;  %RS reduction due to failure of partner DDP.
N_reduction = 0.98;     %percentage reduction in N due to failure of
partner DDP.
PoD_function = 4.5; %Linear relationship between RS and PoD
Fail_Safe_AFHRS = 15000; %maximum AFHRS which the simulation will run
DDP_Var = zeros(DDPnum,Fail_Safe_AFHRS);
FS_PSE_failed=0;
%Block 1 - Establish initial residual strength of DDPs
disp('Establish RS at manufacture');

pse_manufacture = PSE_initialisation %m-script call
for i=1:DDPnum
    SoI(i,2)=pse_manufacture(i);
 %   SoI(i,3)=4;
end
%Time Loop Start- Loop over time
%pse_current = pse_manufacture;
%pse_current is the time-based variable for RS
SoI_intact = 1;
AFHRS = 0;
while SoI_intact > 0
% Establish timescale
% Time step = each loop cycle = 1 AFHR.
```

```
% ***************************************
% Block 2 - Impact of Degradation on RS
for i=1:DDPnum
    currentDDP = SoI(i,2);
    currentType = SoI(i,3);
    % ++++++++++++Damage Variables++++++++++
    % Accidental Damage - Impact
    qdep=rand;                  % establish probability for surface impact
damage
    P_Surf = [0.001139727,0.000375265,0,0];             % Probability of
Surface Damage
    AD1_RS = 0;                 % RS degradation of Surface Damage
    P_Laminate = [0.001650789,0.000470923,0.000277941,0];      %
Probability of Laminate Damage [High/Intermediate/Low]
    AD2_RS = 0.05;              % RS degradation of Laminate Damage
    P_T_Thickness = [0.001004032,0.000872278,0.000134814,0];  % Probability
of Through-Thickness Damage [High/Intermediate/Low]
    AD3_RS = 0.2;               % RS degradation of Through-Thickness Damage
    RS_Impact = 0;      % reset Impact array
    % Environmental Damage - Moisture degradation
    Max_Moisture_Content = 1.2729;     % Maximum moisture content (%)
    Moisture_RS_factor = 0.000001;        % Relationship between moisture
content and residual strength

    Moisture_Content = 1-(0.003/(0.003+0.00291))*exp(-0.003*AFHRS);
%determine current moisture level
    if Moisture_Content < Max_Moisture_Content
    else
        Moisture_Content = Max_Moisture_Content;
    end
    moisture_effect = Moisture_Content*Moisture_RS_factor;
    RS_Moisture = 0;  % reset Moisture array
    % Fatigue Damage
    cycle_AFHR = 10;                    % Cycles per AFHR
    RS_Fatigue = 0;    % reset the Fatigue Damage array

    %Start of Degradation calculations
    % Impact Damage/Degradation
        if qdep<P_Surf(currentType)
            RS_Impact = currentDDP-AD1_RS;
        end
        if (qdep-P_Surf(currentType))<P_Laminate(currentType)
            RS_Impact = currentDDP-AD2_RS;
        end
        if (qdep-
(P_Surf(currentType)+P_Laminate(currentType)))<P_T_Thickness(currentType)
            RS_Impact = currentDDP-AD3_RS;
        end
        if qdep>P_T_Thickness(currentType)
            RS_Impact = currentDDP;
        end
    currentDDP = RS_Impact;
    % ++++++++++++Environmental Damage (ED)++++++++++
    RS_Moisture = currentDDP-moisture_effect;
    currentDDP = RS_Moisture;
    % ++++++++++++Fatigue Damage (FD)++++++++++
    N=NtoFailure(i);
    fatigue_effect = (1/currentDDP)^((AFHRS*cycle_AFHR)/(N-1));
    RS_Fatigue = fatigue_effect*currentDDP;
    currentDDP = RS_Fatigue;
    SoI(i,2) = currentDDP;
```

```
% ***************************************
% Block 3 - Perform maintenance activities (still to do)
% use an input file (txt) to define the maintenance schedule for the
% structure - Maintenance_Schedule.

