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**CORROSION AND MULTIPLE SITE DAMAGE
IN RIVETED FUSELAGE LAP JOINTS**

by

Jason P. Scott, B.Sc.

A thesis submitted to
the Faculty of Graduate Studies and Research
in partial fulfilment of
the requirements for the degree of

Master of Engineering

**Department of Mechanical and Aerospace Engineering
Ottawa-Carleton Institute
for Mechanical and Aerospace Engineering**

Carleton University
Ottawa, Ontario
March 1997.

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“CORROSION AND MULTIPLE SITE DAMAGE
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Thesis Supervisor

Chair, Department of Mechanical and Aerospace Engineering

Carleton University
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ABSTRACT

As aircraft approach or exceed their original design lives, there is an increased risk of fatigue cracking at rivet holes. Fatigue cracks in fuselage lap splices tend to develop as multiple site damage (MSD). MSD can develop into long lead cracks relatively quickly, and can diminish the fail-safe characteristics of a fuselage. Lap joints are also affected by corrosion.

This thesis describes the results of an investigation into the interaction between corrosion and MSD in lap joints using a previously developed coupon specimen. Tests were conducted using two schedules: pre-corrosion followed by fatigue, and alternating corrosion and fatigue. The results were compared with baseline fatigue results. The pre-corrosion plus fatigue schedule reduced life to visible crack initiation by 38% over the baseline tests but produced non-uniform MSD. The alternating corrosion and fatigue schedule produced more severe corrosion and more uniform MSD, but did not reduce life to visible initiation as severely.

In order to assess the corrosion severity in the test specimens, several Non-Destructive Evaluation (NDE) methods were used to measure corrosion pillowing and sheet thickness loss during testing. These methods produced good qualitative agreement but showed differences on the absolute degree of corrosion. Some correlation between the degree of damage along the rivet row and the subsequent crack initiation pattern was seen.

In conjunction with this work, an investigation into splice construction and detail design was made to determine the key factors affecting joint life. The rivet-and-hole interaction and the effect of secondary bending within the splice were found to be critical factors.

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UNITS

In keeping with the practices of the North American aerospace industry and the Institute for Aerospace Research (NRC), the Imperial system of units is employed in this thesis. SI equivalents are provided in brackets within the text, and in the figures where practical. The following conversion factors are useful:

$$1 \text{ inch} = 25.4 \text{ mm}$$

$$1 \text{ foot} = 0.3048 \text{ m}$$

$$1 \text{ lb}_f = 4.4482 \text{ N}$$

$$1 \text{ ksi} = 6.8948 \text{ MPa}$$

1. INTRODUCTION

There were 3 717 jet airliners over the age of 20 years worldwide as of 1 January, 1996. an increase of 328 over 1995 [1]. It is estimated that by the year 2000 there will be 4 474 airliners, manufactured by Boeing, Douglas and Lockheed alone, over the age of 20 [2]. As these aircraft age, several important issues are emerging. Two of these are multiple site damage (MSD) and corrosion, which were identified as the principal causes of the 1988 in-flight fuselage failure of an Aloha Airlines aircraft [3]. MSD has been described by Sampath and Broek as follows [4]:

MSD is caused by fatigue and is manifested in fuselage lap-joints in older commercial airplanes. The phenomenon is characterised by small, longitudinal cracks emanating from successive rivet holes. Their probability of detection associated with conventional Non-Destructive Inspection (NDI) methods is relatively low; yet, they can potentially link-up domino fashion to cause an unstable longitudinal fracture which may override the built in fail-safe design.

Corrosion weakens the structure, making it more susceptible to fatigue damage. Fatigue damage and corrosion, along with inspection and maintenance practices, are receiving the attention of the aircraft industry, the research community and the regulatory authorities. and have been the subject of several international symposia and publications [e.g. 5,6,7,8,9,10].

While extensive research into fatigue damage of lap joints has been conducted in the eight years since the Aloha accident, the interaction of corrosion and fatigue damage remains largely uninvestigated, as do the means of characterising the damage. These deficiencies have been recognised by Hoeppner et al. [11]:

The Aloha accident illustrates how incomplete reporting can heavily influence research trends. In this case, the importance of corrosion and the possibility of fretting as causes of the accident were not vigorously pursued. Instead, the emphasis shifted to the effect: multiple-site damage.

This thesis is part of an ongoing research effort [12,13]. The thesis objectives are:

1. to identify the fundamental causes of MSD cracking in lap joints;
2. to investigate the role of corrosion in MSD crack initiation and growth, focusing on the simulation of aircraft corrosion as it occurs in service; and
3. to identify existing and emerging methods of evaluating and characterising the resultant damage, preferably without disassembly of the joint.

The following chapters begin with an overview of aging aircraft issues leading to, and the concerns raised by, MSD. The construction and behaviour of typical pressurised fuselage lap joints is analysed in Chapter 3 in order to illustrate the mechanisms which cause MSD crack initiation and growth. Chapter 4 investigates the corrosion of aircraft and looks at simulation methodologies, including previous research efforts. Finally the requirements and current capabilities of non-destructive evaluation (NDE) technologies are presented in Chapter 5.

Having established the background, Chapter 6 defines the problem, identifies specific research areas and describes the experimental research program undertaken. Chapter 7 details the equipment and experimental procedures used. Results and discussions are presented in Chapter 8 with conclusions following in Chapter 9.

2. AGING AIRCRAFT AND MULTIPLE SITE DAMAGE

2.1 Aging Aircraft: An Overview

2.1.1 History

Aging aircraft are a major concern in the aircraft industry today because a large portion of commercial and military fleets are flying well beyond their original design life in terms of both flight cycles and flying hours. On April 28, 1988, this problem was brought to public attention when an Aloha Airlines Boeing 737 flying at 24 000 ft (7 300 m) suffered a structural failure in which an 18 ft (5.5 m) long section of the cabin above the floor line between the cockpit and wing was torn from the aircraft with the loss of one life (Figure 2-1) [14]. The aircraft landed safely.

It was an early B-737, production number 152, and was operated by Aloha for its entire life. From delivery, in April 1969, it averaged 13 flights per day, typically only 25 minutes long, flying in the salty, hot and humid environment of the Hawaiian islands. Most flights were pressurised but the maximum operating pressure differential of 7.5 psi (52 kPa) was not always reached.

Aloha's fleet of 11 B-737s was accumulating flight cycles at twice the rate for which it was designed. Originally conceived for an economic service life of 20 years, 75 000 flight cycles and 51 000 flight hours, the accident aircraft was on its 89 681st flight in only its 35 496th hour making it the second oldest aircraft in the world in terms of flights. Two other Aloha B-737s had accumulated 90 051 (the world wide fleet leader) and 85 409 flights at the time of the accident. These aircraft were grounded and subsequently scrapped, along with the accident aircraft, after inspection revealed extensive corrosion throughout their airframes.

The accident report attributed the failure to rapid and catastrophic crack growth caused by lap joint multiple site damage (MSD) in the upper rivet row at stringer S-10L [3]. Figure 2-2 shows MSD found elsewhere on the Aloha aircraft during the post-accident inspection. The fatigue cracks had started at knife-edges in the countersunk rivet holes. The failure was aggravated by lap joint and tearstrap disbonding, corrosion and poor maintenance.

Disbonding of lap joints was common in early production B-737s. The lap joint, shown in Figure 2-3, consists of two 0.036 inch (0.91 mm) thick fuselage skins 'cold' bonded together using an epoxy impregnated 'scrim' cloth and riveted with three rows of countersunk rivets. Cold bonding, also employed on early B-727s, B-747s and Lockheed L-1011s, was intended to carry the fuselage hoop loads with the rivets providing a back-up load path only. The reliability of the cold bond was poor, however, and disbonding and corrosion were identified early in service. Disbonding increased the load on the rivets while the hygroscopic adhesive absorbed moisture leading to corrosion. Boeing had long been aware of disbonding, corrosion and multiple fatigue cracks in B-737 lap joints, having released service bulletins on the subject as early as 1972. From line number 292, it adopted a design with additional hot-bonded doublers but unbonded, primed and sealed joints. A similar construction is still being used on current production aircraft including B-757s and B-767s.

Although the Aloha aircraft was unusual in terms of flight usage and environmental exposure, the accident cannot be attributed to these factors alone. Extensive multiple cracks had already been found on three B-737s with more than 45 000 flights, but unlike the accident aircraft, these exhibited low corrosion [18]. Disbonded tearstraps and almost 30 feet (9.1 m) of lap joint cracking were found.

Nor has MSD been confined to the Boeing 737. Recently a 39.4 inch (1.0 m) long crack in the forward upper fuselage of a DC-9-30 was found during routine visual inspection

[15]. Corrosion was also evident. The aircraft was built in 1969 and had completed 82 300 flights for 70 000 hours.

Aloha's was not the first accident to draw attention to general aging aircraft issues. There were a number of earlier accidents, some of which are briefly described here:

1. On April 14, 1976, a Hawker Siddeley HS 748 crashed in Argentina after the starboard wing separated in flight [16]. The aircraft, built in 1962, had flown 25 759 hours. The investigation concluded that cracking from rivet holes in the wing skin and from an adjacent reinforcing plate near a cut-out had grown to a length of 36 inches (0.91 m) before failure. Aggravating this situation was an imprecise manufacturer's inspection program for the area concerned, making it possible for cracks to go undetected. Subsequent inspection of sister aircraft found cracks in similar locations, including a 27.6 inch (0.70 m) crack in one aircraft.
2. On May 14, 1977, a Danair cargo B-707 crashed near Lusaka in Zambia after the starboard tailplane failed in flight [17]. The aircraft, built in 1963, was believed to have flown less than 50 000 hours and 15 000 flights. The tailplane was recovered and found to have suffered a fatigue fracture. Although designed in accordance with fail-safe philosophy, the crack location was not easily inspectable. There had been no prior history of problems in this location but examination of the world B-707 fleet revealed damage in other aircraft, some with as few as 26 000 flight hours. In fact Boeing had not found a crack in a high-time B-707 it had bought back for fatigue analysis; but inspection after Lusaka revealed it.
3. In August 1981 a B-737-200 suffered a belly-section failure which led to mid-air disintegration over Taiwan. This aircraft had a history of belly bilge corrosion but had completed only 33 000 flights [3,18].

4. In 1987 a 43 year-old DC-3 suffered a wing failure. Although this accident received little media and public attention, it started the Canadian regulatory response to the aging aircraft situation in the year before the Aloha accident [19].

These few examples illustrate the scope of the problem: it affects many types of aircraft and is not limited to particular structural components or design features. These examples also point out the critical role of inspection.

At this point it is appropriate to define aging aircraft. The economic design life of an aircraft is based on the anticipated mix of missions and is defined in terms of operational life in years, number of flight hours and number of flight cycles. Individual aircraft use up these three quantities at different rates. In view of fatigue, life in flight cycles has become the standard measure of age for pressurised fuselages; but it must be remembered that corrosion is a function of time, not flights. For the purposes of this thesis, the following definition will be used [after 20]:

An aging aircraft is one which requires a change to maintenance programs or is subject to operational restrictions in order to fly beyond its economic design life.

Changes in maintenance can include changes in inspection intervals, mandated inspection of particular features, and mandated pre-emptive repairs and cleaning procedures.

2.1.2 Aircraft Design Philosophies and Regulatory Responses

The first generation of pressurised jet transports was designed to *safe-life* principles; that is, the aircraft or component is ‘guaranteed’ for a designated life-time based on analysis, demonstration tests and a suitable scatter factor. At the end of this life the item in question is automatically retired.

Following the ‘infant-jet’ Comet pressure cabin failures of 1954 [17] the *fail-safe* philosophy was adopted. Unlike safe-life design, fail-safe design anticipates likely crack scenarios and provides multiple load paths and crack arrest features to maintain structural integrity; that is, allowing structure to ‘fail safely’. Since the design relies on redundancy and inspection, it must ensure that crack growth is slow with easily achievable inspectability and detectability by inspectors who know where to look.

Federal Aviation Administration (FAA) regulations require aircraft to demonstrate fail-safety by sustaining 80% of their limit loads in spite of a complete or partial failure of any single structural element. For instance, testing of the Boeing 737 showed that the fail-safe design, which includes longitudinal and circumferential tear straps every 10 inches (0.25 m) could sustain a 40 inch (1.02 m) fuselage crack, that is, one extending across two bays. The presence of the frames redirected the crack from the longitudinal direction into the circumferential direction causing the skin to flap and leading to controlled decompression.

The introduction of fail-safe principles was something of a revolution in aircraft design; 1978 brought an evolution. A number of accidents in the 1970’s had revealed that the methods, thresholds and frequency of inspection in use could fail to detect structural damage before catastrophic failure [2]. With Federal Airworthiness Regulations 25.571 [21], the FAA introduced the *damage tolerance* concept as a means of rigourously implementing the fail-safe philosophy. It required that consideration be given to damage growth characteristics at multiple sites and that an inspection program be designed to ensure that damage would be detected before the aircraft residual strength dropped below regulatory fail-safe limits. Existing aircraft were reassessed in order to comply.

