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Milestone Case Histories in Aircraft Structural integrity

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Summary

This report has been prepared as a chapter in an introductory volume of the Elsevier Science treatise "Comprehensive Structural Integrity". The report discusses four milestone case histories in aircraft structural integrity, describing the causes of structural failure and the lessons learned. These case histories are the DeHavilland Comet crashes in 1954, the General Dynamics F-111 crash in 1969, the Dan Air Boeing 707 crash in 1977, and the Aloha Airlines Boeing 737 accident in 1988.

Contents

(25 pages in total)

1. INTRODUCTION

Aircraft structures are assembled mainly from metallic components, though efforts continue for increasing the use of advanced composites and laminates. Figures 1 and 2 give examples of 1990s forecasts of the materials to be used in military and civil airframes. Such forecasts tend to overestimate the rapidity with which new materials replace aluminium alloys. Recently, however, an important decision was made to use the glass fibre/aluminium laminate GLARE for sections of the new Airbus A380 fuselage (Beauclair, 2002).

Be that as it may, since the introduction of all-metal stressed-skin airframe structures in the 1930s the development of aircraft structural integrity has been concerned largely with the service behaviour of high strength metallic materials, particularly aluminium alloys. Broadly speaking, the history of this development 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.

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 tolerance design methods, which recognise that airframe structures must withstand service loads even when damaged and cracked. Safety to be ensured by testing and analysis of damaged structures, pre-service and in-service inspections, and eventual repairs and replacements.

Service failures have been greatly influential in this development. Four case histories are often cited (Schijve, 1994; Blom, 2002; McEvily, 2002) as milestones in the aircraft industry's approach to structural integrity, see table 1.

Table 1 Milestone case histories in aircraft structural integrity (Schijve, 1994)

year	*aircraft failure*	*influence, follow-up*
1954	DeHavilland Comet; two aircraft crashed owing to fuselage explosions.	General awareness of finite aircraft fatigue life as an important issue for passenger safety. Attention drawn to full-scale fatigue testing.
1969	F-111; wing failure due to undetected material flaw.	Aircraft should be damage tolerant. Fatigue cracking due to initial damage should be considered.
1977	Boeing 707; tailplane lost owing to fatigue failure in spar.	Old aircraft become more fatigue-critical, *geriatric aircraft*.
1988	Boeing 737; aircraft lost part of fuselage skin structure owing to multiple fatigue cracks in spar splices.	Multiple Site fatigue Damage (MSD) can occur in *ageing aircraft*, especially in lap joints of the pressurized structure.

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These case histories and their influences on aircraft structural integrity will be discussed in sections 2-5 of this chapter. Section 6 is a summary mentioning ongoing research and development.

2. THE DEHAVILLAND COMET CRASHES

2.1 Case Histories

The DeHavilland Comet was the first commercial jet transport, entering service in 1952. The aircraft's performance was much superior to that of contemporary propeller-driven transports. Apart from its speed the Comet was the first high-altitude passenger aircraft, with 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 1,231 pressurization cycles in service. It was tested in a water tank to minimise damage in the event of failure. After 1,825 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 3. 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 4.

The test aircraft was repaired and strain gauges applied to the *outside* surfaces of several escape hatches and windows. Results for the service and test failure locations are also shown in figures 3 and 4. 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 (Swift, 1987). However, a full-scale test effectively accounts 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 5 shows the basic pressure shell structure and the probable failure origin 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 5b, created a very high stress concentration at the first fastener. In the case of the probable failure origin for Comet G-ALYP the first fastener was a countersunk bolt, as shown in figure 5c. 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 4, combined with the stress concentrations provided by the frame cutout and knife-edge condition of the first fastener hole, figures 5b and 5c.

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 SAFE-LIFE. This means that the entire structure 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 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 firstly – 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.

Verification of FAIL-SAFE design concepts requires much 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 yet required to do full-scale testing. Subsequent experience and knowledge has led to mandatory full-scale testing.

It is important to note here that not all structural 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 helicopters, and some military aircraft.

3. THE GENERAL DYNAMICS F-111 CRASH

3.1 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 United States Air Force (USAF) and others. The F-111 is an unusual aircraft: it is a variable geometry "swing-wing" fighter-bomber; and it uses high-strength steel in major airframe components, namely the wing carry-through box, wing pivot fittings, some of the centre fuselage longerons and the empennage carry-through structure (Buntin, 1977).

On December 22, 1969, just over a year after entering service, F-111 #94 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 3.5g, 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 6. 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 6, a limited amount of fatigue crack growth occurred in service before overload 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 programmes (Buntin, 1977). The overall concerns about structural integrity led to a fracture control programme for the critical steel parts in the airframe. The approach was – and is, unhappily for the sole remaining operator – an expensive one that requires aircraft to be periodically removed from service and the entire wing carry-through structure to be proof tested at -40 °C. Details of the proof test and associated fracture mechanics analyses are given by Buntin (1977).