% count AFHRS until next inspection per DDP
% +++++++++++++++++++++++++
    if SoI(i,11) == SoI(i,10)
        %Block 3A - perform inspection
        %Determine PoD per DDP
        %disp('++++++Performing Inspection+++++++++');
        RRS = 1.5 - SoI(i,2);
        temp1 = RRS*PoD_function; %linear PoD relationship
%         temp1 = (7.0292*log(RRS)+96.494)/100;
        if temp1 > 0.9
            SoI(i,12) = 0.9;
        else
            SoI(i,12) = temp1;
        end
        %+++++++++++++++++++++
        %Was damage detected?
        inspection_prob = rand; %probability of successful inspection
        if inspection_prob < SoI(i,12)
            % damage has been detected, determine response
            if RRS > 0.1
                % perform repair
                %disp('++++++Perform Repair++++++++');
                SoI(i,12) = 0; % reset PoD value.
                pse_repair = PSE_initialisation; %m-script call
                SoI(i,2) = pse_repair(i);
                SoI(i,9)=0;
                SoI(i,8)=0;
            else
                % monitor damage
                %disp('*******Monitor Damage********');
                SoI(i,12) = 1;
            end
        else
            % damage not detected....no further action
        end
        %+++++++++++++++++++++
        SoI(i,11) = 0; %reset the inspection interval counter
    else
        %no further action performed
    end
%***************************************
%Block 4 - Determine loop end
% Time Loop Stop - when structure fails
    if SoI(i,2)<1
        if SoI(i,4)==1 %Fail-safe DDP is allowed to fail and not result in
structural failure.
            %Degradation still occurs to the damaged DDP.
            Failed_PSE = SoI(i,1);
            if SoI(i,8)==0      %initial reduction in RS of partner DDP
                if AFHRS > 1000
                    SoI(i,9)=AFHRS/1000;
                else
                    SoI(i,9)=AFHRS;      %AFHRS for failure
                end

                SoI(i,8)=1;     %this needs to be reset to zero when DDP is
```

```
repaired
                for j=1:DDPnum
                    if SoI(j,5)==Failed_PSE
                        RS_DDP=SoI(j,2);
                        N_Fatigue=NtoFailure(j);
                        SoI(j,2)=RS_DDP-Failed_DDP_RRS;
                        NtoFailure(j)=N_Fatigue*N_reduction;
                        if SoI(j,8)==0
                            disp('FS_PSE1');
                            FS_PSE_failed = 1;
                        end
                    end
                    if SoI(j,6)==Failed_PSE
                        RS_DDP=SoI(j,2);
                        N_Fatigue=NtoFailure(j);
                        SoI(j,2)=RS_DDP-Failed_DDP_RRS;
                        NtoFailure(j)=N_Fatigue*N_reduction;
                        if SoI(j,8)==0
                            FS_PSE_failed = 1;
                            disp('FS_PSE2');
                        end
                    end
                    if SoI(j,7)==Failed_PSE
                        RS_DDP=SoI(j,2);
                        N_Fatigue=NtoFailure(j);
                        SoI(j,2)=RS_DDP-Failed_DDP_RRS;
                        NtoFailure(j)=N_Fatigue*N_reduction;
                        if SoI(j,8)==0
                            FS_PSE_failed = 1;
                            disp('FS_PSE3');
                        end
                    end
                end
                for k=1:DDPnum
                    if SoI(k,1)==Failed_PSE
                         if k==i
                             %this is the DDP which has been identified as
                             %failed.
                         else
                             RS_DDP=SoI(k,2);
                             N_Fatigue=NtoFailure(k);
                             SoI(k,2)=RS_DDP-Failed_DDP_RRS;
                             NtoFailure(k)=N_Fatigue*N_reduction;
                             if SoI(k,8)==0
                                FS_PSE_failed = 1;
                                disp('FS_PSE4');
                             end
                         end
                    end
                end
    %This decision statement is used as a fail-safe
    %determine if fail-safe structure has failed (all DDP in fail-safe
    %partner PSE failed)
                FS_PSE_failed
                if FS_PSE_failed==0
                    SoI_intact = 0
                    disp('fail-safe structure has failed');
                else
                    FS_PSE_failed=0;
                end
            end
```

```
        else
            SoI_intact = 0;
            i
            SoI(i,2)
            disp('^^^^^^^^^^^^^^^Structure Failed^^^^^^^^^^^^^^^^^^');
        end
    end
    temp = SoI(i,11);
    SoI(i,11) = temp + 1;
end

if AFHRS == Fail_Safe_AFHRS %AFHRS fail-safe
    SoI_intact = 0;
    disp('^^^^^^^^^^^^^^^Simulation Timed Out^^^^^^^^^^^^^^^^^^');
else
    AFHRS = AFHRS + 1;
    disp('^^^^^^^^^^^^^^^End of Cycle^^^^^^^^^^^^^^^^^^');
end
end
AFHRS
SoI
```

PSE_initialisation.m (modified for SIFCMsim)