In 1977, following the Danair B-707 accident at Lusaka, Boeing introduced a Supplemental Inspection Program (SIP) to give additional attention to the B-707 tail section. SIP’s were subsequently adopted for many aircraft, sometimes applying to entire

fleets and sometimes to so-called 'lead-the-fleet' aircraft - those with the highest flight hours and cycles. In extreme cases a SIP is given legal force when the FAA reissues it as an Airworthiness Directive (AD).

The industry and regulatory responses to the Aloha accident are covered in detail by Wakeman [13] but are summarised here. The accident stimulated several aging aircraft programs focused on ensuring the continued airworthiness of large transport aircraft [22]. The National Transportation Safety Board (NTSB) investigation recommended that all future turbojet transport category aircraft undergo full-scale structural fatigue testing to a minimum of twice the projected economic service life before certification. Further, all current aircraft types in this category were to be tested and the manufacturers required to identify MSD-susceptible structures and adopt appropriate inspection programs.

As a result of the Aloha and other incidents, Boeing, Douglas, Lockheed, British Aerospace and Airbus Industrie have initiated structural audits for their older aircraft. Responsibility for continuing airworthiness and structural integrity of the world's aging fleet is thus shared among the manufacturers, operators and regulators.

The United States Air Force (USAF) introduced damage tolerance requirements in the early 1970's in response to several accidents involving supposedly fail-safe components. These accidents demonstrated that a structure cannot be fail-safe without an inspection program [23]. The USAF maintains a large fleet of transport-category aircraft and plans to keep existing Lockheed C-5A and C-130 aircraft in service to 2010 at which point they will be 40 to 50 years old. The KC-135 (a military version of the B-707), C-5A and C-141 have experienced extensive MSD and corrosion in their airframe lap splices, leading to significantly degraded fail safety [23].

The KC-135 fleet averaged 34 years of age in 1994, well beyond its design life in years [24]. Aircraft usage, however, averages only 300 cycles per year so that fleet life is well

below the design life in cycles. Due to fiscal constraints the USAF plans to operate some of these aircraft to the year 2040. This is equivalent to keeping World War I era aircraft in regular operational service today!

2.1.3 Status of Aging Aircraft

Boeing's fleet includes the greatest variety and number of high-time aircraft, many of which are flying well beyond their design lives. The B-737 is the best-selling jet in history with over 1500 in service of which 681 are 15 years old or more. Although Boeing's aging aircraft have received the most attention, other manufacturers are affected. Table 2-1 shows the status of the more common commercial transports, 15 years old and more, as of 1 January 1996, with respect to their original design lives. Note that most manufacturers have conducted full-scale tests in beyond design life to ensure that test flights exceed the fleet leaders. Note that the fleet leaders in terms of flight cycles, flight hours and years are usually different aircraft.

Examples of several aircraft have exceeded their design lives. In fact, almost half the Douglas fleet has exceeded its original design life with the median age for the DC-8 and DC-9 fleets being 24 and 20 years respectively in 1991. In 1997 the Boeing 757 and Airbus A310 will join this list of aircraft aged 15 years or greater. The Concorde is a special case since it flies at much greater altitudes than most commercial jets, which imposes a higher pressure differential on its fuselage. Both operators (British Airways and Air France) plan to operate the aircraft until about 2015.

Table 2-1: Design Life and Fleet Status of Some Aging Aircraft; 1 January 1996. [1]

Aircraft Type	Units Aged 15+	Service Entry (Year)	Design Life (cycles)	Fleet Leader	Design Life (hours)	Fleet Leader	Design Life (years)	Fleet Leader
Boeing 707	143	1958	20 000	34 792	60 000	89 920	20	38
Boeing 727	1315	1964	60 000	73 587	50 000	77 640	20	33
Boeing 737	681	1968	75 000	89 039	51 000	81 458	20	29
Boeing 747	452	1970	20 000	33 046	60 000	99 238	20	27
Boeing 767	5	1982	-	28 674	100 000	53 419	40	15
Douglas DC-8	269	1959	50 000	48 146	100 000	87 744	20	38
Douglas DC-9	783	1965	105 000	100 596	75 000	81 383	20	31
Douglas DC-10	314	1971	42 000	37 162	60 000	93 362	20	26
Douglas MD-80	73	1980	-	42 168	30 000	47 523	-	16
Airbus A300	137	1974	36 000	35 844	60 000	48 163	20	23
Lockheed L-1011	168	1972	115 000	33 779	210 000	74 437	20	25
Concorde	13	1976	24 000	6 932	45 000	19 166	12-15	21

2.2 Terminology

Before proceeding further, several terms pertaining to aircraft damage and cracking will be defined. Crack development is conveniently divided into two phases: crack initiation and crack growth. For the purposes of this thesis, the crack initiation phase will be described using the terminology introduced by Wakeman [13]:

Crack Nucleation: the coalescence of micromechanical damage to form a small crack under the rivet head.

Crack Initiation: the visual occurrence of the crack tip emerging from under the countersunk rivet head.

Small Crack Growth: the period between nucleation and initiation. For countersunk rivets in thin skins a crack has propagated at least 0.040 inches (1.02 mm) by the time it becomes visible, and has possibly tunnelled even further (Section 3.6).

To these three definitions will be added the following:

Crack Lifetime: the number of cycles for a crack to grow from nucleation to failure of the component.

Crack Detection: the point at which a crack is detectable by any of several Non-Destructive Evaluation (NDE) methods. This leads to the terms visual detection, ultrasonic detection and so forth. Note that visual detection therefore corresponds to crack initiation.

Definitions of aircraft fatigue damage were produced by the Airworthiness Assurance Working Group (AAWG) Industry Committee on Widespread Fatigue Damage [25] some four years *after* the Aloha accident (which illustrates the complexity of the issues faced

when dealing with aging aircraft). These definitions, which are illustrated in Figure 2-4, will be adopted for the purposes of this thesis:

Widespread Fatigue Damage (WFD or WSFD): The presence of fatigue cracks at multiple sites of the airplane structure such that the interaction of these cracks degrades the damage tolerance capability of the structure more than any one crack acting independently.

Multiple Site Damage (MSD): Simultaneous development of fatigue cracks at multiple sites in the same structural element, such that fatigue cracks may coalesce to form one large crack.

Multiple Element Damage (MED): Simultaneous development of fatigue cracks in similar adjacent structural elements, leading to an interaction of these cracks.

For the remainder of this thesis, MSD will refer to the occurrence of several cracks at adjacent and collinear rivet holes *in a fuselage lap joint*. The terms lap splice, lap joint and skin splice are used interchangeably.

2.3 Concerns Raised by Multiple Site Damage

MSD usually appears first in lap joints which have become disbanded, corroded or are located in higher than normal stress areas, such as near windows and floor beams. It can be confined to one frame-bay, or form in several adjacent bays. The cracks are often of similar length at each side of several rivets across the middle of a bay. An example is shown in Figure 2-5. The series of small undetectable cracks can form several small lead cracks which may link-up to cover one or more frame bays within months or even weeks of normal operational use leading, potentially, to fuselage failure, as was seen in the case of the Aloha accident.

It is this growth which constitutes the threat posed by MSD. The situation is illustrated in Figure 2-6. The threat can manifest itself in two ways:

1. MSD can compromise fuselage residual strength and reduce critical crack length, even without link-up (Figure 2-7). In particular, subcritical cracks in a frame bay adjacent to one containing a long lead crack may prevent crack arrest at the frame, lead to premature failure and/or prevent a controlled flapping failure.
2. Many undetectable cracks within one frame-bay can coalesce into a long lead crack between inspection intervals to form a lead crack spanning one or more bays. The damage can occur before detection because MSD crack lifetimes are much shorter than for individual cracks, due to crack interaction.

At the time many of the now-aging aircraft were designed, the possibility of widespread MSD was not considered, in part because MSD was unknown and in part because small cracks were considered insignificant. The fail-safe philosophy assumed that the structure adjacent to damage was itself *undamaged*; but the development of MSD invalidates this if neither multiple load paths nor crack arrest features can be depended upon to provide sufficient integrity. As a result the responsibility for assessing the health of aging aircraft falls to inspection. Inspection techniques must detect the small crack sizes characteristic of MSD and/or the length of inspection thresholds and intervals must be reduced.

2.4 Summary

The expense of new aircraft combined with the growth in air travel and industry deregulation is encouraging the extended use of many aircraft types, increasing the average age of the world fleet. Further, more and more second- and third-hand aircraft are passing into the hands of third-world operators where compliance with regulations is not being stringently enforced by local airworthiness authorities.

The Aloha B-737 must, in retrospect, be seen to have been a prime candidate for MSD and corrosion but the realities of fatigue suggest that damage could strike younger aircraft experiencing much less severe operational use [26]. The accident showed that longitudinal cracks are not always arrested or turned into the circumferential direction by the presence of tearstraps and/or frames.

MSD is especially troubling because it is not readily detectable and is exacerbated by cracks in adjacent structural elements, by lap-splice and tearstrap disbonding and by corrosion. The effect of MSD on residual strength is one of the most critical issues facing aging aircraft.

3. RIVETED FUSELAGE LAP JOINTS

This chapter reviews lap joint design and focuses on the features which lead to cracking in general and MSD in particular. The review will proceed from a brief overview of fuselage features to a detailed investigation of the effect of the rivet and hole in a joint. The emphasis is on riveted lap joints in commercial transport aircraft.

3.1 General Design Features

3.1.1 Fuselage Design

The construction of modern pressurised commercial aircraft has evolved over several decades to a fairly standard form. The fuselage is typically a semi-monocoque design consisting of a cylinder stiffened internally with circumferential frames and longitudinal stringers. It is subjected to large concentrated reaction forces from the wings, landing gear and empennage. The frames and stringers carry the normal loads, maintain fuselage shape and support the skin. The skin carries shear loads, contains internal pressurisation and plays an aerodynamic role externally. A typical fuselage is shown in Figure 3-1. Stringer pitch is typically 6 to 10 inches (150 to 250 mm) and frames are usually spaced 20 inches (508 mm) apart.

The skin is constructed of large sheets of aluminum alloy, typically 2024-T3, fastened together with rivets in one of several ways (Figure 3-2). At circumferential joints, the sheets are generally butted together with the aid of an internal doubler. This assembly may be fastened to the circular frame beneath (typical Boeing construction) or may stand alone between two adjacent frames (typical Douglas construction). At longitudinal joints the sheets may also be butted together but more usually are overlapped, with or without doublers, and riveted to the stringer beneath.

The skin sheets, which are as large as mill sizes permit to minimise the number of splices, are fastened directly to the stringers. In many aircraft they are also fastened to the frames, which have cutouts through which the stringers pass. Often a doubler called a tearstrap is bonded to the skin at the frame location to assist the frame in slowing or arresting longitudinal cracks. Some aircraft have a floating frame in which the skin is attached only to the stringers which are, in turn, fastened to the frames with shear clips. In these cases tearstraps are typically attached to the skin at the frame locations as a means of crack arrest. Additional tearstraps are sometimes included at the mid-bay location, between frames.

3.1.2 Typical Joint Designs

Before beginning a detailed study of joints, a brief survey of typical longitudinal joint designs used by the manufacturers will be made to illustrate the major design features, variables and terminology.

Boeing uses the single shear lap joint almost exclusively for the longitudinal joints in its aircraft. Figures 3-3 and 3-4 show typical B-737 construction as an example. The joint consists of two sheets fastened with three rows of rivets, typically using 1 inch (25 mm) spacing both longitudinally and circumferentially. A 'tophat' section stringer is attached, invariably at the middle rivet row. Narrow body aircraft use 0.036 inch (0.91mm) and 0.040 inch (1.02 mm) skin extensively with 5/32 inch (4.0 mm) diameter countersunk rivets. Widebody aircraft generally use much heavier gauges. Along the belt line heavier gauges are used to attach to the window frame forgings.

In all B-737s, the skin is reinforced with a rectangular grid doubler, produced by hot-bonding a second sheet to the main skin and chemically milling out the 'waffle' pattern (Figure 3-3) [3]. It doubles the thickness locally at stringers and forms tearstraps at the frames and midbay points. In early B-737s, those with cold-bonded joints (line numbers 1-291), the lap is formed by two single thicknesses of skin (Figure 3-4). When Boeing

switched to unbonded joints (with line number 292), the outer skin thickness was doubled at the joint by extending the existing bonded waffle doubler into this area. The additional thickness was added to eliminate the knife-edge countersink in the outer skin since rivets had replaced the bond as the means of load transfer.

Joint construction similar to that used in the B-737 is found in the B-727 but with detail differences. Smaller rivet heads are used and the stringer has a slightly different profile. Bonded circumferential tearstraps replace the waffle-doubler, eliminating doublers beneath the stringers. Construction of joints in B-747s and newer Boeing aircraft are also similar.