3.2 Lessons Learned

The cold proof test is a specific solution to safe operation of the F-111. However, the loss of F-111 #94, together with early and widespread fatigue cracking in the Lockheed C5-A wing boxes (Mar, 1991), led the USAF to reconsider and abandon its previous policy, which was essentially a SAFE-LIFE approach verified by full-scale fatigue testing to several lifetimes.

After much research the USAF provided and mandated new guidelines to ensure aircraft structural integrity. These guidelines became known as the DAMAGE TOLERANCE philosophy, incorporated in Military Specification 83444 (1974). This approach differs from the original FAIL-SAFE design principles, developed for commercial transport aircraft after the Comet crashes, in two major respects:

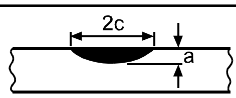
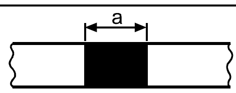
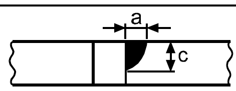
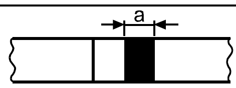
(1) The possibility of cracks or flaws in a new structure must be considered. In fact, Military Specification 83444 makes it mandatory to assume initial damage.

(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:
   – inspectable structures can be qualified as fail-safe or slow flaw growth structures, for which initial damage must grow slowly and not reach a size large enough to cause failure between inspections;
   – non-inspectable structures may still be classified as damage tolerant provided they can be qualified for slow flaw growth, which in this case means that initial damage must not grow to a size causing failure during the design service life.

While the USAF DAMAGE TOLERANCE approach has been effective in ensuring structural safety, it is by no means the last word on designing for aircraft structural integrity. Some general comments on the above two points will be made here.

<u>Initial damage.</u> Table 2 shows the Military Specification 83444 initial damage assumptions for ensuring *safety:*

Table 2 USAF MIL-A-83444 safety requirements for assumed initial damage

types of flaw		***aspect ratio (a/c)***	***flaw size a (mm) to be assumed immediately after inspection***		
description	**geometry**		**pre-service inspection with high standard NDI**		**in-service inspection with special NDI**
			fail-safe	slow flaw growth	
surface flaw		1.0 0.2	1.27	3.18	6.35
through crack			2.54	6.35	12.7
corner flaw at a hole		1.0 0.2	0.51	1.27	6.35 mm beyond fastener head or nut
through crack at a hole			0.51	1.27	6.35 mm beyond fastener head or nut

Although the initial flaw geometries in table 2 are rather arbitrary, their sizes are large enough for fracture mechanics calculations of fatigue crack growth using models based on well-established macrocrack growth behaviour: fracture mechanics is a cornerstone of the DAMAGE TOLERANCE philosophy.

However, Military Specification 83444 also provided guidelines for obtaining initial flaw sizes (usually called Equivalent Initial Flaw Sizes, EIFS) to be used in quantifying the structural *durability*. (At the time, the issue of durability was seen as an economic problem only. Nowadays it is also linked to safety problems in older aircraft, see section 5 of this chapter.) 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 Military Specification 83444 durability requirements concern initial flaw sizes well below 0.5 mm, in the so-called *short crack regime*. The behaviour 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 microstructure (Wanhill, 1986). This means that analytical modelling of short crack growth is problematical.

Another, more fundamental, aspect of durability is that it need not be determined by the immediate and continuous growth of small initial flaws. For example, an extensive investigation of fatigue cracking in pressure cabin lap splices from service aircraft and full-scale test articles showed there were significant initiation periods, up to 75 % of total life, before commencement of a regular process of fatigue crack growth (Wanhill *et al*., 2001).

<u>Non-inspectable structures.</u> 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 (Swift, 1983), which means they are undesirable in terms of safety *and* economics: the guaranteed service lives would be uneconomically short compared to FAIL-SAFE structures, see the earlier remarks in subsection 2.2.

4. THE DAN AIR BOEING 707 CRASH

4.1 Case History

On May 14, 1977, a Dan Air Boeing 707-321C airfreighter 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 in the upper chord of the rear spar of the right-hand horizontal stabilizer, figure 7. Fatigue cracking began at a fastener hole owing to higher loads than those anticipated in the design. The fatigue spread into the upper chord, with overall crack growth being accelerated by large intermittent tensile crack jumps. Fatigue crack growth finally gave way to overload 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 7 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 of this chapter 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) There should be adequate time for inspection when the damage reaches a size detectable by the specified inspection method.

Both these aspects were concerned in the accident. Firstly, periodic inspection of the horizontal stabilizer had a recommended 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. Secondly, 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 inadequacy of the inspection method.

4.2 Lessons Learned

The most immediate lesson from the Dan Air Boeing 707 crash is that a FAIL-SAFE *design concept* does not by itself constitute a FAIL-SAFE design. Inspectability is equally important, as discussed above.

