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Milestone Case Histories in Aircraft Structural Integrity[☆]

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1 Introduction

Notwithstanding the more recent innovative material and structural applications, particularly the use of composites (discussed later, in Section 8.3), the evolution of aircraft structural integrity has been concerned largely with the service behavior of high-strength metallic materials, particularly aluminum alloys. Broadly speaking, the history of this evolution is as follows (Niu, 1988):

1930–1940: Commercial development of metal aircraft for public transport. Design and analysis emphasized static strength, with little or no consideration of airframe fatigue.

[☆]Change History: January 2015. R. Wanhill completely reworked the article and two additional authors were added (L. Molent and S. Barter).

Table 1 Milestone case histories in aircraft structural integrity – from several sources

Category	Year	Aircraft failures	Influence, follow-up
Generally known	1954	De Havilland Comet (2X); pressure cabin disintegrations resulting from fatigue	General awareness of aircraft finite fatigue life and significance of Fail-Safety. Attention drawn to full-scale fatigue testing
	1969	General Dynamics F-111; wing loss owing to fatigue from undetected material flaw	Aircraft should be Damage Tolerant: fatigue due to initial damage should be considered
	1977	Boeing 707; tailplane loss owing to fatigue failure in spar	Older aircraft become more fatigue-critical. The term geriatric aircraft is introduced
	1988	Boeing 737; part of pressure cabin skin structure lost owing to multiple fatigue cracks in skin splices	Widespread Fatigue Damage (WFD) can occur in aging aircraft, especially in pressure cabin lap joints. Corrosion control emphasized
Less generally known	1958	Boeing B-47B (2X) and B-47E (2X); wing losses resulting from fatigue	Need fatigue control, accurate/conservative Safe-Life predictions; design and test methods to avoid future problems. USAF Aircraft Structural Integrity Program (ASIP) initiated
	1990	Aermacchi MB-326H; wing loss owing to fatigue failure in spar	Quantitative Fractography (QF) for fatigue life (re)assessment; lifing methods using realistic initial discontinuity sizes

1940–1955: Increasing awareness of importance of fatigue for airframe safety. Materials with higher static strengths were developed without corresponding increases in fatigue strength. Design became based on both static and fatigue strengths.

1955–present: Development of Fail-Safe and Damage Tolerant design methods, which recognize that airframe structures must withstand service loads even when damaged or cracked. Safety should be ensured by fatigue and fracture testing and analysis of damaged structures, pre-service and in-service inspections, and eventual repairs, replacements or retirement.

Service failures have greatly influenced this development. Four case histories are often cited (Schijve, 1994; Blom, 2002; McEvily, 2002; Tiffany *et al.*, 2010) as milestones in the aircraft industry’s approach to structural integrity, see the upper part of Table 1. Less generally known, but also important, are the case histories in the lower part of this table. The case histories and their influences on aircraft structural integrity will be discussed in chronological order in Sections 2–7. Section 8 provides a summary of the lessons learned and ongoing research and development.

2 The BOAC De Havilland Comet Crashes in 1954

2.1 Case Histories

The De Havilland Comet was the first commercial jet transport, entering service in 1952. The aircraft’s performance was much superior to that of the contemporary propeller-driven transports. Apart from its speed, the Comet was the first high-altitude passenger aircraft, since it had a cabin pressure differential almost double that of its contemporaries (Swift, 1987).

Within two years of entering service, two of the fleet disintegrated while climbing to cruise altitude. Comet G-ALYP was lost on January 10, 1954. Modifications were made to the fleet to rectify some of the items that might have caused the accident. However, Comet G-ALYY was lost on April 8, 1954. The fleet was then grounded. Extensive investigations followed, including most importantly a full-scale repeated pressurization test on an aircraft removed from service, registration number G-ALYU.

The test aircraft had accumulated 1231 pressurization cycles in service. It was tested in a water tank to minimize damage in the event of failure. After 1825 test pressurizations, the pressure cabin failed during application of a proof cycle at 33% higher loading. The failure showed evidence of fatigue cracking that began at the aft lower corner of the forward escape hatch (see Figure 1). Additional investigation of wreckage from Comet G-ALYP also showed evidence of fatigue, in this case commencing from the right-hand aft corner of the rear automatic-direction-finding window (see Figure 2).

The test aircraft was repaired and strain gauges applied to the outside surfaces of several escape hatches and windows. The strain gauge results for the service and test failure locations are also shown in Figures 1 and 2. Swift (1987) pointed out that out-of-plane bending would have caused the inside principal stress to be significantly higher, which could well have contributed to the early fatigue failures. This out-of-plane bending would not have been considered in a design analysis for the Comet, nor indeed for subsequent commercial jet aircraft. However, the full-scale test effectively accounted for it.

Swift (1987) described the Comet pressure cabin structure in more detail, in order to bring out some further important aspects of the service failures. Figure 3 shows the basic pressure shell structure and the probable origin of failure for Comet G-ALYP. The basic shell structure had no crack-stopper straps to provide continuity of the frame outer flanges across the stringer cutouts. The cutouts, one of which is shown in Figure 3(b), created a very high stress concentration at the first fastener. In the case of the probable origin of failure for Comet G-ALYP, the first fastener was a countersunk bolt, as shown in Figure 3(c). The countersink created a knife-edge in both the skin and outside doubler. The early fatigue failure may thus be attributed to high local stresses

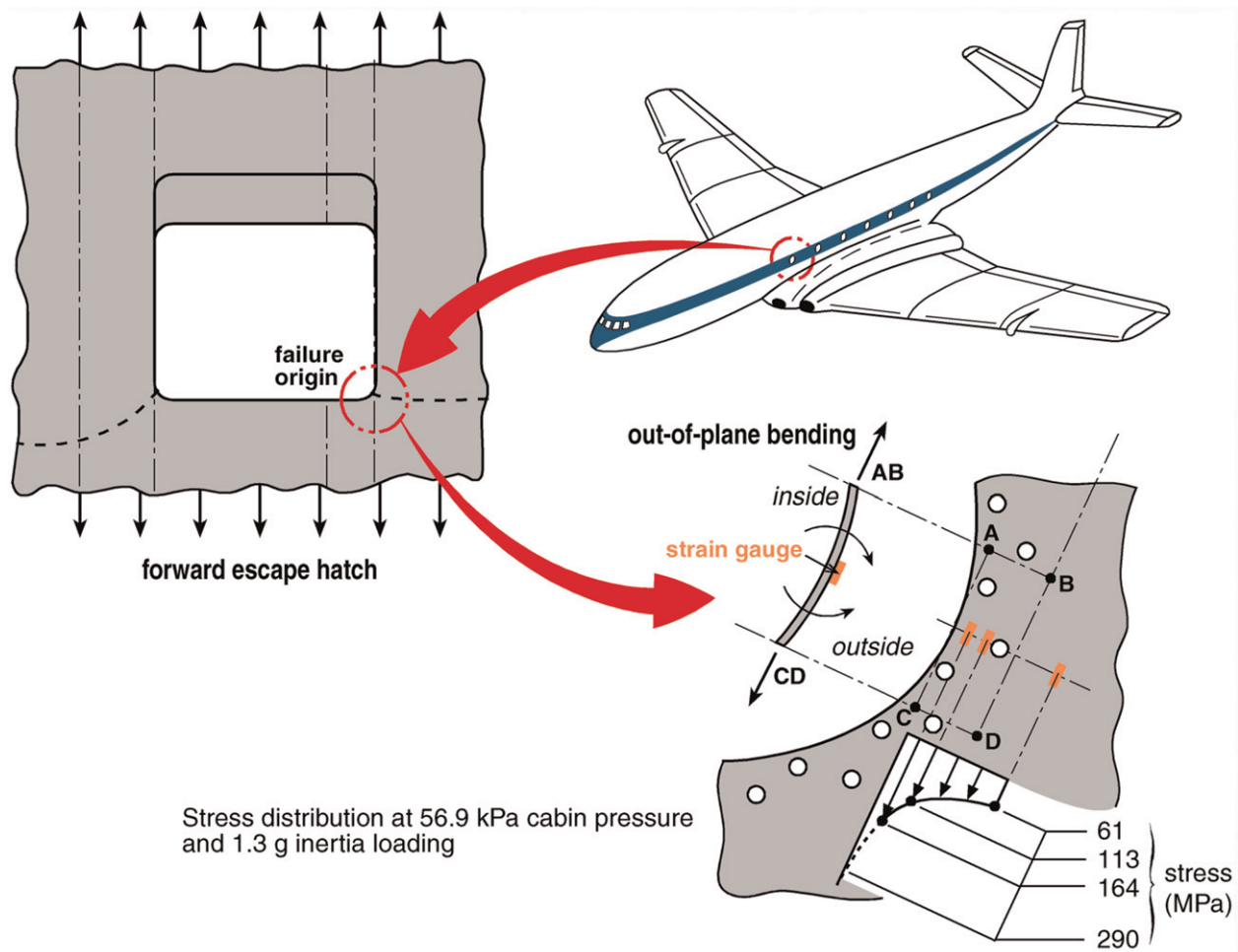


Figure 1 Probable failure origin of test aircraft Comet G-ALYU: stress distribution obtained after repair (Swift, 1987).

(Figure 2), combined with the stress concentrations provided by the frame cutout and knife-edge condition of the first fastener hole (Figures 3(b) and 3(c)).

Once the fatigue crack initiated in Comet G-ALYP, its growth went undetected until catastrophic failure of the pressure cabin. Obviously, this should not have happened, but Swift (1987) provided an explanation from subsequent knowledge. He showed that the basic shell structure of the Comet could have sustained large, and easily detectable, one- and two-bay cracks if they had grown along a line midway between the positions of the frame cutouts. In other words, the basic shell structure would have had adequate residual strength for these crack configurations. However, neither one- nor two-bay cracks would be tolerable if they grew along the line between frame cutouts. For these cases, crack-stopper straps would have been needed to provide adequate residual strength.

