

EVOLUTION OF AIRCRAFT STRUCTURES AND INTEGRITY MANAGEMENT

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ABSTRACT: Current aircraft structural integrity (ASI) management programs are based predominately on the knowledge and experience associated with metallic aircraft structures. Since the 1960s, the percentage of aircraft structures being manufactured from composite materials is steadily increasing. Latest generation of commercial aircraft structures are composed of approximately 50% composites by weight and several new military airframes comprise of more than 75% airframe weight (Tiger and NH-90 helicopters). This paper reviews the evolution of current aircraft structural integrity management with particular interest in the changing design philosophies and thus the impact on how ASI is being managed.

INTRODUCTION

Aircraft structural integrity management programs are aimed at ensuring that aircraft can operate safely and economically throughout its life from a structural integrity point of view. As typified by MIL-STD-1530C, 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

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- d. Providing lessons-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. 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 more than thirty percent by weight of aircraft structure constructed from composites. This percentage is higher for large commercial aircraft currently under development, such as the Boeing 787 Dreamliner, which has a structure, comprised of approximately fifty percent composites by weight. Finally, modern military helicopters such as the Eurocopter Tiger and NH90 have more than seventy-five percent composite structures by weight. This increase in structural usage combined with the decrease in Factors of Safety resulting from the desire to optimize aircraft structure to minimise weight, raises the question of whether current aircraft structural integrity methodologies are appropriate for aircraft structures built largely from composites.

BRIEF HISTORY 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.

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:

1. Uncertainties in loads;
2. Inaccuracies in structural analysis;
3. Variations in strength properties of materials;
4. Deterioration during service life, and
5. 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 age³ of aircraft operated by the United States air force was only 1.7 years (Ramey and Keating, 2009).

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 included the development of the first stress-life (S-N) diagram for a full scale fabricated structure (CA-12 Boomerang wings) and consequently 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).

³ Design age is the timeframe between the first example of an aircraft design entering service and the last

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 the 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 55,000AFHR⁴).

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).

example leaving service.

⁴ This value of AHFRS was determined by using the ratio of AFHRS to flight cycles identified for G-ALYP.

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 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 United States Air Force during 1956 to 1960 had risen to 8.7 years (Ramey and Keating, 2009).

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).

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-3⁵ thereto; the Special Conditions and the provisions 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.’

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.

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

⁵ This amendment added fail-safe into the regulations.

increasing since the 1978 introduction of the McDonnell Douglas MD-80, as shown in Figure 1.

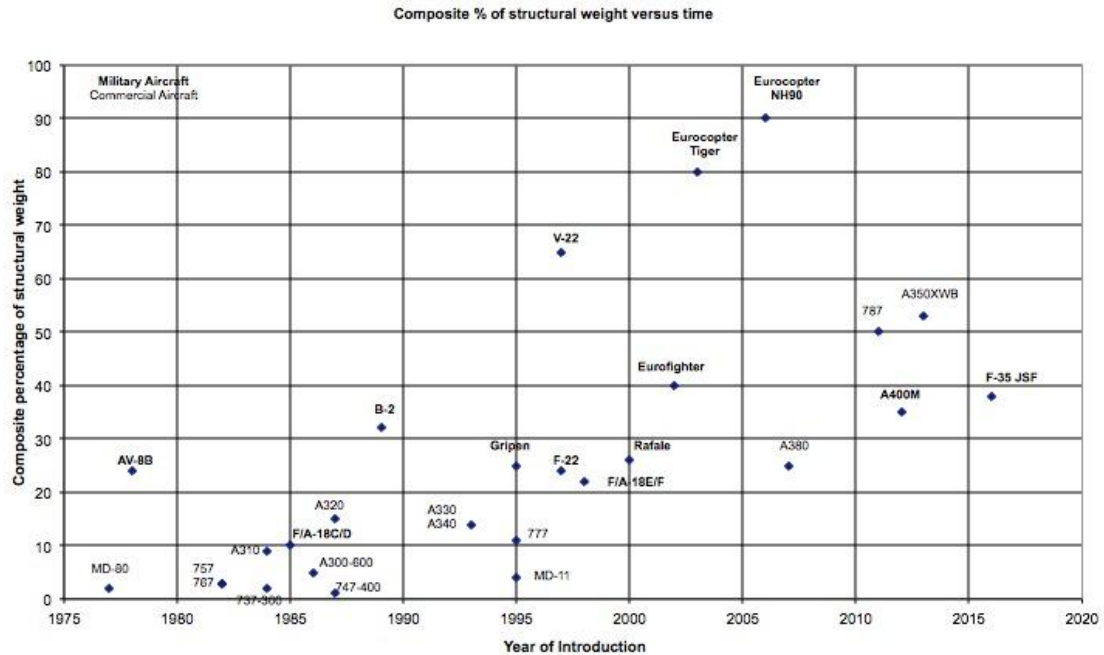


Figure 1. Growth of composite structure on major aircraft programs (1975-2010) as a percentage of weight (adapted from ATSB, 2007)

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:
 1. Removal of obsolete guidance;
 2. Harmonisation with other regulations;
 3. Update based on service and/or certification experience; and
 4. Inclusion of new technology, including materials, engineering methods and maintenance procedures.
- b. AC 20-107B issued on the 8th September 2009, this 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 only corrected minor errors.