```
function pse_manufacture = PSE_initialisation
% PSE_initialisation for SIFCMsim
% Purpose: Set the initial Residual Strength (RS) values of the
% PSE matrix.
% This function will include the impact of design and maufacturing defects.
% ++++++++++++++++++++++++++++++++++
% STATUS 11/5/13
% Remaining actions:
% -
%++++++++++++++++++++++++++++++++++++
% Establish initial RS values (4 Detail Desing Points - DDP)
% Define variables
DDPnum=73; %must be changed to match the same variable value in SIFCMsim.m
RS_initial = 1.5;
pse_ideal = ones(1,DDPnum)*RS_initial;
design_base_defect = 0.02;
BSdesign_base_defect = 0.1;
manufacture_base_defect = 0.01;
BSmanufacture_base_defect = 0.2;
pse_manufacture = pse_ideal;
% *********************************
% Determine effect of Design Defects
%disp('Design-Quality************');
qdep=rand(1,DDPnum);          % establish probability for the defect of
interest
% Quality Design Defect
defect_RS = zeros (1,DDPnum);  % reset the defect array
for i = 1:length(qdep)
    if qdep(i) < 0.1
       defect_RS(i) = rand*design_base_defect;
    end
    if qdep(i) > 0.9
       defect_RS(i) = rand*design_base_defect;
    end
```

```
end
for j = 1:length(pse_manufacture)
    pse_manufacture(j) = pse_ideal(j) - defect_RS(j);
end
% Black Swan Design Defect
%disp('Design-Black Swan***********');
defect_RS = zeros(DDPnum); % reset the defect array
qdep=rand(1,DDPnum);          % establish probability for the defect of
interest
for i = 1:length(qdep)
    if qdep(i) < 0.01
       defect_RS(i) = rand*BSdesign_base_defect;
    end
    if qdep(i) > 0.99
       defect_RS(i) = rand*BSdesign_base_defect;
    end
end
for j = 1:length(pse_manufacture)
    pse_manufacture(j) = pse_manufacture(j) - defect_RS(j);
end

% ************************************
% ************************************
% Determine effect of Manufacturing Defects
%disp('Manufacture-Quality***********');
qdep=rand(1,DDPnum);            % establish probability for the defect of
interest
defect_RS = zeros(1,DDPnum);    % reset the defect array
% Quality Manufacture Defect
for i = 1:length(qdep)
    if qdep(i) < 0.1
       defect_RS(i) = rand*manufacture_base_defect;
    end
    if qdep(i) > 0.9
       defect_RS(i) = rand*manufacture_base_defect;
    end
end
for j = 1:length(pse_manufacture)
    pse_manufacture(j) = pse_manufacture(j) - defect_RS(j);
end
% Black Swan Manufacture Defect
%disp('Manufacture-Black Swan***********');
defect_RS = zeros(1,DDPnum);  % reset the defect array
qdep=rand(1,DDPnum);          % establish probability for the defect of
interest
for i = 1:length(qdep)
    if qdep(i) < 0.01
       defect_RS(i) = rand*BSmanufacture_base_defect;
    end
    if qdep(i) > 0.99
       defect_RS(i) = rand*BSmanufacture_base_defect;
    end
end
for j = 1:length(pse_manufacture)
    pse_manufacture(j) = pse_manufacture(j) - defect_RS(j);
end