2.2 Lessons Learned

The Comet accidents and subsequent investigations changed fundamentally the structural fatigue design principles for commercial transport aircraft. Before and also during the Comet era, the fatigue design principles were based on the so-called Safe-Life philosophy. This means that the entire airframe was designed to achieve a satisfactory fatigue life with no significant damage, i.e., cracking. The Comet accidents, and other experiences, showed that cracks could sometimes occur much earlier than anticipated, owing to limitations in the fatigue analyses, and that safety could not be guaranteed on a Safe-Life analysis basis without imposing uneconomically short service lives on major components of the structure.

These problems were addressed by adoption of the Fail-Safe design principles in the late 1950s. In Fail-Safe design, the structure is designed first – as before – to achieve a satisfactory life with no significant damage. However, the structure is also designed to be inspectable in service and able to sustain significant and easily detectable damage before safety is compromised. These latter requirements were met mainly by employing structural design concepts having multiple load paths, with established residual strength requirements in the event of failure of one structural element or an obvious partial failure.

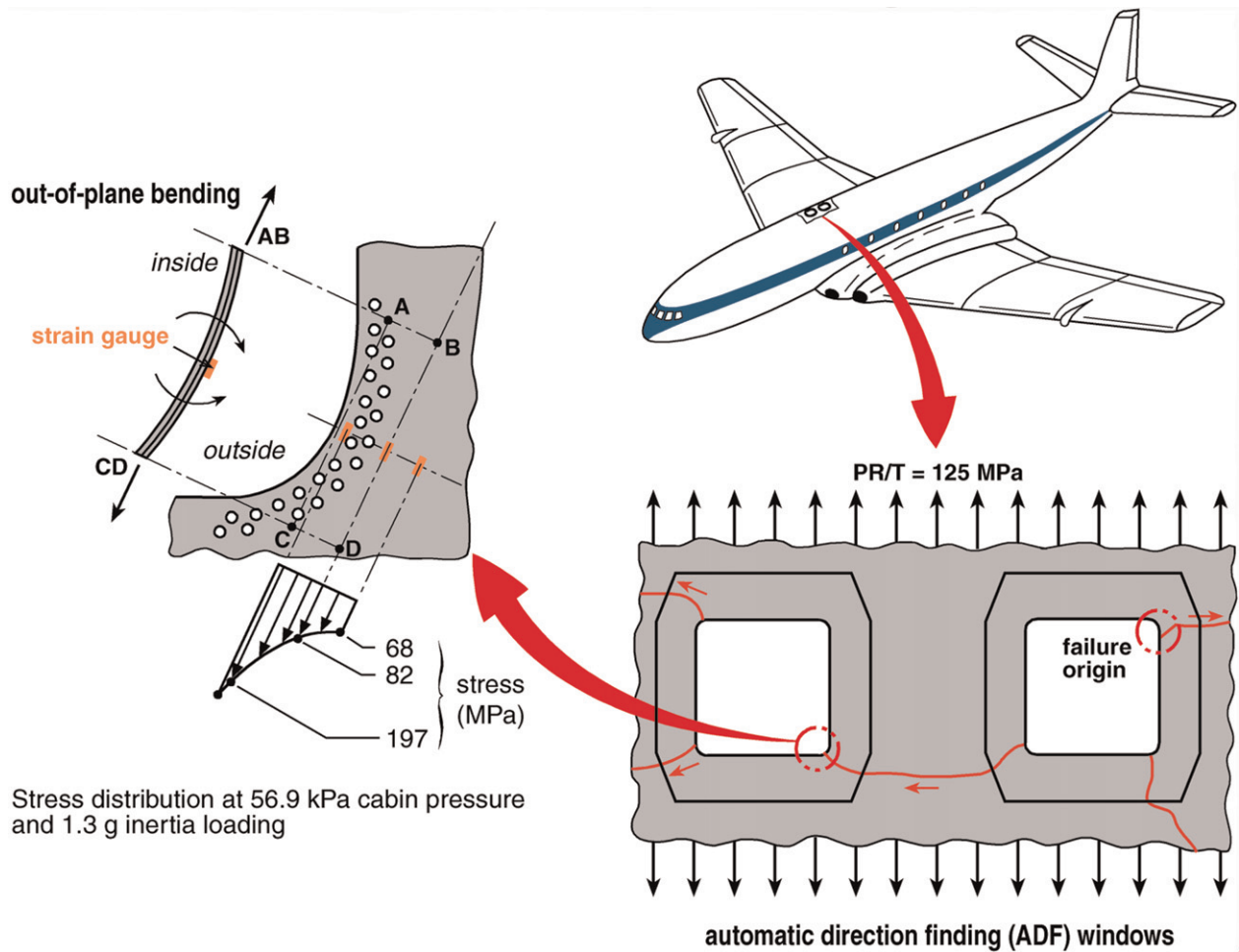


Figure 2 Probable failure origin of service aircraft Comet G-ALYP: stress distribution obtained from repaired test aircraft, Comet (Swift, 1987).

Verification of Fail-Safe design concepts requires substantial fatigue and residual strength testing. An essential part of this verification is the study of fatigue crack growth, its analysis and prediction. However, when the Fail-Safe principles were first adopted, it was not required to do full-scale testing. Subsequent experience and knowledge led to mandatory full-scale testing of some new designs, but others were exempted because they were based on design techniques and analyses from previously tested types. This was known as 'Grandfathering,' and is no longer allowed. In other words, full-scale testing must now be done for both original and derivative designs.

It is important to note here that not all airframe components are amenable to Fail-Safe design. The main exceptions are landing gears, usually made from high-strength steels and designed to Safe-Life principles. Going beyond commercial transport aircraft, Safe-Life design is also used for most general aviation aircraft, and at least some airframe structures and components in helicopters and military aircraft. This is partly due to the extensive use of composites, which are currently designed to a 'no growth' damage criterion analogous to Safe-Life design, see Section 8.3; but also the U.S. Navy, as well as others, continues to require aircraft designs based on the Safe-Life principle.

3 The USAF Boeing B-47 Crashes in 1958

3.1 Case Histories

The Boeing B-47 bomber aircraft was in the late 1940s a radically new design, with six turbojets mounted on pylons beneath a thin and flexible swept-back wing. The prototype first flew in December 1947. The aircraft's high-speed performance exceeded 600 mph, resulting in its selection in 1950 as a high altitude medium bomber by the Strategic Air Command (SAC) of the United States Air Force (USAF). Acceptance for operational use was based on a *static* test in 1950 (Negaard, 1980). This was followed by a limited flight load survey from September 1952 to March 1954, focussing on high loads. Hence the aircraft was validated only for static loads, this being the contemporary design and testing approach.

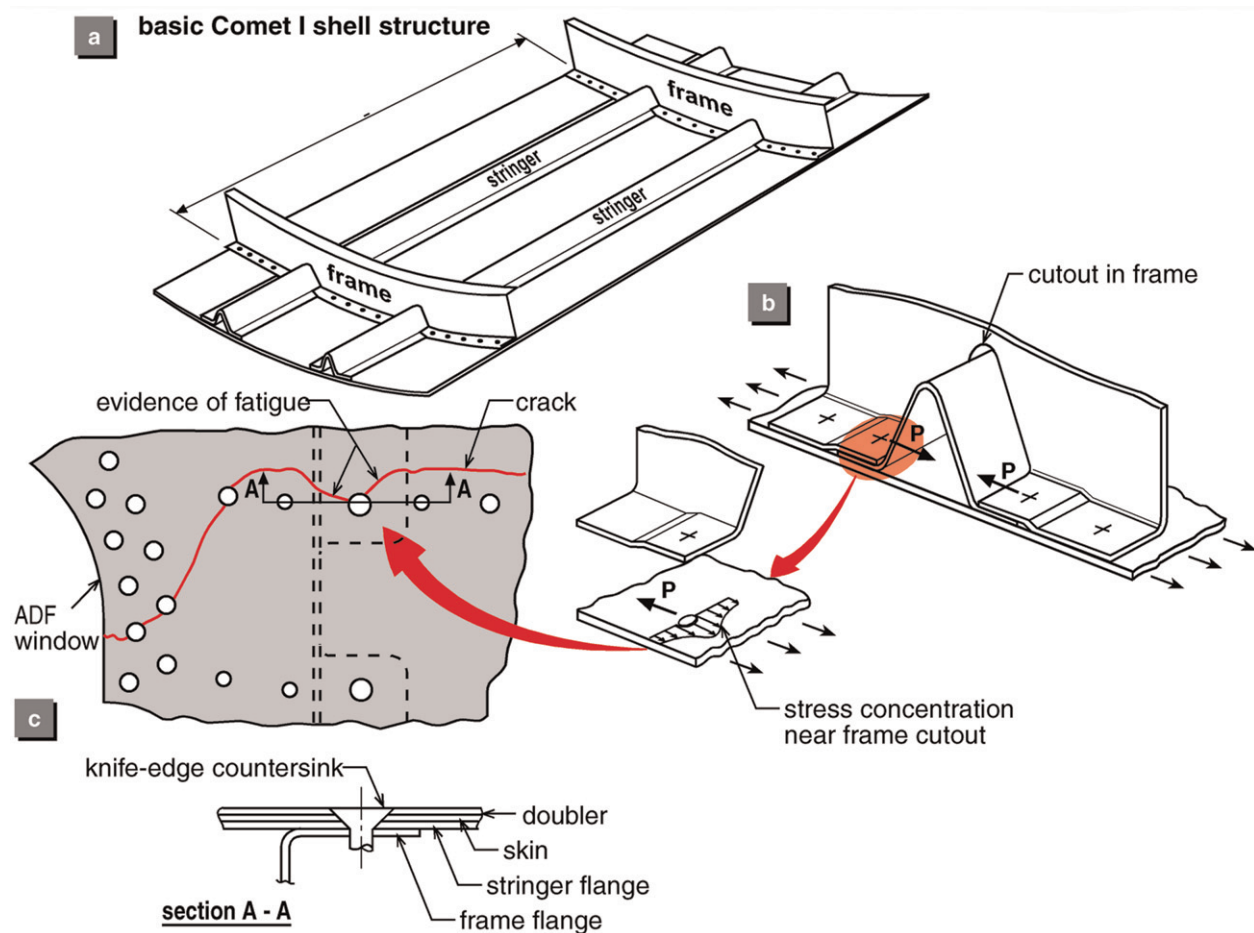


Figure 3 Details of the probable failure origin (large arrow) of service aircraft Comet G-ALYP (Swift, 1987).

