İSTANBUL TECHNICAL UNIVERSITY ★ INSTITUTE OF SCIENCE AND TECHNOLOGY

DAMAGE TOLERANCE ASSESSMENTS OF COMMERCIAL JET TRANSPORT STRUCTURES

M.Sc. Thesis by
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TİCARİ JET UÇAK YAPILARININ HASAR TOLERANS DEĞERLENDİRMELERİ

YÜKSEK LİSANS TEZİ

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FOREWORD

This thesis is dedicated to my beloved mother. Mom, Rest in Peace!
I would like to acknowledge my supervisor Mr. Gökhan İnalhan for his excellent guidance and advice during the project.
Special thanks are truly deserved by my friend, Sevin Okyay, for her excellence in manuscript editing.
Special appreciation goes to my family, my cat “Pati”, my friend Özgür Gürbüz and my company Pegasus Airlines.

December, 2009

Barış KAYA
Aeronautical Engineer

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ABBREVIATIONS

AATF	: Airworthiness Assurance Task Force
AAWG	: Airworthiness Assurance Working Group
A/C	: Aircraft
AC	: Advisory Circular
AD	: Airworthiness Directiveness
ADF	: Automatic Direction Finder
ATA	: Air Transport Association
B.O.A:C.	: British Overseas Airways Corporation
BZI	: Base Zonal Inspection
CP	: Cathodic Protection
CPAC	: Corrosion Prevention and Control
CPC	: Corrosion Preventive Compounds
CPCM	: Corrosion Prevention and Control Manuals
DGAC	: Direction Générale de l'Aviation Civile
DSG	: Design Service Goal
DSO	: Design Service Objective
EIFS	: Equivalent Initial Flaw Size
FAA	: Federal Aviation Admininstration
FC	: Flight Cycle
FH	: Flight Hour
ICCP	: Immersed Current Cathodic Protection
JAR	: Joint Aviation Requirements
LM	: Lochead Martin
MED	: Multiple Element Damage
MSD	: Multiple Side Damage
MS	: Margin of Safety
NDI	: Non Destructive Inspections
NTSB	: National Transportation Safety Board
OEM	: Original Equipment Manufacturar
POD	: Probabilty of Detection
PSE	: Primary Structural Element
RAG	: Repair Assessments Guidelines
RAM	: Repair, Alternations and Modification
SAB	: Special Attention Service Bulletin
SB	: Service Bulletin
SID	: Supplemental Inspection Documents
SRM	: Structural Repair Manual
SSI	: Structural Significant Item
STC	: Supplemental Type Certificate
TC	: Type Certificate
WFD	: Widespread Fatigue Damage

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DAMAGE TOLERANCE ASSESSMENTS OF COMMERCIAL JET TRANSPORT STRUCTURES

SUMMARY

Design of safe and competitive commercial aircraft structures involves a host of significant considerations. Some accidents were milestones along the road. New concepts has been proposed related to structural design, materials, production techniques, inspection procedures and load spectra. The present state of the art has been affected by various conditions associated with interests of the aircraft industry, the aircraft operator and the airworthiness authorities. The current trends in aircraft operations are showing an increasing demand for lower operational and maintenance costs. Practically, this translates into aircraft with longer design lives with longer inspection intervals and shorter inspection downtimes and fleets that are operated beyond intended design life. As a consequence, the damage tolerance aspects of primary aircraft structures are becoming highly important, due to the tighter design requirements for new aircraft on the design table. Damage tolerance assessment and design play a major role in the aerospace industry not only in the design of new structures and components but also their ongoing maintenance and support.

This study is focused on general overview and literature review of damage tolerance assessments and design philosophy, better understanding of design principles (static, safe life, fail safe and damage tolerance), current airworthiness regulations for damage tolerance evaluation of structures, continued airworthiness challenges in the industry; corrosion prevention and control programs, widespread fatigue, repair assessment process and guidelines and supplemental inspection programs.

TİCARİ JET UÇAK YAPILARININ HASAR TOLERANS DEĞERLENDİRMELERİ

ÖZET

Güvenli ve rekabetçi ticari uçak yapıları tasarımı, pek çok önemli faktörü içerir. Kimi kazalar, yol boyunca kilometre taşları (dönüm noktası) olmuştur. Yapısal tasarım, malzeme, üretim teknikleri, kontrol prosedürleri ve yük spektralarına ilişkin olarak yeni kavramlar önerilmektedir. Günümüzdeki gelişme seviyesi, uçak sanayiinin çıkarlarına, uçak operatörüne ve uçuşaelverişlilik otoritelerine bağlı çeşitli durumlardan etkilenmiştir. Uçak operasyonlarındaki şimdiki eğilimler, daha düşük işletme ve bakım masraflarına yönelik artan bir talebi gösteriyor. Pratikte bu; daha uzun tasarım ömürleri, daha uzun kontol aralıkları ve daha kısa kontol arıza süresi olan uçaklar anlamına gelir. Bunun bir sonucu olarak da, birincil uçak yapılarının hasar tolerans analizleri/değerlendirmeleri, tasarım masasında yeni uçaklar için daha sıkı gereksinimler olması nedeniyle, çok önem kazanmaktadır. Hasar tolerans değerlendirmesi ve hasar tolerans tasarımı, hava-uzay sanayiinde, yalnızca yeni yapılar ve parçaların tasarımı açısından değil, onların süregelen bakımı ve desteği açısından da bellibaşlı bir rol oynar.

Bu çalışma, hasar tolerans değerlendirmeleri ve tasarım felsefesine hem üretici hem de operatör açısından genel bir değerlendirme üzerinde odaklanmıştır. Havacılık endüstrisinde dönüm noktası sayılan kazarlardan öğrenilen dersler, günümüz havacılık sanayiinde kullanılmakta olan tasarım ilkeleri (statik, güvenli ömür, arıza emniyeti ve hasar toleransı), yapıların hasar tolerans değerlendirmesi için varolan uçuşaelverişlilik mevzuatı, yaşlanan uçak meselesi, yaygın yorgunluk hasarı, sanayideki süregelen uçuşa elverişlilik sorunları, korozyon önlenme ve kontrolü, onarımların/tamirlerin değerlendirme teknikleri, değerlendirme süreci ve ana esaslarının daha iyi anlaşılmasını amaçlar.

1. INTRODUCTION

Operations economy demands that structures can be operated safely throughout the expected service life. Design of structures is fundamentally a guided interactive process aimed at achieving a practical balance between state-of-the-art structural capability and the intended usage requirements. These capabilities and requirements are typically evaluated against each other through a disciplined design process comprising regulations, methods and analysis, data bases, validation tests etc. Modern airplanes operate in a complex combination of external load sources, environments, human elements and economic requirements. The structure must be designed so that it can satisfy, with a sufficient confidence level, the requirements of the service life. Also, today's economic reality dictates that a fleet must be operated beyond intended design life. The estimated material properties must be evaluated as precisely as possible within all the environmental conditions to avoid failures. For this reason the aircraft structural integrity programs are put in place to provide an integrated and systematic approach to aircraft structural engineering and maintenance. Following chapters are focused on examples of lessons learned from commercial jet transport accidents, historical perspective and details of design philosophies, general overview of damage tolerant design principle and its elements and continued airworthiness challenges in the industry from a damage tolerance point of view.

2. LESSONS LEARNED FROM ACCIDENTS

Many of the lessons learned from tragedies are timeless, and are relevant to today's aviation community. By learning from the past, aviation professionals can use that knowledge to recognize key factors, and potentially prevent another accident from occurring under similar circumstances, or for similar reasons, in the future.

2.1 BOAC Flight 781, de Havilland DH-106 Comet 1, G-ALYP and South African Airways (SAA) Flight 201, de Havilland DH-106 Comet 1, G-ALYY

The de Havilland Comet was the world's first commercial jet airliner to reach production. Developed and manufactured by de Havilland, The design was of the Comet aircraft commenced in September, 1946 it first flew on July 27, 1949 and was considered a landmark in British aeronautical design. The first production aircraft (G-ALYP) flew on January 9, 1951 and it was granted on March 22, 1952. The aircraft was delivered to B.O.A.C. on March 13, 1952 and entered into scheduled passenger service on May 2, 1952, after having accumulated 339 hours. 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 [1].

Within two years after service, on January 10, 1954, a Comet I aircraft (DH 106-1) serial number G-ALYP known as Yoke Peter disintegrated in the air at approximately 30,000 feet and crashed into the Mediterranean Sea off Elba. The aircraft was on a flight from Rome to London. At the time of the crash the aircraft had flown 3680 hours and experienced 1286 pressurized flights. The Comets were removed from service on January 11, 1954. A number of modifications were made to the fleet to rectify some of the items which were thought to have caused the accident and service was resumed on March 23, 1954.On April 8, 1954, only sixteen days after the resumption of service, another Comet aircraft G-ALYY known as Yoke Yoke disintegrated in the air at 35,000 feet and crashed into the ocean near Naples. The aircraft was on a flight from Rome to Cairo. At the time of the crash the aircraft had flown 2703 hours and experienced 903 pressurized flights. Prior to these two accidents, on May 2, 1953, another Comet, G-ALYV had crashed in a tropical storm of exceptional severity near Calcutta. An inquiry, directed by the Central Government of India, determined that this accident was caused by structural failure which resulted from either:

a) Severe gusts encountered during a thunderstorm.

b) Overcontrolling or loss of control by the pilot when flying through a thunderstorm.

After the Naples crash on April 8, 1954, B.O.AC. immediately suspended all services. On April 12, 1954, the Chairman of the Airworthiness Review Board withdrew the certificate of airworthiness. The UK Minister of Supply instructed Sir Arnold Hall, Director of the Royal Aeronautical Establishment, to complete an investigation into the cause of the accidents. On April 18, 1954, Sir Arnold decided that a repeated loading test of the pressure cabin was needed. It was decided to conduct the test in a tank under water. In June 1954, the test started on aircraft G-ALYU, known as Yoke Uncle. The aircraft had accumulated 1230 pressurized flights prior to the test. After 1830 further pressurizations, for a total of 3060, the pressure cabin failed. The starting point of the failure was at the comer of a passenger window. The cabin cyclic pressure was 8.25 psi but a proof cycle of 1.33P was applied at approximately 1,000 pressure cycle intervals. It was during the application of one of these cycles that the cabin failed. Examination of the failure provided evidence of fatigue. Further investigation of Yoke Peter on structure recovered near Elba also confirmed that the primary cause of the failure was pressure cabin failure due to fatigue. The origin in this case was at the corner of the Automatic Direction Finding (ADF) windows on the top centerline of the cabin. Yoke Uncle was repaired and the fuselage skin was strain gauged near the window corners. The peak stresses measured were 43,100 psi for 8.25 psi cabin pressure plus 650 psi for Ig flight and 1950 psi for a 10 ft/sec gust for a total of 45,700 psi. The material was DTD 546 having an ultimate strength of 65,000 psi. Therefore, the IP + Ig stress was 70% of the material ultimate strength. Thus, the cause of the failures was determined to be fatigue due to high stresses at the window comers in the pressure cabin. This investigation resulted in considerable attention to detail design in all future pressure cabins and demonstrated the need for full-scale fuselage fatigue tests [2, 3].

Swift [1], described the Comet pressure cabin structure in more detail, in order to bring out some further important aspects of the service failures. Figure 2.1 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 2.2b, 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 2.2c. 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 2.1, combined with the stress concentrations provided by the frame cutout and knife-edge condition of the first fastener hole, Figures 2.2b and 2.2c. 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 [1] 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.

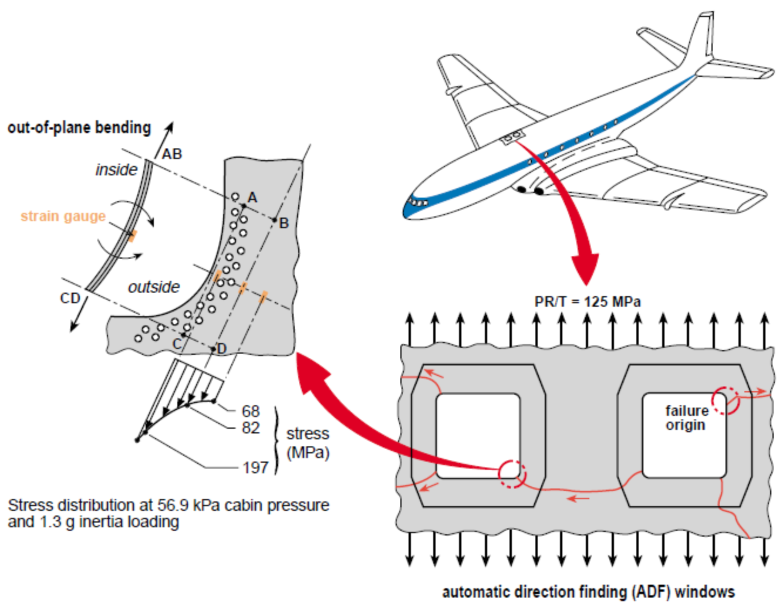

Figure 2.1: Probable Failure Origion of Comet G-ALYP

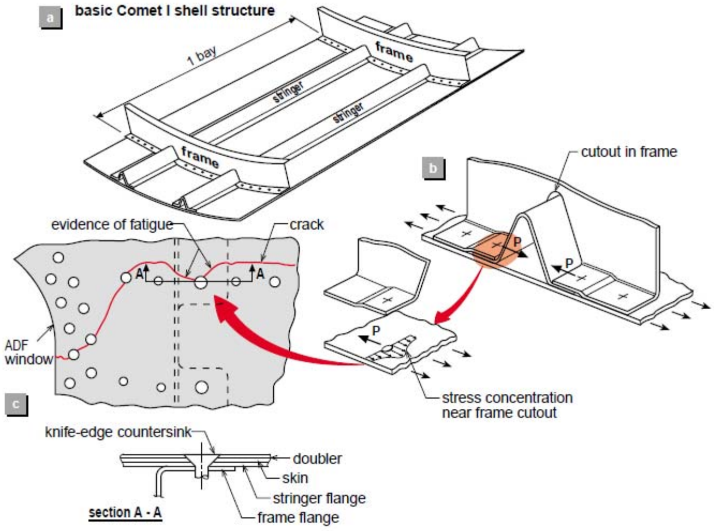

Figure 2.2: Details of the Probable Failure Origin of Comet G-ALYP

2.1.1. Lesson learned

In Comet accident, the fatigue properties of the production airplanes were not well understood, and test results were misleading regarding the airplane's actual fatigue life. The performance of test specimens must be representative of the production airplane, and produce results that are conservative relative to the expected usage and environment. De Havilland's fatigue test setup did not reflect the production fleet. The test specimen had been previously cold worked due to overloaded proof pressure testing. This overload inadvertently enhanced the fatigue life of the specimen, and thus produced results which did not reflect the production configuration. Following the accidents, it was learned that the original test fuselage had been subjected to both the overpressure tests and subsequently, to the fatigue cycle testing. These two tests conducted on the same test article inadvertently changed the material properties in high stress areas of the fuselage (i.e., window corners). Production airplanes, which had not been subjected to repeated overpressure cycling, were found to form fatigue cracks near the corners of the windows at approximately 1000 airplane flight cycles, while the original test specimen was subjected to 16,000 cycles before the first crack was observed. 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. [4]

2.2 Dan-Air Services, Ltd, 707-300, G-BEBP

On May 14, 1977, Dan-Air (G-BEBP) on a non-scheduled international cargo flight 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. G-BEBP was the first aircraft off the 707-300C series convertible passenger/freighter production line [5]. The crash led to the striking but unflattering term “geriatric jet”.