end
```

APPENDIX C – SIFCM_PSE OUTPUT

Variation in Inspection Interval (Linear PoD)

130429 - SIFCM_PSE		linear PoD relationship		PoD = 450*RRS	
Inspection Interval	10 (fail-safe 300)	20	5	5 (fail-safe increased to 1000)	2 (fail-safe increased to 3000)
Sample Run	AFHRS to failure				
1	179	188	300+	499	2151
2	300+	38	300+	364	2087
3	300+	32	14	149	1958
4	179	133	300+	127	2799
5	300+	97	Stop	49	1455
6	143	256		278	2907
7	27	19		789	2555
8	300+	58		794	1627
9	18	115		298	1379
10	81	253		254	2311
11	300+	33		910	2492
12	97	79		654	1689
13	108	36		223	1847
14	98	41		402	1853
15	239	239		103	3000+
16	178	98		179	2927
17	149	56		805	3000+
18	47	16		159	2834
19	59	30		699	2007
20	57	17		304	2451
21	300+	53		1000+	2392
22	258	237		37	2749
23	168	271		187	1739
24	179	187		443	1907
25	35	93		571	3000+
26	194	54		879	2449
27	119	117		12	1528
28	77	178		407	1247
29	300+	26		334	2635
30	300+	115		353	3000+
31	166	99		534	3000+
32	169	194		906	2959
33	117	78		724	2294
34	33	118		262	1905
35	106	300+		303	2981

36	201	158		289	155
37	101	31		624	3000+
38	300+	70		488	2147
39	138	73		934	2335
40	168	16		587	883
	10	20		5	2
Average	125.4193548	102.6153846		433.6666667	2106.882353

Where the cycle-limit was reached (no structural failure), the cycle-limit is identified (e.g. 300+) and that sample is not considered, when calculating the average.

Variation in Inspection Interval (Logarithmic PoD)

Inspection interval = 5		10	2
Sample Run	AFHRS to failure		
1	403	468	225
2	124	119	2725
3	1619	216	316
4	92	271	3118
5	131	93	2053
6	869	436	2465
7	203	348	2272
8	379	107	1189
9	250	175	1429
10	732	57	2629
11	854	148	165
12	573	127	1493
13	548	88	2465
14	509	188	2719
15	776	257	2217
16	446	135	1479
17	784	66	1620
18	1067	89	1825
19	309	418	771
20	294	76	2982
21	123	299	597
22	224	37	2889
23	560	215	2578
24	314	81	2663
25	529	53	1873
26	511	472	302
27	538	158	1543
28	638	46	2146
29	454	158	2601
30	187	24	3383
31	493	427	1943
32	880	27	399
33	784	125	3257
34	63	58	2567
35	248	277	275
36	409	236	3413
37	12	49	1711

38	379	75	2897
39	726	199	2089
40	354	159	1561
	5	10	2
Average	484.7	176.425	1921.1

Variation in Moisture Content (Logarithmic PoD)

	Variation in Moisture Content Factor for RS degradation					
	Baseline (II = 2, log PoD)			Baseline (II = 10, log PoD)		
Sample Run	10^{-6}	10^{-7}	10^{-5}	10^{-6}	10^{-7}	10^{-5}
1	1781	1291	1169	468	21	234
2	1227	3033	2037	119	21	39
3	830	3335	1585	216	46	28
4	1249	2068	2175	271	188	196
5	2263	124	2619	93	286	276
6	3229	2029	118	436	340	61
7	1030	1697	2477	348	118	57
8	1868	2916	1265	107	271	299
9	2536	599	2385	175	67	439
10	2177	2889	2416	57	87	49
11	1373	2578	1281	148	87	258
12	1995	2663	903	127	46	149
13	1965	1873	1363	88	187	29
14	2059	302	1431	188	164	349
15	1924	1543	1931	257	169	387
16	1995	2146	2329	135	188	108
17	1434	2601	649	66	19	319
18	1825	3383	1328	89	247	91
19	2582	1943	1785	418	92	335