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

A more specific lesson is worth noting. 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 aluminium 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).

5. THE ALOHA AIRLINES BOEING 737 ACCIDENT

5.1 Case history

On April 28, 1988, Aloha Airlines flight 243, a Boeing 737-200, 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 floorline separated from the aircraft, see the photograph in figure 8. 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 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 access to warm, humid, 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 8. This type of failure is called Multiple Site fatigue Damage (MSD). Somewhat poignantly, Swift discussed the then *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:

(1) Skin splice configuration. The pressure cabin longitudinal skin splice had been cold bonded, using an epoxy-impregnated woven scrim cloth, see figure 8, 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.

(2) Cold bonding production difficulties. The early service history of production 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 flight 243 but well before the accident.

(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, issuing documents is one thing, living up to them is quite another, see below.

The way these 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 though the rivets. These had countersunk heads causing knife-edges in the upper skin, see figure 8, 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. This they did without hindrance by the disbonded tear straps. In other word, 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.

5.2 Lessons Learned

The Aloha Airlines Boeing 737 accident prompted worldwide activities to ensure the safety and structural integrity of ageing 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 organising several ageing 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, though 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.

Widespread Fatigue Damage (WFD). There are two types of WFD: Multiple Site fatigue Damage (MSD) – as in the Aloha Airlines Boeing 737 – 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 longstanding 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 ageing aircraft inspections, this "traditional" practice is regarded less favourably, though it is still a potential option.

Corrosion. The Aloha Airlines Boeing 737 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 (Akdeniz, 2001).

Corrosion control programmes have been set up for commercial transport aircraft and ageing military aircraft (Paone, 1993; Nieser, 1993; Akdeniz, 2001). These programmes 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 programmes is assessed from the "levels" of corrosion found during inspections. These levels are 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
- level 3 : immediate airworthiness concern.

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

6. SUMMARY

The four case histories discussed in this chapter are often considered to be milestones in the development of aircraft structural integrity. Lessons learned from these accidents, and others, have greatly influenced and improved our knowledge and perception of the problems involved in ensuring safety and durability.

Ongoing research and development aims to improve structural analysis capabilities and the methods for fatigue life and crack growth prediction. The combined effects of fatigue and corrosion are also receiving much attention. Efforts to increase the use of advanced composites and laminates, particularly by replacing all-metal structures in civil airframes, are providing new challenges and rethinking of well-proven design principles and methods.

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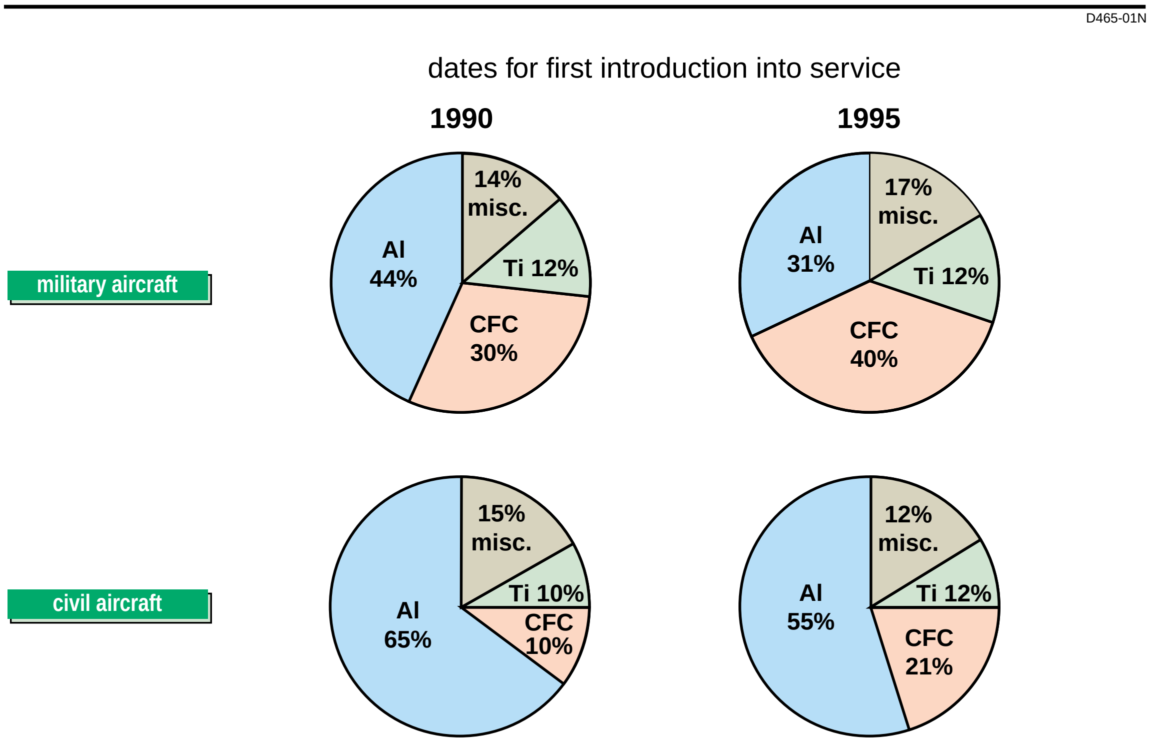