Douglas uses single and double shear butt splices extensively (Figure 3-5) in addition to lap splices. It is the only manufacturer to employ finger- or scallop-doublers, which are intended to reduce load-transfer at the first rivet row, to improve fatigue initiation life and to ensure skin crack detectability [27]. Douglas's design results in a more complex joint: the standard single-shear DC-9 joint (Figure 3-6) has five components, including the stringer, as compared with three in the B-737. The DC-9 seems to have better fatigue resistance than ordinary single shear lap joints, although it must be noted that hoop stress in the DC-9 is among the lowest in commercial aircraft.

Airbus Industrie also uses single shear laps (Figure 3-7) but with some design differences from those used by Boeing and Douglas. In constructing the widebody A300 only two rivet rows are used [28]. The skin is 0.063 inches (1.6 mm) thick and 3/16 inch (4.8 mm) diameter titanium rivets are spaced 0.87 inches (22 mm) apart. An asymmetrical-section stringer is attached to the upper rivet row. Bonded doublers 0.024 inches (0.6 mm) thick are used in several combinations. In the A340 (a newer widebody), three rivet rows are used with 0.94 inch (24 mm) spacing in 0.087 inch (2.2 mm) thick sheet [29].

The Lockheed L-1011 includes single shear lap joints similar to Boeing joints but with larger 3/16 inch (4.8 mm) diameter rivets and the stringer attached at the upper row [30].

Figure 3-8 illustrates two such lap joints plus a butt joint used at the crown of the fuselage which include a combination of doublers on both sides of the joint.

3.1.3 Effects of Fatigue Loading and Fuselage Construction

Above about 8000 ft (2440 m) commercial aircraft cabins are pressurized for passenger comfort. As the aircraft climbs beyond this height to its cruising altitude, often 35 000 to 45 000 ft (10 700 to 13 700 m), internal pressure is maintained at the 8 000 ft (2 400 m) altitude level. At cruising altitude, therefore, a pressure differential exists between the cabin and external environment. For instance, at 35 000 feet an aircraft pressurised to 8 000 ft is subject to a pressure differential of 7.4 psi (51.0 kPa).

Every time an aircraft is pressurized the skin and structural joints are loaded. This load is applied gradually from 8 000 ft as a function of the rate of climb. Pressure difference changes due to change in cruise altitude and mild turbulence can and do occur during long flights but are relatively small. More extreme changes of pressure, due to high turbulence, abrupt flight maneuvers, or abnormally high descent rates, are estimated to happen less than once in the life of an average aircraft [27]. While such an event must be accounted for in the design of an aircraft, the effect on fatigue life of an average aircraft will be small.

In an unreinforced cylinder, pressurisation induces the well-known hoop stress of $\Delta p R/t$ and longitudinal stress of $\Delta p R/2t$ (where Δp is the pressure differential, R is the fuselage radius and t is the skin thickness). The operating $\Delta p R/t$ stresses for commercial aircraft range between 10.1 ksi (69.6 MPa) for the narrowbody DC-9 and 16.2 ksi (112 MPa) for the widebody B-747 [33]. Because these stresses are proportional to radius, widebody aircraft require thicker skins. Reinforcement by frames and stringers reduces the stresses somewhat. Hoop stress across the middle of a framebay is relatively flat and ranges between 75% and 90% of the $\Delta p R/t$ stress for stiff frame-skin connections and floating

frames, respectively [29]. Hoop stress falls off near the frames and tearstraps which carry some of the load [31]. Typical hoop stress profiles are shown in Figure 3-9.

The longitudinal stress distribution is similar; flat between the stringers and falling off near them. Tearstraps and doublers which locally increase the skin's thickness will have a similar but smaller effect. Strain gauge measurements on a B-737 have shown the longitudinal stress in the centre of a frame bay to be about 60% of the hoop stress [32], compared to the value of 50% found in an unreinforced cylinder.

At a lap joint the skin thickness is doubled (more if doublers are used) which will lower the average longitudinal stress, as will the attached stringer. This lower longitudinal stress is expected to lengthen the life to crack nucleation by reducing the stress concentration in the hoop direction at the fasteners. Flat panel experiments at the National Aerospace Laboratory (NLR) of the Netherlands have studied the effect of this longitudinal stress on lap joints [29]. They found similar crack nucleation and growth characteristics with and without longitudinal loading, but there was insufficient data to firmly conclude that inclusion of longitudinal loading yielded longer fatigue lives. Nevertheless, the longitudinal stress was not thought to significantly affect joint behaviour or fatigue performance.

In addition to the hoop and longitudinal stress variations, the frames and stringers produce pressurisation pillowing, in the following way: under pressure, the skin expands radially outwards but is restrained at the stringers (and frames, if the skin is fastened directly to the frames) which leads to local out-of-plane bending at these stiffeners. The bending stresses are tensile on the inner surface and compressive on the outer. The skin-to-stringer connection resembles an elastically built-in edge with the degree of elasticity a function of the stiffness of the fastening method. At lap joints, stringer stiffness will influence the bending within the joint. This effect will be investigated in Section 3.5.

The two outer rivet rows are labelled the upper and lower critical row where upper and lower refer to the on-aircraft position. Pressurisation pillowing, therefore, induces tensile stresses on the faying surface at the upper rivet row which will add to the hoop stress. The hoop stress and out-of-plane bending induced by pressurisation are illustrated in Figure 3-10. Note that the pressure load is resisted by forces and moments at the stringers.

Another component of skin stress is that due to fuselage curvature. Skins can either be pre-formed to the appropriate radius or flat skins can be riveted in place. In the latter case compressive stresses are induced on the inner surface and tensile stresses on the outer surface. The stresses are directly proportional to skin thickness and inversely proportional to fuselage radius. Most older Boeing aircraft used flat skins which will spring off when the rivets are removed. For a narrowbody fuselage with 0.036 inch (0.9 mm) thick skin on a 72 inch (1830 mm) radius, these stresses will be about 2.6 ksi (17.9 MPa) on the surfaces. On the inner surfaces this compressive stress will reduce the hoop stress by about 25%; on the outer surfaces it will be similarly increased.

The skin also experiences stresses due to fuselage bending and twisting. Although the frames and stringers are designed to support these loads, the attached skin is affected by them, as are the skin joints. These shear stresses result in curving cracks. In fatigue testing, these shear stresses are allowed for by raising the applied stress by 30 to 40% over the nominal hoop stress [34].

A comprehensive analysis or simulation of the combined effect of all these stresses is difficult and does not appear to have been attempted in the literature. In general, a fuselage joint is treated as experiencing roughly constant amplitude loading at the rate of one cycle per flight. The load is applied and removed gradually and peak load is held for the duration of the cruising altitude portion of the flight.

Like any structure which is repeatedly loaded and unloaded, pressurized fuselage lap joints can suffer from fatigue. Although the loads are small relative to the yield strength of the aluminum alloys in use, and the number of cycles completed by an aircraft small relative to many other structures, lap joints can show cracking damage in as few as 20 kcycles, which is comparable to the design life in cycles of long-range widebody aircraft such as the B-747 and is significantly less than that of many short-haul jets. Cracking occurs because stress concentrations can raise local stresses above the yield strength of the alloy. Repeated cyclic loading causes local plastic deformation resulting in microcracks and microvoids. These propagate and coalesce until one crack dominates and grows perpendicularly to the local imposed principal stress. Figure 3-11 shows the fatigue life of typical joint designs. Those joints which experience out-of-plane bending due to eccentricity perform less well than double-shear joints. The reasons for this difference are explored in the remainder of this chapter.

3.2 Design of a Riveted Lap Joint

Several factors affect the design of a pressurised lap joint [27]. They are:

1. Joint performance: Four performance criteria must be satisfied, namely ultimate strength, fatigue resistance, damage tolerance and corrosion prevention. The overall structural loading and the design life of the aircraft must be known in order to quantify these four criteria.
2. Weight: The four performance criteria must be satisfied within the constraints of a lightweight structure. Thus the amount of material required must be optimised
 - a) to prevent structural failure from one of the extreme loading conditions (ultimate strength),
 - b) to delay onset of widespread cracking until the aircraft has reached its useful life (fatigue resistance), and
 - c) to maintain structural integrity with certain levels of hidden or undiscovered damage (damage tolerance); all while minimising weight.

3. Inspectability: Despite designing for high fatigue resistance, a joint must also be easily inspected since a crack-free life cannot be guaranteed. This means that the predominant mode of failure should occur in a structural component visible from outside the aircraft. This mode is typically determined by fatigue testing.

The design is an iterative process to determine the stress level limits, materials and design features which satisfy all requirements. Due to the lack of reliable analytical tools for accurately predicting fatigue life, joint design draws heavily on empirical test results.

Faying surface corrosion can ruin the most carefully designed and tested joint, often without warning. This is why corrosion prevention must be considered from the initial design stages. Douglas has found that the only successful means of preventing moisture entrapment in faying surfaces is to use faying surface sealants to exclude moisture [27].

The particular stress conditions imposed on joints by internal pressurization and fuselage loading were presented in Section 3.1. The joints must handle these imposed stresses but are also affected by several local features peculiar to joints. These can be loosely collected into three categories:

1. Fretting: This is the term given to small amplitude oscillatory motion between two contacting surfaces [35]. Fretting can be particularly dangerous in riveted fuselage lap joints because it can play a role in crack initiation near rivets.
2. Stress Concentrations Due to Rivets and Holes: Joining of the sheets produces stress concentrations where the load path is interrupted or shifted. Load transfer in a riveted joint occurs by both friction and bearing. The residual stresses around the hole as a result of riveting and the stiffening effect of the rivet itself both affect joint behaviour.
3. Secondary Bending: Joint eccentricity is the result of two sheets being joined on a line other than their centre-line, as occurs in any single shear joint. As a result.

the load path must shift laterally; since the load path tries to align itself, the joint bends. This is commonly referred to as secondary bending and is typically quantified in terms of the effect it has on the nominal stress in the joint using stress concentration factors. The resultant bending stress distribution leads to peak stresses on the faying surface, making it the likely fatigue crack nucleation point.

These three effects will be treated in detail in subsequent sections. It should be noted that while fretting and stress concentrations are found in all joints, secondary bending is limited to single shear lap and butt joints. Several other detail design features will be seen to add to the complexity of lap joint design, any of which may have a significant effect on joint life. These include, but are not limited to:

- Joint characteristic dimensions, such as sheet thickness, overlap length and rivet spacing;
- Rivet-and-hole characteristic dimensions including countersink depth, hole fit, head dimensions and rivet squeeze force;
- Load transfer characteristics at and between rivets due to clamping, bearing and friction (i.e. ratios of transfer to bypass);
- Presence of stringers as integral parts of a joint; and
- Adhesive bonding layers, primers and sealants between the faying surfaces.

3.3 Fretting

Fretting is thought to be the result of thin surface oxides or films being abraded allowing the opposing surfaces to contact and form intermetallic joints. This adhesive contact may begin after as few as 20 cycles and transmits stresses between the two surfaces where previously they slid without load transfer. Adhesive contact creates debris as small asperities break off and can lead to material transfer if the asperities break at a location

other than where they first bonded. Often this process exposes bare metal which oxidizes and becomes debris. Because of the small amplitudes of motion between sheets (typically $<25\text{ }\mu\text{m}$), debris may be trapped between the faying surfaces.

The contact leads to plastic deformation and eventually cracking to relieve the stresses. Where two surfaces are in partial contact only, a compressive stress is formed ahead of the contact patch and a tensile stress behind [36], which also encourages crack nucleation. There seems to be no consensus on what fraction of life is spent in nucleation and early growth.

When fretting removes surface protection, such as oxides or primers, the faying surfaces and debris are subject to corrosion. The combination of fretting and corrosion resembles corrosion-fatigue in that there is a synergistic effect [37]. Although fretting occurs in inert environments and vacuum there is evidence to suggest that oxidation and corrosion aggravate the process. For instance, it has been found, in an oxidizing environment, that cycles-to-failure increase as frequency is increased [38], the trend seen in other environmentally-influenced fatigue phenomena. Alternatively, it may be that higher frequencies reduce the formation of intermetallic bonds and hence the creation of localised stress concentrations. This would go some way to explaining the shorter fatigue lives seen in actual fuselages than in test articles (which are typically tested at relatively higher frequencies).

Fretting can be particularly dangerous in riveted fuselage lap joints because it can accelerate crack nucleation at one or several sites, contributing to MSD development. The clamped region around the rivet-hole is the area of concern (Figure 3-12). Short cracks can grow and link-up under the combined actions of fretting and fatigue and subsequently grow due to fatigue only [39, 40]. The combined actions of fretting, fatigue and corrosion can reduce component life by an order of magnitude over that due to any single mechanism [41].

Two features which have a bearing on the present investigation should be highlighted:

1. Fretting occurs in lap joints where it can be aggravated by fatigue and corrosion;
and
2. Fretting can lead to initiation at sites which are initially undamaged and/or are initially free of obvious stress raisers.