Examination of the detached stabilizer revealed evidence of a fatigue failure of the top chord of the rear spar, initiating at the 11th fastener hole, which is used by both the rear spar upper chord and upper skin structure. The location of the fastener was 14.25 inches outboard of the attachment of the stabilizer attachment pin. The cracking progressed in fatigue over approximately 60% of the chord, and then began a series of several tensile jumps, separated by small periods of fatigue.

The total number of flights between the initiation of the fatigue crack and the final failure of the upper chord was estimated by the investigators to have been approximately 7,200 flights, with 3,500 of the flights being the duration to grow the crack across the exposed surface of the top chord. Failure of the top chord was followed by fracture of the upper web, center chord, lower web and lower chord, leading to loss of the stabilizer and loss of control of the aircraft.

The investigation discovered no unique feature leading to the cracking at the 11th fastener hole other than high stresses existing in the entire inboard area of the rear spar. The location of the 11th fastener hole is indicated by an arrow in Figure 2.3 [4].

Figure 2.4 shows that the rear spar consisted of discrete elements. These were linked together by fasteners.

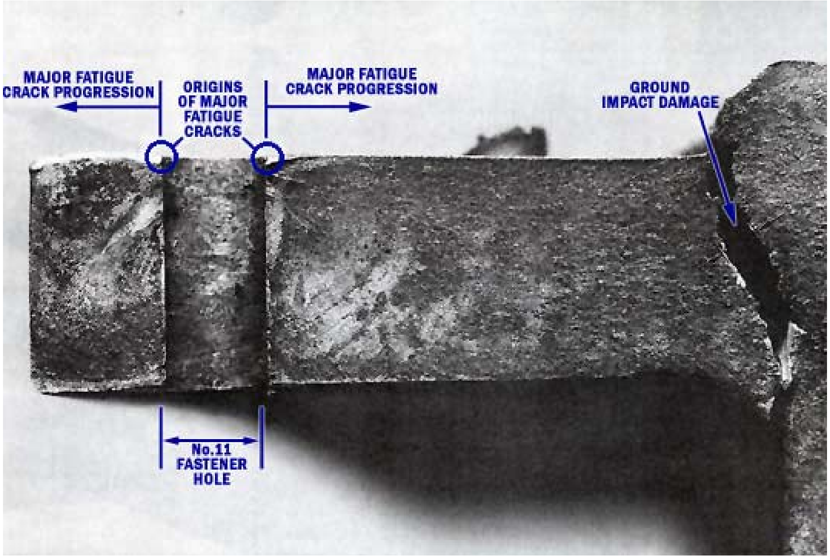

Figure 2.3: Photo of Fracture Face Emanating from the 11th Fastener

This configuration was intended to be a fail safe design and 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 that sustainable significant damage should be large enough to be found by the specified inspection method and there should be adequate time for inspection when the damage reaches a size detectable by the specified inspection method.

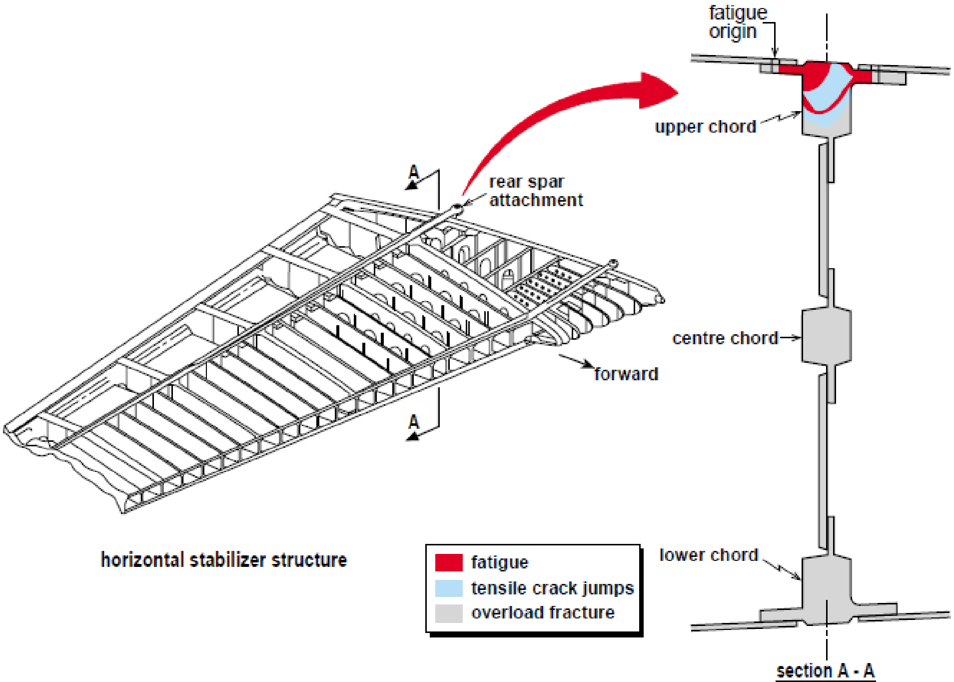

Figure 2.4: Fatigue Origin of the Dan Air Boeing 707

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 Thus although the manufacturer had designed the horizontal stabilizer to be failsafe, in practice it was not, owing to the inadequacy of the inspection method.

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 [5, 6].

2.2.1. Lesson learned

The horizontal stabilizer of the 707-300 was a "scaled-up" design of the 707-100/200. The structural characteristics of the two designs were assumed to be similar, and extrapolations of assumptions to the new design were considered valid. Results from the fatigue tests conducted on the 707-100/200 were used to validate the assumptions on the 707-300 design, and new testing was not conducted. The accident demonstrated that the basic similarity assumptions were incorrect, and that testing to validate those assumptions should have been conducted in order to properly understand the changes to structural properties inherent in the new design. The Dan-Air accident is considered by many to be the final catalyst for the issuance of regulations requiring establishment of damage tolerance based inspections, whenever practical, for transport category aircraft. At the time of the accident, there were on-going discussions within the aircraft industry regarding the need for an alternative to fail-safety for protection against fatigue in older aircraft and for new type designs. [4]

2.3 Aloha Airlines Flight 243

On April 28, 1988, a Boeing 737-297 serving the flight suffered extensive damage after an explosive decompression during climb out at cruise altitude, but was amazingly, able to land safely at Kahului Airport on Maui. About 18 foot /5.5 m long section of the upper fuselage and supporting structure aft of the cabin entrance door and above the passenger floorline suddenly departed the aircraft sweeping a flight attendant overboard, Figure 2.5. The accident airplane, N73711, a Boeing 737-297, serial number 20209, was manufactured in 1969 as production line number 152 and had since accumulated 89,680 flight-cycles and 35,496 flight-hours at the time of the accident [7].

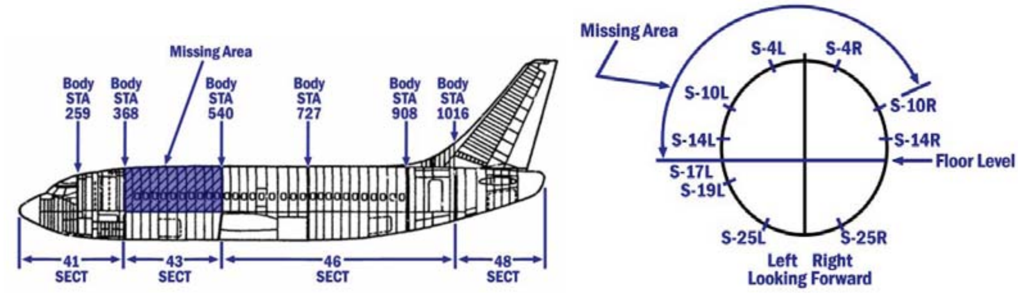

Figure 2.5: Illustration Showing the Missing Fuselage Section

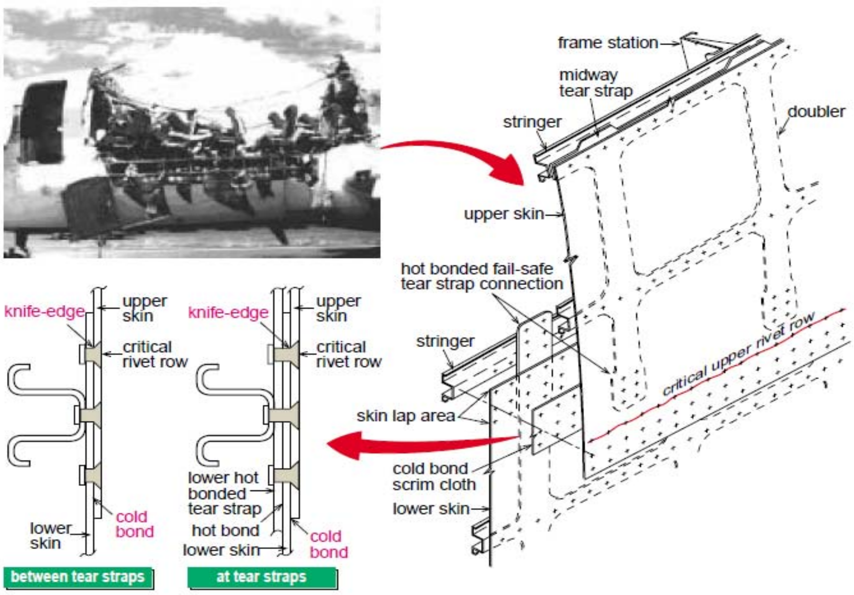

Figure 2.6: Structural Apects of Aloha Airlines Boeing 737 Accidents

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 Figure 2.6. 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 [1,6]. Major contributions to the accident can be summarized as follows [4]:

<u>Lap Splice (Joint) Design:</u> Adjacent fuselage panels are joined longitudinally by overlapping the edge of the skin of the upper panel about three inches over the edge of the skin of the lower panel. On the early 737s (up to line number 291), the overlapping skins were bonded together with an adhesive and were fastened with three rows of rivets. The fuselage pressurization (hoop) loads were intended to be transferred through the adhesive bond, rather than through the rivets. This design used a cold bond adhesive (a scrim cloth is impregnated with an adhesive that cures at room temperature and must be kept at dry ice temperature until shortly before its use to prevent premature curing). The cold bond process had manufacturing difficulties (surface preparation quality, condensation in the joint during assembly, and premature curing of the adhesive). These difficulties led to the random appearance of bonds with degraded adhesion, with susceptibility to corrosion, and with some areas that did not bond at all. Disbonded areas were then subject to in-service corrosion due to moisture wicking, which leads to further disbonding.

<u>Widesprad Fatigue (WFD):</u> Once disbonding of the lap splice occurs, the fuselage pressurization loads that were intended to be transferred by the adhesive bond are instead transferred by the rivets. Since the countersink for the rivet head went through the entire thickness of the upper skin (creating a knife edge), a higher than typical stress concentration resulted . The combination of effects from the high stress concentration, the rivet load transfer and the far field stress levels led to the development of fatigue cracks at many adjacent or neighboring rivet locations [Multi-Site Damage (MSD)].

An insidious feature of MSD is that many small, hard-to-detect cracks can link up rather suddenly to form a long, critical crack. The advanced stages of MSD, which occurred on this airplane, can result in Widespread Fatigue Damage (WFD), a condition where the airplane structure is no longer able to sustain the required residual strength loads.

Maintenance and Surveillance: The Aloha Airlines maintenance program used a D-check (heavy maintenance and inspection check) interval of 15,000 flight-hours, which appears acceptable compared to the 20,000 flight-hour interval recommended by manufacturer. However, due to the unusually short flights in the Aloha Airlines flight schedule, flight-cycles were accumulated at about twice the rate that Boeing considered when it produced its maintenance recommendations. In pressurized fuselage structure, the initiation of fatigue cracks and the subsequent rate of crack growth are predominated by the accumulation of flight-cycles, not flight-hours. This fact was not sufficiently regarded when the Aloha Airlines maintenance program was produced and then approved by the FAA (some maintenance tasks should have been more frequent). After the accident, visual inspection of the exterior of the airplanes in the Aloha Airlines 737 fleet was conducted. Swelling and bulging of skin, dished fastener heads, pulled or popped rivets, and blistering, scaling, and flaking paint were present at many sites along the lap joints of almost every airplane. According to the NTSB, Aloha Airlines did not produce evidence that it had in place specific severe operating environment corrosion detection and control programs as outlined in the manufacturer Corrosion Prevention Manual. The NTSB noted that "it appears that even when Aloha Airlines personnel observed corrosion in the lap joints and tear straps, the significance of the damage and its criticality to lap joint integrity, tear strap function, and overall airplane airworthiness was not recognized [7]. It was further noted that "the overall condition of the Aloha Airlines fleet indicated that pilots and line maintenance personnel came to accept the classic signs of on-going corrosion damage as a normal operating condition."

2.3.1 Lesson learned

The Aloha Airlines Boeing 737 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 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. 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 [6].

3. FOUR MAJOR DESIGN PHILOSOPHY

The modern aeronautical engineering of aircraft design has been an evolutionary process accelerated tremendously in recent times from the demanding requirements for safety and the pressures of competitive economics in structural design. The relatively short span during which aircrafts were designed and manufactured witnessed four different design and analysis philosophies [8], Figure 3.1:

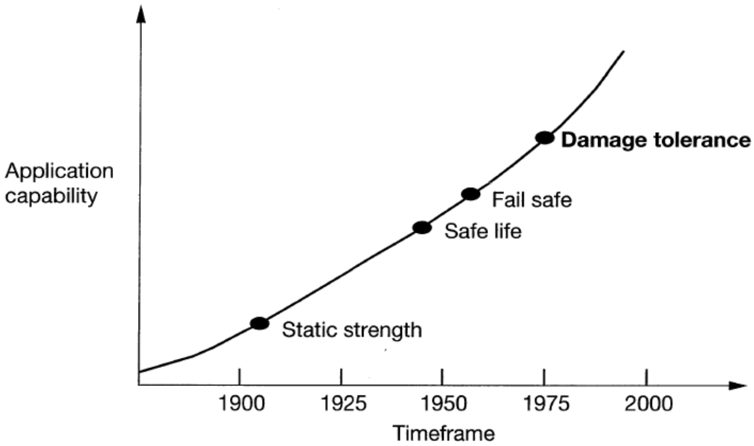

Figure 3.1: Timeframe for Four Major Design Philosophy

Static Strength Design (cca 1900-1950) was based on limiting the allowable stresses to some "safe" fraction of the static strength and /or fatigue limit. The damage attributed to accidental loads, corrosion and/or fatigue was summarily ignored. The design limits and limits loads inferred directly from test.

Safe Life Design (cca 1950-1960) was based on assessment of the finite fatigue life during which an initially flaw-free part develops a crack of critical size. Repeated load testing was performed on Comet jets and it was for the first time recognized that initial cracks are nucleated at fraction (less than a quarter) of the structural fatigue life.