Fig. 1 A mid-1990s forecast for airframe materials [1].
CFC = Carbon Fibre Composites (also known as GRP = Graphite Reinforced Plastics)

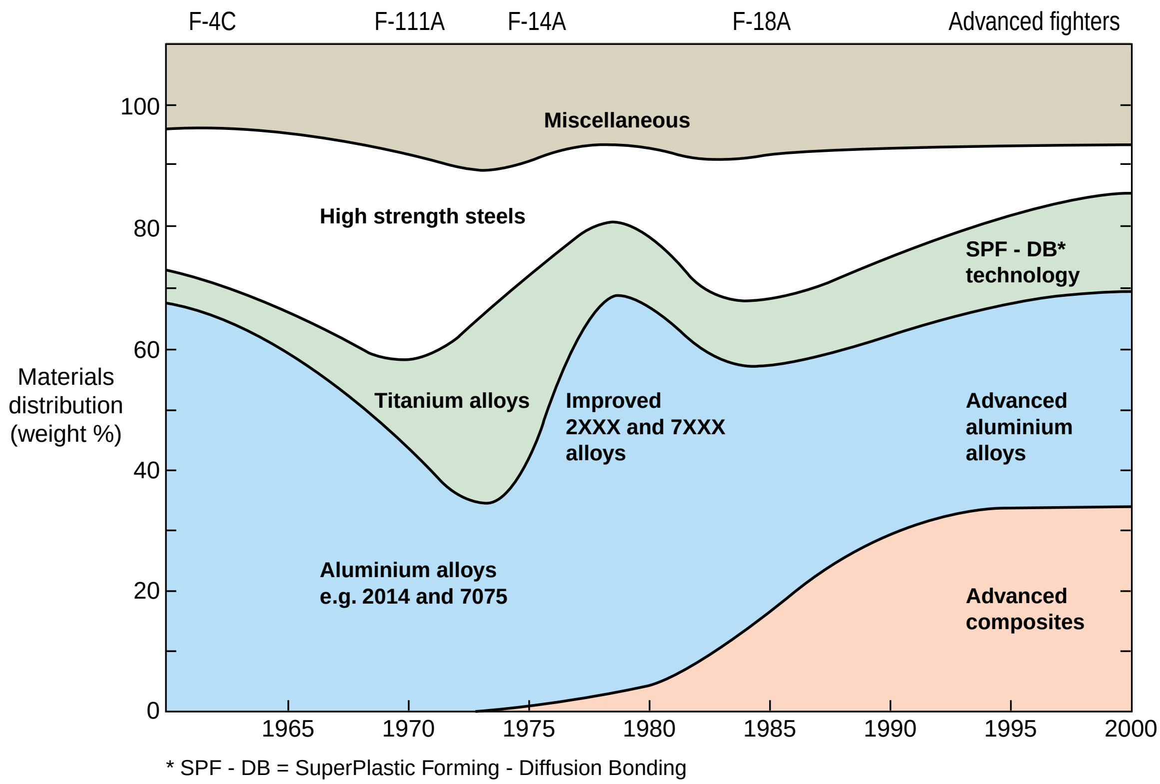

Fig. 2 A 1992 forecast for military airframe materials [2]