3.4 Rivet-and-Hole Interaction

3.4.1 Holes

A stress concentration factor (SCF) of exactly 3.0 exists at a hole in a large two-dimensional sheet under in-plane loading [42]. For a three-dimensional sheet some through-thickness variation occurs but the SCF remains close to 3.0 for sheets which are thin relative to the hole diameter. Countersinking increases the hole SCF significantly at the shoulder of the countersink, up to ~ 4.1 ; at the sheet surfaces the SCF is reduced below 3.0, provided the countersink depth is less than the sheet thickness. For a countersink depth equal to the sheet thickness, the hole SCF increases to ~ 4.0 at the resulting knife-edge surface [43].

In bending, a straight-shank hole in a thin sheet has a SCF of ~ 2.0 on the sheet surfaces. For a countersunk hole, the SCF is almost unaffected except for the knife-edge case, where the SCF increases to ~ 2.5 .

In addition, the stress concentration is affected by biaxial stresses and by adjacent holes. For a biaxiality ratio of 0.5 (i.e. for pressurised cylinders), the simple hole SCF is reduced to 2.5. As described earlier the longitudinal stress in the overlap area may be somewhat less than one-half the hoop stress which means the SCF will not be reduced as much. An infinite row of collinear holes has a negligible effect for the diameter-to-pitch spacing ratios typically found in lap joints [44].

Finally, the effects of pressure applied to the hole surface must be considered. Such pressure will introduce tangential tensile stresses and radial compressive stresses equal in magnitude to the applied pressure [42]. Similar stresses are induced during the riveting process (Section 3.4.5).

The above SCF's are for empty holes. It should be noted that the shape of the hole becomes non-cylindrical under in-plane and bending loads.

3.4.2 Fasteners

When two sheets are fastened together, the fastener stiffens the assembly. In a bolted joint there is clearance between the bolt shank and the hole wall. Load can only be transferred by friction due to tightening. As long as there is no slip between the bolted members, no bearing between the bolt shank and hole walls occurs. As loads increase, the hole distorts and the shank bends which may lead to contact.

In pin-connected holes, a pin passes through the two holes but does not clamp the two sheets together; load transfer is by bearing. The hole wall is restrained somewhat by the presence of the pin. Without load transfer, a pin-filled hole with no interference reduces the SCF to ~ 2.0 [2]. Interference in the fit can reduce this further. When the pin is loaded, as in a joint, Shivakumar and Newman [43] calculate an SCF of ~ 1.0 , presumably because of the compressive stresses introduced over the contact patch.

A rivet combines the features of bolting (frictional load transfer) and pinning (bearing load transfer) and adds residual stress effects due to the interference created in the riveting process. In a riveted joint, the rivet expands to fill the hole and so can transfer load directly by rivet bearing, however friction remains an important load transfer mechanism. The entire rivet-sheet combination can distort under loading.

3.4.3 Fastener Flexibility

Fastener flexibility is a concept used in design of structural joints to determine the distribution of internal loads and the residual strength of cracked structures. The fastener flexibility (also called the bolt constant) is defined as the total deflection in the joint due to the fastener [45]. This deflection is caused by a combination of 1) plastic deformation of fastener and hole; 2) fastener bending and 3) fastener tilt and sheet bending (in single shear joints). These three factors are affected by the squeeze force applied during riveting. Consider, for instance, a single shear lap joint with one row of fasteners (Figure 3-13). The total elongation is

$$\delta = \frac{PL}{EA} + f P$$

where P is load, L is length between measuring points, E is sheet material modulus, A is sheet cross-sectional area and f is fastener flexibility. All the load must be transferred by the single row of fasteners. For a multiple row joint a distinction must be made between bypass load, P_{bp} , and load transfer, P_{lt} , for each row. The elongation of a three row single shear joint becomes

$$\delta = \sum_{i=1 \text{ to } 4} \left\{ \frac{P_{bp} L}{EA} \right\}_i + \sum_{j=1 \text{ to } 3} \left\{ f P_{lt} \right\}_j$$

where the joint has now been divided into four segments i divided by three fasteners j , all of which must be evaluated separately. As expected a fastener with zero flexibility (i.e. infinitely stiff) makes no contribution to joint elongation.

A high squeeze force is expected to increase the stiffness of the fastener. A well-fastened rivet-and-hole assembly can be considered to be a quasi-continuous structure quite unlike sheets which are attached at several discrete points (as they might by spot welds for instance and as they are often approximated in models).

3.4.4 Riveting

The riveting process is akin to forging; the relatively soft rivet material deforms under pressure to form the driven-head. The distortion of the driven-head can be quite large; the maximum strain to failure in compression is about 80%, versus less than 20% in tension. The driven and manufactured heads clamp the sheets together. The riveting process is shown in Figure 3-14 and was summarised by Müller [29] as follows:

- a. At the beginning of riveting the axial compression of the (undeformed) shank leads to radial expansion. As the squeeze force increases the shank diameter grows until the clearance is eliminated. (The hole is typically drilled 0.1 mm larger than the original rivet diameter.) The shank nearest the driving anvil expands more easily than that near the manufactured head and the manufactured head itself. At this point the rivet is already deforming plastically.
- b. Stresses within the sheet begin to build due to the expanding rivet shank. The hole diameter increases, with the deformation initially elastic. Compressive radial and tensile tangential stresses grow in the sheet at the hole edge. The unsupported portion of the shank can deform freely.
- c. At some point the sheet deformation around the hole becomes plastic and continues until the riveting operation is complete. The unsupported shank deforms into the characteristic barrel-shaped driven-head and begins to compress the sheets. At this point some material is still being driven into the hole (that is, the driven-head volume is decreasing).
- d. Finally the squeeze force is removed allowing the rivet and sheets to relax elastically. Hole expansion reduces slightly while sheet thickness and the rivet length increase slightly. If the sheet thickness springback is greater than that of the rivet, the rivet shank will be in tension and contributing to clamping.

It is clear that the magnitude of squeeze force plays a large role in determining not just the shape of the driven-head but the (complex three-dimensional) stress state around the hole. Squeeze force has recently been identified as a critical factor in understanding the behaviour of riveted joints in the comprehensive work of Müller [29]. Specifically, it affects the residual stresses in sheet material around the rivet, the interference between the rivet and the hole, and the rivet flexibility.

Riveting was originally performed manually but is now highly automated, as are the drilling and countersinking operations. It is generally performed using displacement-controlled riveting machines. Set-up of the machine is an iterative process by which the operator arrives at the final anvil height to yield a given driven-head diameter for the rivet size and sheet thicknesses in question. Manufacturers allow a wide range of driven-head diameters during manufacturing, typically between 1.25 and 1.8 times the nominal shank diameter. Driven-head diameter variation is thought to account for a fatigue life variation up to a factor of five [29].

Driven-head diameter is affected by differences in the undriven length and in rivet batch material properties. Müller found that force-controlled riveting produced repeatable residual stresses at the hole despite these differences and thus concluded that squeeze force is the more useful standard for ensuring constancy in rivet installation.

3.4.5 The Installed Rivet

Some general observations may be made about the installed rivet. Riveting creates an interference fit which applies radial pressure to the hole boundary. This induces local radial compression and tangential tension in the skin around the hole leading to local plastic deformation. When the squeeze force is removed, springback may result in tangential compressive stresses which serve to reduce the effective SCF. Through-thickness stresses are also produced due to the clamping. Hole expansion of a countersunk rivet is

not uniform along the length of the hole, being greatest near the driven-head and least at the shoulder of the countersink.

Results based on elastoplastic finite element (FE) modeling reported by Müller can be used to examine the effect of squeeze force on each of the principal stress components and thus to identify susceptibility to crack nucleation. Müller modeled a driven-head in a single sheet, which corresponds by symmetry to a slug head rivet. In the model, he used the ratio of driven-head diameter to shank diameter (D/D_0) as the variable on the understanding that material properties in an FE model are fixed. For the purposes of analysis Müller used squeeze force exclusively, which is proportional to D/D_0 for a fixed set of material properties.

Through-the-thickness stress: Figure 3-15 shows that clamping (σ_z) is largely confined to the 'shadow' of the driven-head. Both the compressive peak stress and the volume of material in compression increase with the squeeze force (as indicated by the driven-head diameter). The resulting annular clamped region is the site of frictional load transfer in the joint and will be sensitive to fretting. The extent of clamping contributes to a shift of crack nucleation sensitivity away from the hole-edge for higher squeeze forces.

Tangential Stress: For high squeeze forces this stress (σ_θ) is compressive under the rivet head but tensile beyond it (Figure 3-16) which shifts the crack-sensitive location away from the hole. Increasing the squeeze force does not increase the peak compressive stress at the hole edge but does increase the volume of material in compression. The tensile stress peak beyond the compressed region increases significantly, but this occurs somewhat away from the rivet where bending stresses are expected to be less severe. For low squeeze forces no compressive region is formed at the hole edge.

Radial Stress: This stress (σ_r) is a function of the interference between the rivet and hole (Figure 3-17). The compressive region extends well beyond the driven-head, especially for higher squeeze forces (Figure 3-18). Crack initiation is favoured at the mating surface since interference is lowest there.

Some results for a countersunk head are shown in Figures 3-19 and 3-20. Müller found that the compressive residual stresses are significantly larger in the non-countersunk sheet than in the countersunk sheet, possibly because the countersunk head does not distort much during riveting.

The residual tangential stresses are, on average, more compressive in the non-countersunk sheet and less so in the cylindrical hole portion of the countersunk sheet. Small tensile forces exist in the conical portion of the countersunk hole although these might be removed for a slightly proud rivet head which would induce compressive stresses directly into the countersunk region. Nevertheless, the countersunk sheet is more susceptible to crack initiation.

Radial stresses are generally more compressive in the non-countersunk sheet but near the mating surface are about even, on average. Since the residual stresses are small around the countersunk head, this region of the countersunk hole will be the first to lose interference, reducing the bearing load area in the countersunk sheet to the cylindrical portion of the hole only. Since the load transfer in bearing must be identical in both sheets at a given rivet, the same bearing load applied to this smaller area will make the countersunk sheet more susceptible.

Precise details of the hole countersink, the rivet dimensions and the riveting process determine stress state, and therefore eventual nucleation sites. In general the critical locations are at or near the holes on the faying surfaces. The countersunk sheet, where present, is more susceptible than the driven-head sheet. Overall, higher squeeze forces lead to higher residual compressive stresses, which will improve fatigue life.

3.4.6 *The Loaded Joint*

In a lap joint, a hoop stress is superimposed on the residual stresses induced by riveting. Figure 3-21 shows the estimated first principal stresses on the faying surfaces of a three-row lap joint due to a 10.5 ksi (72.4 MPa) applied hoop stress [46]. The upper plot is the outer (countersunk) sheet with the (upper) critical row at right; the lower plot is the inner (driven-head) sheet with the (lower) critical row at left. The scale is plotted in psi. These results are from an FE model consisting of two 0.045 inch (1.1 mm) thick sheets. Present in the model but not shown are rivets to simulate clamping, and a stringer. The effects of riveting were introduced by applying a 136 ksi (938 MPa) compressive load to the heads of the rivets

Immediately around the rivets the skin is relatively unloaded or remains in compression. This is the clamped area beneath the rivet heads. Outside this area the stresses increase rapidly and are highest at the critical rows. The peak stress of 29.4 ksi (203 MPa) at the upper critical row is almost three times the remote stress and lies on a line approximately 45° from the rivet centreline, just beyond the edge of the clamped area. The high stress region curves outwards towards the adjacent columns of rivets. Cracks often follow a similar curve but not always: the stress distribution can be altered by the growing crack.

3.4.7 *Load-Transfer*

The fraction of the load transferred by friction versus the fraction by bearing at each row of a joint varies with joint design. Swift [2] has calculated the total load transfer at each row in a three-row joint with 0.040 inch (1.0 mm) skins and 3/16 inch (4.8 mm) diameter rivets. Based on displacement compatibility requirements he suggests that 37.5% of the load is transferred at each of the two outer rows and the remaining 25% by the middle row. Müller obtained similar results with FE modeling.

Müller estimated that 20% of the external load is carried by friction and the remainder by bearing, based on FE models of a single shear lap joint. Jarfall [45] found experimentally

that 38% of the external load on a double shear joint was carried by friction at the first peak maximum load, but that the friction component grew to 72% of the load transfer after 200 kcycles, indicating an almost complete reversal of the fractions of load attributable to bearing and to friction. He attributed this to a permanent set developing during early cycling and noted that most fastener flexibility data is measured in the first few cycles.

Sealants, primers and adhesives all have different load transfer and aging characteristics. Sealants and primers can reduce friction which increases bearing loads. This may be a reason why actual aircraft joints perform much less well than lap joint specimens, which are typically tested without primers or sealants. Adhesive bonding should produce much more even stress transfer across the joint. Joint eccentricity remains and is in fact increased by the thickness of the bond layer itself which is usually an epoxy-impregnated woven scrim cloth. Disbonded adhesive reduces frictional load transfer carrying capacity which must be taken up by fastener bearing, and may act as a compliant layer within the joint removing rigidity of the joint overlap.