Fail Safe Design (cca. 1960-1975) acknowledged the inevitability of the occurance and presence of fatigue cracks in aircrafts structures. The design emphasizes the redundancy (multiple load paths) of the structure to promote the stress redistribution away from a cracked component and avoid brittle failure.

Sufficent opportunity is allowed for a timely detection of the damage and the role of accidentally incurred damage and corrosion was the first time incorporated into analyses. The criteria for the inspection intervals and detection of cracks was prescribed eventhough the limits on the maximum risk were not expilicitly defined.

Damage Tolerant Design (cca. 1975 to present) assumes that fatigue cracks are inevitable. Fracture mechanics methods are applied to predict the number of load cycles leading fatigue failure in order to prescribe safe inspection intervals. Special attention is devoted to the determination of the residual strength of a damaged structure, rate of damage growth and multiple damage site nucleation and growth mechanism These concerns provide the basis for codes which must be used for the design of the new aircrafts and for the certification of the continued exploitation of aged aircrafts.

4. STATIC STRENGTH DESIGN

The most common type of structural analysis is based on static strength. Static strength analysis predicts the strength or margin of safety of a structure, for a given loading condition and material strength. The loads are usually limit, ultimate, or 'equivalent' dynamic loads. Many of these loads are specified by regulatory agency requirements. Static strength analysis is still the basis for structural certification, although additional types of analysis are now also required. There is a little regulatory guidance for choosing a static strength analysis method. The method chosen is usually left up to the manufacturer. Many are based on classical text book methods, but may have some slight modifications or use different data (based on additional testing). Many of these methods and data are proprietary to the manufacturer.

The load relationship between limit design loads and ultimate design loads prescribed in the strength requirements is:

Limit design load X Factor of safety (1.5) =Ultimate design load **(4.1)**

Limit loads are the actual (maximum) loads applied to airplane structure for stipulated flight and ground conditions. If deflection under load significantly changes the distribution of external or internal loads, it has to be accounted for. The structure must be able to support limit loads without detrimental, permanent deformation. Under a rapid application of a load, structure, such as a wing, will deflect momentarily to a position beyond that for an equal static loading. This added dynamic effect and the resulting dynamic stresses must be considered. The ultimate design loads are the limit loads multiplied by a factor of safety. Airplane structure must sustain ultimate loads without failure. Except for certain special requirements, the required factor of safety is 1.5. It is intended to provide reserve strength over that required to sustain expected limit loads. This reserve strength requirement is based on experience. It covers material factors such as the ratio of yield to ultimate strength, variation in mechanical properties, and size tolerances. It provides for unknown variations in loads, stresses, and stiffness.

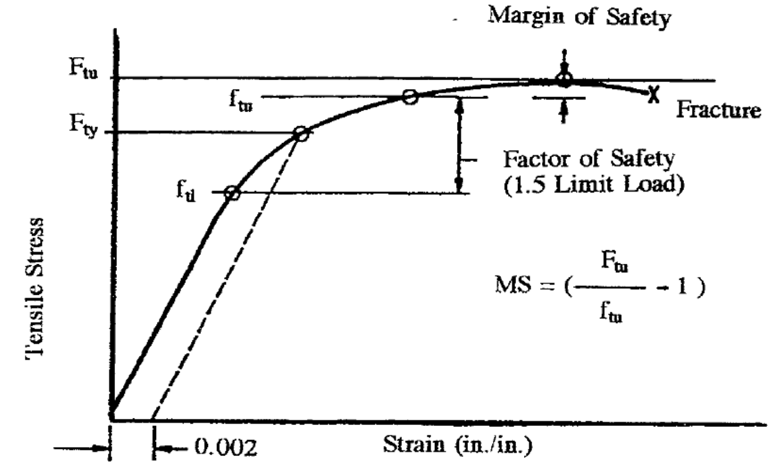

Figure 4.1: Stress-Strain Curves for Aluminum Alloy (Fty>0.67Ftu)

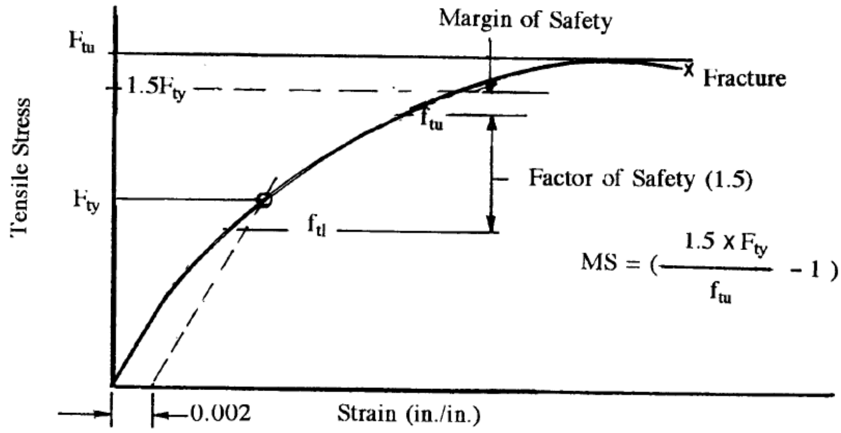

Figure 4.2: Stress-Strain Curves for Aluminum Alloy (Fty<0.67Ftu)

f_{tl}: Actual stress at limit load

f_{tu}: Actual stress at limit load

F_{ty}: Materail yield strenght at 0.002 in./in.

F_{tu}: Materail ultimate strenght

MS: Margin of Safety

The static strength requirements of metallic structure can best be explained through the use of stress-strain curves as shown in figure 4.1 and 4.2 [9]. The modulus of elasticity (E) is the slope of the curve in the elastic range during which the strain is directly proportional to the stress. The tensile yield strength (F_{ty}) is the stress level at which the metal reaches its maximum measurable elastic limit. The maximum measurable elastic limit is defined as a permanent deformation of .002 in/in. The ultimate tensile strength (F_{tu}) is the maximum stress level observed prior to fracture or failure of the metal. Since the metal must be able to carry limit loads without any detrimental permanent deformation, the resulting design limit stress (f_{tl}) is generally less than the yield strength of the metal (F_{ty}). Similarly, the structure must withstand ultimate loads without failure. The resulting design ultimate stress (f_{tu}) must not exceed the ultimate tensile strength (F_{tu}). In most instances, F_{tu} will be the critical stress, but when 1.5 F_{ty} is less than Ftu the critical stress will be 1.5 F_{ty}. The margin of safety for a simple linear case is:

$$MS = (F_{tu}/f_{tu} - 1) \text{ X } 100 \qquad \textbf{(4.2)}$$

For all other cases the margin of safety is based on the allowable load (P_{all}) and the applied load (P_{app}):

$$MS = (P_{all}/P_{app} - 1) \text{ X } 100 \qquad \textbf{(4.3)}$$

The magnitude of the alowable damage on a component is a direct function of the margin of safety in the original design.

4.1 Deficiencies of Static Strength Analysis

Experience shows that static strength analysis does not satisfy all the requirements for modern airplane structural integrity. For example, static strength cannot predict the fatigue life of the structure. Static strength does not predict the loss of strength, due to the presence of cracks or other stress concentrations (residual strength).

In the early days of aviation, the deficiencies of static strength analysis did not pose a major hazard. This was partially due to low yield strength of the available materials. The low yield strength also meant a high fracture toughness, which minimizes the effects of fatigue cracking and residual strength. In addition, early airplane usually had short operational lives. This also minimized the effects of fatigue cracking and residual strength. Today's airplanes last much longer, accumulating more fatigue cycles. The usual design service objective for a commercial airliner is 20 years. Airplanes are remaining in service longer. These longer lives necessitate methods of analysis to ensure structural integrity as the airplane ages.

5. SAFE LIFE DESIGN

In aerospace, fatigue life evaluation has been specifically based in the past on what is defined as safe-life design. Safe-life means that the component/aircraft is designed such that it is virtually able to withstand its whole design-life inspection free. Once the design life has expired, the component has to be removed, irrespective of having fractured or not [10]. Advisory Circular 25.571-1C defines safe-life as: "Safe-life of a structure is that number of events such as flights, landings, or flight hours, during which there is a low probability that the strength will degrade below its design ultimate value due to fatigue cracking." [11].

Reliance of safe-life principles for continued airworthiness of early commercial airplanes were to some degree successful. This was primarily due to rapid technology developments rendering airplanes obsolete before serious challenges of the established life limits. Conversion of World War II bombers to airliners caused some airworthiness authority concerns which resulted in limits of operational lives and/or initiated measures for nondestructive testing. In the 1950s, it became clear that static strength criteria had to be supplemented by estimated replacement times for some critical structural elements such as spar beams on numerous one-spar and two-spar wings. Many such configurations had evolved during the military bomber type developments during World War II. It became clear that fatigue failures would likely be due to use of high strength aluminum alloys without corresponding increase in fatigue strength. Further compounding the problem was improved stress analysis methods coupled with detailed and full-scale static testing of structural components, which often would eliminate past hidden static strength margins.

The knowledge of actual operating conditions also became more extensive which provided more precise static strength analyses based on rational ultimate design conditions. Important lessons were learned and fatigue test requirements emerged. Repeated load testing was for instance performed on the Comet I in 1950. These tests were carried out on the same wings used for ultimate static strength tests.

The influence of these high loads on cumulative fatigue damage is today self evident but not recognized at the time. It was also recognized through experience that first defects in the fleets could occur at less than a quarter of the test demonstrated life. The attempts to design for a certain life was gradually changed to control fatigue life by limiting major component service lives.

The use of imprecise and inaccurate fatigue analyses coupled with inherent material scatter characteristics often resulted in unnecessarily short lives and many sound structures were retired prematurely. Implementing safe-life principles often resulted in political problems for some airplane types in service in different countries. The overall problem with the safe-life principles were indeed that an acceptable commercial airliner safety standard could not be economically achieved [12].

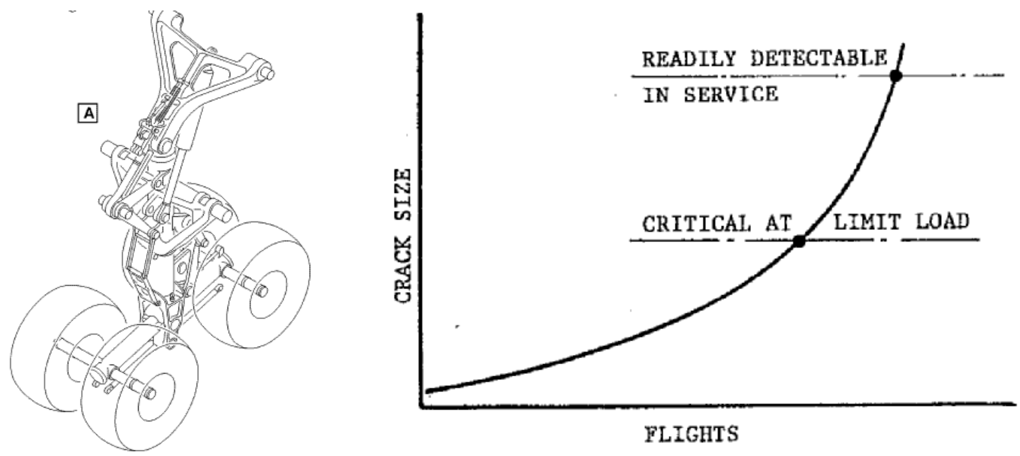

Figure 5.1: Example of Safe Life Structure

An example is given in figure 5.1, [13]. Single-load path structures are sometimes more simple in production (economic aspect), but in view of fatigue high safety factors are required, which implies a lower allowable design stress and a heavier component [14].

A structure designed as safe life contains a single load path only and the inspectable crack length may be in the range of the critical crack length. Consequently inspection intervals to monitor the structure cannot be defined.

A failure of one of the structural elements leads to the complete failure of the safe life structure and possibly to significant consequences for the aircraft.

A fatigue resistant design of safe life structure is based on fatigue life calculations for all structural elements during the design phase and is justified by full scale fatigue test with the complete safe structure. The fatigue life calculations are performed using the linear damage accumulation according to Palmgren-Miner considering relevant load spectra and material (S-N) data. The calculated fatigue life as well as the achieved test life are divided by relevant scatter factors. [15]

5.1 Deficiencies of Safe Life Design

The safe-life design principle has a very conservative approach. Cracking is not allowed during a component's design life, therefore safe-life structures require extensive fatigue testing. This is expensive and time consuming. These requirements result in structures which are heavier than parts designed to fail-safe design principles. So, safe-life design is usually limited to structures which cannot be fail-safe, or where adequate inspections are impractical. Today, safe-life design principles are typically limited to ground loaded structures such as high strength landing gear steel components for which substantial fatigue test verification is required. Safe-life principles can also result in the large economic impacts. If major structural components must be replaced often, the airplane may not be cost effective to operate.

6. FAIL SAFE DESIGN

Advisory Circular 25.571-1C [11] defines fail safety as: “The attribute of the structure that permits it to retain its required residual strength for a period of unrepaired use after the failure or partial failure of a principal structural element.” This definition is applicable to all structural concepts. Residual strength is simply “the strength of a damaged structure.” The phrase “required residual strength” relates to the capability of the damaged structure being consistent with the required load level.

The fail-safe design principle uses multiple load paths to ensure structural integrity. If one load path cracks completely through, or sustains accidental damage, the remaining load paths carry the additional load. Examples include: Multiple stringers and ribs in wings, Multiple wing panels, Multiple stringers and frames in fuselage construction (This construction also breaks the fuselage skin into redundant panels), Bonded and bolted fittings (often called 'back-to-back‘ fittings), and bonded and bolted landing gear beams.

Static strength analysis of these structures, with one element severed, satisfies the certification requirement for fail-safety. The loads for fail-safe design conditions are called 'fail-safe loads'. These usually correspond to limit load conditions.

The fail-safe principle also requires that any damage will be detected during an inspection, and then repaired. Some types of damage produce effects that are obvious, such as flapping fuselage skin panels, or wing fuel tank leaks. This obvious damage is considered part of the fail-safe inspections. Fail safe design philosophy has been a fundamental design requirement for aircraft manufacturers. Fail safe designs provide inherent robustness in the event of significant structural damage from several possible sources. Sources include fatigue damage, environmental deterioration, accidental damage, maintenance errors, manufacturing flaws and discrete events such as engine burst and impact damage.

The application of the fail-safe concept to structural design is based on the fundamental idea that a failure or obvious partial failure of a single principle structural element shall not cause the loss of the aircraft while in flight through [16]:

• Complete structural element collapse.

• Large deflections resulting in loss of control such as

- Large wing deflections which may make flight impossible

- Flap assymmetry

• Flutter, whether it is of a fixed or movable surface.

• A component failure such as a turbine engine bleed duct which in turn could blow up a wing, fuselage or empennage.

• A change in the aerodynanuc characteristics such that continued flight is impossible.