20	735	399	2307	76	268	117
21	3805	3257	3205	299	428	267
22	1011	2567	1944	37	58	458
23	2343	275	2707	215	97	253
24	2679	3413	1849	81	69	117
25	746	1711	2478	53	117	94
26	3283	2897	515	472	372	237
27	2047	2089	1915	158	59	219
28	1823	1561	2137	46	69	328
29	1925	1949	3475	158	54	145
30	3217	859	716	24	152	185
31	3623	2967	2667	427	398	48
32	2031	897	2561	27	128	478
33	2193	2341	1915	125	419	333
34	437	1674	1808	58	439	458
35	2043	2196	1667	277	334	16

36	2339	1793	2087	236	148	68
37	1444	2035	1935	49	35	117
38	1171	2210	2739	75	94	107
39	2040	1149	1093	199	154	49
40	2633	2091	1028	159	136	205
	10^{-6}	10^{-7}	10^{-5}	10^{-6}	10^{-7}	10^{-5}
	1972.75	1983.575	1849.85	176.425	166.8	200.05

Variation in Impact Damage Reduction in Residual Strength (Logarithmic PoD)

Variation in Impact RS Degradation			log PoD/0.000001 Moisture/10 II					
	Through-Thickness	0.25	0.2	0.1	0.1		0.2	0.2
Sample Run	Laminate Damage	0.05	0.05	0.05	0.025		0.025	0.1
1		219	468	548	859		284	59
2		128	119	928	556		136	106
3		237	216	381	958		407	291
4		216	271	217	908		233	219
5		108	93	816	295		25	238
6		16	436	568	976		294	103
7		246	348	449	803		84	27
8		202	107	595	539		149	318
9		74	175	685	211		175	49
10		149	57	306	529		296	48
11		82	148	709	773		283	95
12		136	127	264	699		187	127
13		47	88	567	706		199	104
14		75	188	468	591		196	189
15		27	257	389	648		144	67
16		78	135	893	489		88	57
17		41	66	553	509		339	47
18		14	89	869	1098		29	157
19		95	418	597	715		417	9
20		17	76	768	419		366	19
21		128	299	559	552		49	187

22		189	37	360	508		65	24
23		105	215	449	1073		316	196
24		22	81	630	666		99	189
25		256	53	985	659		136	38
26		89	472	746	474		458	28
27		68	158	575	808		129	277
28		16	46	486	565		65	149
29		248	158	758	767		87	158
30		339	24	626	1077		298	39
31		31	427	808	782		329	179
32		34	27	579	891		206	197
33		217	125	974	197		337	39
34		29	58	429	978		214	189
35		90	277	482	597		482	336
36		12	236	618	616		234	157
37		146	49	397	1155		229	124
38		217	75	892	911		279	312
39		88	199	517	722		116	119
40		59	159	307	558		379	207
		0.25	0.2	0.1	0.1	0.2	0.2	0.2
		0.05	0.05	0.05	0.025	0.05	0.075	0.1
Average		**114.75**	**176.425**	**593.675**	**695.925**	**176.425**	**220.95**	**136.825**

APPENDIX D – SIFCMsim INPUT FILES

Structure of Interest 1 Input File

1	1	1	0	0	0	0	10000000	50
2	1	1	0	0	0	0	10000000	50
3	1	1	0	0	0	0	10000000	50
4	1	1	0	0	0	0	10000000	50
5	2	1	0	0	0	0	10000000	500
6	2	1	0	0	0	0	10000000	500
7	2	1	0	0	0	0	10000000	500
8	2	1	0	0	0	0	10000000	500
9	3	4	0	0	0	0	10000000	500
10	3	4	0	0	0	0	10000000	500
11	3	4	0	0	0	0	10000000	500
12	3	4	0	0	0	0	10000000	500
13	4	2	0	0	0	0	10000000	100
14	4	2	0	0	0	0	10000000	100
15	4	2	0	0	0	0	10000000	100
16	4	2	0	0	0	0	10000000	100
17	5	4	0	0	0	0	10000000	100
18	5	4	0	0	0	0	10000000	100
19	5	4	0	0	0	0	10000000	100
20	5	4	0	0	0	0	10000000	100
21	6	2	0	0	0	0	10000000	100