3.5 Secondary Bending

Bending moments due to external forces typically produce bending stresses distributed over a wide area. These stresses can be satisfactorily evaluated analytically. Local bending due to eccentricity of lines of force acting on discontinuities such as cross-sectional transitions produces small areas of high local stress variation which are much more difficult to evaluate. This is the case with secondary bending, where the neutral axis shifts laterally by a distance equal to the sheet thickness. The bending is concentrated at the outer rivet rows. Some variation in bending also occurs between columns of rivets (i.e. longitudinally on the aircraft). The secondary bending factor (SBF), defined as

$$SBF \equiv \frac{\sigma_{bending}}{\sigma_{applied}},$$

is a common means of comparing bending in joints, where σ_{bending} is the local bending stress and σ_{applied} is the remote applied stress. When the remote stress is not available, it is often replaced by the local membrane stress, σ_{local} , to yield the secondary bending *ratio* (SBR).

To make a full three-dimensional (3D) analysis of secondary bending requires a detailed understanding of the rivet-and-hole interaction which does not appear to be available. Means of estimating it are by finite element methods and by measuring stresses or strains in actual joints. FE models are complicated by the large number of elements required to accurately model the situation coupled with the need to keep skin and rivet elements from passing into one another. Stress and strain measurements are largely confined to outer surfaces and suffer from geometric constraints.

A two-dimensional (2D) analysis of secondary bending in lap joints was made by Hartman and Schijve in 1969 [47, 48]. Using beam theory they modeled the splice as a series of offset beams joined by rigid perpendicular links (Figure 3-22). The links replace the rivets and match the slopes of the neutral axes. The joint overlap is modeled as a single beam with thickness equal to the two sheets and length equal to the distance between the outermost fasteners. The maximum tensile bending stress occurs on the convex surface of the sheet at the rigid link (i.e. rivet) location. Due to the forced localisation of bending stresses at this point in the model, the calculated bending is more severe than occurs in a joint. The Schijve model results in the following equation for the SBF at the rivet location:

$$SBF = \frac{6e/t_1}{\left[1 + \frac{\alpha_1 T_2}{\alpha_2 T_1} \right]}$$

where $T_j = \tanh \alpha_j m_j$ and $\alpha_j = \sqrt{\frac{12\sigma t_1}{Et_j^3}}$ for $j = 1, 2$. E is the sheet material modulus,

σ is the applied stress, e is the eccentricity between beams and the remaining terms are as defined in Figure 3-22. This equation is plotted in Figure 3-23 for a simple lap joint with two rows 2.0 inches (50.8 mm) apart in 0.040 inch (1.02 mm) thick Al2024-T3 for an applied load of 0 to 15 ksi (0 to 103 MPa). These values correspond to the test specimen used throughout this thesis.

The model has some limitations, namely:

1. rivet tilting does not occur, i.e. fastener flexibility is zero;
2. maximum bending is highly localised to the outer fastener location due to the assumed beam thickness mismatch;
3. only the two outermost rivet rows are included;
4. the mechanics of load transmission between rivets and sheets are ignored; and
5. stress concentrations due to the rivet-and-hole are ignored;
6. variation in secondary bending along the joint (i.e. longitudinally) is ignored.

Although the model is limited by its simplicity and probably yields worst-case bending results, it allows quantitative analysis of design features such as sheet thickness, eccentricity, outer rivet row pitch and the distance to the next stringer. Several design variables pertinent to this thesis were examined with the model and listed in Table 3-1. An applied stress of 14 ksi (96.5 MPa) was used which yields a calculated SBF of 0.914.

Table 3-1: *Influence of Design Features on Secondary Bending Factor*

Design Feature	Value [in(mm)]	SBF	Percent Difference
Thickness (t)	0.036 (0.91)	0.884	-3.3
	0.040 (1.02)	0.914	-
	0.045 (1.14)	0.953	+4.3
	0.071 (1.80)	1.164	+27
Eccentricity (e)	0.020 (0.51)	0.914	-
	0.021 (0.53)	0.960	+5.0
	0.0225 (0.57)	1.028	+12
	0.025 (0.64)	1.143	+25
Rivet Pitch (n)	1.00 (25.4)	0.914	-
	1.12 (28.4)	0.882	-3.5
	1.24 (31.5)	0.858	-6.1
	1.36 (34.5)	0.840	-8.1
Rivet Tilt (β)	0°	0.914	-
	0.1°	0.856	-6.3
	0.5°	0.626	-32
	1°	0.338	-64

Several points should be noted:

- Sheet Thickness (t): SBF increases with thickness as expected since the maximum bending location is on the outer surface.
- Eccentricity (e): The effect of adhesive or sealant layers can be treated by increasing the eccentricity. Small increases in eccentricity over the baseline value of 0.020 inches (0.51 mm) (i.e. one-half sheet thickness) significantly increase SBF.
- Rivet Pitch (n): The effect of rivet clamping on the effective length of the overlap section can be addressed by increasing the rivet row pitch (n) in the model. If it is assumed that rivet row stiffening extends over the area in the ‘shadow’ of the

driven-head (Section 3.4.5) then the effective rivet pitch would be lengthened by one driven-head diameter. Increasing the 1 inch pitch to 1.24 inches (for a typical driven-head) reduces secondary bending factor only slightly.

- Rivet Tilt (β): A limitation of the model is the constraint on rivet tilting due to the eccentric links remaining perpendicular to the sheets. Schijve modified the model to allow for rivet tilt caused by plasticity at the rivet-and-hole [48]. Allowing the rigid link to tilt by small amounts has a large effect.

Some of these variables produce significant changes, but all are small compared to the influence of rivet tilt. Thus no reasonable estimate of the secondary bending can be made without first estimating the rivet tilt in the joint which in turn requires knowledge of the joint deflection (as characterised by fastener flexibility). Jarfall [45] reported a comparison of two- and three-layer joints with a single fastener in which it was found that most of the joint deflection caused by rivet flexibility was due to fastener tilt rather than fastener bending. This is not surprising for a single-fastener joint but highlights the problem. It is possible that accounting for tilt will allow reasonable estimates of secondary bending to be made without otherwise adjusting the 2D model.

Schijve also treated the effect of misalignment with his model [48]. Under loading, the two ends lie on the same neutral line, as expected. By clamping the sheet ends, he showed that an offset between the ends equal to the sheet thickness (i.e. no bending in the unloaded condition) had a negligible effect in a joint with dimensions typical of an aircraft joint.

The Schijve model implies that only the two outer rows carry load since the neutral line lies at the sheet interface between these rows. Müller addressed this limitation by accounting for neutral line step changes at each rivet row [29]. This required knowledge of the load transfer at each rivet, which he estimated from measurements of rivet flexibility and FE modeling. The results for two-, three- and five-row joints are shown in Figure

3-24. The overlap length is the distance between the outer rows. Additional rows reduce slightly the bending at the outermost rivet position. When a third row is added to the model used above (2 inch (50.8 mm) overlap), the SBF falls from 0.914 to about 0.83, a reduction of 9.2%.

Actual aircraft joints invariably include a stringer at one of the fastener rows. The effects of this stringer on secondary bending can be examined as follows: a stringer constrains the midpoint of the splice which has a negligible effect compared to an unsupported joint since the support alone does not eliminate the joint eccentricities [47]; addition of an unloaded doubler will stiffen the joint. A stringer acts as a doubler and constrains not just the lateral position of the joint but also its rotation since it is itself constrained in rotation at the circumferential frames. A stringer therefore reduces secondary bending because of the doubler effect and reduced rotation.

Very little data on secondary bending in lap joints exists in the open literature. Schütz and Lowak [49] compared the calculated maximum secondary bending to measurements made midway between fasteners on the hole tangent in a two-row lap joint. They found that the basic Schijve analysis overestimated secondary bending by almost 100%. They attributed this overestimation to the deformation of the rivet-and-hole (i.e. rivet tilt). There is also likely to be some difference between the bending value at the rivet (where the Schijve analysis was made) and between the rivets (where the measurements were made).

Phillips and Britt made strain gauge measurements at a point 0.16 inches (4.1 mm) from the upper lap edge (i.e. about 0.6 inches (15 mm) from the critical rivet row tangent) on a pressurised B-737 [32]. Their measurements suggest a secondary bending ratio (SBR) of about 0.3. Unfortunately they question the data recorded for one of the gauges at this location.

3.6 Crack Development in a Lap Joint

Crack nucleation does not depend just on the ‘bulk’ properties of the material, such as fatigue crack-growth resistance and fracture toughness. Nucleation which occurs at a surface also depends on surface features (e.g. defects, damage, roughness), fretting effects, residual stresses in the surface layer and material quality [50]. Because bearing load is localised near the hole, its effect is limited to the period of nucleation and early growth. All these factors serve to increase the scatter in life-to-nucleation, making it the major factor in overall fatigue life [2].

Cracks will nucleate at the locations of highest stress. As seen in Section 3.3, the hole filling, stiffening, clamping and residual stress characteristics of rivets all help to reduce the stress concentration compared to an open hole. The rivet-and-hole combination remains, however, the principal stress concentration and is in a location subject to high load transfer, high secondary bending and fretting. Cracks are therefore expected to nucleate in this location and this is where they are indeed found (Figure 3-25) [13, 51]. The small crack is likely to form on the faying surface at the hole edges (poor clamping) or a short distance away (good clamping) [52]. Cracks are less likely to form on the hole or countersink surface because the stresses here are lower (recall that bending stresses are highest at the faying surface). From the faying surface nucleation site, the crack may grow along the rivet hole edge and along the countersink. A knife-edged, countersunk hole might be expected to show more severe cracking than one with a blunter edge. Müller suggests, however, based on FE analysis, little actual difference in SCF as a result of good hole filling.

Due to the stress variation through the thickness, the crack tip at the faying surface tends to lead that on the outer surface, producing an effect called tunnelling. The shape of the crack front and the amount of tunnelling is the subject of some speculation [29]. Eastaugh [12] estimated tunneling in 0.040 inch (1.0 mm) 2024-T3 to be about equal to the skin thickness. This suggests that just prior to visible initiation at a countersunk rivet

head, the crack tip on the faying surface is already 0.040 inches beyond the edge of the rivet head, in addition to having covered the area under the countersink. Away from the clamped area the degree of tunnelling reduces as the bending stress falls.

The effect of cracking in the early stages is of great interest. Once the crack front extends to the hole surface, the stiffening effect of the rivet will begin to reduce as the hole begins to open allowing rivet tilt and relaxation of the residual stresses. The reduction in stiffness is likely to be greater in the countersunk sheet due to the lower residual stresses, the lower interference and the lower stiffness of the countersunk head. With clamping reduced, the crack front can extend from the faying surface up to the countersink face.

Once the crack tip emerges from the clamped area beneath the countersunk head, crack growth proceeds more quickly. Very soon after emerging adjacent cracks can begin to affect one another as the size of the shed load increases. This is particularly true when cracks on both sides of a rivet are sufficiently large to eliminate rivet clamping. At this point not only is load being shed immediately at the crack due to the discontinuity in the skin, it is also being shed in the areas ahead of and behind the crack due to the loss of frictional load transfer capability between the cracked (outer) sheet and the uncracked (inner) sheet.

After two adjacent cracks link-up the bending in the remaining uncracked sheet will increase due to the transferred load and the adjacent crack tips. Both crack tip bulging at the transition from the unbent free edge to the bent loaded sheet and that due to internal pressurization will become significant. Fuselage skin curvature also affects the crack behaviour because it changes the stress distribution. All these effects are of relatively little import in the discussion of small cracks considered previously.

3.7 MSD Development

3.7.1 Where MSD Develops

As discussed, MSD tends to develop in the outer rivet rows. The problem of MSD tends to be less severe in the lower critical row (i.e. that on the inner sheet) due to the lack of a countersink, the presence of the driven-head, and the effect of pressurization pillowing to somewhat counter out-of-plane bending. Cracking has been found in this rivet row by Schijve [50] for joints with non-knife-edge rivet holes. Inner skin cracking is of concern to the industry because it is difficult to detect. It is desirable, therefore, to ensure wherever possible that cracking occurs in the more easily inspected outer skin.

Moreover, evidence suggests that the middle row of rivets can become critical if the upper row is replaced with Universal or button-head rivets. This is a standard Boeing repair for detected short MSD and is common on aging aircraft. Several middle row cracks were found in repaired joints on the Aloha aircraft [3] as well as by an FAA study on the effectiveness of this repair [53].

MSD cracking is generally uniform in the middle of a frame bay decreasing to little or no damage near the frames due to the variation in hoop stress across a typical frame bay. Cracks typically appear at or slightly ahead of the rivets due to the differences in clamping and are often found at both sides of a given rivet.