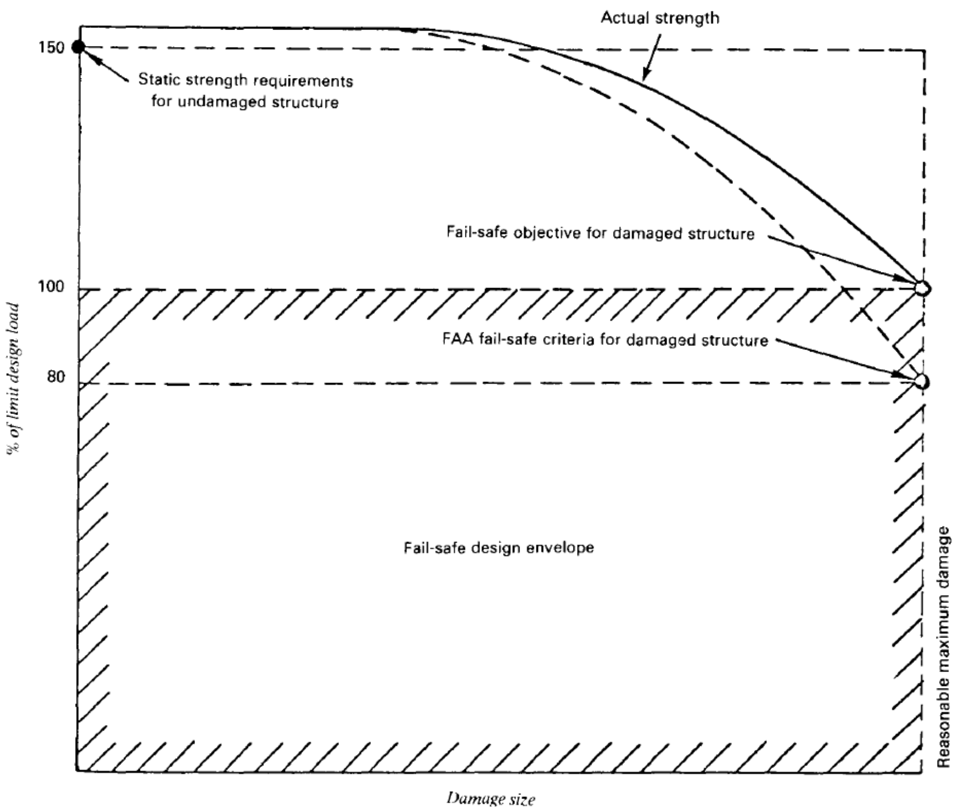

Figure 6.1: Fail Safe Design Requirements

An example could be complete loss of a wing leading edge where a partial loss would be survivable. Structural failures can be result of a wide variety of misfortunes. The degree or of a single principle structural element failure has not been defined, however some interpretations have been made thereto. If a structural assembly such as a wing box were made up of a relatively large number of elements such as integrally stiffened planks comprising the wing surfaces, a single failure would be the complete severence of one "plank". Similarly, a single failure could involve the severence of a beam cap or a shear web of the wing beam. Fail-safe design requirements are illustrated in Figure 6.1. Wing fail safety example is shown in Figure 6.2 [17]. Spanwise joints provide capability to sustain failure of any one skin panel.

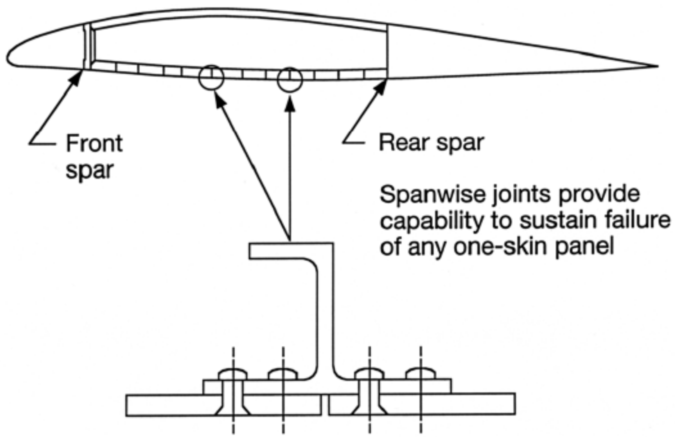

Figure 6.2: Wing Fail Safety

6.1 Fail Safe Design Deficiencies

Experiences have shown that fail safe design produces structure with a credible but imperfect safety record. Fail-safe design is a good philosophy, and worked well for many decades. In fact, fail-safe design still provides the basis for most new airplane designs. However, operational experience shows that some of the assumptions of fail-safety do not hold true. Cracks usually develop in several elements at the same time, making the alternate load paths weaker. This is called multiple site cracking. A/C manufacturers cannot always rely on obvious damage to alert operators to structural problems before they become catastrophic. Corrosion weakens alternate load paths, and accelerates crack growth. To compensate for these deficiencies in fail-safe design, the damage tolerance philosophy was developed.

7. DAMAGE TOLERANT DESIGN

Damage tolerance is the attribute of the structure that permits it to retain its required residual strength for a period of use after the structure has sustained a given level of fatigue, corrosion, accidental or discrete source damage [11]. Another word, it is the ability of the structure to sustain damage in the form of cracks, without catastrophic consequences, until such time that the damaged component can be repaired until the economic service life is expired and the airplane or component retired [18].

Since the likely development and growth of the damage may be established by analysis, it is possible to develop a structural maintenance programme, with a series of scheduled inspections and replacements to ensure that catastrophic structural failure will be avoided throughout the operational life of the aircraft.

The damage tolerance evaluation required by FAR 25.571 is apparently intended to consider the effects of fatigue damage, environmental damage (e.g. corrosion) and accidental damage. Design precautions are taken to minimize the risk of corrosion damage, and any corrosion that does occur is generally in areas not sensitive to fatigue, such as the lower part of fuselage, and can be removed as soon as discovered by an adequate maintenance programme. As a result, the impact of corrosion on the fatigue characteristics of the structure is not considered, and the fatigue and corrosion inspection programmes are developed separately, both taking account of the accentuated deterioration due to likely accidental damage.

The damage tolerance evaluation can therefore be understood with a crack propagation and residual strength analysis for structural elements subject to ‘natural’ or ‘accidental’ fatigue damage only. The assessment involves consideration of the probable damage locations, the extent of damage, crack initiation mechanisms, crack growth time histories and crack detectability.

The results of these analyses, which are supported by extensive fatigue test evidence, are subsequently used to establish a suitable programme of inspections for each fatigue-critical area of the structure, which should consist of the method of inspection, the inspection threshold, and the repeat inspection interval.

Damage tolerance can be achieved more easily by incorporating fail-safety features, such as redundancy, multiple load paths and crack arresters. Fail-safe structures can sustain larger damage, but if unattended this damage will eventually still cause a catastrophic failure. Hence, fail-safety features by themselves do not prevent fracture: the partial failure (e.g. the failed load path) still must be detected and repaired; even if the structure is fail-safe, inspection is essential to achieve safety. Without fail-safety features the structure can still be damage tolerant, provided cracks are detected and repaired before they impair the safety. Fail-safety features merely alleviate the inspection problem.

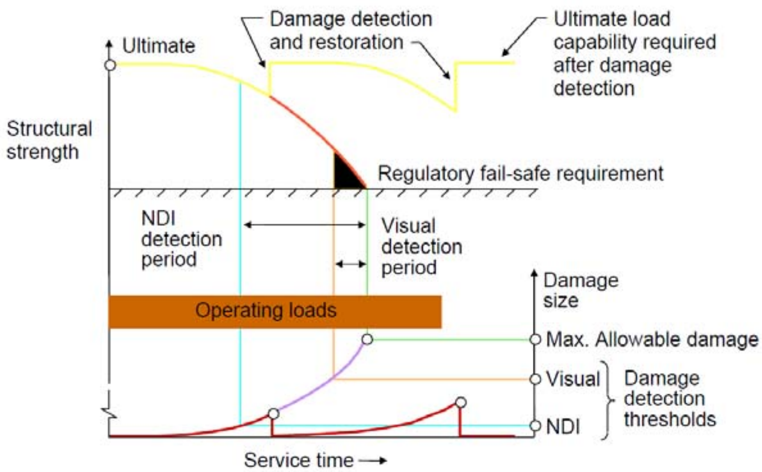

Figure 7.1: Strength Requirements for Damage Tolerant Structure

Damage tolerant analysis begins with the same static strength calculations as non-damage tolerant structure. This includes analyses for ultimate, limit, and failsafe design conditions.

A/C manufacturers perform fatigue analysis to ensure the economic design life goals. Finally, they perform crack-growth and residual strength predictions. Crack growth prediction assumes initial cracks which are too small to detect and then they predict how much the cracks grow during normal airplane operations.

When the cracks grow to a certain size, you can detect them. Residual strength predictions tell how the presence of the fatigue cracks affect the strength of fail-safe elements. They also tell how the presence of undetected accidental damage affects the strength of remaining fail-safe elements.

Manufacturers use the crack growth and residual strength predictions to establish maximum crack sizes. They also establish how many inspection opportunities exist to find a given crack. This ensures that airplane damage is detected, and repaired before the strength is below a minimum level.

Certification of commercial jet transports mandates damage tolerant designs in all instances where it can be used without unreasonable penalty. The technical capability has evolved to relate inspection requirements to damage growth which, in the past, were based on service experience. Primary airframe components are designed to meet specific static and dynamic loading conditions that greatly exceed normal operating loads. Graphical representation of damage tolerance is shown in Figure 7.1. Maximum structural strength capability occurs at the beginning of an airplane's life. The operating loads are much smaller than the ultimate strength. As the airplane ages, the strength slowly reduces, due to crack growth and/or corrosion damage. Before the strength becomes less than the residual requirement, the damage is detected and repaired back to original capability. Note that there are several opportunities for damage detection. This process continues throughout the life of the airplane.

Laboratory developed probability of detection (POD) curves are often relied upon in service environments beyond what is justified by experimental evidence. This is an even more serious problem when these methods are called upon to search an area for unknown defects rather than to confirm the presence of a specified type and location. Cracks missed during inspections are often not properly accounted for in POD data. Visual inspection has been and will continue to be the main source of initial detection of previously unknown damage in most commercial jet transport structures. The lack of interest and resolve in the research community to characterize and quantify visual POD data is indeed perplexing.

As the airplane progresses through its service life, damage may occur and reduce static strength capability (residual strength). Structure is damage tolerant if damage that may occur, can be discovered and repaired before the residual strength falls below the regulatory failsafe capability. The damage detection period is dependent on structural characteristics as well as maintenance and inspection procedures. Once damage has been detected, strength must be restored to the ultimate design level. Aging airplane structure may be affected by widespread fatigue damage that alters the detection requirements. This is due to the effect of local damage at multiple sites on residual strength capability [12].

7.1 Single Load Path – Damage Tolerant

It is the structure with a single load path, with a residual strength under limit load with a crack of a bigger size than the detectable size. The certification authorities allow these structures, although they are not recommended. The inspection interval is based on the growth of the crack, from its detectable size up to its critical size. Therefore, these structures are designed so that possible cracks grow slowly enough to increase the growth period and provide reasonable inspection intervals. An example of a structure of this type is a wing intrados with integrated stringers as the one shown on figure 7.2 [13,15].

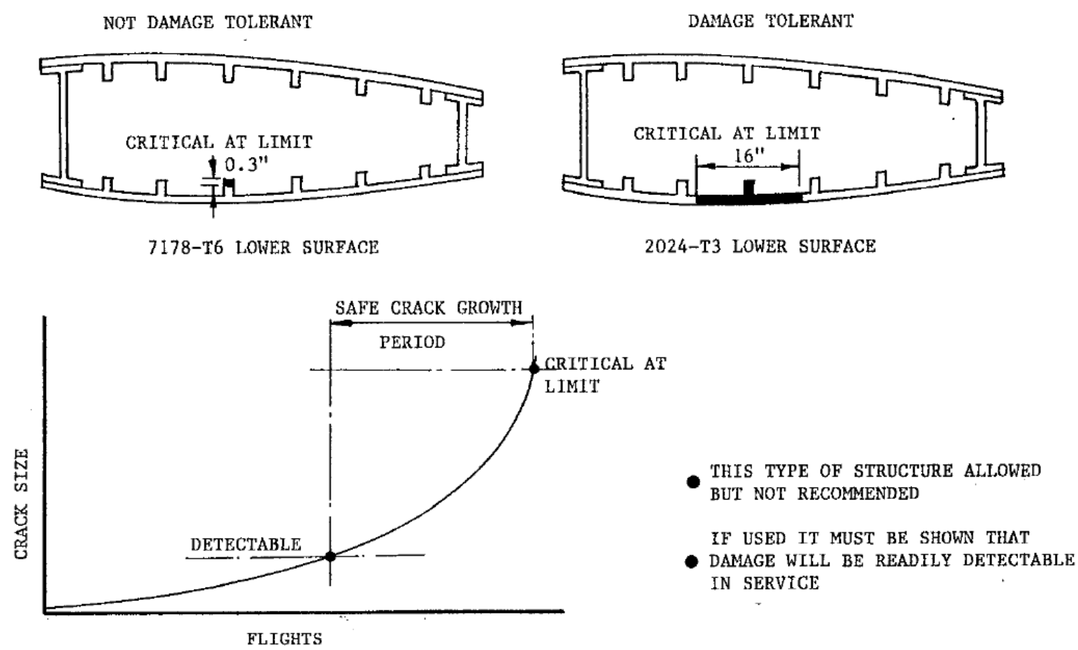

Figure 7.2: Example of Single Load Path Structure – Damage Tolerant

7.2 Multiple Load Path – Damage Tolerant

It is a structure with more than one load path that can withstand the limit load with one of them completely broken. There are three kinds of structures, depending on its inspection way.

7.2.1 Structure externally inspectable

In this type of structures one of the load paths (primary) is located on the internal part of the structure and due to the difficulty to inspect it, either because of the access or by the inspection method, it is not possible to detect cracks on it. Therefore, the secondary load path is externally inspected. The crack on the primary path grows up to its critical size with no detection possible, and once the primary path is broken, the load is re-distributed, and the crack on the secondary path keeps on growing up to its detectable size and further its critical size. The inspection interval is based on the growth of the crack on the secondary path from its detectable size up to its critical size. This type of structures is recommended for most of the primary structures formed by rigid panels (e.g., wing and fuselage skins). It is also recommended that the critical size is bigger than or equal to two distances between stiffeners (two bay cracks) with the central one broken. An example of this type of structure can be a skin with riveted stringers where there is no possibility to inspect the stringer externally, figure 7.2 [13,15].