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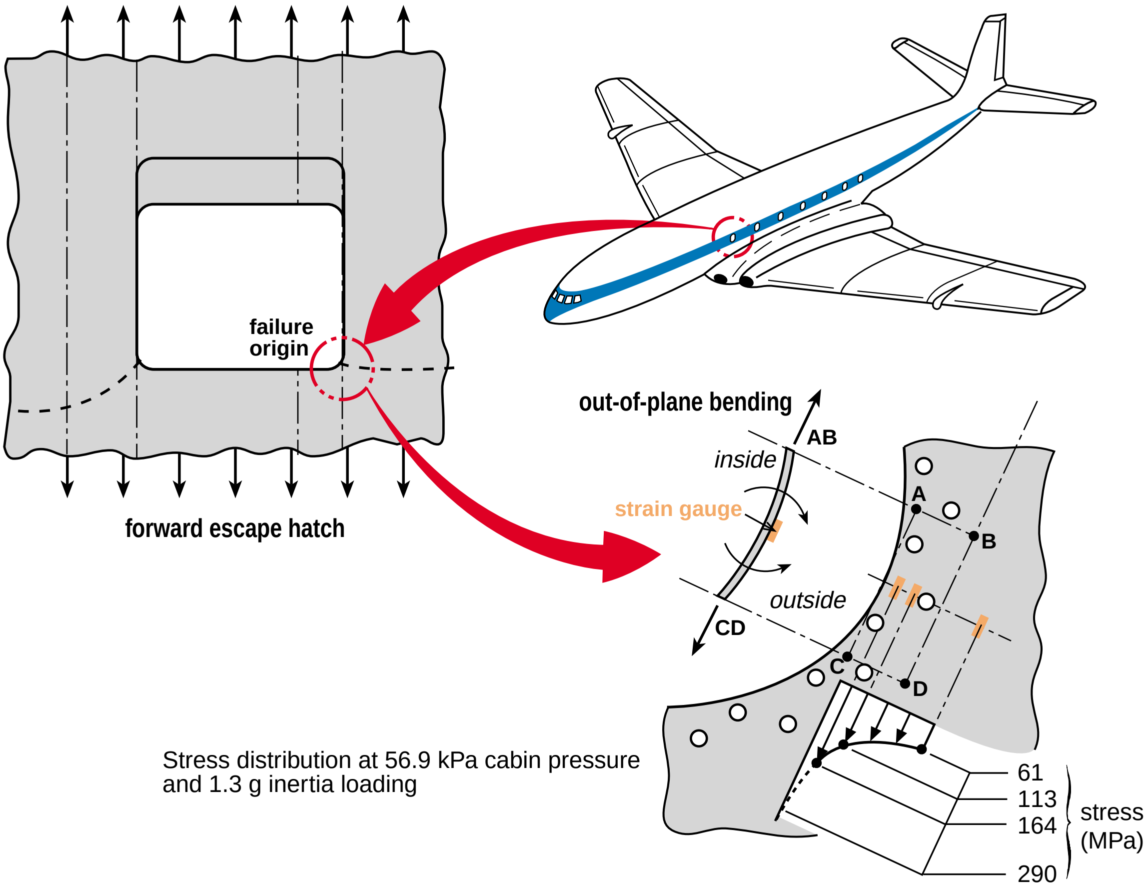

Fig. 3 Probable failure origin of test aircraft Comet G-ALYU: stress distribution obtained after repair (Swift, 1987)

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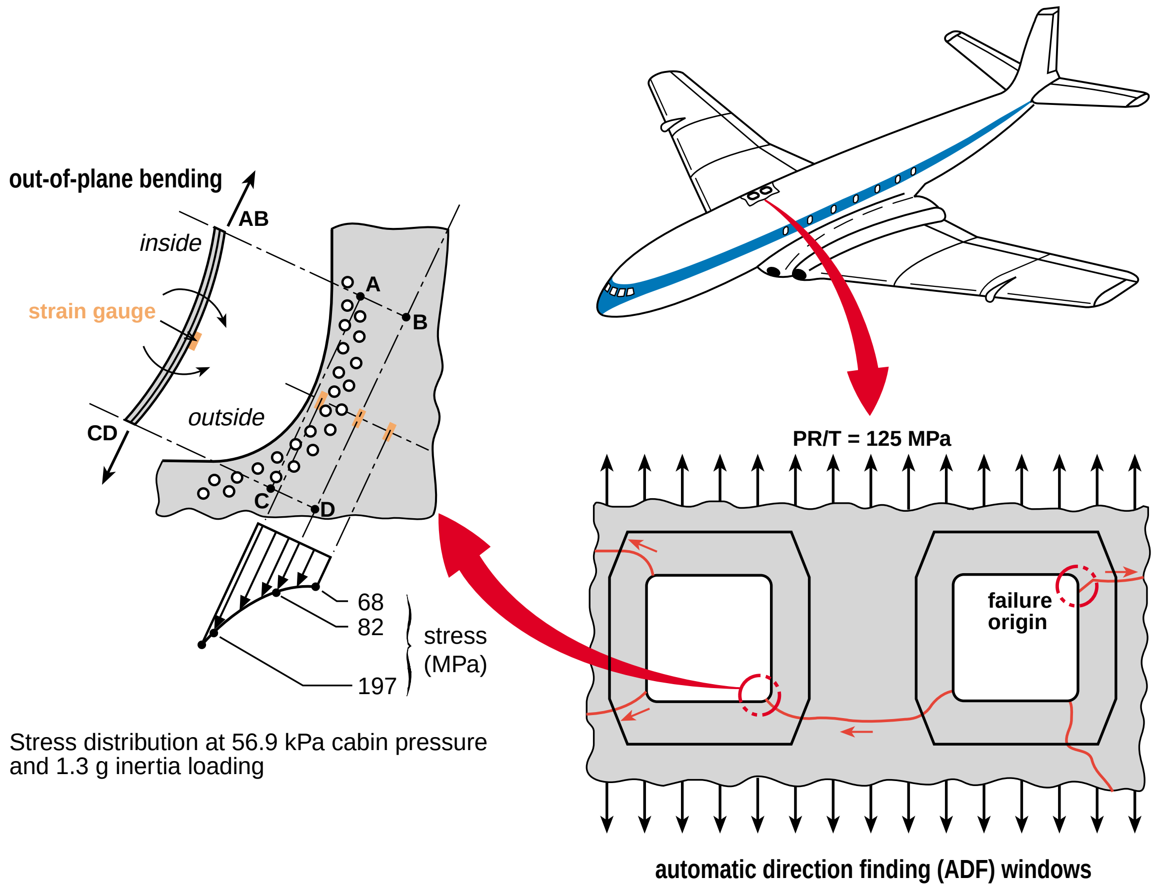