Table 2 B-47 catastrophic fatigue failures in early 1958 (Negaard, 1980)

Aircraft	Date	Failure location	Flight hours
B-47B	March 13	Center wing, BL 45	2077
TB-47B	March 13	Left wing lower surface, BL 35	2419
B-47E	March 21	Disintegration	1129
B-47E	April 10	Aft wing-to-fuselage fitting, FS 515	1265

The development, production and service history of the B-47 are well-documented (Negaard, 1980; Knaack, 1988). Quantity production began in 1951 and ended in 1957. The USAF did not specify a service life, but the intention was to keep the aircraft in service until 1965. In fact, most were retired by 1966, though a few remained in service until 1969.

The shortcoming of validation based solely on static loads was tragically demonstrated by four catastrophic fatigue failures in early 1958, two of which occurred after less than 1300 flight hours, Table 2. (Another crash, but this time attributed to overload, occurred on April 15.) The fatigue failures resulted in an emergency program of inspection and repair that began in May 1958. The program focussed on the critical areas (both right hand (R.H.) and left hand (L.H.) indicated in Figure 4. The critical wing-to-fuselage area included Body Line (BL) 35, BL 45, and Fuselage Station (FS) 515, see Table 2; the critical wing area was at Wing Station WS-354 (Negaard, 1980). The two small red arrows in Figure 4 show the positions of two of the four wing-to-fuselage connecting pins. These pins (both R.H. and L.H.) and the surrounding structure were the primary locations to be inspected for cracks.

In the meantime, and during the emergency program, the SAC's concern was to keep the fleet flying. This could be done only by applying flight restrictions, which came to include aircraft gross weight, airspeed, maneuvers, stalls, buffet, flights through turbulence, touch-and-go landings, and a ban on low-level flying except for takeoff and landing (Knaack, 1988). These severe restrictions were applied to all B-47s not previously inspected for cracks in the critical areas.

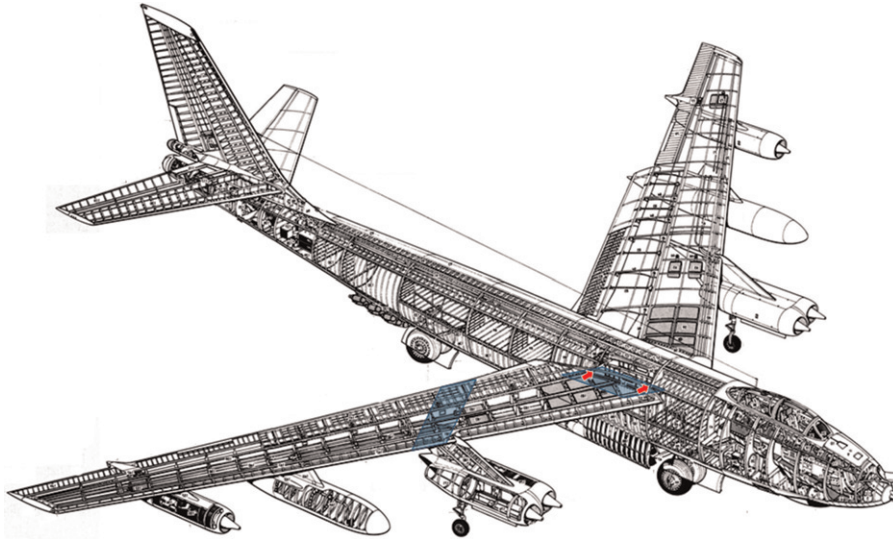


Figure 4 Blue-shaded fatigue-critical areas (only R.H. side indicated) determined from the B-74 crashes in 1958. Some inference was required by the present authors. The small red arrows point to the positions of two of the four wing-to-fuselage connecting pins.

The first task of the emergency program was an interim fix comprising disassembly of the suspect areas; reaming each bolt hole to oversize; detailed non-destructive inspection (NDI) followed by additional reaming of any holes with crack indications; and reassembly using oversize bolts. The interim fix was soon superseded by Boeing's 'ultimate' fix. This was much more extensive, requiring repairs and modifications that included the area where the lower wing skins met the fuselage; the inner-to-outer wing splices at WS-354 (see the diagonal blue-shaded area in [Figure 4](#)); and reaming out the aft wing-to-fuselage fittings and installing oversize connecting pins. These (steel) pins were the size and shape of a quart-sized (approximately 1 l) U.S. glass milk bottle. This eventually caused the emergency program to be dubbed 'Project Milk Bottle' ([Negaard, 1980](#); [Knaack, 1988](#)).

Most of the fleet was modified by October 1958, and by January 1959 all aircraft had been inspected and reworked at least once. When the program ended in June 1959, only a few interim-fixed aircraft still needed work. This was a truly remarkable feat, involving more than 1600 aircraft ([Negaard, 1980](#)).

3.2 Lessons Learned

As in the case of the Comet accidents, the B-74 crashes in 1958 caused a paradigm shift in how aircraft – in this case military aircraft – were designed and validated for service. It became evident that there was a need to control fatigue, and to make accurate and conservative Safe-Life fatigue predictions based on analysis and testing in the development stages of future aircraft. However, this change in philosophy did not (yet) include Fail-Safety, which had become a cornerstone of commercial aircraft design principles.

More specifically, 'Project Milk Bottle' accomplished much more than keeping the B-74s airborne. Firstly, Boeing had to develop a *structural integrity program* to assess their service life ([Knaack, 1988](#)). This program included full-scale fatigue tests by Boeing, Douglas and the National Aeronautics and Space Administration (NASA). These tests, the 1958 crashes, and other problems, revealed many gaps in contemporary knowledge about high-performance aircraft. In turn, this led the USAF to initiate a long-term program, the Aircraft Structural Integrity Program (ASIP), which continues with regular updating to the present day.

The early history of ASIP is described elsewhere ([Negaard, 1980](#)), but it is interesting to note the program's extent already by early 1959. ASIP was initially divided into 11 sub-program areas ([Negaard, 1980](#); [Miedler et al., 2002](#)): (1) design criteria, (2) mission profile data, (3) static test, (4) flight load survey, (5) low altitude gust environment, (6) fatigue test, (7) sonic fatigue, (8) high temperature structure, (9) interim service load, (10) velocity, gravity and height (VGH) life history recording, and (11) 8-channel service load recording.

Over the next several years the basic ASIP document remained little altered, although there was a large increase in Military Specifications (MIL SPECS) to support implementation of the program ([Miedler et al., 2002](#)). But, as discussed in Section 4, the F-111 crash in 1969 resulted in a major change to the USAF's structural integrity policy: namely, the introduction of the Damage Tolerance approach.

The current scope of ASIP is well described in an extensive review by [Callagher \(2007\)](#), which includes the current requirement to again consider redundant load paths where practicable (i.e., Fail-Safety).

4 The USAF General Dynamics F-111 Crash in 1969

4.1 Case History

In 1964 the General Dynamics Corporation was awarded a contract for the development and production of the F-111 aircraft, subsequently to be procured by the USAF and the Royal Australian Air Force (RAAF). The F-111 was an unusual aircraft: it was a variable-geometry 'swing-wing' fighter-bomber; and it used high-strength steel in major airframe components, notably the wing carry-through box, wing pivot fittings, some of the center fuselage longerons and the empennage carry-through structure (Buntin, 1977).

On December 22, 1969, just over a year after entering service, F-111A #67-0049 lost the left wing during a low-level training flight. The aircraft had accumulated only 107 airframe flight hours, and the failure occurred while it was pulling about 3.5 g, less than half the design limit load factor (Mar, 1991). An immediate on-site investigation revealed a flaw in the lower plate of the left-hand wing pivot fitting (Figure 5). This flaw had developed during manufacture and remained undetected despite its considerable size: 23.4 mm × 5.9 mm. As can be seen from Figure 5, a limited amount of fatigue crack growth occurred in service before fast fracture of the plate, which resulted in immediate loss of the wing.

This accident could conceivably have been considered an 'isolated case' in view of the most unusual flaw that caused it. However, fatigue and fracture problems were also encountered during the airframe test programs (Buntin, 1977). The overall concerns about structural integrity led to a fracture control program for the critical steel parts in the airframe. The approach was an expensive one that required aircraft to be periodically removed from service and the entire wing carry-through structure to be proof tested at -40°C . In fact, the entire aircraft was subjected to the proof test, such that all D6ac steel components were tested. Details of the proof test and associated fracture mechanics analyses are given by Buntin (1977). Besides the proof tests, inspectable areas were checked by standard NDI techniques to limit the possibility of any cracks reaching critical sizes between the tests.