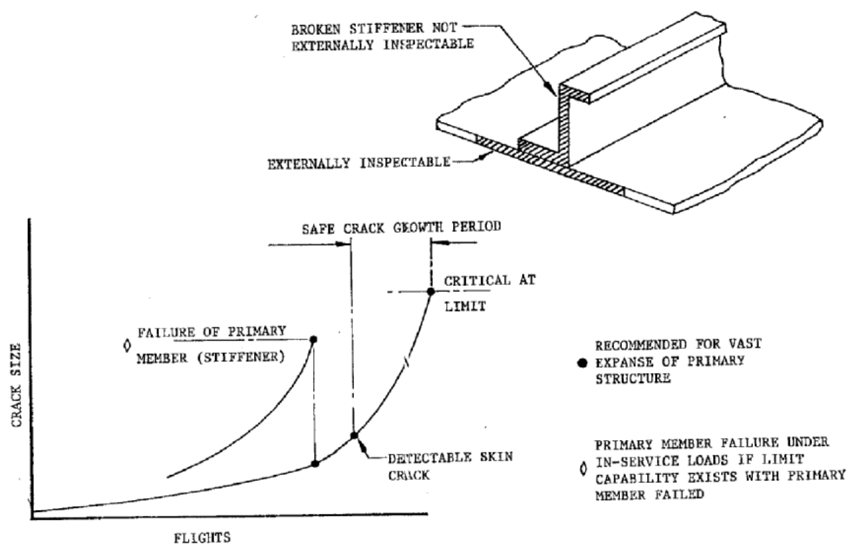

Figure 7.3: Example of Multiple Load Path Structure Externally Inspectable

7.2.2 Non inspectable for less than one load path failure

In this type of structures, the primary load path is such, that the crack can only be detected when the load path is broken. Therefore, the crack of the primary path grows up to its critical size and then it breaks; at that moment it becomes detectable. From that moment onwards, the load is re-distributed and the crack on the secondary path keeps on growing up to its critical size. Then, the inspection interval is based on the growth of the crack of the secondary path, from the time the primary breaks up to its critical size. An example of this type of structure can be that of double lugs (e.g., the joint of horizontal stabilizer to the fuselage), figure 7.3 [13,15].

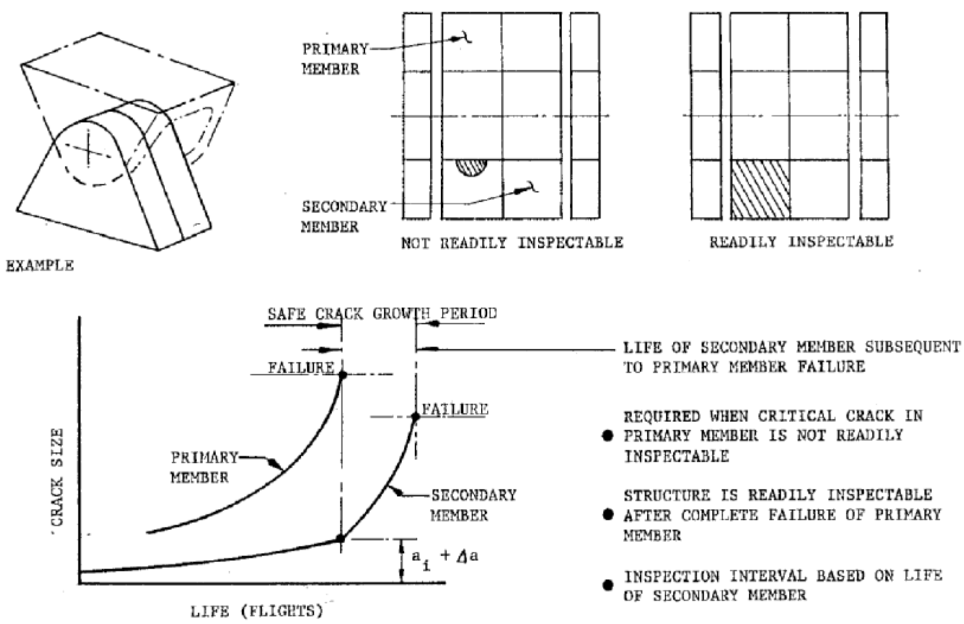

Figure 7.4: Example of Multiple Load Path Structure Not Inspectable for Less than Load Path Failure

7.2.3 Inspectable for less than one load failure

They are structures with load paths that can be inspected with a detectable size lower than the element breakage size. Therefore, a crack can be detected on the primary element before it breaks. Once the primary path is broken, the load is redistributed and the crack on the secondary path keeps on growing up to its critical size. The inspection interval is based on the growth of both cracks from the moment the primary is detected up to the critical size of the secondary path (with the primary path broken). An example of this type of structure can be that of an integrated skin formed by several plates riveted, as shown on figure 7.4. [13,15].

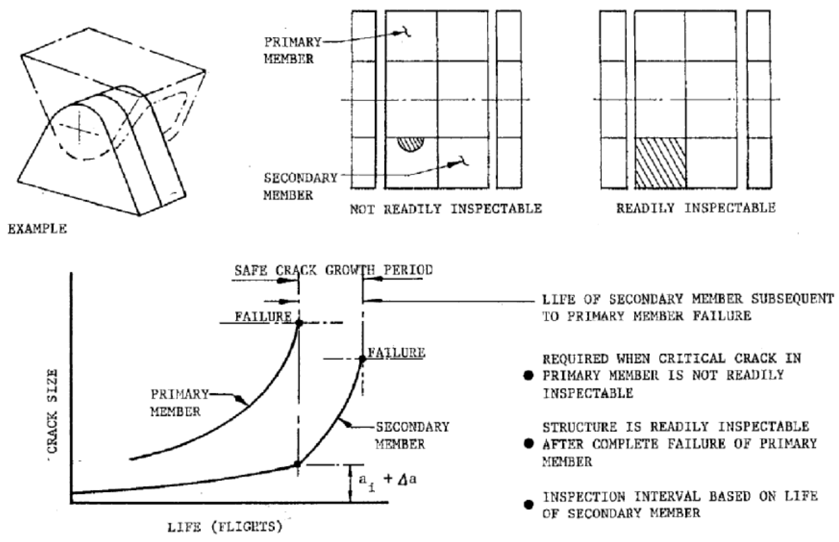

Figure 7.4: Example of Multiple Load Path Structure Inspectable for Less Than Load Path Failure

7.3 Elements of Damage Tolerance

Damage tolerance has three main elements (Figure 7.5) interrelated and any change to one of these elements has an effect on the other two elements. [2]

- Residual (allowable) Strength Predictions
- Crack (damage) Growth Predictions
- Inspection Programs (damage detection)

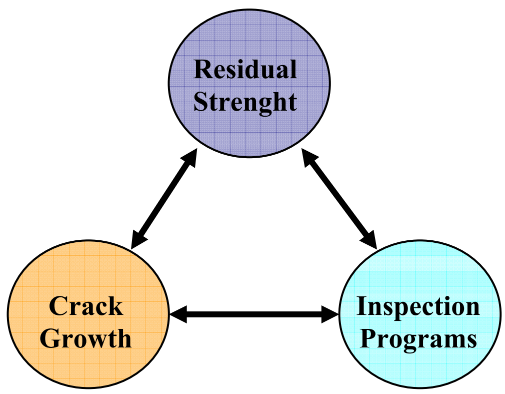

Figure 7.5: There Main Elements of Damage Tolerance

7.3.1 Residual strength (Allowable damage)

The static strength of a simple structural element is usually defined as F_{tu} x A. However, this definition is not valid where cracks exist in the element. The stress concentration produced by the crack makes the load carrying capability much less than the net area multiplied by the allowable stress.

To prevent catastrophic failure, one must evaluate the load carrying capacity that will exist in the potentially cracked structure throughout its expected service life. The load carrying capacity of a cracked structure is the residual strength of that structure and it is a function of material toughness, crack size, crack geometry and structural configuration. The determination of residual strength for uncracked structures is straightforward because the ultimate strength of the material is the residual strength.

A crack in a structure causes a high stress concentration resulting in a reduced residual strength, as shown in figure 7.6 . When the load on the structure exceeds a certain limit, the crack will extend. The crack extension may become immediately unstable and the crack may propagate in a fast uncontrollable manner causing complete fracture of the component. In general, unstable crack propagation results in fracture of the component. Hence, unstable crack growth is what determines the residual strength. In order to estimate the residual strength of a structure, a thorough understanding of the crack growth behavior is needed. Also, the point at which the crack growth becomes unstable must be defined and this necessitates the need for a failure criterion. There are several criteria available; these criteria are tailored to represent the ability of a material to resist failure.

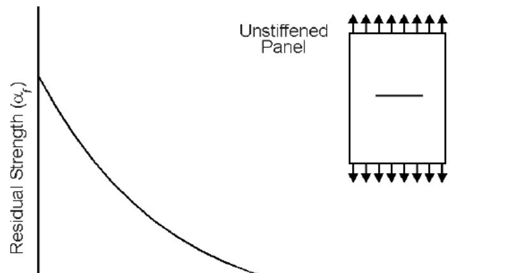

Figure 7.6: Residual Strength vs Crack Lenght

7.3.2 Crack growth (Damage growth)

Classical fatigue analysis can determine how many cycles occur before a metal specimen initiates a crack. It does not tell the rate or size of crack growth. Residual strength analysis determines the critical crack size. Crack growth analysis tells the number of flights for a small (undetectable) crack to grow to the critical crack size. Crack growth rate depends on the material and the loading spectrum. Materials with high fracture toughness usually have slower crack growth rates than materials with low fracture toughness.

The interval of damage progression from lengths below which there is negligible probability of detection to an allowable size determined by residual strength requirements. A crack in a structure will increase in size in response to application of cyclic loads.

As shown schematically in figure 7.7, growth is negligible when the crack is very small. Since these effects are nearly impossible to observe, it can be argued that some tiny flaws are always present in a structure. An alternative interpretation is that a small crack is initiated in perhaps 5% of the time range of the diagram due to a manufacturing flaw or material inclusion and then grows during the greater part of the time range to failure. As the crack increases in size, increments of extension get larger until a critical dimension is attained at which the structure fractures in the course of a single cycle of loading.

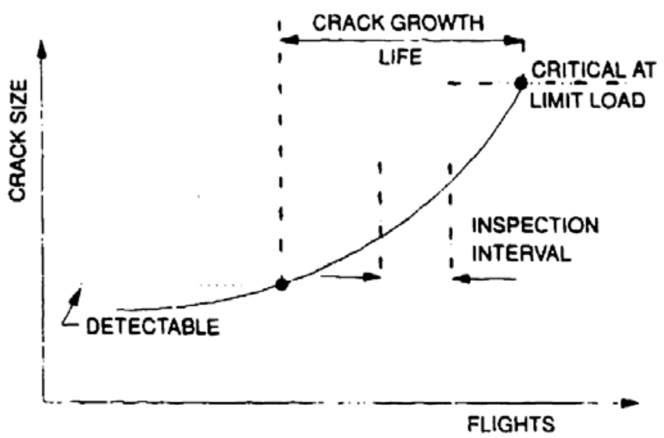

Figure 7.7: Crack Growth in Response to Cyclic Loads

7.3.3 Damage detection (Inspection program)

The purpose of damage tolerance inspections is to discover damage, before the structures residual strength capability falls below the residual strength requirement. Residual strength prediction tells what size the cracks correspond to minimum strength (critical crack length). Crack growth prediction tells how quickly cracks will grow to the critical length. A/C manufacturers choose the inspection interval so that a crack smaller than the detection threshold can't grow to critical length before the next inspection period. This is called the damage detection period. The detectable crack size depends on the inspection method. Some methods can find very small cracks (high frequency eddy current, for example). Other methods can only find larger cracks (general visual inspection, for example). The detectable crack size (and therefore the inspection method) affects how often the inspection is required.

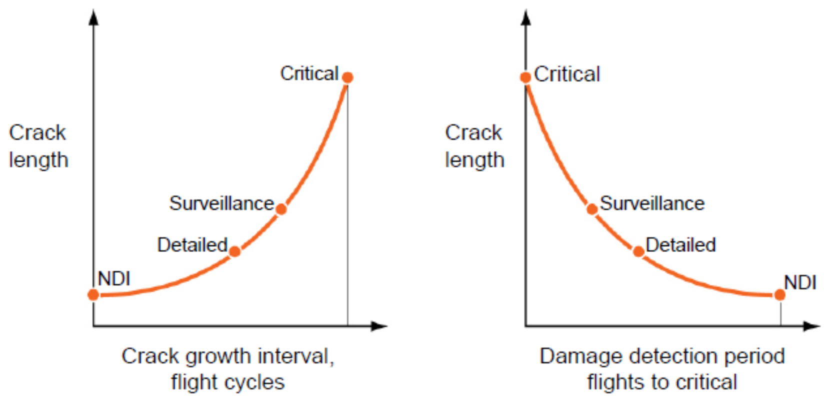

Figure 7.8: Crack Lenght Versus Inspection Methods

Crack growth life is the time (measured, for example, in terms of number of flights) it takes a crack to grow from some initial length to a critical size that reduces the strength margin to zero. An initial size at which the crack can be detected marks the start of this time scale. The purpose of damage tolerance analysis is to ensure that crack growth life is greater than any accumulation of service loads that could drive a crack to a dangerous size. This objective can be achieved with an inspection program that detects cracking initiated by fatigue, accident, or corrosion before propagation to failure. Inspection frequencies must be at intervals that are fractions of expected growth life to afford the opportunity for corrective action that maintains structural safety if cracks are found. The economic feasibility of an inspection plan must consider the cost trade-off between inspection methods and intervals, figure 7.8.

8. CONTINUING AIRWORTHINESS CHALLENGES

Each new airplane model enters service in airworthy condition, built to meet all the design goals set for its anticipated service life. However, variations in operating environments, different maintenance practices, and the ability to remain in service much longer than originally anticipated can lead to significantly different structural performance over time.

Continuing airworthiness concerns for aging jet transports has received attention over the last two decades. Supplemental structural inspection programs were developed in the late 1970s to address fatigue cracking detection in airplanes designed to the fail-safe principles. These evaluations were performed in accordance with updated damage tolerance regulations to reflect the state-of-the-art in residual strength and crack growth analyses based on fracture mechanics principles. Damage at multiple sites was also addressed in terms of dependent damage size distributions in relation to assumed lead cracks in different structural members. Structural audits were performed in the mid 1980s to ascertain whether these supplemental inspection programs addressed independent multiple site damage in similar structural details subjected to similar stresses. The safe decompression concepts were challenged in these reviews of different manufacturer damage tolerance philosophies but no major changes occurred.

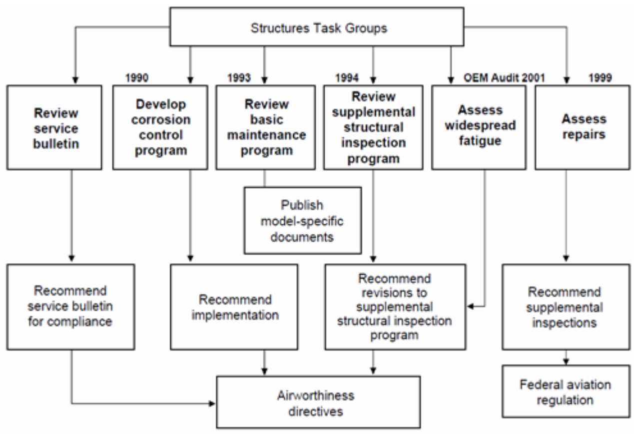

Figure 8.1: Industry Aging Fleet Initiatives

The Aloha Airlines failure identified a need to pay particular attention to aircraft which had accumulated a high number of flight cycles. As invariably a high number of flight cycles were associated with older aircraft, the aftermath of the Aloha incident rise to the so-called Aging Aircraft Program which was launched in the early 1980s. To assist their deliberations on how best to address the aging aircraft problem, the FAA recruited the help of a number of industry experts and formed the Airworthiness Assurance Task Force (AATF) in 1988. This group was subsequently renamed the Airworthiness Assurance Working Group (AAWG) of the Aviation Rulemaking Advisory Committee (ARAC) which was formed by in 1991. Many of the aging aircraft in existence at the time the AAWG was formed had not been designed on a damage tolerant basis as this had not been made a regulatory requirement until the introduction of FAR 25.571 Amendment 45 in December 1978. Consequently, the AAWG decided that their first priority was to implement programs for fail safe certified aircraft that would subject them to an equivalent level of inspection and maintenance scrutiny as that to which damage tolerance certified aircraft were subjected. The AAWG defined a number of programs, some of which have been subsequently implemented on the different aging aircraft fleets by aircraft specific Structures Task Groups (STGs), as shown in figure 8.1.