Fig. 4 Probable failure origin of service aircraft Comet G-ALYP: stress distribution obtained from repaired test aircraft, Comet G-ALYU (Swift, 1987)

D465-01N

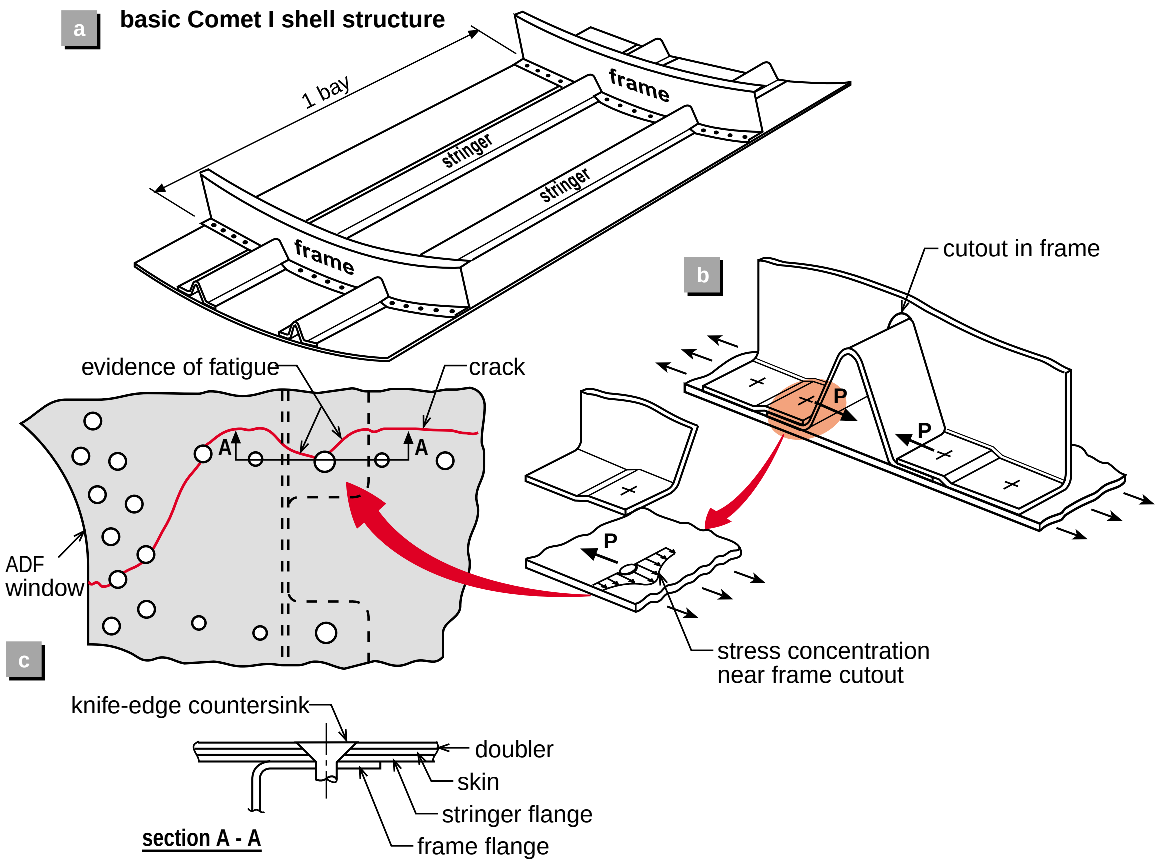

Fig. 5 Details of the probable failure origin of service aircraft Comet G-ALYP (Swift, 1987)