1988 - Aloha Airlines Boeing 737 Accident

On the 28th April 1988, after reaching its 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 safely 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).⁶

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.

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 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 United States Air Force 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

⁶ Thus the aircraft design did not include the Damage Tolerance philosophy which was introduced at

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 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.

Summary

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

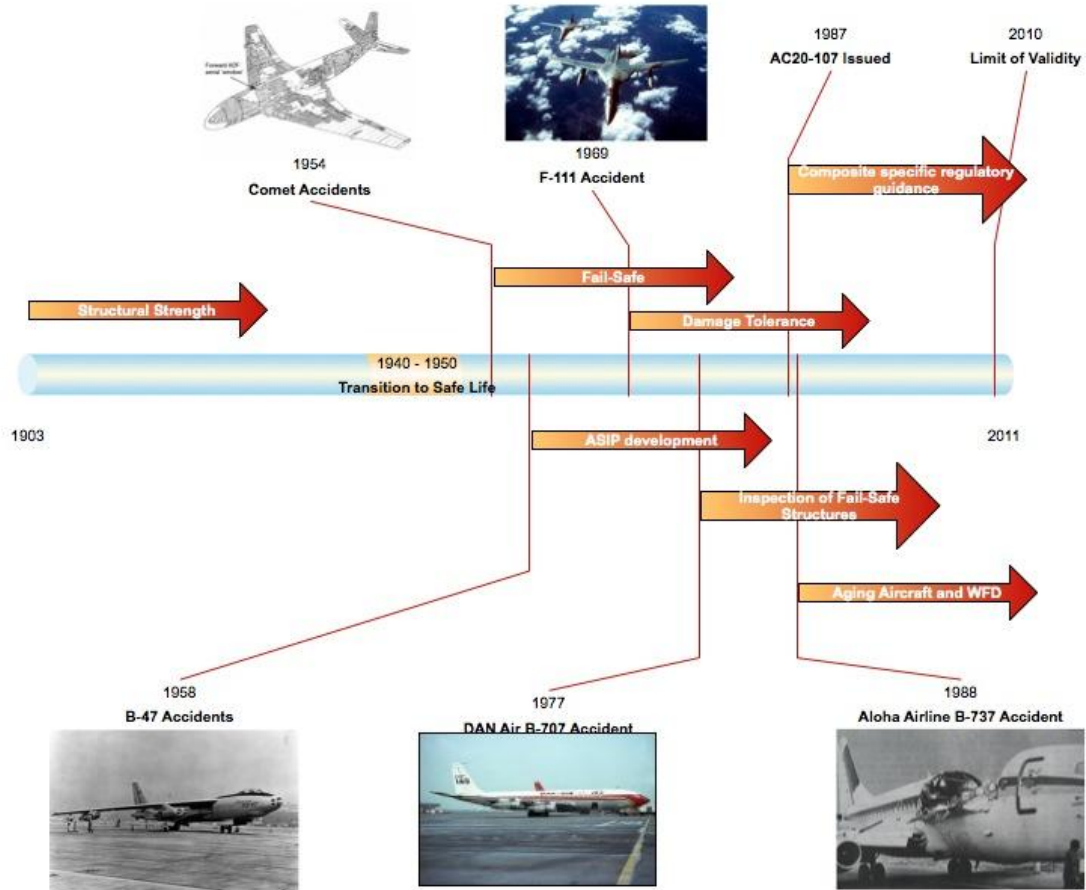


Figure 2. Milestones in Aircraft Structural Integrity.

STRUCTURAL INTEGRITY PHILOSOPHIES

Structural Strength

The Structural Strength philosophy was used to ensure structural integrity with fatigue, environment degradation and other aging factors accounted for by ensuring that the structure had sufficient excess strength to ensure that as these factors reduced the strength of the structure, failure would not occur during the an aircraft's life of type.

Safe Life

The concept of Safe-Life utilizes non-redundant load paths (i.e. single point of failure), which have a finite life, specified, above which probability of failure is

unacceptably high. The intent being that the structure will not fail due to fatigue within the service life of the aircraft. An advantage of the Safe-Life concept, as identified in DEF-STAN 00-970, is that it minimizes the number of inspections required to be performed in-service.