8.1 Service Bulletin Reviews

Continuing airworthiness of jet transport structures designed to the fail-safe principles has traditionally been ensured by inspection programs. In the event of known, specific fatigue cracking and/or corrosion problems that if not detected and repaired, had the potential to cause a significant degradation in airworthiness, the normal practice in the past was to introduce a service bulletin.

The net result of this process was to carry out inspections of all affected airplanes until damage was detected and then to perform the repair. Thus, continuing structural airworthiness was totally dependent on repetitive inspections. Aging airplane concerns prompted reassessment of the viability of indefinite repetitive inspections. [12] With increasing in-service experience, the type certificate holder or supplemental type certificate holder may find ways to improve the original design resulting in either lower maintenance costs or increased performance.

These improvements (normally involving some alterations) are suggested through service bulletins to their customers as optional (and may be extra cost) items. The customers may exercise their discretion whether or not to incorporate the bulletins. Evaluation process of this service bulletins may vary between operators. Different types of service bulletins are issued by TC holder/STC holder/OEMs.

For example, Boeing issues special attention, alert and standard type service bulletins to recommend inspection procedures method, threshold and interval which were designed to ensure with high (but undefined) degree of probability that the structural damage would be detected and be repaired before significant degradation in structural airworthiness occurred.

Frequently these service bulletins would also specify modifications/rework procedures that would eliminate the cause of the cracking problems and provide an alternative to repetitive inspections as a means of ensuring continued structural integrity. The inspection parts of the service bulletins were sometimes mandated by Airworthiness Directives. Most of the Alert Service Bulletins will result to issue of an AD.

Boeing's recommended compliance times in alert service bulletins are based on many factors considered during the safety analysis process, managing the risk being the most prominent of those considerations.

In conjunction with the lead airline review process, the FAA also reviews, comments on, and approves Boeing's mitigation plan and compliance recommendations. The FAA has recognized that there is additional time allocated to comply if the AD release is significantly later than Boeing's alert service bulletin release. The FAA's actions are based on their risk assessment, whether qualitative or quantitative, and the FAA's risk assessment governs their establishment of compliance periods. On the other hand, Special Attention Service Bulletin's (SAB) are reserved for less urgent safety issues (quite often personal safety) or significant economic issues. They remain a heritage Puget Sound-unique variation of standard service bulletins.

The additional tagging and verbiage associated with SABs are meant to increase operators' attention to consider our recommendations for incorporation. Sometimes ADs will result from SABs; for these cases, as with Alert SBs, the FAA's Compliance requirements apply.

Following lines provides details of the criteria used by Airbus to issue compliance recommendations for service bulletins. In accordance with recommendations contained in the ATA 2200 specifications, Airbus service bulletins are issued with a reason paragraph giving background information on the modification as well as guidance for implementation decision thus enabling Operators to establish retrofit requirements with respect to their own fleet experience. In addition, Airbus allocates a compliance classification as well as an accomplishment timescale which are derived from the decision process specified in ATA 2200.

Mandatory: This category is allocated if the accomplishment is rendered mandatory through an Airworthiness Directive (Consigne de Navigabilité) issued by the French DGAC to ensure adequate safety level in accordance with JAR39. The timescale/limitation contained in the relevant Consigne de Navigabilité is referenced under accomplishment timescale within the compliance paragraph.

Recommended: This category has been developed to incorporate major enhancements regarding aircraft operation:

Operation: Modifications which significantly improve the level of airworthiness compliance regarding combined failure modes/system issues identified during aircraft operation.

Maintenance: Modifications which provide terminating action on disruptive inspections, mandatory or otherwise; or prevent costly repairs. Airbus strongly encourages that operators/owners accomplish service bulletins in this category in accordance with the recommended timescale so as to upgrade the affected aircraft to the latest available standard.

Desirable: This category is allocated if the service bulletin offers improvements in terms of dispatch reliability, maintenance cost, maintenance program reduction, operational benefits, crew/passenger comfort or regulated environmental nuisance. The Accomplishment Timescale is left to the Operator's entire decision.

Optional: This category is allocated if the service bulletin has been developed on a specific operator/owner request the accomplishment timescale is left to the operator's own decision.

8.2 Corrosion Prevention and Control Programs

Corrosion defines as “the electrochemical deterioration of a metal because of its chemical reaction with the surrounding environment.” [19]. Corrosion has always been recognized as a major factor in airplane aintenance. Each airline/operator addresses it differently according to its operating environment and needs. It became apparent that without effective corrosion control programs, the frequency and severity of corrosion are increasing with airplane age and, as such, corrosion was more likely to be associated with other forms of damage such as fatigue cracking.

Corrosion control in the aerospace industry has always been important, but is becoming more so with the ageing of the aircraft fleet. As the aircraft ages, fleets experinece increased number of structural failures and excessive maintenance cost and increased downtime. If the corrosion is not detected and treated at earlier stages, catastrophic consequences would likely to occur. For that reason, aircraft manufacturers have published Corrosion Prevention and Control Manuals (CPCM) and guidelines to assist the operators/airlines. Commercial aircraft age in service, the structures experience many forms of corrosion (details of corrosion types can be found in [19]), dependent on the alloy and temper, the finish, the assembly design and location on the airplane, the drainage, and the internal and external environments.

Unattended, these may become serious airworthiness issues. In the United States, the Federal Aviation Administration (FAA) requires that the manufacturer “… show that catastrophic failure due to fatigue, corrosion or accidental damage, will be avoided throughout the operational life of the airplane …” [20] for that the airline operator to have a plan to address corrosion problems and reduce their severity between inspections. “Each certificate holder is primarily responsible for the airworthiness of its aircraft … the performance of the maintenance … in accordance with its manual and the regulations …” [21]. This, if allowed to continue, could lead to an unacceptable degradation of structural integrity, and in an extreme instance, the loss of an airplane. The working groups have recognized the need for a universal baseline minimum corrosion control program for all airplanes to prevent corrosion from affecting airworthiness. Maximum commonality of approach within and between each manufacturer to ensure consistent and effective procedures throughout the world have been a key objective for the working groups. The program requirements apply to all airplanes that have reached or exceeded the specified implementation age threshold for each airplane area. The specific intervals and thresholds vary between models, but all programs follow the same basic philosophy.

8.2.1 Corrosion prevention and control methods

There are a number of diffenet approaches to improve the ability of corrosion resistance of a material, component, or systems. Corrosion prevention and control (CPAC) methods, can be divided up into six main groups [22]:

<u>Organic or Inorganic coatings:</u> Coatings are the easiest, most effective, and least expensive Corroison Prebevtion and Control method available. Durable metallic, inorganic, and organic coatings are frequently used for providing short or longterm corrosion protection of metals from various types of corrosive media.

There are two main types of coatings: barrier coatings and sacrificial coatings. A barrier coating acts as a shield, which isolates the metal being protected from the surrounding corrosive environment. Barrier coatings are typically unreactive, resistant to corrosion, and also protect against wear. Sacrificial coatings preferentially corrode, which is an effective mechanism for protecting the cathodic substrate.

<u>Surface Treatments:</u> These treatments modify a material's surface to improve its corrosion resistance. Chemical conversion coating and anodizing techniques employ chemical reactions to create a stable, corrosion resistant oxide film on the metal's surface. Shot peening is a mechanical process that induces compressive residual stresses on the surface, thus improving resistance to stress corrosion cracking and corrosion fatigue. Laser treatments can modify the surface characteristics of a materialincluding its hardness and morphology.

As a result, certain forms of corrosion may be mitigated or prevented. Laser shock peening is a process somewhat analogous to shot peening in that it too can induce compressive residual stresses within a metal's surface region to increase its resistance to stress corrosion cracking and corrosion fatigue.

Corrosion Preventive Compounds (CPCs): CPCs offer temporary protection against corrosion and as such, are materials that need to be reapplied during scheduled maintenance. They come in four basic types [23]:

- Water displacing soft films
- Water displacing hard films
- Non-water displacing soft films
- Non-water displacing hard films

Water displacing CPCs penetrate small cracks and crevices, force out any water that may be present, and leave behind a protective film. These films can be oil, grease, solvent, or resin based. They repel water or they can contain corrosion inhibitors.

Inhibitors: Inhibitors are chemicals that either react with the surface of a material to decrease its corrosion rate, or modify the operational environment to reduce its corrosivity. Inhibitors may be dissolved in an aqueous solution or dispersed in a protective film. For instance, they can be injected into a completely aqueous recirculating system to reduce the corrosion rate in that system.

They may also be used as additives in coating products, such as surface treatments, primers, sealants, hard coatings, and CPCs. Inhibitors are usually grouped into five different categories:

- Passivating
- Cathodic
- Film forming
- Precipitation
- Vapor phase.

Cathodic and Anodic Protection: Cathodic protection (CP) is a widely used electrochemical method for protecting a structure or important components of a system from corrosion. There are two main classes of cathodic protection: active and passive. Active cathodic protection, also called impressed-current cathodic protection (ICCP), uses an external power supply to provide an electrical current to the surface of a metal. The excess electrons at the surface feed the corrosive medium, thereby protecting the metal from being stripped of its electrons, which would otherwise result in corrosion of the metal. Passive CP systems are simpler than ICCP systems and involve the galvanic coupling of the metal being protected to a sacrificial anode, which protects the adjoining surface by freely giving up electrons and in the process preferentially corroding.

Anodic protection is a more recently developed but less frequently used method of corrosion control. Using an applied electrical current, this method actively creates a passive film on the surface of the material being protected. For some applications, the current can be sustained during the material's service life, in order to maintain the passive film. Passive films are very non-reactive and consequently materials that possess them are resistant to corrosion. The method is typically employed to protect materials that are exposed to strongly alkaline or acidic environments. Anodic protection can only be used on metals capable of forming a passive film such as stainless steels.

Strategic Methods: Design, maintenance, and material selection can be strategically used to minimize the extent to which a material, component or system corrodes during its lifetime. Simple elements incorporated into a design can lessen the risk of corrosion. For example, allowing for drainage of water that otherwise might become trapped can effectively reduce corrosion problems. Similarly, employing seals to preclude water from entering a component or system will improve its corrosion resistance. Another effective way to minimize corrosion is to use gaskets to electrically insulate two dissimilar metals, thus eliminating the possibility of galvanic corrosion. Employing a maintenance schedule where a vehicle or structure is periodically cleaned will help reduce corrosion.

Salt and debris buildup on vehicles or structures, for example, can accelerate corrosion. Therefore, routinely washing them to remove the contaminants is a sound practice. In addition, regularly touching up protective coatings or reapplying CPCs will help reduce instances of corrosion. Perhaps the most important way to minimize corrosion from occurring in the first place is to make appropriate material selection decisions by carefully considering the application, environment, and potential corrosion problems that might occur. Selecting the appropriate materials and associated corrosion prevention and control practices can reduce maintenance costs and system downtime over the life-cycle of a system.

8.2.2 Corrosion control program guidelines

The possibility of an in-flight mishap or excessive downtime for structural repairs necessitates an active corrosion prevention and control program.

The type and aggressiveness of the corrosion prevention and control program depend on the operational environment of the aircraft. Aircraft exposed to salt air, heavy atmospheric industrial pollution, and/or over water operations will require a more stringent corrosion prevention and control program than an aircraft that is operated in a dry environment.

In order to prevent corrosion, a constant cycle of cleaning, inspection, operational preservation, and lubrication must be followed. Prompt detection and removal of corrosion will limit the extent of damage to aircraft and aircraft components. The basic philosophy of a corrosion prevention and control program should consist of the following: [19]

- Adequately trained personnel in the recognition of corrosion including conditions, detection and identification, cleaning, treating, and preservation;
- Thorough knowledge of corrosion identification techniques;
- Proper emphasis on the concept of all hands responsibility for corrosion control surfaces,
- Inspection for corrosion on a scheduled basis,
- Aircraft washing at regularly scheduled intervals,
- Routine cleaning or wipe down of all exposed unpainted surfaces,
- Keeping drain holes and passages open and functional,

- Inspection, removal, and reapplication of preservation compounds on a scheduled basis,
- Early detection and repair of damaged protective coatings,
- Thorough cleaning, lubrication, and preservation at prescribed intervals,
- Prompt corrosion treatment after detection,
- Accurate record keeping and reporting of material or design deficiencies
- Use of appropriate materials, equipment, and technical publications.

8.3 Baseline Maintenance Program Reviews

Operators/airlines must develop and document an approved maintenance program before introducing a new or derivative airplane into service. A/C manufacturers prepare a maintenance planning document (MPD) to assist operators in developing their own programs. Included in this document arc all maintenance review board (MRB) reporting requirements as well as manufacturer recommended tasks. Each operator has the final responsibility for its maintenance plan and adherence to maintenance schedules.

8.4 Supplemental Inspection Program Reviews

The manufacturer, in conjunction with operators, is expected to initiate development of a supplemental structural inspection program for each airplane model [24]. The purpose of this program was to ensure continued safe operation of the aging fleet by timely detection of new fatigue damage locations. The original Supplemental Inspection Documents (SID) normally is based on predictions or assumptions (from analyses, tests and/or service experience) of failure modes, time to initial damage, frequency of damage, typically detectable damage, and the damage growth period. These documents have been updated on a regular basis to reflect service experience and operator inputs. In the light of current aging fleet concerns, these inspection programs were to ensure adequate protection of the aging fleet. The major focus of these reviews were [12]:

• Adequacy of the present fleet leader sampling.

• Inclusion/deletion of principal structural elements (PSE).

Such a program must be implemented before analysis, tests, and/or service experience indicates that a significant increase in inspection and/or modification is necessary to maintain structural integrity of the airplane. In the absence of other data as a guideline, the program should be initiated no later than the time when the high-time or high-cycle airplane in the fleet reaches one half its design service goal. The initial candidate fleet leader samples comprised those airplanes exceeding 50% of the design objective in flight cycles when the typical fleet leader reached 75%.

Basic structural reassessments of compromised of the identification of structural parts or components which contribute significantly to carrying flight, ground, pressure or control loads whose failure could affect the structural integrity necessary for the safety of the airplane and whose damage tolerance or safe-life characteristics it is necessary, therefore, to establish or confirm. These are called Structural Significant Items (SSIs).

The calculation of residual strength, with multiple site damage and interactive crack growth under typical flight and ground loading, such that the airplane structure can sustain the load conditions stated for failsafe qualification under the current FAR 25.571 [20]. The establishment of a procedure for developing inspection programs that provide a high probability of detecting fatigue damage before residual strength falls below the fail-safe requirements.