3.7.2 When MSD Develops

The fatigue life of lap joints is neither well understood nor well documented in the open literature. Because lap joints are designed for a finite fatigue life in view of weight considerations, it is only a matter of time before any joint succumbs to MSD [51]. A number of full-scale fuselage tests have been conducted by Boeing. These produced cracks which typically emerged after 65 and 80% of test life although in some cases, as early as one-half of test life [54, 55].

3.7.3 How MSD Develops

The period of undetected small crack growth, from nucleation to initiation, is slow and relatively independent. Changes in the overall stress field may affect all the rivets across a joint but these small cracks are not affected by one another.

The period of growth after initiation to the end of joint life, the crack growth phase, can itself be subdivided. Boeing data reported by Pelloux et al. [56] suggests that visible cracks grow at a fairly constant rate up to a length of 0.25 inches (6 mm) from the edge of the fastener hole with growth rates between 5×10^{-6} and 2×10^{-5} in/cycle (0.13 and 0.5 $\mu\text{m}/\text{cycle}$). Beyond this point adjacent cracks begin to influence one another and will accelerate towards one another. About 25% of joint life is spent between initiation and the first link-up of two small cracks to form a short lead crack. The same effect occurs if the two cracks overlap instead of physically linking.

After link-up, the new lead crack dominates, growing very quickly, especially where MSD exists at the adjacent rivets. The lead crack links up with this MSD, growing much faster than it would alone. Often, a pre-existing crack is seen to initiate under the influence of a rapidly approaching lead crack.

3.8 Summary: Fatigue of Riveted Lap Joints

Since riveted lap joints are designed for a finite fatigue life, cracking will occur. Furthermore, it is likely that scatter in initiation at the rivets within a given joint will be low and that simultaneous crack growth will occur at several rivets leading to an MSD situation. The fatigue strength of riveted lap joints is relatively poor because of:

1. Fretting between the overlapping sheets inside and around the rivet holes;
2. High stress concentrations due to rivets and holes; and
3. Secondary bending.

Secondary bending (3) is a particular problem because:

- a. High stresses have a decisive effect on fatigue strength behaviour even when localised in a small area; and
- b. Cross-sections in danger of fatigue-failure usually lie at discontinuities.

Note that fatigue strength is not a simple function of calculated peak stresses since the complications of load transfer and fretting alter the stress state. A more accurate means of calculating secondary bending will not therefore necessarily lead to improved prediction of fatigue life. Estimating the secondary bending in a proposed joint and reducing it if possible do, however, remain worthwhile design objectives.

As pointed out in Section 3.6, a practical definition of fatigue life of a joint may not be life-to-failure but rather life-to-nucleation. In practice, initiation may be more useful since it is more easily identified (Section 2.2). Once a small fatigue crack is detected, one can be fairly certain that part-through cracks exist at adjacent rivets in a typical lap joint [50]. Obviously the implementation of any such definition depends on possibilities for crack detection and inspection procedures. Lap joint design always aims to make the critical row in the outer skin more susceptible to cracking than the inner row. The safety of only inspecting the outer skin rests on the success of this design principle.

Several design features have been encountered in the course of this chapter. Table 3-2 summarises their likely effect on the fatigue life of a joint, and the presence or absence of the feature in typical fuselage lap joint designs. As pointed out in the pertinent sections, the effect on fatigue life of some features is unclear.

Table 3-2: *Design Features in Pressurised Fuselage Lap Joints.*

Feature	Effect on Fatigue Life	Present in Fuselage Lap Joint?
Secondary Bending	Reduce	Yes
Pressurisation Pillowing	Reduce	Yes
Stringer/Stiffener	Improve	Yes
Fuselage Curvature	Improve	Yes
Sealant/Primers/Paints	Reduce (?)	Depends on Aircraft
Knife-Edge Countersink	Reduce (?)	Depends on Aircraft
Longitudinal Stress	No Effect (?)	Yes

4. CORROSION OF AIRCRAFT LAP JOINTS

This chapter describes the interaction of corrosion (a result of the environment in which an aircraft operates) and fatigue (the result of flight cycles) on a lap joint. The object is to present suitable simulation methodologies for use in the lab.

4.1 Corrosion of Aircraft

Corrosion can begin to attack an aircraft during assembly and certainly once it leaves the manufacturer. The amount of deterioration is a function of detail design, preventive measures employed and the operating environment of the aircraft. Corrosion damage typically increases with time and, more importantly, can accelerate fatigue damage. The effect on aircraft life is presented schematically in Figure 4-1. Corrosion can be loosely divided into time-dependent and time-independent processes (Figure 4-2).

The cost of corrosion, in terms of material lost and prevention undertaken, is tremendous. In 1978 the total cost of corrosion in the United States was estimated at US\$ 70 billion [57]. It has been identified as the single most expensive structural cost item for the USAF [23]. In 1990 the cost of corrosion maintenance to the USAF was US\$ 185 000 per aircraft [24]. These costs are due to both an incomplete understanding of corrosion and a failure to consider adequately its ravages. Selection of materials was often made without due regard to their resistance to corrosion; paints, primers and sealants provided incomplete protection and deteriorated with time and cycling; and designs failed to provide adequate drainage. There are also human and social factors: many standard corrosion inhibitors have been eliminated due to environmental and health concerns; some operators have neglected the corrosion-prevention programs set out by the manufacturers. Exacerbating the problems are the limited capabilities of the non-destructive evaluation (NDE) techniques available; the limited capability to predict corrosion; and the fact that aircraft are being operated well beyond the lives envisioned by their designers. Cur-

rently, the major focus is on prevention and detection. It is anticipated that newer aircraft will suffer fewer problems than the current aged fleet because of design and protection improvements [59].

4.1.1 Corrosion Types

The types of corrosion, their areas of attack and typical product, damage and distribution are treated thoroughly by Wakeman [13], but a brief review will be included here. All aluminum alloys are susceptible to corrosion. The Cu alloys (i.e. 2xxx) show better corrosion resistance than the Zn-Mg (7xxx series) alloys but at slight cost in strength and fracture toughness. Existing aging aircraft skins are almost invariably 2024-T3. General corrosion of the skins typically only occurs when paint quality is significantly degraded but this rarely occurs since aircraft are cleaned and repainted frequently; any damage is easily spotted and removed. The following corrosion types are typically seen in aircraft fuselage skins and adjacent structure:

Pitting corrosion: Pitting is often the first sign of corrosive attack and is common in aluminum and its alloys. The pitting process and an example of early pitting are shown in Figures 4-3 and 4-4 respectively. Pitting is primarily a galvanic process in which particles and matrix corrode electrochemically. It first appears as general staining and discolouring as the protective oxide breaks down. Severe pitting can become self-sustaining in tight stagnant areas and thus is often found within lap joints. The mechanism of pitting in aluminum alloys is treated very well in Reference 60.

Intergranular corrosion: This typically attacks the exposed ends of rolled materials, such as at sheet edges and holes. Proceeding along the grain boundaries, intergranular corrosion weakens the grain structure of the alloy. The process and an example are shown in Figures 4-5 and 4-6.

Exfoliation corrosion: A more severe form of intergranular corrosion, exfoliation occurs when the corrosion product builds up sufficiently to force grains of the alloy apart. It is particularly damaging at rivet holes, where flaking skins reduce the capacity of the rivet to transfer load.

Crevice corrosion: This term is often used when small volumes of electrolyte are trapped within a crevice such as that between faying sheets or at the rivet/hole interface. Crevice corrosion in lap joints typically results in pitting on the faying sheet surfaces and intergranular and exfoliation attack at the sheet edges and the hole surfaces.

4.1.2 Corrosion and Cracking

Corrosion damage and fatigue can interact to nucleate cracks in at least two ways: 1) corrosion damage such as pits and exfoliation can act as stress raisers; and 2) surface defects created by fatigue are likely sites for corrosive attack. Although the exact mechanisms are unknown, it has been suggested that the main effect of pits is to encourage crack nucleation by producing high local hydrogen concentrations on suitably oriented crystallographic planes [61]. After the nucleation stage, crack growth can be accelerated by the higher local stresses induced by thinning, and by stress concentrations due to damage in the crack path.

4.1.3 Corrosion Damage, Product, and Distribution

The most serious corrosion is that within the lap joint itself. An example of damage to a lap joint in an aged Boeing 727 is shown in Figure 4-7. Moisture and corrosive compounds can enter a joint at the lap edge after preventive measures, such as paint, primer and sealants, have broken down even slightly. Corrosion may often be attributed to a combination of poor design, faulty manufacturing, harsh operating conditions and poor maintenance

and tends to concentrate in lap splices, both longitudinal and circumferential, for several reasons:

- crevices encourage moisture ingress;
- fretting fatigue breaks down protective films which allows corrosive attack; and
- disbonding (in bonded joints) tends to encourage moisture absorption.

Intergranular and exfoliation corrosion are generally only found once the environment gains access to rivet hole surfaces. In practice rivet-filling of holes seems to be sufficient to delay moisture ingress [62]. Most corrosive attack is therefore due to environmental penetration at the lap edge. Corrosion in lap joints is often evenly distributed between rivets but can vary from undamaged to heavily corroded in the space of a few rivets.

Joints are typically oriented on aircraft to prevent moisture accumulation on the external surface. Consequently, when moisture is trapped within the fuselage between the outer skin and the internal insulation it can collect on the inner lap edge. The sealants used here often provide incomplete protection and deteriorate as previously noted. The possible role in corrosion attack of condensation at this inner edge appears to have been largely ignored.

It should also be noted that occasional repainting of aircraft temporarily delays corrosion by protecting the lap edge, until the new paint itself begins to deteriorate [62]. Once moisture regains access, corrosion will continue at existing sites. In typical corrosion removal procedures, the joints are wedged open to physically scrape and grind the faying surfaces. It is difficult therefore to guarantee complete corrosion removal. Of additional concern is the mechanical damage done during the cleaning process. The relative impossibility of complete cleaning is highlighted by the fact that joints with a history of corrosion tend to corrode again at the same sites [63].

Faying surface corrosion generally produces oxides and hydroxides of aluminium. The final product has a volume approximately 6 times greater than the aluminum alloy consumed [64] and is largely insoluble meaning that once formed within the joint, the product cannot escape. Instead the skins of the lap joint are forced apart leading to ‘corrosion pillowing’, a bulged appearance between the rivets. Figure 4-8 shows the surface deflection caused by 5% thickness loss throughout a joint according to finite element modeling [46]. (The model is geometrically identical to that used in Section 3.4.6.) This corrosion pillowing has recently been identified as a major stress raiser at rivet hole locations [65], significantly adding to the concerns already raised with respect to corrosion. Cracks due to pillowing stresses have been found in both B-727 and L-1011 fuselage *skins* [66] and KC-135 lap joint *rivets* [62] at these locations.

4.1.4 Corrosion Characterisation

The severity of corrosion damage must be defined in such a way that inspection can accurately and efficiently determine the required maintenance. The average thickness loss of metal is currently the most common means of damage characterisation (Figure 4-9). A thickness loss of 10% is often quoted because this level is detectable by current non-destructive evaluation (NDE) methods. As a result, it is also the threshold at which many repairs are mandated [e.g. 67]. Unfortunately, this approach has several limitations. The measured amount of material lost is necessarily approximate and does not include any measure of the distribution or the location of the damage. For instance, does the damage cover a wide portion of the joint or is it localised? Is the damage in the critical row, or elsewhere? Is the damage concentrated in one sheet? Further, it is unknown whether the 10% value, even if suitably defined, is critical. Finite element modeling has shown that the loss of about 6% of total thickness throughout a joint leads to stresses equal to the yield stress of 2024-T3 near the critical-row rivets [65].

A joint in which both sheets have lost 5% of their thicknesses has the same amount of product and resulting distortion as a joint consisting of an undamaged sheet and one having lost 10% of its thickness. In accordance with current maintenance practices, however, only the latter joint would be repaired, and this only if the damage were in the outer, more easily inspected, sheet. Damage on inner layers will progress much further before being detected by NDE methods in current use.

It may prove easier and more useful to determine severity of corrosion damage by examining the extent of corrosion pillowing. Surface deflection caused by pillowing can be seen in appropriate lighting conditions but requires experience. The D-Sight Aircraft Inspection System [69] provides a controlled means of viewing this pillowing (Figure 4-10). Numerical modeling and ray-tracing methods are being used to generate images of skin pillowing produced by known amounts of corrosion. These images can then be compared with images from aircraft to estimate the corrosion levels [65].

A distinction should be made between indications of corrosion which can be used in service and those which cannot. Pillowing and skin thickness change, for instance, can be determined from the outside of an aircraft. By contrast, surface roughness and the size, depth and distribution of pits within a joint are of interest but can only be determined after disassembly and cleaning.