The Safe Life of a structure is determined using:

- a. Basic material fatigue properties as defined by S-N curves and modified Goodman diagrams;
- b. Factors to account for design, fatigue quality index (used to account for local structural stresses) and scatter factor (factor of safety used to make the S-N curve more conservative), and
- c. Miner's Rule for fatigue accumulation and the loading spectrum of the aircraft.

As stated by FAA (1993) when applied correctly the Safe-Life approach provides a conservative estimate of an aircraft's useful economic life.

However, as identified by Grandt (2004) one of the primary failings of the safe-life approach was its inability to adequately account for unanticipated structural or material damage.

Fail-Safe

The concept of Fail-Safe utilizes redundant load paths to carry loads for failed structures until damage is detected via inspection. The residual strength of the structure shall be maintained post-failure via damage arrest or secondary load paths for period sufficient to be detected visually.

Accident Investigation Branch (1978) defines a Fail-Safe design as 'one in which there are one or more redundant structural elements which are capable, in the event of a

failure of one of the primary load members, of carrying flight loads. However, a Fail-Safe design is only fail-safe whilst the degree of redundancy is sufficient to cater for a failure.’

Once a failure has occurred, the Safe-Life methodology becomes applicable, as it is necessary to detect the failure before the fail-safe (redundant) structure becomes weakened by fatigue or other environmental effects.

Damage Tolerance

Damage Tolerance was defined by Gallagher, Glessler, Bernes, Engle and Wood (1984) as the ‘ability of a structure to successfully contain damage over a specified life increment without adversely affecting safety of flight.’ Damage tolerance relies on the assumption that flaws of finite size are present from manufacture and fatigue cracks will develop from these flaws, which will be either:

- a. Detected by inspection; or
- b. Not grow to critical size during the service life.

As such, this philosophy is founded upon crack growth theory, which allows the growth rate of cracks to be predicted, with less scatter than that present in the Safe-Life philosophy. Gallagher et al. (1984) identified the following factors as impacting crack growth rates:

- a. Quality – the size of flaws present in the structure at manufacture;
- b. Usage – the loading history for the structure, for example the number of flight cycles;
- c. Material – the material properties, for example the fracture toughness (K_{Ic}), and
- d. Geometry – the structural properties, for example, Stress Intensity Factors.

STRUCTURAL INTEGRITY MANAGEMENT

There are a number of airworthiness regulatory authorities, which regulate aircraft airworthiness and associated Aircraft Structural Integrity (ASI). The United States Department of Defence provides comprehensive regulation of ASI Management (ASIM) via MIL-STD-1530C. As such, this standard will be used for comparison against the applicable ASI regulations of the United Kingdom Ministry of Defence (MoD) and the FAA.

United States Department of Defence

The United States Department of Defence (DoD) requirements for ASIM (termed ASI Program (ASIP)) are documented within MIL-STD-1530C. This standard states that the purpose of ASIP is to ensuring that aircraft can operate safety 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 lesson-learnt to be applied to the next generation of aircraft design/modifications.

The following five tasks are performed to implement these pillars.

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-516⁷ 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)⁸ for the certification of aircraft structures. The guide 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, damage tolerance. Assessments are performed regarding production Non-Destructive Inspection (NDI) capability and a risk assessment to demonstrate that risks to structural integrity have been adequately mitigated. The design service spectra

⁷ Section 5

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.' 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

⁸ Annex A, Section A.3.12.1 contains details specific to composites.

(inspections, maintenance, modifications, damage assessments, risk analysis, etc.) when the aircraft enters the inventory.’

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) such as descriptions of aircraft structures, critical design conditions, minimum margins and structural limitations, and
- d. Development of plan for the structural management of the aircraft once in service (known as a Force Structural Management Plan in MIL-STD-1630C).

This document contains details of:

1. Inspection and structural maintenance programs, including the methods and inspection intervals.
2. Structural maintenance database, and
3. 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.’ 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.

United States Civil

FAA Federal Aviation Regulations (FAR)

The FAA requires either damage tolerance, or if impractical, safe-life evaluations to be performed for all fixed wing aircraft structures certified in accordance with either FAR 23 (FAR23.573 and 23.574) and FAR 25 (25.571). The FAA requirements for commercial aircraft (FAR 25) most directly associated with the MIL-STD-1530C tasks are identified in Table 1.