These documents includes the type of damage being considered, and likely sites; inspection access, threshold, interval, method and procedures; applicable modification status and/or life limitation; and types of operations for which the SID is valid.

8.5 Widespread Fatigue Damage (WFD)

In 1988, the industry experienced a significant failure of the airworthiness system involving probable WFD. This failure allowed an airplane to fly with significant unrepaired fatigue damage. As a result the airplane, an Aloha Airlines 737, experienced a rapid fracture and loss of a portion of its fuselage in night. As a direct consequence of this accident, the FAA hosted the International Conference on Aging Airplanes in June 1988 in Washington. D.C.

As an outcome of the conference, an organization of operators, manufacturers and regulators was formed under the U.S. Federal Advisory Committee Act to investigate and propose solutions to the problems evidenced by the accident. This group is now known as the Airworthiness Assurance Working Group (AAWG). Since the 1988 Aloha incident, the industry has identified three new WFD conditions in the fleet per year. The follow-on actions to the conference were published in a final report in June 1999 [25].

Widespread fatigue damage (WFD) in a structure is characterized by the simultaneous presence of cracks at multiple structural components where the cracks are of sufficient size and density that the structure will no longer meet its damage tolerance requirement. The two sources of WFD are;

Multiple Site Damage (MSD), characterized by the simultaneous presence of fatigue cracks in the same structural element; and

Multiple Element damage (MED), characterized by the simultaneous presence of fatigue cracks in similar adjacent structural elements.

An industry committee on WFD identified 16 generic types of structure susceptible to WFD as shown Appendix A.

Since the Aloha accident in 1988, extensive research has been carried out on MSD/WFD. A brief survey of available MSD/WFD analysis guidelines and methods was performed by focusing on the major authorities and aircraft industries. In general, some useful technical guidelines were found in working documents from AAWG (Airworthiness Assurance Working Group) [26] and FAA (Federal Aviation Administration) [27] for MSD/WFD analysis/evaluation. In 1994, Airbus developed a Monte Carlo method for the WFD evaluation of the A300 fuselage, where random initial crack scenarios were determined by a lognormal distribution developed from simple coupon fatigue lives, and subsequent crack growth and failure was analyzed using fracture mechanics models [28]. In 2006, Lockheed Martin (LM) Aerospace developed an equivalent initial flaw size (EIFS) based Monte Carlo method to perform risk analyses of C-130 build-up structures [29]. On the subject of risk assessment as it might apply to managing MSD/MED situations, it was acknowledged that considerable advances had occurred in this area recently.

The effect of MSD is shown on figure 8.2 . In the presence of MSD adjacent to a lead crack the residual strength is reduced drastically.

The drop of the residual strength from the capability of the intact structure to the capability to withstand the design loads occurs in a much shorter time compared to the case of a local damage. Furthermore the crack growth for an MSD scenario is increased compared with the single crack.

Together with the reduced critical crack length this results in a significantly reduced crack growth period between the detectable and the critical situation. The Industry Committee on WFD has evaluated the experience o the participating manufacturers based on the results of large component and full scale fatigue tests as well as in service experience in order to identify the locations potentially susceptible to WFD. From this compilation of data each area was assessed for its susceptibility to WFD and was then characterized as either multiple element and/or multiple site damage.

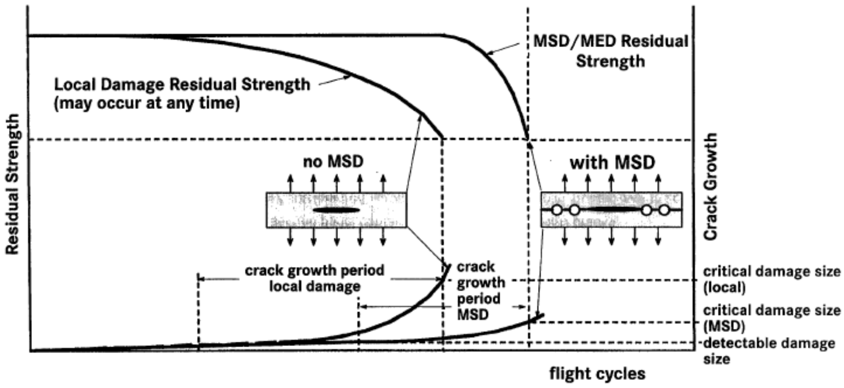

Figure 8.2: Effect of Multiple Side Damage

8.5.1 Structure susceptible to MSD/MED

Susceptible structure is defined as that which has the potential to develop MSD/MED. Such structure typically has the characteristics of multiple similar details operating at similar stresses where structural capability could be affected by interaction of multiple cracking at a number of similar details. The following list contains known types of structure susceptible to MSD/MED. [25]

Structural Areas:

- Longitudinal Skin Joints, Frames, and Tear Straps (MSD/MED)
- Circumferential Joints and Stringers (MSD/MED)

- Lap joints with Milled, Chem-milled or Bonded Radius (MSD)
- Fuselage Frames (MED)
- Stringer to Frame Attachments (MED)
- Shear Clip End Fasteners on Shear Tied Fuselage Frames (MSD/MED)
- Aft Pressure Dome Outer Ring and Dome Web Splices (MSD/MED)
- Skin Splice at Aft Pressure Bulkhead (MSD)
- Abrupt Changes in Web or Skin Thickness - Pressurized or Unpressurized Structure (MSD/MED)
- Window Surround Structure (MSD, MED)
- Over Wing Fuselage Attachments (MED)
- Latches and Hinges of Non-plug Doors (MSD/MED)
- Skin at Runout of Large Doubler (MSD) - Fuselage, Wing or Empennage
- Wing or Empennage Chordwise Splices (MSD/MED)
- Rib to Skin Attachments (MSD/MED)
- Typical Wing and Empennage Construction (MSD/MED)

Widespread fatigue damage (WFD), affecting large aging transport aircraft over 75,000 pounds, is being addressed by FAA rulemaking. The WFD concern relates to a scenario where a series of small fatigue cracks, in the absence of highly-reliable small crack detection techniques, could lead to unacceptable reduction in residual strength below the damage tolerance safety requirements. The objectives of a WFD program are to identify the primary structures susceptible to WFD, to predict the time the WFD is expected to occur and to establish additional maintenance actions (inspections and modifications/replacements) as necessary to preclude WFD. The maintenance requirements that were developed and became mandatory have created an extensive, but not unbearable, workload for operators of large transport category aircraft. However, it was felt that compliance with these additional maintenance activities were needed to ensure the continual structural airworthiness of these aircraft as they aged; the efforts of the working groups that developed these additional programs have been vindicated. These additional maintenance requirements were initially applicable only to 11 specific large transport category aircraft models.

8.6 Structural Repair Assessments

One of the ways is periodically updating the structural repair manuals (SRMs). Working through the repair assessment program, manufacturer task groups developed programs to address repairs that may affect the damage tolerance characteristics of the original airplane structure. For each model, structural repair manuals (SRM) assist the operator in ensuring that typical repair action maintains the airframe structural integrity. Repairs that are fallen out of SRM are handled by individually prepared and approved engineering drawings/sketches. These repairs have primarily focused on static strength and fail-safe aspects. If they are on primary structural elements, damage tolerance analysis would be required.

8.6.1 Repair assessments guidelines (RAG)

RAGs are airplane manufacturer's model-specific guidelines on how to perform the repair assessment. The fuselage is more sensitive to structural fatigue than other airplane structure because its normal operating stresses are closer to its limit design and more prone to damage from ground service equipment and corrosion than other structures and requires repair more often, therefore the repair assessment is currently limited to the external fuselage pressure boundary in figure 8.3. Uniformity/similarity of these repair assessment procedures is important to simplify operator workload.

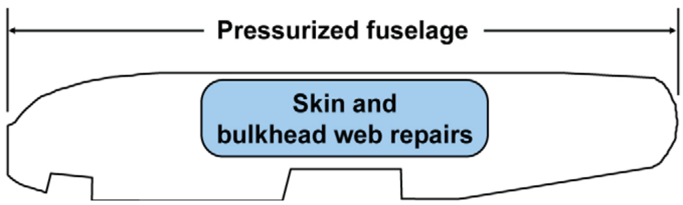

Figure 8.3: Fuselage Pressure Boundary Repairs

The objective of these guidelines can be summarised as follows:

- To provide a practical methodology in the form of guidance material to allow repairs to be evaluated by operators without complex analysis.

- To direct the repair evaluation process towards showing that an installed repair, which replaces the strength and durability of the original structure is found damage tolerant.
- To develop a repair evaluation process such that it will identify when and what supplemental inspections (threshold, interval, method, location) are required.

The SRMs and RAGs describes rationale for repair Categories A, B and C [1] as shown Figure 8.4.

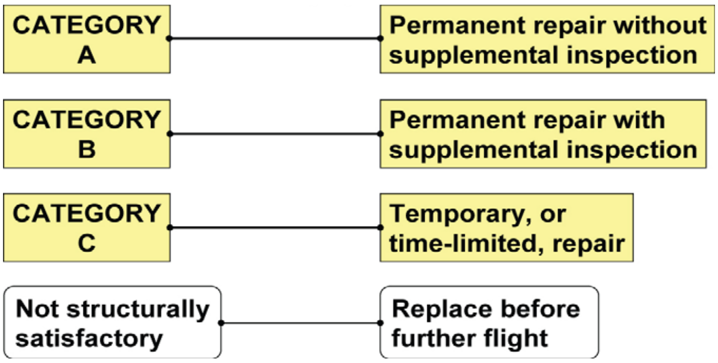

Figure 8.4: Repair Categorization

Category A: A permanent repair for which the Base Zonal Inspection (BZI) is adequate to ensure continued airworthiness (inspectability) equal to the unrepaired surrounding structure. The operator should demonstrate to the regularity agency that its maintenance or inspection program is at least as rigorous as the BZI.

Category B: A permanent repair that requires supplemental inspections to ensure continued airworthiness.

Category C: A temporary repair that will need to be reworked or replaced before an established time limit. Supplemental inspections may be necessary to ensure continued airworthiness before this limit.

There are two principal techniques that can be used to accomplish the repair assessment when using the manufacturer's RAGs.

The first technique involves a three-stage procedure. This technique could be well-suited for operators of small fleets. The second technique involves the incorporation of the RAGs as part of an operator's routine maintenance program.

This approach is well-suited for operators of large fleets and would evaluate repairs at predetermined, planned maintenance visits as part of the maintenance program. The first technique generally involves the execution of the following three stages [30] as shown figure 8.5.

Stage 2: The repair categorization is determined by using the data gathered in Stage 1 to answer simple questions regarding structural characteristics. Well-designed repairs in good condition meeting size and proximity requirements are Category A. The process continues for Category B and C repairs.

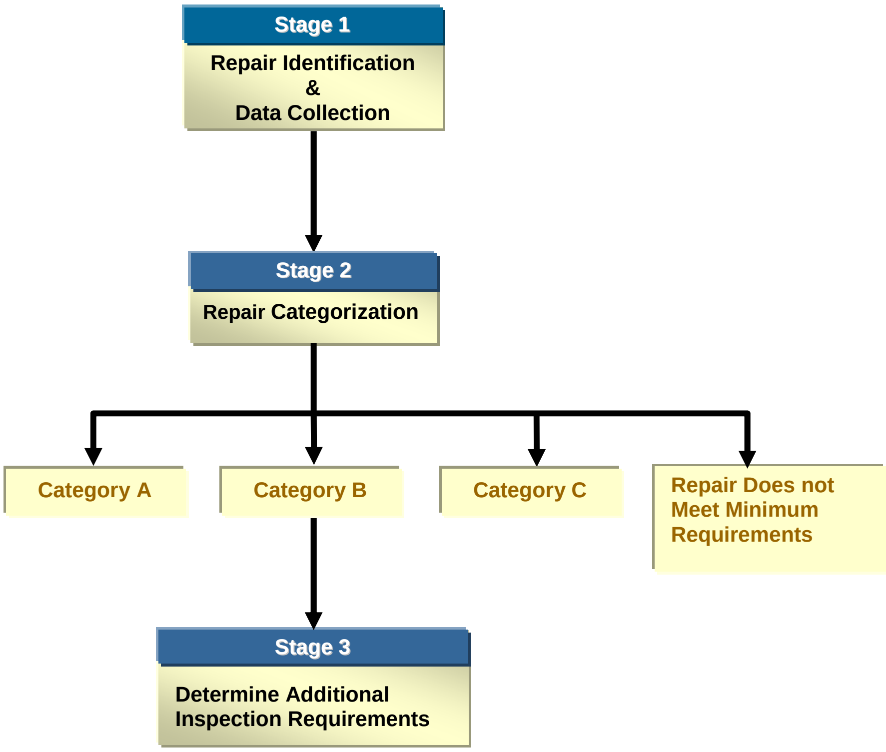

Figure 8.5: Repair Assessment Stages

Stage 3: The supplemental inspection and/or replacement requirements for Category B and C repairs are determined in this stage. Inspection requirements for the repair are determined by a simple calculation or by using manufacturer specific predetermined values incorporating the supplemental inspection requirements into the operators' maintenance program completes the repair assessment process.

8.6.2. Continued airworthiness of structural repairs

In the early 1990s, today the Airworthiness Assurance Task Force (AATF), now known as the Airworthiness Assurance Working Group (AAWG) was aimed/tasked with assessing the general condition of repairs on transport airplanes and developing recommendations concerning whether new or updated requirements and compliance methods for structural repair assessments of existing repairs should be instituted and made mandatory for several specific transport category airplanes. In 1996, a final report was released detailing the data they collected during two surveys of repairs done in 1992 and 1994 [31]:

<u>1992 Survey:</u> A survey of lower fuselage repairs on stored airplanes was conducted at aircraft storage and salvage locations in Mojave, CA and Amarillo, TX. The work was conducted by teams of engineers from FAA Aircraft Certification Offices, FAA Flight Standards Offices, operators, and original equipment manufacturers (OEMs). The teams surveyed 30 airplanes and assessed a total of 356 repairs.

<u>1994 Survey:</u> A survey of repairs on in-service airplanes was conducted at various operators' maintenance bases by only engineers from OEMs, which included Airbus, Boeing, Douglas, and Lockheed. The teams surveyed 35 airplanes and assessed a total of 695 repairs.

The scope of the studies was limited to an external visual observation of external lower fuselage plating repairs. The objectives were to gain first-hand observations of typical repairs, and to sample the numbers, types, proximities, and conditions of repairs. The quality of each repair was assessed based on AAWG repair criteria and with OEM size and proximity limits. Each repair was examined and classified into three categories [31]:

Category A: A permanent repair for which the baseline zonal inspections are adequate to ensure continued airworthiness (inspectability) equivalent to unrepaired surrounding original structure.

Category B: A permanent repair that requires supplemental inspections to ensure continued airworthiness.

Category C: A temporary repair that will need to be reworked or replaced prior to an established time limit. Supplemental inspections may be necessary to ensure continued airworthiness prior to this limit.

Table 8.1 [31] shows a breakdown of the models and numbers surveyed in 1992 and 1994. In all, a total of 65 airplanes (13:727, 9:737, 7:747, 3:DC-S, 10:DC-9, 4:DC-10, 9:A300, 2:L-1011 8:F-28) and 1051 repairs were assessed. Average of 16 repairs per airline. The study found that about 40% of repairs met the criteria for a Category A repair. The 60% that fell into Category B or C were primarily due to size and/or proximity criteria.