22	6	2	0	0	0	0	10000000	100
23	6	2	0	0	0	0	10000000	100

24	6	2	0	0	0	0	10000000	100
25	7	3	0	0	0	0	10000000	100
26	7	3	0	0	0	0	10000000	100
27	7	3	0	0	0	0	10000000	100
28	7	3	0	0	0	0	10000000	100
29	7	3	0	0	0	0	10000000	100
30	7	3	0	0	0	0	10000000	100
31	8	4	0	0	0	0	10000000	100
32	8	4	0	0	0	0	10000000	100
33	8	4	0	0	0	0	10000000	100
34	8	4	0	0	0	0	10000000	100
35	8	4	0	0	0	0	10000000	100
36	8	4	0	0	0	0	10000000	100
37	9	4	0	0	0	0	10000000	100
38	9	4	0	0	0	0	10000000	100
39	9	4	0	0	0	0	10000000	100
40	9	4	0	0	0	0	10000000	100
41	9	4	0	0	0	0	10000000	100
42	9	4	0	0	0	0	10000000	100
43	10	4	0	0	0	0	10000000	100
44	10	4	0	0	0	0	10000000	100
45	10	4	0	0	0	0	10000000	100
46	10	4	0	0	0	0	10000000	100
47	10	4	0	0	0	0	10000000	100
48	10	4	0	0	0	0	10000000	100
49	10	4	0	0	0	0	10000000	100
50	11	3	0	0	0	0	10000000	100
51	11	3	0	0	0	0	10000000	100
52	11	3	0	0	0	0	10000000	100

53	11	3	0	0	0	0	10000000	100
54	11	3	0	0	0	0	10000000	100
55	11	3	0	0	0	0	10000000	100
56	12	3	0	0	0	0	10000000	100
57	12	3	0	0	0	0	10000000	100
58	12	3	0	0	0	0	10000000	100
59	12	3	0	0	0	0	10000000	100
60	12	3	0	0	0	0	10000000	100
61	12	3	0	0	0	0	10000000	100
62	13	1	0	0	0	0	10000000	50
63	13	1	0	0	0	0	10000000	50
64	13	1	0	0	0	0	10000000	50
65	13	1	0	0	0	0	10000000	50
66	14	1	0	0	0	0	10000000	500
67	14	1	0	0	0	0	10000000	500
68	14	1	0	0	0	0	10000000	500
69	14	1	0	0	0	0	10000000	500
70	15	4	0	0	0	0	10000000	500
71	15	4	0	0	0	0	10000000	500
72	15	4	0	0	0	0	10000000	500
73	15	4	0	0	0	0	10000000	500

Structure of Interest 2 Input File

1	1	1	1	2	0	0	10000000	50
2	1	1	1	2	0	0	10000000	50
3	1	1	1	2	0	0	10000000	50
4	1	1	1	2	0	0	10000000	50
5	2	1	1	1	0	0	10000000	500
6	2	1	1	1	0	0	10000000	500
7	2	1	1	1	0	0	10000000	500
8	2	1	1	1	0	0	10000000	500
9	3	4	1	4	0	0	10000000	500
10	3	4	1	4	0	0	10000000	500
11	3	4	1	4	0	0	10000000	500
12	3	4	1	4	0	0	10000000	500
13	4	2	1	3	0	0	10000000	100
14	4	2	1	3	0	0	10000000	100
15	4	2	1	3	0	0	10000000	100
16	4	2	1	3	0	0	10000000	100
17	5	4	1	6	0	0	10000000	100
18	5	4	1	6	0	0	10000000	100
19	5	4	1	6	0	0	10000000	100
20	5	4	1	6	0	0	10000000	100
21	6	2	1	5	0	0	10000000	100
22	6	2	1	5	0	0	10000000	100
23	6	2	1	5	0	0	10000000	100
24	6	2	1	5	0	0	10000000	100
25	7	3	1	8	9	0	10000000	100
26	7	3	1	8	9	0	10000000	100
27	7	3	1	8	9	0	10000000	100

28	7	3	1	8	9	0	10000000	100
29	7	3	1	8	9	0	10000000	100
30	7	3	1	8	9	0	10000000	100
31	8	4	1	7	9	0	10000000	100
32	8	4	1	7	9	0	10000000	100
33	8	4	1	7	9	0	10000000	100
34	8	4	1	7	9	0	10000000	100
35	8	4	1	7	9	0	10000000	100
36	8	4	1	7	9	0	10000000	100
37	9	4	1	7	8	0	10000000	100
38	9	4	1	7	8	0	10000000	100
39	9	4	1	7	8	0	10000000	100
40	9	4	1	7	8	0	10000000	100
41	9	4	1	7	8	0	10000000	100
42	9	4	1	7	8	0	10000000	100
43	10	4	1	11	0	0	10000000	100
44	10	4	1	11	0	0	10000000	100
45	10	4	1	11	0	0	10000000	100
46	10	4	1	11	0	0	10000000	100
47	10	4	1	11	0	0	10000000	100
48	10	4	1	11	0	0	10000000	100