Table 1: *FAR 25 ASI-related regulations*

MIL-STD-1530C ASI Tasks	FAR 25 Regulation	Description
Task 1 - Design Information	Nil – implied requirement	The structural design requirements are defined in the FAR.
Task 2 – Design Analysis and DT&E	301 305, 307 603, 605, 609, 613 and 619 571 629	- Loads - Proof of Structure - Static - Material and Fabrication Development - DT and Fatigue - Proof of Structure - Flutter
Task 3 – Full Scale Testing	571, 629	Sufficient level of full-scale testing
Task 4 – Certification and Force Management	1529	Operational Limitations and Information
Task 5 – Force Management Execution	FAR 119	Regulations covering the operation and maintenance of aircraft are not covered in FAR 25.

As illustrated in Table 1, FAR25 (and other FAR in general) are not as specific as MIL-STD-1530C in relation to ASI requirements. Additional guidance for the compliance with the regulations identified in the above table is provided by the following FAA Advisory Circulars:

- a. AC 25-21 Certification of Transport Airplane Structures, dated 1 Sept 1999,
and
- b. AC 20-107B (Change 1) Composite Aircraft Structure, dated 24 August
2010.

United Kingdom Military

DEF STAN 00-970⁹ details the UK MoD (and is used by the Australian Defence Force as a benchmark for) requirements for aircraft structures (including ASI) requirements and identifies the following preferred philosophies:

- a. Safe Life, with the fatigue life to be no less than the required life of the aircraft.
- b. Inspection-Based, equivalent to the Damage Tolerance philosophy and is to be used where the component is subject to possible impact damage. The residual strength of the component must not fall below 0.8 Design Ultimate Load (DUL) and the fatigue life of the component must be at least half the required life of the aircraft.

The DEF-STAN 00-970 requirements most directly associated with fixed wing aircraft structural integrity are identified in Table 2.

⁹ DEF STAN 00-970 Part 5/1 Section 3

Table 2

DEF STAN 00-970 ASI-related regulations

MIL-STD-1530C ASI Task	Part 1 ¹⁰ Regulation	Description
Task 1 - Design Information	3.2.8 to 3.2.1, 4.1	Material Selection
Task 2 – Design Analysis and DT&E	3.1.2 3.2.11 to 3.2.14 3.2.15/16 3.2.17 to 3.2.20	- Critical design cases - Safe Life Substantiation - Residual Strength - Inspection-Based Substantiation
Task 3 – Full Scale Testing	3.1.11 3.1.12	- Ultimate and Proof strength demonstration -Measurement of loads on development aircraft
Task 4 – Certification and Force Management	3.2.21/22	- In-service monitoring of aircraft fatigue life.
Task 5 – Force Management Execution	3.1.12	Measurement of loads on aircraft in-service.

DISCUSSION

Resilience of ASI

EUROCONTROL (2009) defines resilience as ‘The ability of a system to succeed under varying and adverse conditions.’ and Woods (2006) identifies that resilience refers to the art of managing the unexpected. Taking the viewpoint that 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.

Over the course of ASIM evolution, the resilience of the ASIM frameworks has increased, as demonstrated by the following examples:

¹⁰ Fixed Wing

- a. Transition to Damage-Tolerance: the inability of Safe-Life to ‘succeed’ when an unanticipated event (manufacturing defect) was encountered lead to Damage Tolerance.
- b. Implementation of Limit of Validity: the inability of then current regulations to adequately address WFD resulted in the implementation of the LoV by the FAA, which essentially mandates a Safe-Life from the effects of WFD.

Lessons to be learnt

Current ASIM frameworks have been developed based on the knowledge and experience with predominately metallic aircraft structures. It is anticipated that the development of ASIM for composite aircraft may follow a similar route, with initial changes the result of aircraft accidents (reactive) with more proactive approaches (e.g. introduction of LoV) being implemented as service experience and design knowledge increases.

The adverse effect of aircraft aging on structural integrity is anticipated to be an issue for composite aircraft structures as it is for metallic structures. The difference will be the mechanisms by which the aging effects manifest:

- a. For metals, fatigue and corrosion are the primary threats to structural integrity.
- b. For composites, corrosion and fatigue (at current design stress levels for operational aircraft) are less of a concern. However, physical impacts on composite structures are more likely to be damaging due to the inability of composites to absorb impact via plastic deformation like metals. Thus, composites are more susceptible to impact damage which can lead to

delamination of the composite plies, cracks and punctures through the composite component.

- c. Composite structures with delaminations are more of a concern with compression fatigue crack propagation than under tensile loading.

FINAL COMMENTS

Based on the above discussion, “Is the current ASI program approach applicable to composite airframes?”. The investigation into the impact of composites on ASI will be the subject of future work. This future work will aim at assisting the transition to a proactive composite ASI program, eliminating the reactive phase encountered with metallic structures.

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