The overall goal was to develop a qualitative opinion of repairs as a safety concern. In general, the AAWG concluded there was a need for repair assessment evaluations; although no immediate safety concern was observed. They cited a need for the following,

(1) Updates to structural repair manuals and new guidance materials for repairs,
(2) OEM developed repair assessment procedures for existing and new repairs, and
(3) Repair assessments training's necessity for operator and regulatory agencies
In addition, the AAWG found the majority of repairs were on the fuselage pressure shell and older airplanes generally have more repairs. As a result of the work, the AAWG recommended certain regulatory changes that have led to implementation of the Repair Assessment Program [32,33].

Table 8.1: AAWG Fuselage Surveys Statistics

AIRPLANE MODEL	AIRPLANES SURVEYED (92/94/TOTAL)	REPAIR CLASSIFICATION		
		REPAIRS REQUIRING NO ADDITIONAL ACTION (CATEGORY A) (92/94/TOTAL)	REPAIRS REQUIRING SUPPIEMENTAL INSPECTIONS (CATEGORY B OR C) (92/94/TOTAL)	TOTAL REPAIRS SURVEYED (92/94/TOTAL)
727	6/7/13	39 /100/ 139	66/109/175	105 / 209/ 314
737	5/4/9	41 /17 / 58	49/66 I 115	90 /83/173
747	2/5/7	13 /37/ 50	32/130/162	45/ 167/ 212
DC-8	0/3/0	0/56/56	0/43/43	0/99/99
DC-9	6 /4/10	21 /37 / 58	32/16 /48	53 / 53 /106
DC-10	0/4/4	0/12/12	0/ 21 /21	0/33 / 33
A-300	9/0/9	17/0/17	18 I 0 118	35/0/35
L-1011	2/0/2	12 1 0 I 12	16 I 0 I 16	28/0/28
F-28	0/8/8	0/10/10	0/41/41	0/ 51 / 51
TOTAL	30 I 35 165	143 1 269 1412	213/426 I 639	356 1695 I 1051

There is a current project on the way. In comparison to the projects conducted in 1992 and 1994, the objective of the this new project is to determine the risk of structural repairs, alterations, and modifications (RAM) RAMs developing WFD. While the objective of the current project is more specific, the net effect is similar in that the results will be used to support decisions on rulemaking activities. Because the AAWG work provided a benchmark for earlier rulemaking activities, the initial goal of the project was to survey at least as many RAMs identified as being surveyed in the 1996 report [31]. It was anticipated that the bulk of the RAMs would be repairs, and most of those would be found on fuselage skins, as it has been the structure of most concern in terms of WFD occurrence. While the fuselage pressure boundary was the primary concern, other structures including wing and empennage were considered [34].

9. CONCLUSION AND RECOMMENDATIONS

Aircrafts, like humans, deteriorate physically during their long lives. As the airplanes ages, operator's workload increases. Accumulating stresses make it important to have regular checkups and, as the years go by, more frequent visits to specialists. Like geriatric medicine, aircraft aging science has become more proactive. Therefore, operators/airlines and Maintenance and Repair Organizations (MROs) should have qualified engineers on site, not only to detect and repair damage, but also to delay or even prevent the onset of certain problems. Also, by learning from the past, engineers can use the knowledge to recognize key factors, and potentially prevent another accident from occurring under similar circumstances, or for similar reasons, in the future.

As discussed in this study, damage tolerance assessment and design play a major role in todays aerospace industry, not only in the design of new structures and components but also their ongoing maintenance and support. Composite material usage on aircraft is increasing rapidly, due to its light weight and fatigue resistance. To study on damage tolerance assessments of composite aircraft structures would be a challenging problem, because of lack of practical standards and educational materials of relevance to industry applications

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APPENDICES

APPENDIX A : Types of Structure Susceptible to WFD [25]

APPENDIX A

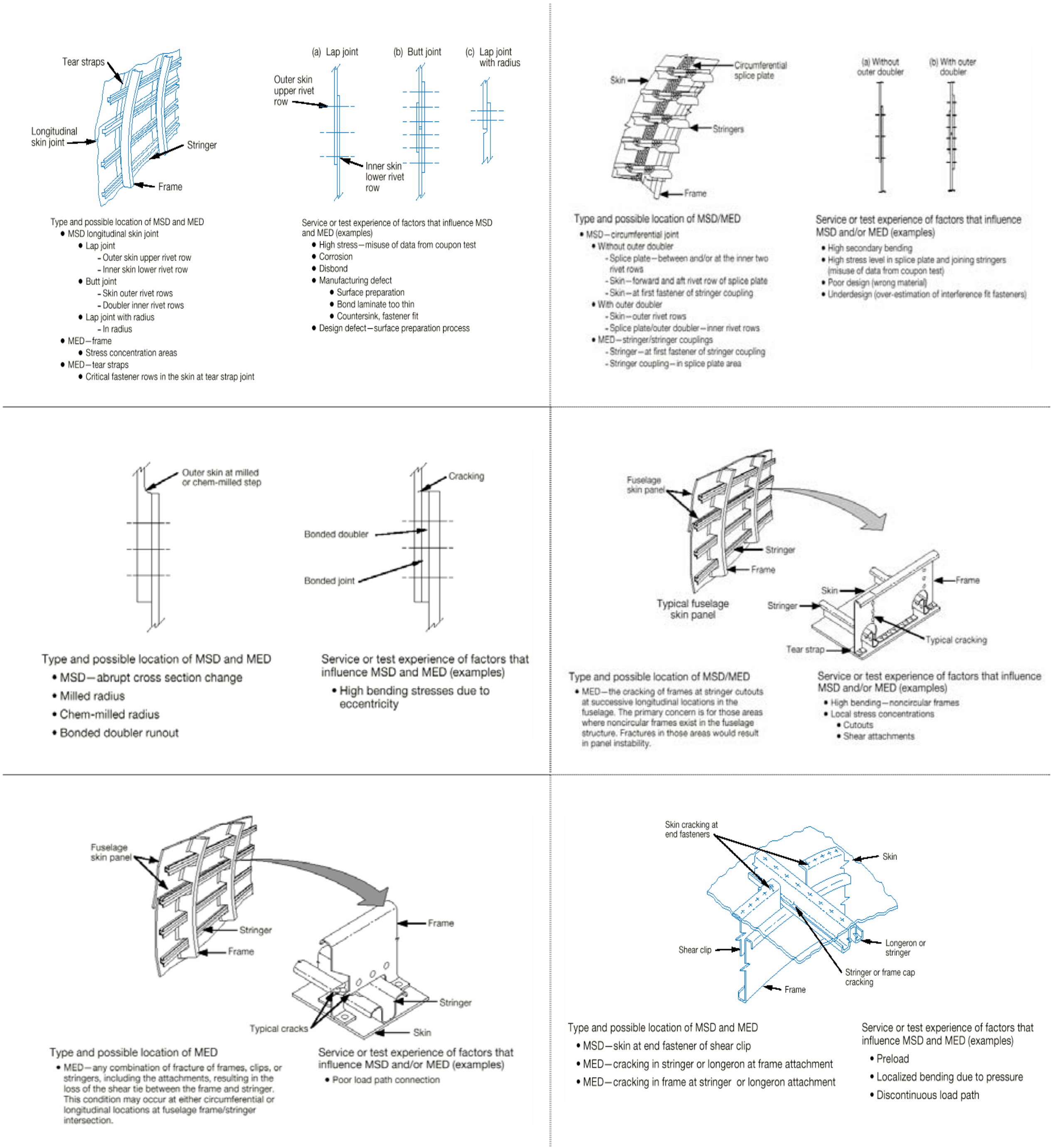

Figure A.1: Types of Structure Suspectible to WFD

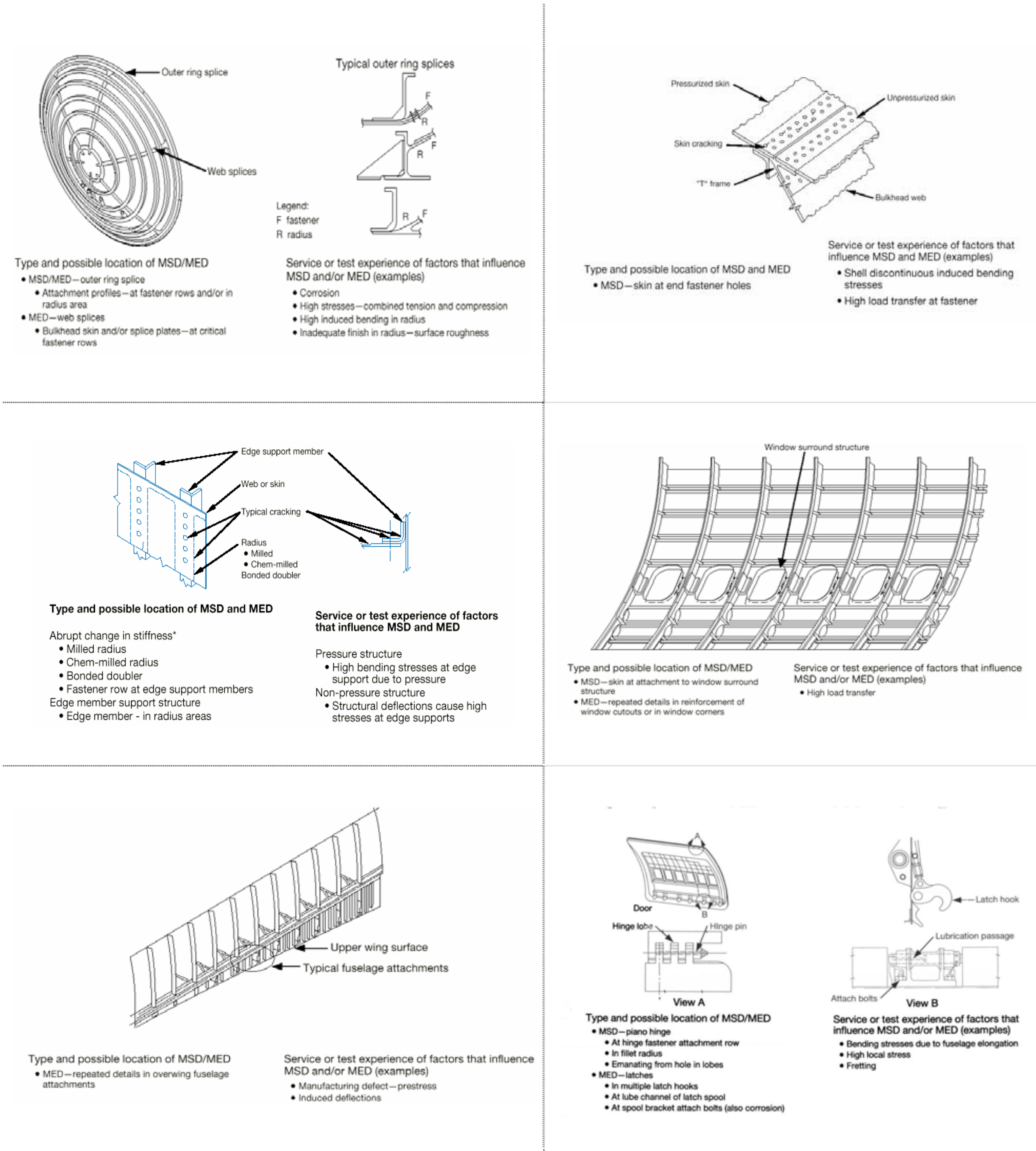

Figure A.1: (contd) Types of Structure Suspectible to WFD

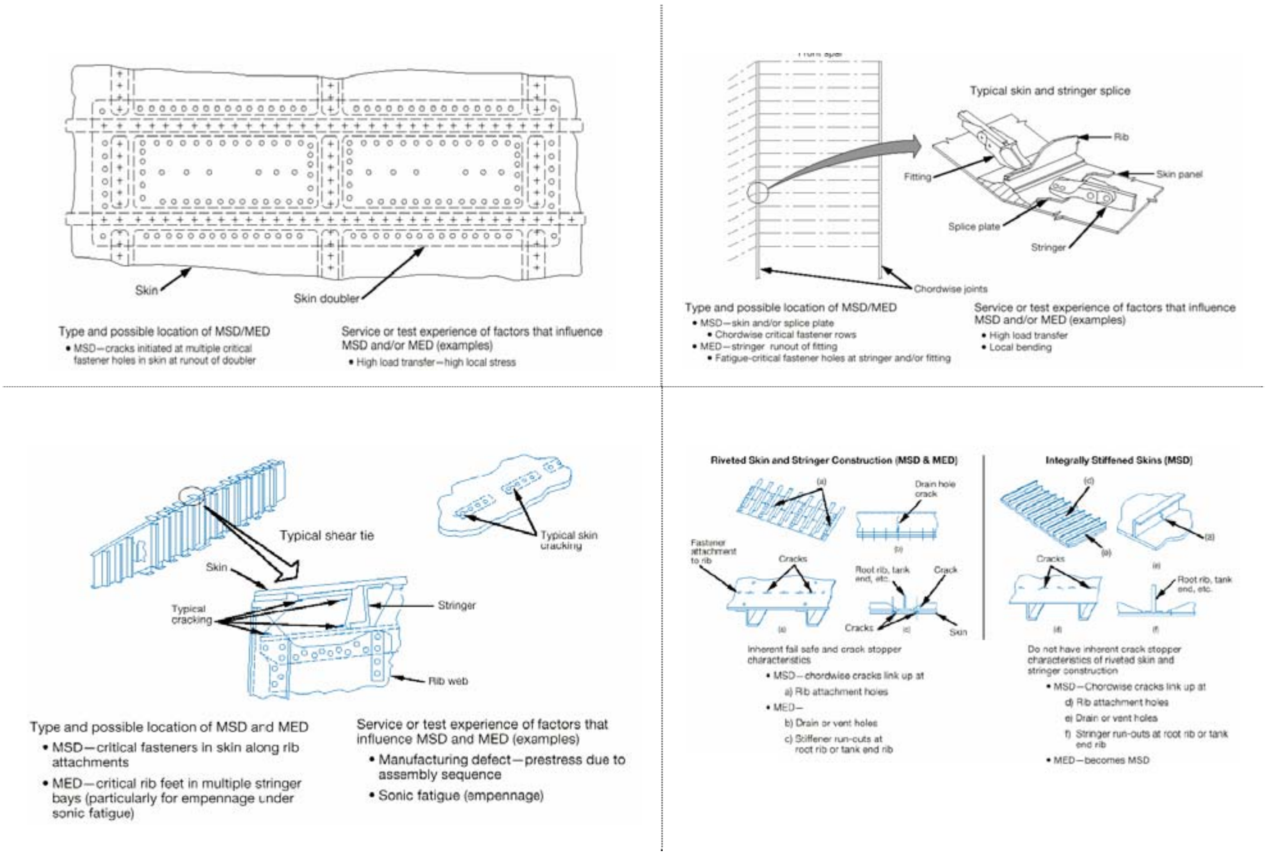

Figure A.1: (contd) Types of Structure Suspectible to WFD

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