49	10	4	1	11	0	0	10000000	100
50	11	3	1	10	0	0	10000000	100
51	11	3	1	10	0	0	10000000	100
52	11	3	1	10	0	0	10000000	100
53	11	3	1	10	0	0	10000000	100
54	11	3	1	10	0	0	10000000	100
55	11	3	1	10	0	0	10000000	100
56	12	3	1	13	0	0	10000000	100

57	12	3	1	13	0	0	10000000	100
58	12	3	1	13	0	0	10000000	100
59	12	3	1	13	0	0	10000000	100
60	12	3	1	13	0	0	10000000	100
61	12	3	1	13	0	0	10000000	100
62	13	1	1	12	0	0	10000000	50
63	13	1	1	12	0	0	10000000	50
64	13	1	1	12	0	0	10000000	50
65	13	1	1	12	0	0	10000000	50
66	14	1	1	15	0	0	10000000	500
67	14	1	1	15	0	0	10000000	500
68	14	1	1	15	0	0	10000000	500
69	14	1	1	15	0	0	10000000	500
70	15	4	1	14	0	0	10000000	500
71	15	4	1	14	0	0	10000000	500
72	15	4	1	14	0	0	10000000	500
73	15	4	1	14	0	0	10000000	500

Structure of Interest 3 Input File

1	1	1	1	2	3	0	10000000	50
2	1	1	1	2	3	0	10000000	50
3	1	1	1	2	3	0	10000000	50
4	1	1	1	2	3	0	10000000	50
5	2	1	1	1	3	0	10000000	500
6	2	1	1	1	3	0	10000000	500
7	2	1	1	1	3	0	10000000	500
8	2	1	1	1	3	0	10000000	500
9	3	4	1	1	2	0	10000000	500
10	3	4	1	1	2	0	10000000	500
11	3	4	1	1	2	0	10000000	500
12	3	4	1	1	2	0	10000000	500
13	4	2	0	0	0	0	10000000	100
14	4	2	0	0	0	0	10000000	100
15	4	2	0	0	0	0	10000000	100
16	4	2	0	0	0	0	10000000	100
17	5	4	0	0	0	0	10000000	100
18	5	4	0	0	0	0	10000000	100
19	5	4	0	0	0	0	10000000	100
20	5	4	0	0	0	0	10000000	100
21	6	2	0	0	0	0	10000000	100
22	6	2	0	0	0	0	10000000	100
23	6	2	0	0	0	0	10000000	100
24	6	2	0	0	0	0	10000000	100
25	7	3	0	0	0	0	10000000	100
26	7	3	0	0	0	0	10000000	100
27	7	3	0	0	0	0	10000000	100

28	7	3	0	0	0	0	10000000	100
29	7	3	0	0	0	0	10000000	100
30	7	3	0	0	0	0	10000000	100
31	8	4	0	0	0	0	10000000	100
32	8	4	0	0	0	0	10000000	100
33	8	4	0	0	0	0	10000000	100
34	8	4	0	0	0	0	10000000	100
35	8	4	0	0	0	0	10000000	100
36	8	4	0	0	0	0	10000000	100
37	9	4	0	0	0	0	10000000	100
38	9	4	0	0	0	0	10000000	100
39	9	4	0	0	0	0	10000000	100
40	9	4	0	0	0	0	10000000	100
41	9	4	0	0	0	0	10000000	100
42	9	4	0	0	0	0	10000000	100
43	10	4	0	0	0	0	10000000	100
44	10	4	0	0	0	0	10000000	100
45	10	4	0	0	0	0	10000000	100
46	10	4	0	0	0	0	10000000	100
47	10	4	0	0	0	0	10000000	100
48	10	4	0	0	0	0	10000000	100
49	10	4	0	0	0	0	10000000	100
50	11	3	0	0	0	0	10000000	100
51	11	3	0	0	0	0	10000000	100
52	11	3	0	0	0	0	10000000	100
53	11	3	0	0	0	0	10000000	100
54	11	3	0	0	0	0	10000000	100
55	11	3	0	0	0	0	10000000	100
56	12	3	0	0	0	0	10000000	100

57	12	3	0	0	0	0	10000000	100
58	12	3	0	0	0	0	10000000	100
59	12	3	0	0	0	0	10000000	100
60	12	3	0	0	0	0	10000000	100
61	12	3	0	0	0	0	10000000	100
62	13	1	1	14	15	0	10000000	50
63	13	1	1	14	15	0	10000000	50
64	13	1	1	14	15	0	10000000	50
65	13	1	1	14	15	0	10000000	50
66	14	1	1	13	15	0	10000000	500
67	14	1	1	13	15	0	10000000	500
68	14	1	1	13	15	0	10000000	500
69	14	1	1	13	15	0	10000000	500
70	15	4	1	13	14	0	10000000	500
71	15	4	1	13	14	0	10000000	500
72	15	4	1	13	14	0	10000000	500
73	15	4	1	13	14	0	10000000	500