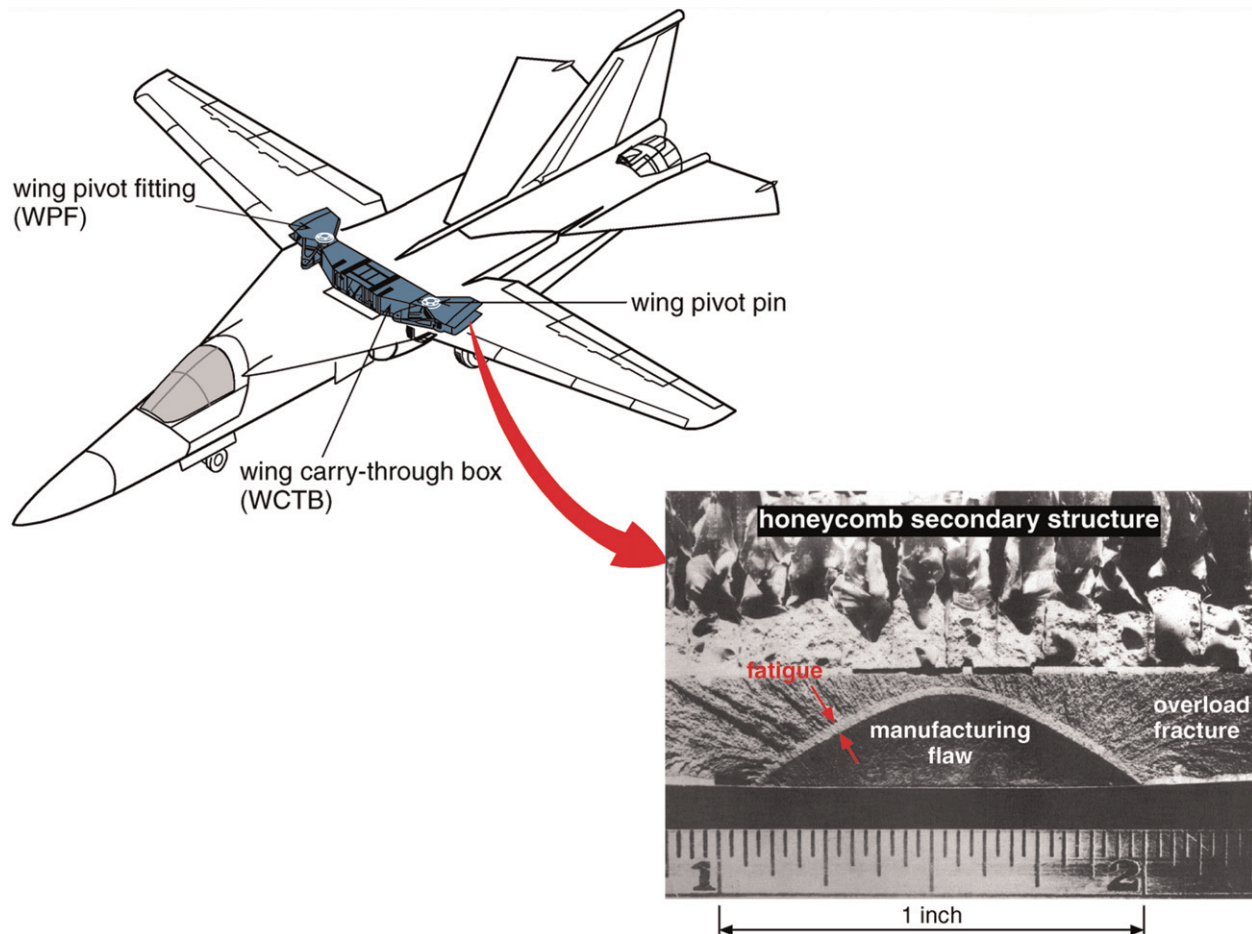


Figure 5 Origin of failure of F-111A #67-0049 a manufacturing flaw in the high-strength D6ac steel lower plate of the left-hand wing pivot fitting (note: 1 in = 2.54 cm).

4.2 Lessons Learned

The cold proof test was a specific solution to safe operation of the F-111. However, the loss of F-111A #67-0049, together with early and widespread fatigue cracking in the Lockheed C-5A wing boxes (Mar, 1991), led the USAF to reconsider and abandon its previous structural integrity policy, which was essentially a Safe-Life approach verified by full-scale fatigue testing to several lifetimes. (This approach had been introduced after the B-47 crashes discussed in Section 3.)

After much research, the USAF provided and mandated new guidelines to ensure aircraft structural integrity that focussed on safety. These guidelines became known as the 'Damage Tolerance philosophy,' incorporated in [Military Specification A-83444 \(1974\)](#) and [Military Standard 1530A \(11\) \(1975\)](#). This approach differs from the original Fail-Safe design principles, developed for commercial transport aircraft after the Comet crashes (see Section 2.2), in two major respects:

1. The possibility of cracks or flaws in a *new* structure must be considered. In fact, MIL SPEC A-83444 made it mandatory to assume initial, possibly undetectable, damage as a design requirement.
2. Structures may be inspectable or non-inspectable in service, i.e., there is an option for designing structures that are not intended to be inspected during the service life:
 - (a) Inspectable structures can be qualified as Fail-Safe or slow flow growth structures, for which initial damage must grow slowly and not reach a size large enough to cause failure within two inspection periods; and
 - (b) Non-inspectable structures may still be classified as damage tolerant provided they can be qualified for slow flow growth, which in this case means that the assumed initial damage must not grow to a size that would cause failure during a period equivalent to two design service lives.

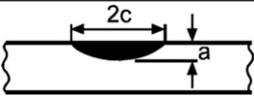
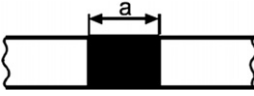

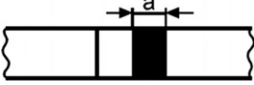
While the USAF Damage Tolerance approach has been effective in ensuring structural safety (its prime purpose), it is not the definitive solution to designing for aircraft structural integrity, see Section 8. Some general comments on the above two points will be made here.

4.2.1 Initial damage

Table 3 shows the MIL SPEC A-83444 initial damage assumptions for ensuring structural *safety*. Although the initial flaw geometries in **Table 3** are considered surrogates for observed damage, their sizes are large enough for fracture mechanics calculations of fatigue crack growth using models based on well-established macrocrack growth behavior. This is important because fracture mechanics is one cornerstone of the Damage Tolerance philosophy (the other one is *testing* of structures with *cracks* in critical locations).

MIL SPEC A-83444 also provided guidelines during the 1970s for obtaining estimates of initial flaw sizes for structural *durability* analyses, i.e., ensuring that an economic life would be achieved. The initial flaw sizes for durability analyses were set for all fastener holes as 0.254 mm. Alternative choices for initial durability crack sizes were allowed if the contractor could demonstrate by fractography (i.e., experimentally) that any crack-like discontinuities were actually smaller. This approach was referred to as the equivalent initial flaw size (EIFS) approach for quantifying the durability. At that time the issue of durability was seen as primarily an economic problem. However, durability has since been linked to safety problems in older aircraft (see Section 6).

Table 3 USAF MIL-A-83444 Damage Tolerance *safety* requirements for assumed initial damage

Types of flaw		Flaw size a (mm) to be assumed immediately after inspection			
Description	Geometry	Aspect ratio (a/c)	Pre-service inspection with high standard NDI		
			Fail-Safe	Slow flow growth	In-service inspection with special NDI
Surface flaw		1.0	1.27	3.18	6.35
Through crack		0.2	2.54	6.35	12.7
Corner flaw at a hole		1.0	0.51	1.27	6.35 mm beyond fastener head or nut
Through crack at a hole		0.2	0.51	1.27	6.35 mm beyond fastener head or nut

Besides differing from the original Fail-Safe approach, where the structure is designed to be durable by achieving a satisfactory life without significant damage, the MIL SPEC A-83444 durability requirements concern initial flaw sizes well below 0.5 mm, in what is now referred to as the *short crack regime*. The behavior of short cracks is greatly influenced by many factors, including local stress-strain fields at notches and fastener holes, contact surface fretting, fastener fit and hole preparation, and material micro-structure (Wanhill, 1986). This means that analytical modeling of short crack growth is problematical, as has been shown by much Quantitative Fractography (QF) following the Macchi MB-326H crash in 1990, see Section 8.2.

Another, more fundamental, problem – at least for transport aircraft – is that durability need not be determined by the immediate and continuous growth of fatigue cracks from small initial flaws or discontinuities. For example, an extensive investigation of fatigue in pressure cabin lap splices from service aircraft and full-scale test articles showed that there were damage accumulation ‘initiation’ periods up to 75% of total life before a regular process of fatigue crack growth began (Wanhill and Koolloos, 2001). On the other hand, many QF data for high-performance aircraft have shown that crack growth can begin almost immediately, even from initial discontinuities smaller than 0.1 mm (Barter *et al.*, 2010).

4.2.2 Non-inspectable structures

The USAF acceptance of non-inspectable structures as damage tolerant, on the basis of slow flaw growth, is not followed by civil aviation authorities. Non-inspectable structures are placed firmly in the Safe-Life category by civil aviation authorities (Swift, 1983), which means they are undesirable in terms of safety and economics: the guaranteed service lives (in the absence of representative full-scale testing) would likely be uneconomically short compared to Fail-Safe structures, see the earlier remarks in Section 2.2.

5 The Dan Air Boeing 707 Crash in 1977

5.1 Case History

On May 14, 1977, a Dan Air Boeing 707-321C air freighter lost the entire right-hand horizontal stabilizer just before it would have landed at Lusaka International Airport. The aircraft had been manufactured in 1963 and had since accumulated 47 621 airframe flight hours and 16 723 landings (Mar, 1991). In view of the design life goals, 60 000 flight hours and 20 years, this aircraft was past its prime. In fact, the crash led to the striking but unflattering term *geriatric jet* (Ramsden, 1977).

Investigation traced the accident back to fatigue failure from a fastener hole in the upper chord of the rear spar of the right-hand horizontal stabilizer (Figure 6). This was unexpected: the designers had presumed that any fatigue cracking could (or would) occur in the rear spar attachment lugs (Eastin and Bristow, 2003). In-service inspections were therefore focussed on these lugs, which were designed for Fail-Safety. However, there was (very) limited provision for inspection of the horizontal stabilizer, see the last paragraph of this Section.

The available evidence from the investigation indicated that fatigue cracking was due to unanticipated high local stresses and significant oscillating loads on the horizontal stabilizer during normal landing roll, owing to speed brake deployment (Eastin and Bristow, 2003). Fatigue spread into the upper chord, and the overall crack growth was rapid with numerous large intermittent tensile crack jumps. The fatigue crack growth finally gave way to fast fracture down through the entire rear spar, and this resulted in the stabilizer separating from the aircraft (Howard, 1986).

The section A–A in Figure 6 shows that the rear spar consisted of discrete elements. These were linked together by fasteners. This configuration was intended to be a Fail-Safe design. It will be recalled from Section 2 that a Fail-Safe design should be able to sustain significant and easily detectable damage before safety is compromised. The key to the Dan Air Boeing 707 crash is ‘easily detectable.’ This means:

1. Sustainable significant damage should be large enough to be found by the specified inspection method.
2. When the damage reaches a size detectable by the specified inspection method there should be adequate time for inspection by the specified inspection technique before the damage becomes critical.