4.1.5 Summary: Concerns Raised by Corrosion

Corrosion damage is initially innocuous and is, unfortunately, often treated as such. Undetected and unrepaired, it can quickly become a threat to structural integrity. Such damage is insidious and, if unchecked, will progress to the point where it is uneconomical to repair [59]. The factors which make corrosion such a threat are emphasized here:

1. Corrosion accelerates with time and the damage is cumulative;
2. Corrosion can act as a stress raiser for crack nucleation and growth;

3. Corrosion can aggravate and accelerate fatigue-related damage;
4. The cost of repair and prevention is large and growing; and
5. The capabilities of existing NDE techniques for detection and quantification are limited.

The first two items have just been discussed; the third will be investigated in the next section. The fourth item is not treated in this thesis but should be borne in mind; the last is the subject of Chapter 5.

4.2 Corrosion and Fatigue in Aircraft

As previously mentioned, corrosion and fatigue both occur in aircraft, and together can produce severe damage. Understanding the interaction is necessary to predict the behaviour of aging aircraft joints. The first step is to distinguish between ‘corrosion and fatigue’ and ‘corrosion-fatigue’. Extensive information on corrosion-fatigue exists in the literature. It deals with *simultaneous* corrosion and fatigue of materials. Aircraft fuselage joints, however, do not generally experience corrosion and fatigue simultaneously. Rather, environmental exposure to corrosive contaminants is generally confined to times when the aircraft is on the ground, unloaded. Only during flight is the fuselage pressurised and the joints loaded. This suggests that corrosion and fatigue occur *sequentially* in skin joints.

During flight, ambient atmospheric conditions are not conducive to corrosion. The temperature is low enough that any moisture will freeze. Most chemical reaction rates are reduced significantly when the temperature drops below freezing. Thus, when the cracks are open, corrosive attack at the crack tip and on the crack faces probably does not occur. Short cracks in particular can close very tightly when unloaded, allowing no opportunity for corrosive attack. Larger cracks which cannot close completely may be subject to limited attack, but this generally comes so late in joint life as to play a minor role.

The atmospheric conditions experienced in a worst-case scenario were reviewed by Wakeman in considering corrosion and fatigue simulation methodologies for lap splice testing. This study considers an Aloha-type operation in a severe corrosive environment (such as the Hawaiian Islands). The atmospheric profile is reproduced in Figure 4-11 and shows the low temperatures and humidities encountered in flight.

The potential severity of operational conditions can be seen in the operational data for Aloha's B-737 fleet. After the 1988 accident Aloha replaced all its high-time aircraft with newer leased B-737s, however Aloha's operational usage appears to be even more severe now, as a comparison of recently obtained information with 1988 data shows.

Table 4-1: *Aloha Boeing 737 Operational Data*

	1988 [3,17]	1996 [68]
Flight Type	Short Haul (Island Hopper)	
Average Flight Time	24-25 mins.	20-35 mins.
Average Flights per Day	13	17-18
Cruising Altitude	35 000 ft (10 700 m) max.	24 000 ft (7 300 m) avg.
Hours per Day in Flight	5.2 (22 %)	8.0 (33 %)
Hours per Day on Ground	18.8 (78 %)	16.0 (67 %)

4.3 Simulation of Aircraft Corrosion and Fatigue

The complexity of corrosion in aircraft makes the development of a realistic MSD corrosion and fatigue simulation methodology difficult. A complete analysis was presented by Wakeman (Figure 4-12); the highlights are reviewed here. Note that there are three areas of importance which must be considered within the simulation:

1. Lap Joint: the design must yield realistic bending and cracking behaviour;
2. Corrosion: the path of attack and its product, damage and distribution must be matched to actual occurrence; and
3. Aircraft Life: the occurrence of fatigue and exposure should match actual aircraft experience.

Any degree of departure from aircraft experience affects the entire test. As outlined, the actual corrosion and fatigue loading of aircraft is sequential rather than simultaneous. This means that standard corrosion-fatigue tests are not adequate. Further, the results of such a test might not even form a conservative lower boundary because corrosion-fatigue tests typically use extremely corrosive solutions that do not adequately simulate the environment in which aircraft operate. As shown earlier, the significant corrosion damage is that occurring within the faying area of the joint. Harsh corrosion tests can do extreme damage to the exterior of joints leading to failure before significant internal damage is done.

Komorowski et al. [69] have developed an accelerated corrosion procedure which produces realistic product and damage in lap joints. It is a modification of the CASS Test [70], a standard ASTM procedure for corrosion of aluminum. The procedure is less harsh than the commonly used EXCO (exfoliation corrosion) test [71] and produces the insoluble corrosion products found in naturally corroded aircraft joints [64].

Wakeman concluded that alternating corrosion and fatigue programs were the most practical means of simulating actual aircraft corrosion and fatigue experience. In such a test, specimens are subject to alternating periods of fatigue cycling and environmental exposure. The simplest test consists of one exposure block and one fatigue block. This approach, which was employed by Wakeman, is expected to produce a conservative lower estimate of the effect of a given amount of corrosion since the splice begins the fatigue test in a pre-corroded state. Adding blocks of exposure and fatigue should produce increasingly realistic simulation. For instance, starting the test with a short fatigue block (before any corrosion) simulates the life of an aircraft from new until degradation of the protective system (paint, primer and sealants) allows environmental access. Further refinements can be made by using more shorter blocks. Ultimately, short exposure periods alternate with single fatigue cycles, to match the experience of aircraft. Since this approach would result in lengthy tests, a practical test has to compromise by accepting fewer, longer blocks.

Such a test could be conducted using a temporary chamber on the load frame to apply the corrosive medium, or the specimen may be corroded in a dedicated corrosion chamber and remounted in the fatigue testing rig for each cycling block. Prior to each fatigue cycling block, a rigorous purging and drying procedure is required to prevent corrosion during loading which would corrupt the results [72].

4.4 Review of Previous Research in MSD and Corrosion

4.4.1 MSD Testing

A large amount of testing has been conducted using sheet specimens containing open holes and/or starter cracks [e.g. 73, 74, 75]. Although this approach may be useful for investigating fatigue crack growth beyond initiation and the resultant loss of residual strength, it is of no use in the study of nucleation and short crack growth in joints because the complexities of load transfer which give rise to crack-nucleating stress concentrations

and fretting sites are ignored. Testing of small coupon-type lap joints has been conducted with artificially-introduced MSD [e.g. 76, 77, 78] to determine and/or predict residual strength. None of this testing realistically simulates the crack nucleation *as it occurs in lap joints*.

The most extensive work on lap joints in general and MSD in particular has come from the National Aerospace Laboratory of the Netherlands (NLR) under the guidance of Schijve. Testing a variety of simple lap joint designs at the NLR has produced naturally-initiated MSD at one or both critical rows and varying degrees of scatter [e.g. 29]. Unfortunately, as the cracking progresses in a standard lap joint, the net section stress increases as the remaining uncracked material takes up the shed load. Thus although simple lap joints (i.e. two overlapping sheets) are useful for studying initiation they are less helpful for studying growth after initiation as it occurs in aircraft.

To realistically simulate the entire crack life, a test specimen must actively account for the stresses experienced by an aircraft lap joint (Chapter 3). Other than the work of Eastaugh [12] and Wakeman [13], there appear to be no results in the open literature for small-scale specimens with this capability. Most work in this direction is conducted using full-scale test articles [e.g. 54, 79, 80]. It is possible that the results from simple lap joints and pre-cracked single sheets could be combined to describe the cracking of a joint but whether reliable results could be inferred from such a combination is debatable.

It should be noted that specimen testing is conducted at frequencies (10 Hz is typical) much higher than those used in full-scale tests, or which occur in aircraft operation. There is general agreement in the literature that this frequency difference has no effect on crack growth, provided environmental effects are absent. Less clear is whether there is any effect on crack nucleation, especially where fretting is present.

4.4.2 Corrosion Testing

The most comprehensive corrosion test programs involving actual joints are the Corrosion Fatigue Cooperative Testing Programme (CFCTP) [72] and its successor, the Fatigue in Aircraft Corrosion (FACT) Programme [81]. Eight separate laboratories conducted tests using two stress levels to investigate the effects of corrosion prior to and during fatigue cycling. Testing used a 1-½ dog-bone specimen of 1/8 inch (3.2 mm) thick 7075-T76 with two countersunk holes and Hi-Lok fasteners, to simulate an aircraft joint. The authors noted that the overall effects of the environment on fatigue were significant and consistent at the two stress levels, however, the results showed that corrosion before fatigue (which will be referred to as pre-corrosion hereafter) led to a greater reduction in fatigue life at the higher of the two stress levels tested, a result contrary to the previous literature.

As part of the follow-up FACT Programme, NLR repeated the tests using Alclad 2024-T3 and 7075-T6 to investigate the effect of alloy and temper. The 2024-T3 specimen results were equivalent to the CFCTP core (i.e. 7075-T76) results while the 7075-T6 specimens had significantly shorter fatigue lives. As a result of both studies, the authors concluded that in order to correctly assess the effect of environment on fatigue, any future testing must include structurally-realistic joints in terms of geometry, loading, materials (including heat treatments), load histories, stress levels and environment. The last point should be emphasised, given the known lack of equivalency between various test environments, a situation aggravated by the lack of emphasis on corrosion characterisation. The studies also highlighted the need to properly dry splices after corrosive exposure to prevent continued corrosion during fatigue cycling.

Other tests using sheet specimens with open holes have examined environmental effects. Of particular interest are studies of the effects resulting from the introduction of corrosion at different points in specimen life because such studies begin to resemble the sequential corrosion and fatigue process identified in Section 4.2. There are only two such studies

known, both comparing two-block (pre-corrosion plus fatigue) and three-block (pre-fatigue plus corrosion plus fatigue) schedules.

The first, by Fadrugas et al. [82], used a small 0.040 inch (1.016 mm) thick Alclad 2024-T3 panel with a single open 5/32 inch (3.97 mm) diameter countersunk hole and a salt-spray environment. They subjected two batches of specimens to different test schedules: 1) two weeks of pre-corrosion followed by fatigue-to-failure; and 2) 100 kcycles of pre-fatigue, two weeks of corrosion and fatigue-to-failure. Comparing these results to previous fatigue-only tests, the authors found that the corrosion had a significant effect when applied before any cycling but a negligible effect when applied after 100 kcycles. What the authors do not consider is that the lives of the fatigue-only specimens lay between 100 and 200 kcycles. It is possible therefore that the corrosion was added too late in the life to have a noticeable effect.

The second study, by Du et al. [83], used Alclad 2024-T3 dog-bone specimens (i.e. no holes) and immersion in a solution of 3.5 wt% NaCl and 10 vol% H₂O₂ for 48 hours. In this investigation the length of the pre-fatigue period was varied and the results compared to previous fatigue-only and pre-corrosion plus fatigue results. The authors noted that the longer the pre-fatigue period, the greater the damage, characterised by surface roughness, done during the subsequent exposure period. Such damage is consistent with corrosion attacking surface defects created by fatigue (Section 4.1.2). Their results showed that even a small amount of pre-fatigue (50 kcycles for a specimen with a fatigue-only life of ~380 kcycles) significantly extended the life over the pre-corrosion specimens. Longer periods of pre-fatigue (greater than half the fatigue-only life) actually lengthened the overall life of the specimens. The authors speculate that prior to exposure these specimens had microcracks which were blunted by the corrosion leading to the observed longer lives.

4.5 Results of Ongoing MSD Coupon Test Program

An extensive research program at the Institute for Aerospace Research (IAR) of the National Research Council (NRC) has developed a coupon-type specimen for general MSD testing (Figure 4-13). It was developed by Eastaugh [12] as an inexpensive specimen to simulate generic lap joints and has demonstrated that MSD can be produced in a relatively simple specimen without using starter cracks or resorting to full-scale barrel or fuselage tests. Certain concessions are necessarily made in the simplification. The joint experiences uniaxial (i.e. hoop stress) loading only and does not model the effects of fuselage curvature or pressurisation pillowwing. It does account for the shedding of load to surrounding structure and, as such, can model MSD from nucleation through link-up to growth of a lead crack. The program was continued by Wakeman [13] and is also the subject of this thesis.

The specimen consists of two sheets of 0.040 inch (1.02 mm) Alclad 2024-T3 and three rows of 5/32 inch (3.97 mm) diameter countersunk rivets made of 2117-T4. No adhesive, sealant or primer has been used to date. The riveted sheets are reinforced with bonded sidestraps and doublers to simulate the structure around a typical fuselage lap joint. Most importantly, the sidestraps can carry load once the sheet begins to crack, simulating the load transfer characteristics from skin to frames in an actual aircraft joint.

Six plain fatigue and three pre-corrosion tests were conducted by Eastaugh and Wakeman. In the baseline fatigue-only tests, cracks initiated after 75 to 80% of specimen life (defined as cracking through all eight rivets in the critical row). The first link-up between two cracks occurred after a further 20% leaving, typically, just 2% of life to the end of the test. Specimen life to initiation and to the fully cracked condition showed a scatter factor of around 2 in the fatigue-only tests.