D465-01N

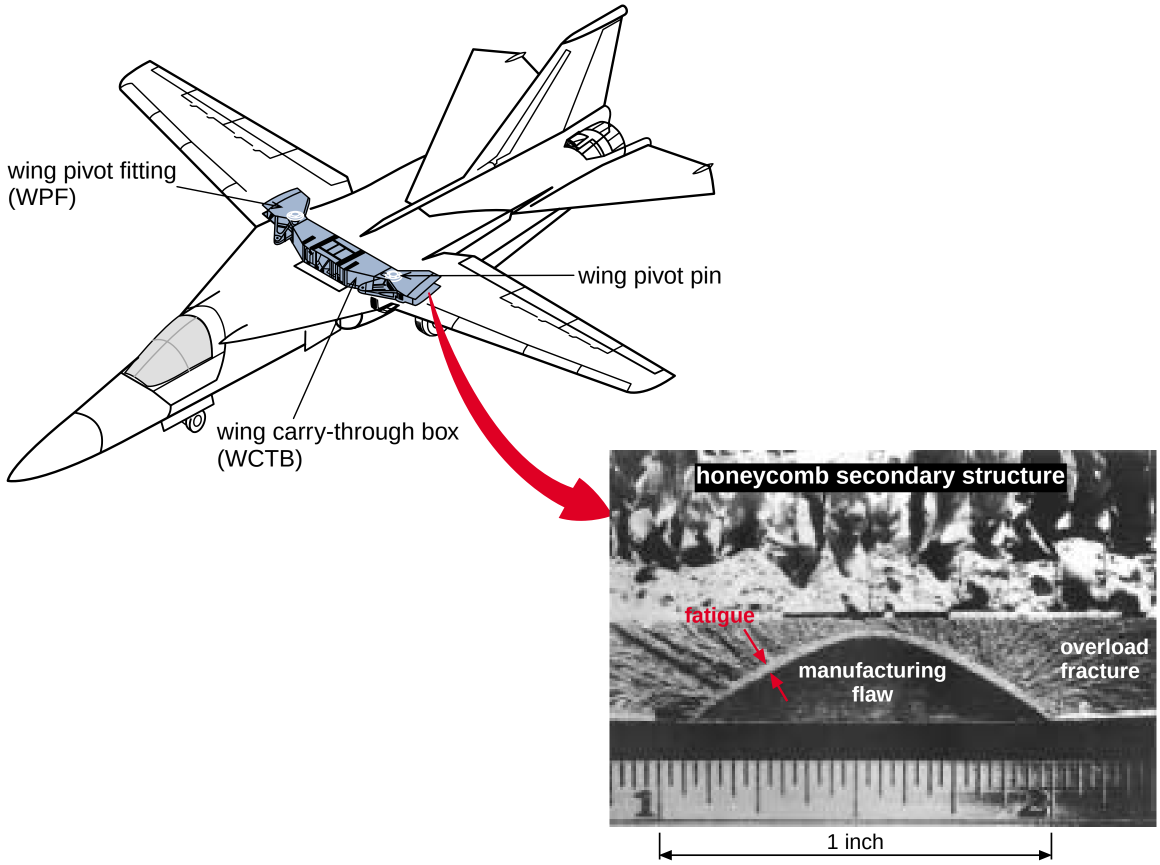

Fig. 6 Failure origin of F-111 #94: a manufacturing flaw in the high-strength steel lower plate of the left-hand wing pivot fitting