Both these aspects were concerned in the accident. First, periodic inspection of the horizontal stabilizer had a recommended inspection time less than half an hour. This suggests visual inspection, which – as subsequently demonstrated by post-accident fleet inspection – would not have detected a *partial* failure of the upper chord of the rear spar. Second, once the upper chord had failed completely, enabling the damage to be detected visually, the structure could not sustain the service loads long enough to enable the failure to be detected (Aircraft Accident Report 9/78, 1979). Thus, although the manufacturer had designed the horizontal stabilizer to be Fail-Safe, in practice it was not, owing to the unanticipated failure location, unanticipated loads, and the inadequacy of the inspection method for this location.

5.2 Lessons Learned

The most immediate lesson from the Dan Air Boeing 707 crash is:

1. A Fail-Safe design concept does not by itself constitute a Fail-Safe design. Inspectability is equally important, as discussed in the last paragraph of Section 5.1.

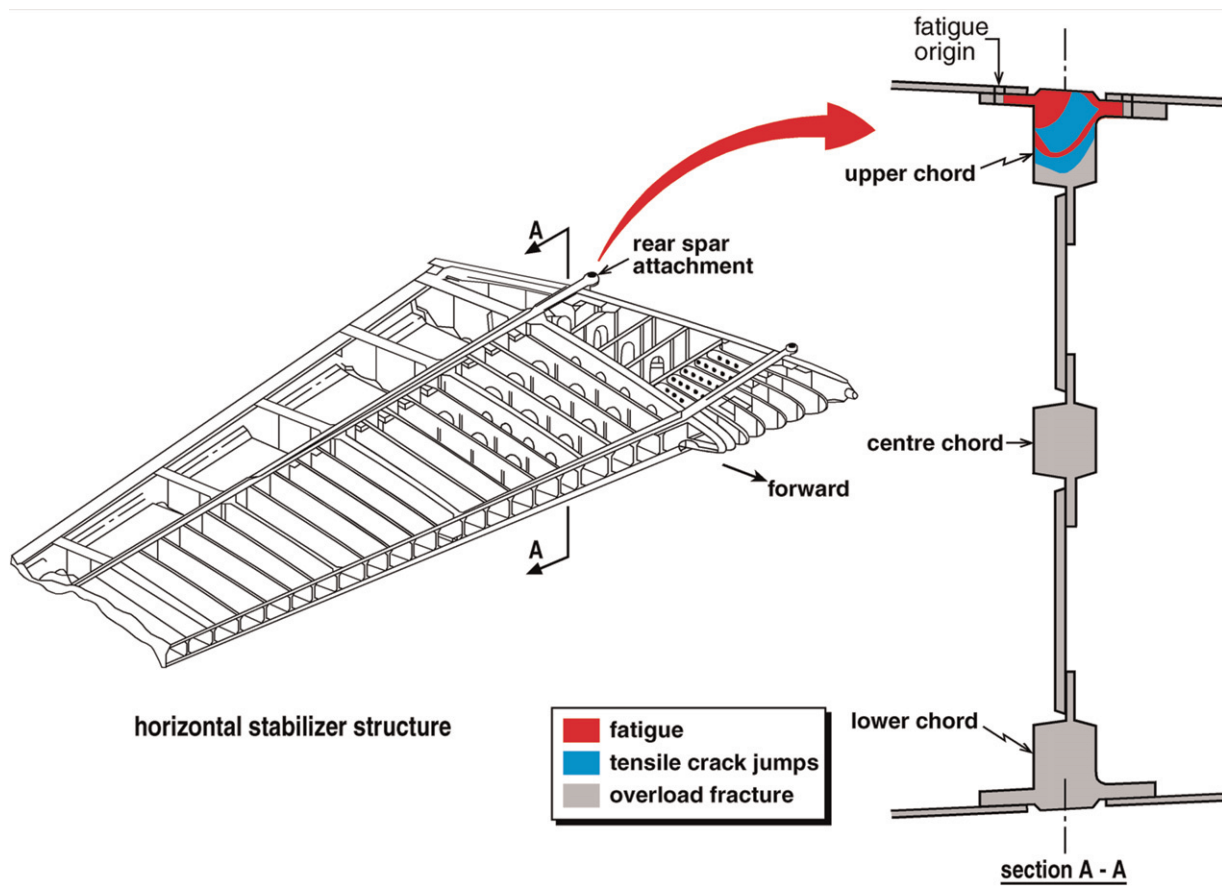


Figure 6 Origin of failure in the Dan Air Boeing 707: a fastener hole in the upper chord of the rear spar of the right-hand horizontal stabilizer (after Howard, 1986).

Two more important lessons came out of this investigation:

2. Tiffany *et al.* (2010) state: "It is essential that load surveys be performed to develop accurate loads and load spectra (including both ground and flight loads) for use in damage tolerance (i.e., crack growth) analyses, fatigue analyses and fail-safe residual strength and life analyses. These load spectra are also very important for the full-scale fatigue test."
3. The manufacturer modified the horizontal stabilizer design for the Boeing 707-300 series in order to increase the torsional stiffness. This was necessary because of an overall increase in aircraft weight (a frequent result of series development). The modification was a material change from an aluminum alloy to a stainless steel for a large part of the top skin attached to the front and rear spars (Howard, 1986). Unfortunately, this modification was not checked by a full-scale fatigue test, which was not required by the contemporary regulations. However, after the Dan Air Boeing 707 crash a full-scale test on a modified horizontal stabilizer reproduced the service failure (Schijve, 1994).

All three lessons point to *the need to do full-scale fatigue testing*.

The Dan Air Boeing 707 crash also prompted airworthiness authorities to reconsider the fatigue problems of older aircraft. It became clear that some existing inspection methods and schedules were inadequate, and that supplementary inspection programs were needed to prevent older aircraft from becoming fatigue-critical.

6 The Aloha Airlines Boeing 737 Accident in 1988

6.1 Case History

On April 28, 1988, Aloha Airlines flight 243, a Boeing 737-297, experienced an explosive decompression during climb out at cruise altitude. About 5.5 m of the pressure cabin skin and supporting structure aft of the cabin entrance door and above the passenger floor-line separated from the aircraft (see the photograph in Figure 7). Amazingly, the damage did not result in disintegration of the aircraft, and a successful emergency landing was made.

The aircraft had been manufactured in 1969 and had since accumulated 35 496 airframe flight hours and 89 680 landings (Aircraft Accident Report, Aloha Airlines, Flight 243, 1989). Owing to the short distance between destinations on some Aloha

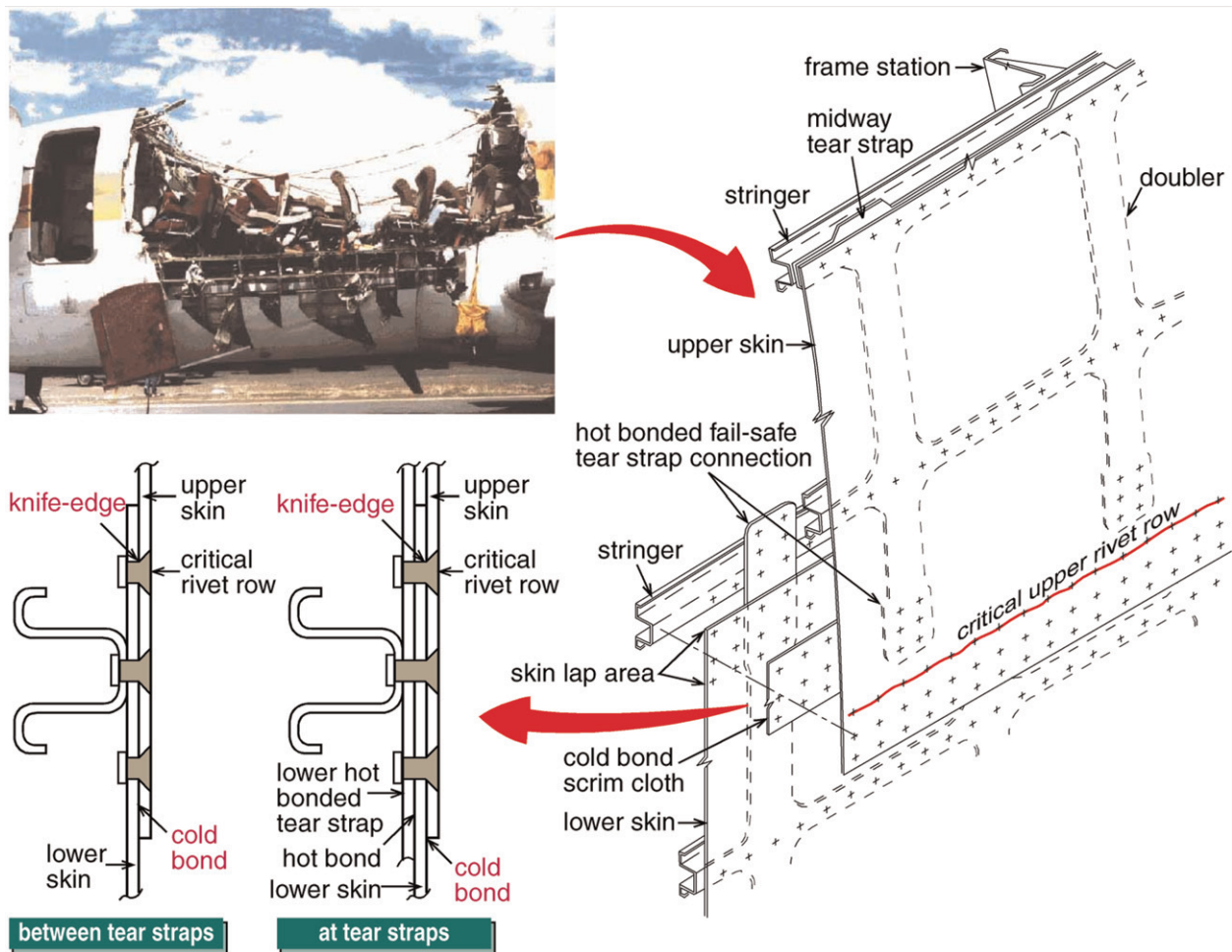


Figure 7 Structural aspects of the Aloha Airlines Boeing 737-297 accident: MSD occurred in the outer (upper) skin, commencing from the knife-edges of the rivet holes along the upper rivet row.

Airlines routes, the maximum pressurization differential was not reached in every flight. Thus, the number of equivalent full pressurization cycles was significantly less than 89 680. Nevertheless, the aircraft was nearly 19 years old. It was also operating with long-term exposure to warm, humid, and maritime air.

Investigation showed the large loss of pressure cabin skin was caused by rapid link-up of many fatigue cracks in the same longitudinal skin splice. The fatigue cracks began at the knife-edges of rivet holes along the upper rivet row of the splice (see the diagrams in Figure 7). This type of cracking is called multiple site fatigue damage (MSD). Somewhat poignantly, Swift discussed the potential dangers of MSD less than a year before the accident (Swift, 1987).

In more detail, the Aloha Airlines Boeing 737 accident occurred because of several factors and their interrelation. These factors are:

6.1.1 Skin splice configuration

The pressure cabin longitudinal skin splice had been cold bonded, using an epoxy-impregnated woven scrim cloth (see Figure 7), as well as riveting. This should have resulted in a safe and durable structure, whereby the pressure cabin loads would be transferred through the bonded splice as a whole, rather than via the rivets only. The splice design was based on this integral load transfer: hence the use of relatively thin skins, absence of a doubler in the splice, and acceptance of rivet hole knife-edges.

6.1.2 Cold bonding production difficulties

The early service history of Boeing 737s with cold bonded skin splices revealed difficulties with the bonding process. These problems resulted in random occurrence of bonds with low environmental durability (i.e., susceptible to corrosion) and with some areas that had not bonded at all (Aircraft Accident Report, Aloha Airlines, Flight 243, 1989). Cold bonding was discontinued in 1972, after production of the Aloha Airlines accident aircraft, and obviously well before the accident.

6.1.3 Maintenance and surveillance

Owing to the cold bonding problems, Boeing issued service bulletins in 1972, 1974, and 1987, and the Federal Aviation Administration issued an Airworthiness Directive in 1987. These documents called for skin splice inspections at regular intervals, and repairs if necessary. However, there was a lack of proper maintenance by Aloha Airlines.

The way the above three factors were involved in the accident is as follows. Defective cold bonding allowed moisture to enter the skin splice during service. This led to corrosion-induced disbonding, both in the cold bonded skin splice and the associated hot bonded tear straps. The loss of skin splice integrity meant that the pressure cabin loads were transferred through the rivets. These had countersunk heads causing knife-edges in the upper skin (see [Figure 7](#)), and the knife-edges caused *mechanically-induced* MSD fatigue of the upper skin along the upper rivet row of the splice. The disbonding and fatigue cracking remained undetected (but not undetectable if there had been proper maintenance and surveillance) until the cracks linked up rapidly. They did so without hindrance by the disbonded tear straps. In other words, the tear straps were unable to provide fail-safety via controlled decompression of the pressure cabin. The result was explosive decompression with separation of a major part of the pressure cabin, as mentioned earlier, and it was only by great good fortune that the aircraft did not disintegrate and remained controllable. Even so, the post-mishap performance of the crew was exemplary.

6.2 Lessons Learned

The Aloha Airlines Boeing 737-297 accident prompted worldwide activities to ensure the safety and structural integrity of aging aircraft. Manufacturers, operators, and airworthiness authorities have collaborated to develop new regulations and advisory circulars, or extend existing ones. The FAA joined with NASA in organizing several aging aircraft conferences, and research funding was provided for investigation of many aspects of the problem.

In all these activities, the emphasis has been on Widespread Fatigue Damage (WFD) in pressure cabins, although the wings and empennage are included ([Goranson, 1993](#)). However, another major issue is corrosion. Soon after the Aloha Airlines Boeing 737 accident, an Airworthiness Assurance Working Group (AAWG) was formed to establish a common approach to corrosion control in commercial transport aircraft ([Paone, 1993](#)). Some general points on WFD and corrosion will be made here.

6.2.1 WFD

There are two types of WFD: MSD – as in the Aloha Airlines Boeing 737-297 – where fatigue cracks occur at many locations in the same structural element; and multiple element fatigue damage (MED), which is characterized by the simultaneous presence of fatigue cracks in adjacent structural elements.

WFD is a major issue because it can rapidly decrease the residual strength, with a loss of Fail-Safe capability both in terms of residual strength and adequate time for inspection. Avoidance of WFD requires identifying susceptible areas, based on tests and service experience; fatigue analyses linking safety and durability; assessment of inspection possibilities; and terminating actions (repair, replacement, or retirement). Much more information is given by [Goranson \(1993\)](#). However, it is noteworthy that more consideration is being given to the terminating actions of replacement or retirement. There has been a long-standing practice of ensuring safety by repetitive inspections and necessary repairs, and also repairs of repairs. Following the Aloha Airlines Boeing 737 accident, and in the light of subsequent investigations and aging aircraft inspections, this ‘traditional’ practice is regarded less favorably, although it is still a potential option.

It should be noted that in 2011 the FAA introduced the concept of Limit of Validity (LOV) for transport aircraft over 75 000 lb (34 000 kg). The LOV concept requires full-scale fatigue testing to determine the onset of WFD ([Federal Aviation Administration, 2011](#)), and as such effectively sets a Safe-Life equivalent for these aircraft.

6.2.2 Corrosion

The Aloha Airlines Boeing 737-297 accident brought fuller recognition of the deleterious effects of corrosion and combinations of corrosion and fatigue on aircraft structural integrity, especially for older aircraft. Severe corrosion can significantly affect the damage tolerance capability by reducing the residual strength. In combination with fatigue, there is a risk of increased and accelerated WFD in transport aircraft ([Akdeniz, 2001](#)).

Corrosion control programs have been set up for commercial transport aircraft and aging military aircraft ([Paone, 1993](#); [Nieser, 1993](#); [Akdeniz, 2001](#)); and many aspects of corrosion and corrosion control have been examined for aging military aircraft ([RTO/NATO, 2011](#)). The corrosion control programs require inspections and maintenance based on calendar intervals, unlike fatigue-oriented inspection and maintenance. However, it is impractical to separate the two types of inspection and maintenance. Many commercial aircraft operators have, therefore, elected to modify the structural fatigue inspection schedules to fit the corrosion inspection intervals.

The effectiveness of corrosion control programs is assessed from the ‘levels’ of corrosion found during inspections. These levels have been defined as follows ([Paone, 1993](#)):

- Level 1: Corrosion, local or light, can be reworked or blended out;
- Level 2: Local repair or partial replacement, widespread reworks or blendouts; and
- Level 3: Immediate airworthiness concern.

Only level 1, or better, is considered acceptable for an *effective* corrosion control program.

7 The RAAF Aermacchi MB-326H Crash in 1990

7.1 Case History

The MB-326 was a single jet-engined trainer designed and originally built by Aermacchi. The first flight was in December 1957. In 1965 the RAAF selected this aircraft for its trainer requirements. From 1967 to 1972 the RAAF acquired a total of 97 MB-326Hs. The first 20 were assembled mainly from Italian components, with the remainder built under license by the Commonwealth Aircraft Corporation (CAC). During the mid-1980s the fleet underwent a Life Of Type EXtension program (LOTEX), which included CAC replacing the wing center sections and spars.

On November 22, 1990, MB-326H #A7-076 lost the left wing during a 6.5 g maneuver off the east coast of Australia. Since LOTEX the aircraft had accumulated 1904 flights, representing 2188 airframe flight hours. This was only about 70% of the proclaimed Safe-Life, which was derived from full-scale fatigue testing in 1974/5. The wreckage was recovered from the sea, and examination showed that the left wing had separated by fatigue and fracture of the 7075-T6 aluminum alloy lower spar cap close to the center section attachment fitting; and that the fatigue cracking had begun at a poorly-drilled bolt hole (Figure 8). This hole should have been flat-bottomed and reamed, but the drill used in manufacture during LOTEX had penetrated through to the inside surface of the spar flange, resulting in severe stress concentrations (Clark *et al.*, 1997).

Further examination of the failed spar revealed manufacturing flaws in many holes, suggesting that similar flaws could exist in the remainder of the fleet. Hence an NDI procedure was developed for fleet-wide examination of possibly-flawed bolt holes in critical locations. This huge effort was intended to enable a continuing safety-by-inspection procedure, with a conservative inspection period derived from QF analysis of the fatigue cracking shown in Figure 8.

Unfortunately, the problem turned out to be much more serious. The fleet-wide NDI revealed that four other wings contained fatigue cracks growing from bolt hole manufacturing flaws, including the right wing of MB-326H #A7-076, as might be expected. However, detailed examination of this right wing also showed that (1) the safety-by-inspection procedure could be unreliable, and (2) there were many cracks growing from *normal build quality* structural details (Athiniotis *et al.*, 1991; Clark *et al.*, 1997). This meant that not only was there cracking from LOTEX-related manufacturing anomalies, but also that the Safe-Life of the entire fleet was questionable and needed to be reassessed.