D465-01N

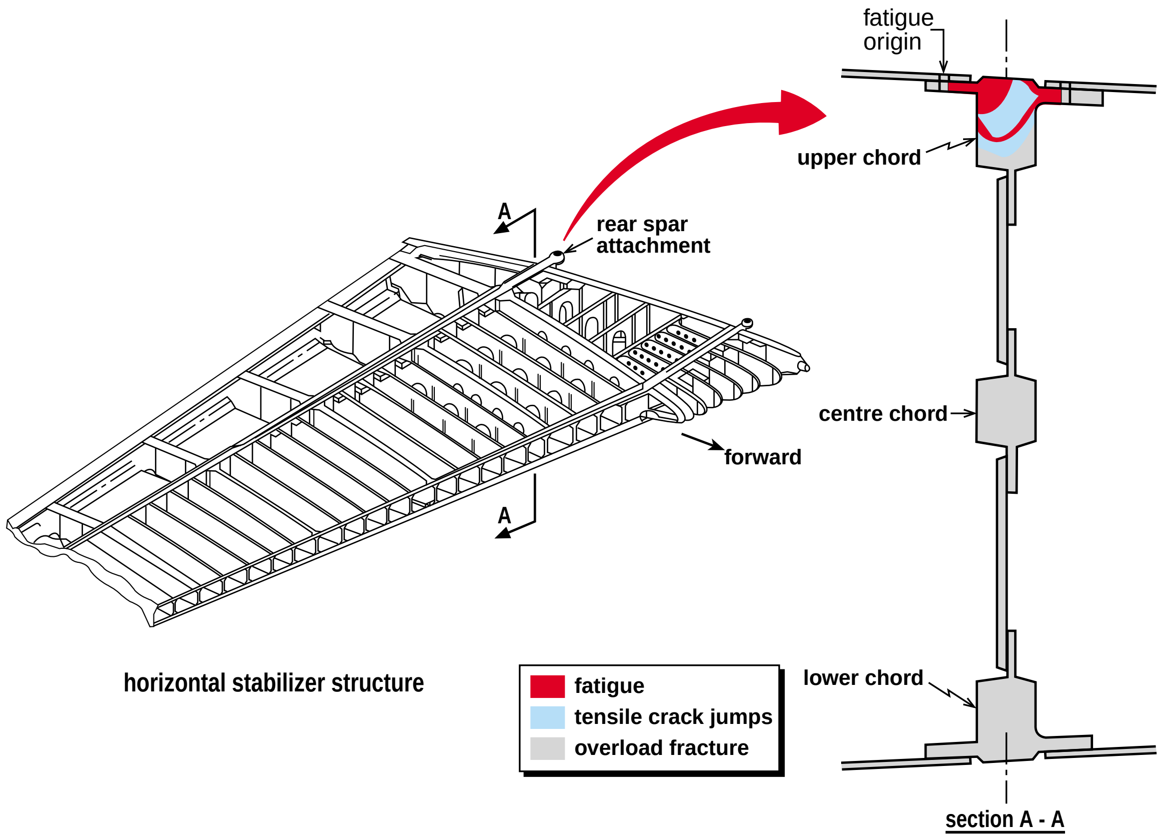

Fig. 7 Failure origin of the Dan Air Boeing 707. After Howard (1986)

D465-01N

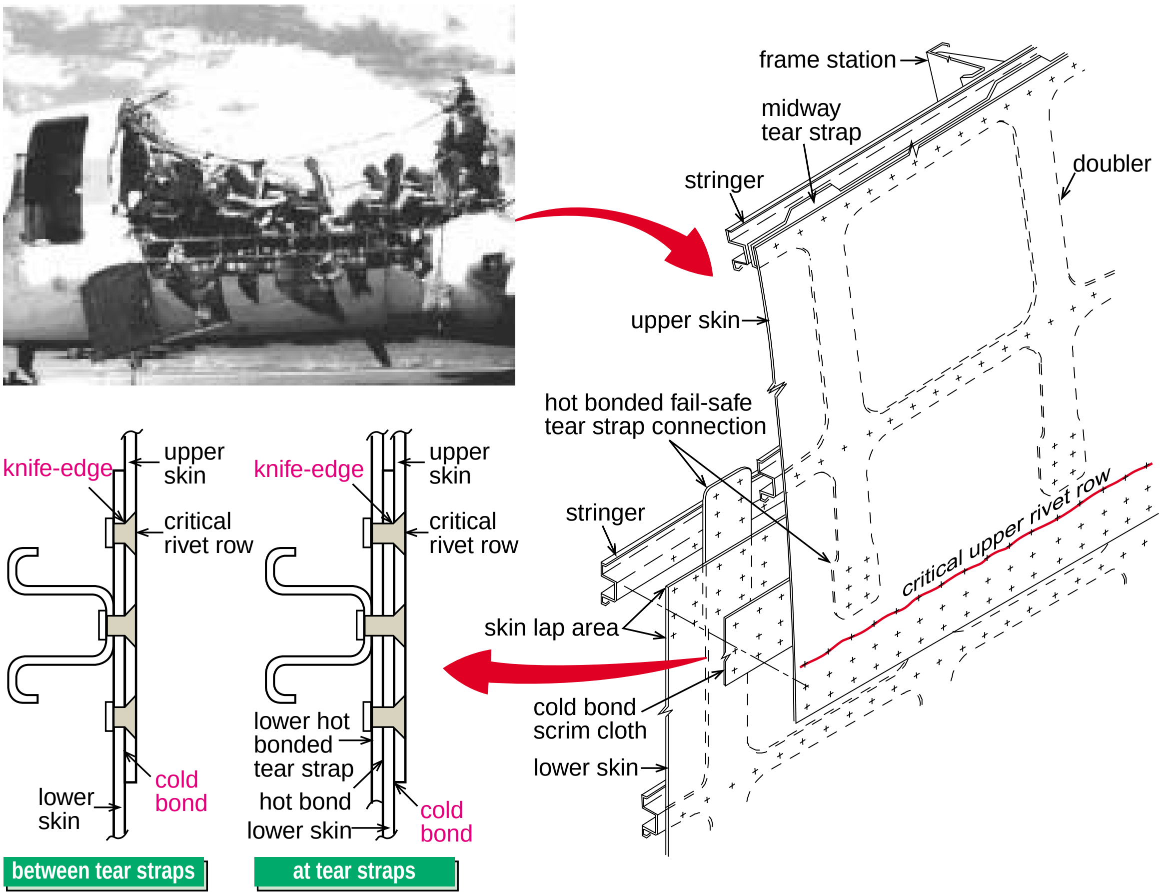

Fig. 8 Structural aspects of the Aloha Airlines Boeing 737 accident: Multiple Site fatigue Damage (MSD) occurred in the outer (upper) skin, commencing from the knife-edges of the rivet holes along the upper rivet row