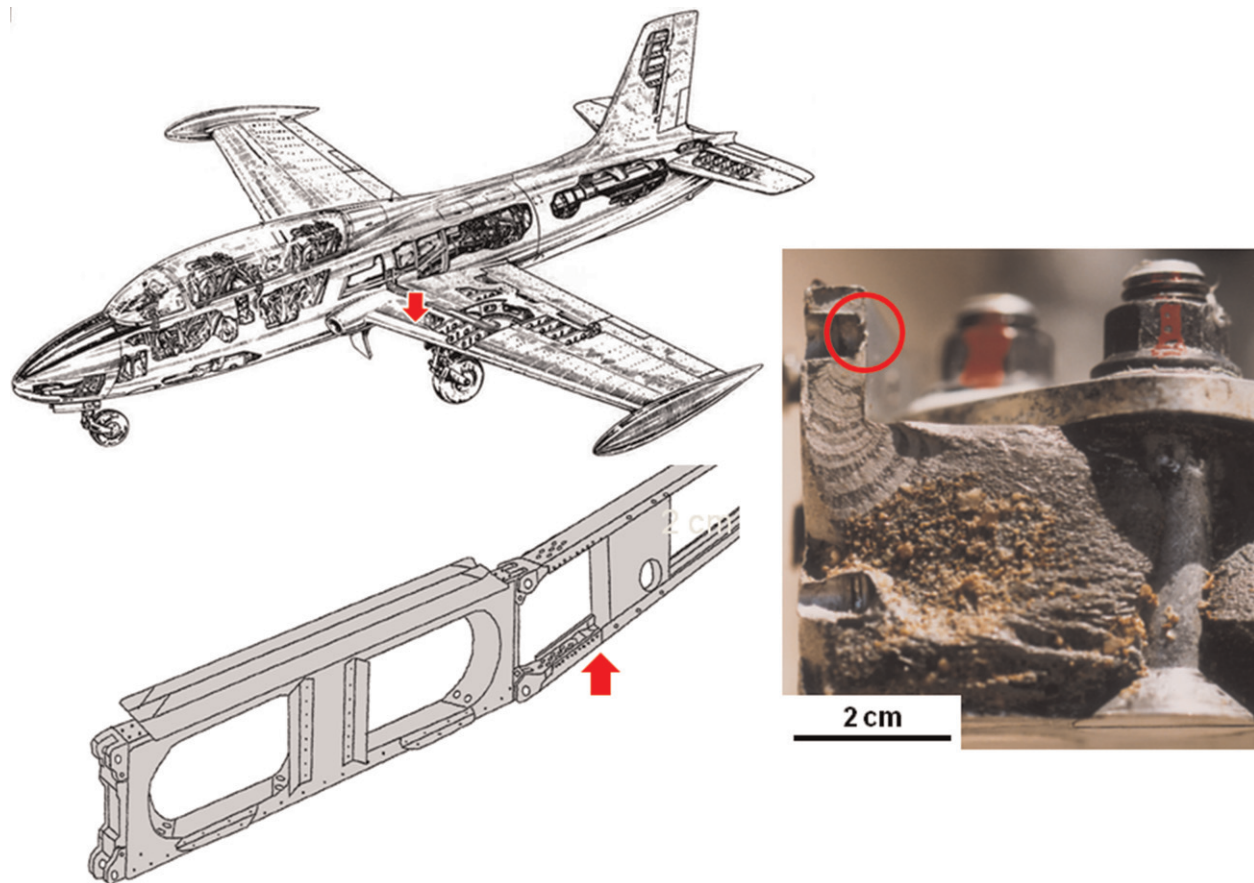


Figure 8 Origin of failure of Macchi MB326H #A7-076: a poorly-drilled bolt hole (circled) in the flange of the left wing lower spar cap. Fatigue crack growth progression markings are clearly visible.

7.1.1 Safe-Life reassessment

This was based on teardown, NDI, and QF of fatigue cracks in nine high-life wings as well as the MB-326H #A7-076 wings. The QF data were used to estimate 'worst case' fatigue crack growth lives from the time the LOTEX-refurbished aircraft entered service, i.e., assuming that cracking began immediately. The crack growth lives were converted into the Safe-Life damage units used for the full-scale test, thereby accounting for the differing load histories of individual aircraft. Pooling these damage estimates, and applying an appropriate scatter factor, resulted in a reduced life limit of only 47% of the original Safe-Life. Application of this reduced life limit would reduce the airworthy fleet size from 69 to 11, which was unacceptable (Clark *et al.*, 1997). Consequently a fleet recovery program was initiated.

7.1.2 Fleet recovery program

This is described in detail by Clark *et al.* (1997), so only a summary is given here. The following steps were taken:

1. Teardown of additional wing spars with over 1000 bolt holes examined by NDI: QF data obtained for 103 fatigue cracks.
2. New and previous QF data were used to develop an exponential crack growth equation describing fleet-wide cracking from the bolt holes.
3. An exponential equation was used to estimate the crack growth lives. As before, these were converted into Safe-Life damage units and then used to reassess the reduced life limit. The result was 52%, close to the first estimate of 47%. A value of 50% was adopted for actual lifing.
4. The safety-by-inspection procedure was scrutinized. Inspection difficulties showed that this approach, *on its own*, was inadequate for extending the service life.
5. Another full-scale fatigue test to obtain more confidence in the life limit was considered. However, close agreement between the first and second estimates of the reduced life limits, and the long lead time for such a test, led to rejecting this option.
6. Wing repair was considered. This would require NDI of thousands of holes and refurbishment of many of them. Even so, there would be no guarantee that all critical areas had been dealt with. Thus this option was also rejected.
7. Wing replacement remained as the only feasible method of fleet recovery. Thirty new wing sets were purchased to allow continued fleet operation.

The question remained whether the new wings would have similar lives to those of the old wings. In October 1994 a mid-air collision between two Macchi aircraft with new wings enabled assessing the build quality of the (damaged) new wings (Clark *et al.*, 1997). Although there were some differences compared to the old wings, the overall build quality was similar. Hence a similar reduced life limit was adopted.

7.2 Lessons Learned

The MB326H #A7-076 wing failure caused the RAAF's structural integrity policy to change similarly to that of the USAF after the B-47 crashes in 1958 (Section 3). The effort (and urgency) required for the Macchi recovery program demonstrated the need to establish a rigorous structural integrity program (ASIP) early in the life of an aircraft fleet. Since 1990 the RAAF has established comprehensive ASIPs for each fleet. This has been a major undertaking, since the RAAF operates a wide variety of aircraft.

Several other points were made by Clark *et al.* (1997) and are summarized here.

7.2.1 Safe-Life full-scale fatigue test

The fleet recovery program revealed two main problems with the fleet Safe-Life estimate. After the wing failure a detailed review of the estimate showed it to be optimistic. Secondly, the original estimate did not account for variations in build quality, which were unforeseen. These problems meant that the original Safe-Life fatigue test data were inappropriate for the recovery program.

7.2.2 Safety-by-inspection

As mentioned earlier, sole use of this approach was found to be inadequate. NDI crack-detection difficulties prevented determining inspection intervals that would guarantee the structural integrity. (Authors' comment: This is not very surprising, since the aircraft was not designed to be inspectable.)

7.2.3 Hybrid lifing approach

The teardown-based Safe-Life reassessment mentioned in Section 7.1 was combined with safety-by-inspection for some critical locations. This required considerable assessment and 'engineering judgement.'

In the context of this historical review the foregoing points and establishment of ASIPs for RAAF fleets are not generally innovative. However, with the hindsight of more than 20 years it seems fair to state that the introduction of extensive QF for fatigue life (re)assessment represents a true milestone, see Table 1. The future importance of this approach was recognized by Clark *et al.* (1997), and extensive DSTO research since then has demonstrated that QF-based fatigue crack growth analyses are essential, both during full-scale and component testing and in subsequent structural integrity management of the fleet (Barter *et al.*, 2010).

7.3 Additional Remarks: Corrosion and Fatigue

Besides the wings, the fatigue life of the center sections was reassessed from teardowns and NDI. Corrosion pitting was present in screw holes connecting the steel spars to the aluminum alloy webs, and the pitting was also associated with fatigue cracking (Clark *et al.*, 1997). Although by no means unique, this *direct* association between corrosion and fatigue in in-service aircraft (in contrast to the Aloha Airlines Boeing 737-297, see Section 6.1) is important and merits further discussion, see Section 8.2.

It was decided to refurbish the center section holes as necessary, and to use a safety-by-inspection approach until cracking was detected in unrefurbishable bolt holes or until the life limit for the center sections was reached (Clark *et al.*, 1997). In addition, the RAAF included the possibilities and opportunities for detecting and dealing with corrosion in the overall structural integrity strategy (Wilson, 1995).

8 Summary and Ongoing Developments

8.1 The Case Histories

The Comet, F-111, 707, and 737 case histories are often considered to be milestones in the development of aircraft structural integrity; and the B-47 and MB326H case histories are certainly regarded as such by the USAF and RAAF. Lessons learned from all these accidents have greatly influenced and improved the aerospace community's knowledge and perception of the problems involved in ensuring safety and durability. However, while these accidents have been especially important and influential for changing design approaches and procedures and specifications, there have been many other failures contributing to the overall developments in aircraft structural integrity. Indeed, these developments continue to the present day, and will likely do so for the foreseeable future.

A particularly poignant failure category is accidents due to improper repairs. There have been tragic accidents owing to improper repairs. These include crashes by the Japan Airlines Flight 123 Boeing 747SR in August 1985, 520 fatalities; a USAF HC130P Lockheed Hercules in April 1986, 11 fatalities; and the China Airlines Flight 611 Boeing 747-209B in May 2002, 225 fatalities. These disasters, and others, have shown the need to consider the damage tolerance of repaired structures as well as the original structures. More information is given in Tiffany *et al.* (2010).

8.2 Developments in Fatigue Analysis of Conventional Airframes

Ongoing research and development aims to improve structural analysis capabilities and the methods for fatigue life and crack growth prediction. Since the Aloha Airlines accident this has included the possible and actual effects of corrosion on fatigue and the combined action of corrosion and fatigue. These developments are reflected in the evolution of fatigue life assessment methods, which is a highly complex process. Table 4 attempts to capture the main aspects of these methods, but is necessarily subject to interpretation since there are many adaptations of these methods. Two of the main developments are summarized here.

8.2.1 Short crack growth

A major problem with the Damage Tolerance approach is that the predicted early crack growth, especially for durability analyses, is highly questionable. This is because early crack growth is derived from back-extrapolation of (1) variable-amplitude long crack growth data or (2) variable-amplitude growth curves derived from long crack constant-amplitude data, with both methods using analytical models 'tuned' to the behavior of long cracks. Also, as discussed in Section 4.2, the initial damage assumptions for durability predictions are well within the *short crack regime*, where analytical modeling is anyway problematical.

When the Damage Tolerance approach was introduced, in the early 1970s, there was a dearth of actual data for short crack growth. Owing to advances in QF this is no longer the case, particularly since the MB326H accident in 1990. There are now many data for early crack growth in high-performance aircraft (Barter *et al.*, 2010), including the types and sizes of fatigue-initiating initial discontinuities (Barter *et al.*, 2012). Analysis methods have been developed to use these data for life predictions and reassessments, including the use of equivalent pre-crack sizes (EPS) to represent realistic initial discontinuities (Barter *et al.*, 2010).

8.2.2 Corrosion and fatigue

Table 4 shows that the holistic approach (called the HOListic Structural Integrity Process, HOLSIP) includes corrosion (Bellinger and Liao, 2009). This is an obvious addition, since it is well-known that corrosion is an important source of fatigue cracking in aircraft structures. However, what is not obvious is whether corrosion only provides a 'kick-start' to fatigue crack growth, as in the Aloha airlines 737 (indirectly) and MB326H center sections (directly); or whether corrosion and fatigue act in combination. It is noteworthy that this uncertainty was recognized more than 40 years ago (Anderson, 1972).

Since the Aloha Airlines 737 accident there has been considerable effort to develop life-prediction models combining corrosion and fatigue. However, only HOLSIP has taken case history data into account. Other models have not done so, most notably those investigating corrosion and fatigue of the industry standard aluminum alloy AA2024-T3. Laboratory experiments and modeling for this alloy were intended to be applicable to corrosion and fatigue problems in transport aircraft pressure cabins, specifically the longitudinal lap splices. However, the basic modeling assumption that fatigue initiation would be due to corrosion pitting is

Table 4 Evolution of methods for aircraft fatigue life assessment (Barter *et al.*, 2010)

-
- Stress–life ($S-N$)
 - fatigue limits, S_e ; unnotched and notched (K_t); constant amplitude (CA) data
 - modifications to S_e
 - mean stress effects (R)
 - linear damage rule, also for variable R
 - scatter factors
 - Strain–life ($\epsilon-N$)
 - strain – life equation, unnotched data, $R = -1$
 - cyclic stress–strain curve analysis
 - rainflow cycle counting (closed hysteresis loops)
 - stress–strain at critical location (notch analysis)
 - mean stress effects (R) via equivalent strain equations, leading to equivalent strain amplitudes
 - damage accumulation rule
 - Damage Tolerance (DT)
 - specified equivalent initial flaw sizes (EIFS) based on NDI capabilities
 - back-extrapolation of long crack growth data to derive short crack growth
 - LEM long crack growth models (non-interaction, yield zone, crack opening, strip yield) to derive variable amplitude (VA) crack growth from constant amplitude (CA) data
 - possible use of crack opening model for short cracks (FASTRAN); differences in long and short crack thresholds need to be included
 - mainly deterministic: stochastic approach becoming accepted
 - Holistic approach (HOLSIP): proposed
 - fatigue initiation mechanisms (also as functions of notch stress concentrations, K_t)
 - fatigue initiation lives ($S-N$ and/or $\epsilon-N$ assessments)
 - evaluation and selection of marker load strategies for Quantitative Fractography (QF) of short-to-long crack growth
 - actual short-to-long long crack growth using marker loads and QF
 - establishment, validation and choice of appropriate crack growth models and ‘laws,’
 - deterministic (‘upper bound’) and stochastic approaches
 - environmental effects, notably corrosion
 - DSTO approach: proposed and implemented for the RAAF
 - actual initial discontinuity/flaw sizes and their equivalent pre-crack sizes (EPS)
 - actual short-to-long crack growth data using Quantitative Fractography (QF)
 - data compilations to establish empirical relationships describing crack growth behavior
 - deterministic (‘upper bound’) estimates of *lead crack* growth
 - scatter factors
-

incorrect: this was shown by investigating lap splices taken from service aircraft (Wanhill and Koolloos, 2001). Also, it appeared that in practical terms there was either a corrosion problem or a fatigue problem: any corrosion fatigue crack growth was secondary to an overall corrosion problem. Similarly, recently reported service experience with a high-performance aircraft suggests that while corrosion pitting may be responsible for causing fatigue cracking, the action of corrosion is predominantly a ground-based phenomenon, while fatigue occurs mainly during flight (Barter and Molent, 2013; Molent, 2014).

From the foregoing it must be concluded that the question whether corrosion and fatigue act in combination or independently requires further investigation per aircraft type and operational conditions. This is an important issue, since it can have a major impact on service life management (Barter and Molent, 2013).

8.3 Developments in Materials and Implications for Structural Integrity

Aircraft structural alloys must have outstanding combinations of engineering properties and also enable the manufacturing of lightweight and durable aircraft structures (i.e., the airframe). Aluminum alloys have predominated in aircraft manufacturing since introduction of the Boeing 247 (1933) and Douglas DC-2 (1934), but composites (mainly carbon fiber reinforced plastics, CFRP) and titanium alloys nowadays provide competition for certain applications. However, it is important to note that the choices of materials for different types of aircraft vary greatly:

- Transport aircraft: Aluminum alloys still account for about 60% of the airframe structural weight, with the notable exceptions of the Boeing 787 *Dreamliner* and Airbus A350 XWB, which use 50–53% composites and only 19–20% aluminum.
- Tactical aircraft: The emphasis tends to be on composites and lesser percentages of titanium, aluminum, and steels. For example, the British Aerospace/DASA/CASA/Alenia Eurofighter and Lockheed Martin F-35 both use 35–40% composites, most of which are in the external skin and substructure. There are exceptions, notably the Lockheed Martin/Boeing F-22, which uses only 24% composites and 20% aluminum. Much of the F-22 structure (42%) is titanium, which can tolerate the relatively high service temperatures, especially in the aft fuselage.

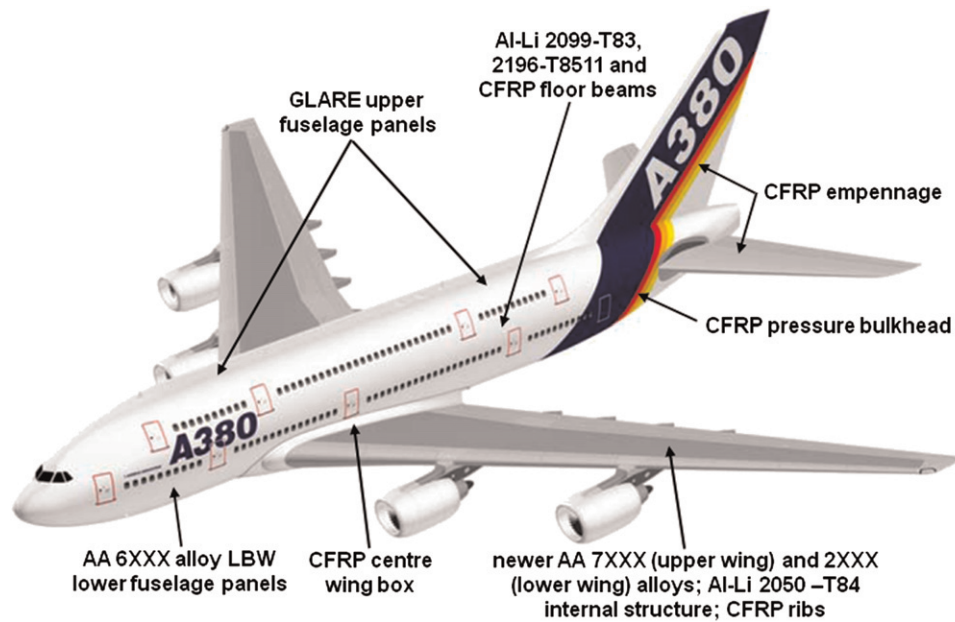


Figure 9 Advanced materials and processing for major structural areas of the Airbus A380 (Wanhill, 2013): CFRP=Carbon Fiber Reinforced Plastic; GLARE=GLASS REinforced aluminum laminates; AA 2XXX, 6XXX, and 7XXX=conventional aluminum alloys; Al-Li=Aluminum-Lithium alloys; LBW=Laser Beam Welding.

- Helicopters: Owing to the importance of weight savings for vertical lift aircraft, even higher percentages of composites are being used. For example, the Bell-Boeing V-22 Osprey airframe consists of nearly 80% composites (Deo *et al.*, 2003) and the NHIndustries NH90 airframe is over 90% composites.

The modern trend is to design and build hybrid airframe structures, not least because of the availability of third generation aluminum-lithium (Al-Li) alloys. Al-Li alloys provide practical structural efficiencies that can rival those of CFRP composites, resulting in similar weight savings with respect to conventional aluminum alloys (Wanhill, 2013). The Airbus A380 represents an important example of hybrid airframe construction, Figure 9.

The structural integrity implications of using the newer materials are summarized here:

1. Replacement of legacy aluminum alloys by the current (third) generation Al-Li alloys will hardly change the fatigue and fracture structural analysis approaches, since these alloys have reached a similar level of reliability in their engineering properties (Wanhill, 2013).
2. GLARE (GLASS REinforced aluminum laminates) is a metal-based composite used extensively in the Airbus A380, see Figure 9. GLARE has fatigue and fracture properties basically amenable to conventional analyses for all-metal airframes. There are some differences, and these have led to the development of a GLARE design 'toolbox' addressing and demonstrating compliance with current airworthiness regulations for metallic structures (Beumler, 2004).
3. CFRP composites have very different properties. The design principles for metallic aircraft structures cannot be used for CFRPs. They have high fatigue resistance when free from defects and stress concentrations, but are susceptible to impact damage and subsequent fatigue cracking and delamination. The growth of this damage is difficult to predict, and this also makes it difficult to validate repairs. Other related problems are inspection reliability, sensitivity to stress concentrations, and difficulties in analyzing complex components to predict the onset of failure. The overall result of these problems is that safety is currently ensured by over-designing CFRPs according to the 'no growth' Damage Tolerance principle (Federal Aviation Administration, 2009; Military Composite Materials Handbook, 2013).

However, this does not mean that there are no further developments. On the contrary: the FAA, for example, has been actively sponsoring research into several key areas, including structural substantiation, damage tolerance and maintenance practices, materials control and standardization, and advanced material forms and processes (Ilicewicz, 2007). These activities provide the basis for continual updating of the safety and certification requirements for composite structures (Ilicewicz, 2007, 2009).

Looking more to the future, exploitation of the full potential of CFRP composites in the next generation of aircraft, i.e., operation at higher stress levels, may reveal difficult or even unacceptable structural integrity issues. There is also a current lack of full-scale fatigue testing programs directed at the composites themselves (rather than the metallic components) under realistic conditions. These conditions include environmental effects (temperature and humidity) on the composites properties.

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