The three pre-corrosion specimens were exposed to the modified CASS (Section 4.3) fog for a total of 52 days before being fatigue tested. Wakeman developed a numerical char-

acterisation of corrosion damage using the output of eddy-current data for the specimens tested. He found a clear trend between degree of damage and reduction in life. He also noted a reduction in the uniformity of the MSD produced in the pre-corrosion specimens: cracks tended to initiate at about half as many locations as for the fatigue-only specimens, and generally at the most severely corroded rivet locations. The average effect of pre-corrosion was to reduce initiation life by 30% and overall fatigue life by 23%.

4.6 Summary

The following points should be highlighted:

1. Aircraft experience sequential environmental exposure and fatigue, not simultaneous corrosion-fatigue.
2. Corrosion and fatigue interact even when occurring sequentially.
3. Alternating exposure and fatigue results in the open literature are limited.
4. The importance of realistic specimens and test protocols when investigating complex synergistic processes should not be underestimated.
5. The importance of characterising corrosion in order to interpret results is not well appreciated in the literature. This is partly due to the limited capabilities of existing NDE technologies (Chapter 5).

5. NON-DESTRUCTIVE INSPECTION AND EVALUATION

5.1 Inspection of Aging Aircraft: The Need for NDE

A number of methods are used to detect, locate and/or quantify corrosion and cracking in aircraft. These non-destructive inspection (NDI) and evaluation (NDE) techniques are critical for maintaining aircraft fleets yet are a jumble of empirical qualitative technologies [84]. Inspection remains the weak link of a damage tolerance philosophy [33]. Referring to the Aloha accident, Ramsden and Marsden highlighted this problem [17]:

After 150 million Boeing narrowbody hours there is no suggestion of a basic design fault; the talk is of inspection, and how evidence of so gross a structural failure could be missed.

The likelihood of structural damage increases with the age of operational aircraft. Thus the inspection of aging aircraft has become much more onerous than for newer aircraft because safety is now dependent on detection of the very small cracks associated with MSD. To secure the safety of aging aircraft and avoid the additional costs of unscheduled or unnecessary repairs, a systematic approach to overall inspection requirements must be adopted which accounts for the statistics of damage occurrence, detection techniques, probabilities of detection, inspection intervals, human factors, repair methods and cost benefits. A principal component of such an approach is a very strong NDE program [84].

Aircraft design cannot prevent cracking while maintaining the light weight crucial to operation. Instead, aircraft are inspected at intervals defined by the FAA and necessary remedial action undertaken according to FAA and manufacturer bulletins. Although cracking has been anticipated to varying degrees in aircraft design since the Comet disasters of the 1950's, the effect of MSD has only been recognised much more recently. A component is judged to be safe if any cracks are smaller than a pre-determined critical

size *and* will not grow to that size during operational service before the next inspection. Thus reliable methods of damage detection and characterisation are required to tolerate these sub-critical cracks.

Preparation for corrosive attack has been less universal. Existing splice designs are intended to prevent environmental access in the first place with various protective measures, typically a combination of paints, primers and sealants. If these fail corrosion of the splice interior can proceed and seriously degrade the integrity of the splice. In many aircraft these preventive measures are known to have failed; significant structural corrosion exists in the world-wide fleet.

Fleet operators, therefore, need to inspect for two different features: corrosion and cracking. Similarly, any experimental investigation of aircraft structures needs these capabilities, however, maintenance and experimental requirements are themselves somewhat different. While experimenters willingly sacrifice time and effort for precision, maintenance practice typically demands simple standardised methods to assess cracking and corrosion with little demand made of the operator to interpret the results and to decide if and what repairs are necessary. This chapter discusses the requirements for inspection systems, the current capabilities and some emerging technologies. For the purposes of this thesis the term *evaluation* will be used loosely to include detection, location and quantification of cracks or corrosion.

5.2 NDE Requirements

On aircraft undergoing overhaul and inspection, NDE methods which can inspect for small MSD cracks and corrosion quickly, accurately and inexpensively are desirable in order to minimise downtime of commercial aircraft while allowing for timely detection and repair. This means that inspection of an intact splice needs to be achieved without removing paint or internal aircraft fittings.

5.2.1 Crack Evaluation

As detailed in Chapter 3, splice joint design attempts to make the outer critical rivet row more susceptible to cracking than the inner critical row. This eliminates the need for removing internal fittings during inspection of fuselage structure under the assumption that if the outer row is found to be intact, the inner row will also be intact. The areas to be inspected total several hundred metres of splice on an average commercial transport aircraft, with rivets typically spaced every 20 to 25 mm (i.e. 40 to 50 rivets per metre). There are ten or more lap splices around the circumference of the fuselage.

Since cracks typically accelerate with increasing length, especially in the presence of other cracks (i.e. MSD), the sooner a crack can be detected the better. Sophisticated and costly NDE technology is needed to improve significantly upon visual detection, but the cost could be somewhat offset because repairs can be carried out earlier and, being less extensive, at lower cost. More important for operators, the length of inspection intervals could then be increased, further reducing costs which may exceed US\$ 1 million dollars per aircraft per inspection. To keep costs down, lowering of the so-called threshold of detectability is important. Any system must also minimise false calls (either positive or negative).

5.2.2 Corrosion Evaluation

A distinction should be made between inspection of large areas to assess overall condition of a joint, fuselage or entire aircraft and examination of small areas. Maintenance requirements demand a simple assessment of large areas. Once corrosion has been identified, localised examination may prove necessary to determine any required repairs. Experimental needs require that corrosion be characterised on a small scale in order to determine the effect on fatigue behaviour.

5.3 Current Capabilities and Emerging Technologies

The primary inspection method for cracks and corrosion is visual examination of the fuselage exterior along every splice. Indeed it has been reported that 80% of all inspections of aircraft structure are carried out visually [84]. The method is simple but highly dependent on the individual operator. The repetitive nature means that inspection is affected by human factors, such as fatigue. Most other NDE methods require a skilled operator to interpret results and to examine each rivet in the inspected area individually. In general an aircraft is inspected visually and any suspect areas are re-inspected using one or more of the available NDE techniques.

Cracks are difficult to find for several reasons:

1. During inspection aircraft are unpressurised and cracks closed. Short cracks, the faces of which are not damaged by fretting, can close extremely tightly;
2. Paint reduces the visibility of cracks. In the case of short cracks, the paint itself may be flexible enough not to have cracked. Paint also masks distortion which may exist ahead of a crack emerging from beneath the rivet head; and
3. Since cracks typically tunnel (due to initiation on the inner faying surface caused by secondary bending), they cannot be visually detected until they are quite close to the surface when visible plastic distortion may indicate their presence. A relatively long internal through crack may be much shorter on the outer (visible) surface.

Corrosion tends to affect large regions of the splice, although detection of intergranular and exfoliation corrosion at rivets requires the same detailed, rivet-by-rivet inspection as is required for crack detection. As with cracks, corrosion effects can also be obscured by paint, particularly repainting, and by repairs. The primary difficulty, however, remains

the lack of well-defined means of characterising corrosion with the result that different inspectors come to different conclusions after examining the same aircraft.

Several NDE techniques are used to augment visual inspection. The following paragraphs describe many of them. Much of the information is drawn from Reference 84.

1. Eddy-Current

Principle of Operation: An applied magnetic field induces eddy currents in a metal. These currents can be measured and their intensity used as an indication of the integrity of the metal. A flawless structure induces eddy currents of a known intensity; cracks or corrosion reduce the ability of eddy currents to flow. These relative differences can be detected and interpreted. A range of frequencies (typically 1 to 100 kHz) is used.

Application: Detection of cracks, pits and corrosion.

Advantages: Fast, sensitive, portable.

Disadvantages: Requires skilled (and therefore expensive) technical personnel and reference standards for accurate results. Probe-type and 'lift-off' of the probe influence accuracy.

Capabilities: Can detect sub-surface defects and tightly closed cracks near fasteners but is less successful beneath the rivet head due to probe size. Reliably indicates 10% average corrosion loss.

Comments: Consistent results were achieved by Wakeman [13] using Portascan and MIZ-40 equipment to quantify the thickness-loss of intact MSD coupon specimens. This technique produced data which could be easily interpreted to quantify levels of corrosion within the joint due to outer layer corrosion. Standard eddy-current techniques cannot distinguish between outer- and second-layer

thickness-loss due to variation in spacing between the two sheets due to corrosion product accumulation. Pulsed eddy-current and frequency mixing techniques are being developed which should be able to discern the thickness loss in each of the two sheets by comparing the results of scans at two different frequencies [e.g. 85, 86].

2. Ultrasound

Principle of Operation: Sound propagates through a given alloy at a known fixed velocity and is reflected off any features within the structure including internal flaws, cracks, holes, the edge of the sheet or the opposite face of the sheet. The reflected signal is detected and the time-of-flight interval together with the known velocity yields the distance to the feature.

Application: Characterisation of defects; thickness gauging of single layers.

Advantages: Easy to operate, reliable, fast, sensitive, portable.

Disadvantages: Requires a skilled operator and reference standards for accurate results. Some prior knowledge of the flaw type is required to select probe-type and frequency. Typically requires a liquid couplant.

Capabilities: Detects sub-surface cracks beyond the rivet head. Gauges thickness to within 0.001 inches (0.025 mm) or better, depending on the application.

Comments: For the purposes of crack detection a wavelength approximately equal to the skin thickness is used. The beam can be reasonably tightly focused but the range of usefulness is short because the signal attenuates rapidly with distance and because more and more reflected signals begin to overlap.

Ultrasound signals applied perpendicularly to a surface are already used to measure thickness, and hence thickness loss (if the original thickness is known)

caused by corrosion within pipes. In thin sheets, the time of flight between emission and reflection of the signal is so short ($< 1 \mu\text{s}$) that it is difficult to resolve the thickness losses, which are typically on the order of thousandths of an inch.

3. *X-ray Radiography*

Principle of Operation: X-ray absorption is a function of thickness and density variation. An x-ray image is a map of these changes.

Application: Detection of internal flaws and corrosion; thickness gauging of single sheets.

Advantages: Can be used on built up structures; highly sensitive; permanent film record of the image.

Disadvantages: Radiation hazard; requires a skilled operator and film processing equipment.

Capabilities: Detects cracks emanating from rivet holes provided that the x-rays are aligned parallel to the crack faces. Thickness gauging to about 1% of sheet thickness has been achieved.

Comments: In principle x-rays can be used to detect cracks under rivet heads however the flux requirements are extremely large.

4. *Liquid Penetrant Inspection (LPI)*

Principle of Operation: A fluid which easily penetrates cracks and other surface irregularities is employed. After immersion, the object is typically viewed in ultraviolet light to see the liquid that has remained within the joint.

Application: Detection of surface-breaking cracks.

Advantages: Simple, reliable, fast.

Disadvantages: Flaw must be surface-breaking.

Capabilities: Detects surface cracks but is easily obscured by any adjacent corrosion damage features that also absorb the penetrant.

5. *Shadow Moiré*

Principle of Operation: This is an enhanced visual technique which highlights changes in surface height [87]. When collimated light is reflected from an uneven surface, the scattered, reflected light produces interference patterns indicative of the surface undulations when viewed through a fine grating. The result is a contour-like plot of alternating dark and light lines.

Application: Detection of corrosion pillowing and surface damage.

Advantages: Simple, fast, easy, reliable.

Disadvantages: Grating must be very close to the surface which requires a flat, smooth surface.

Capabilities: The density of the grating determines the resolution. 200 lines/inch (8 lines/mm) provides a sensitivity of 0.005 inches (0.13 mm). Because of the alternating peaks, surface features can be distinguished to about half this distance. A point on the surface with a known height is required to calibrate the image.

6. *DiffRACTo-Sight (D-Sight)*

Principle of Operation: D-Sight is an enhanced visual technique which highlights surface detail using oblique lighting and a retroreflective screen [69]. Figure 5-1 shows the light-path within the D-Sight sensor. The resultant image shows accentuated surface detail. Figure 4-10, which shows the corrosion pillowing in a

B-727, is such an image. The technique is currently being refined to allow quantitative interpretation of the image [65].

Application: Detection of corrosion pillowing and surface damage such as composite delamination.

Advantages: Permanent image record; simple, easy, reliable, portable.

Disadvantages: Requires skilled operator to interpret results.

7. Thermographic Imaging

Principle of Operation: Different parts of an object cool or heat at different rates depending on material and geometry.

Application: Detection of anomalies, such as disbonds, in thin metal skins and composites.

Advantages: Full-field images; can be used at a distance.

Disadvantages: Requires a skilled operator for accurate interpretation.

Capabilities: The detection of corrosion between layers and of cracks is being investigated.

Comments: Images can be recorded on video for later analysis

Two further methods of inspection are of interest in experimental crack-detection work. They are destructive in the sense that the specimen or aircraft must be cyclically loaded to obtain results.

8. Acoustic emission

Principle of Operation: As a crack propagates, small acoustic signals over a band of frequencies from the audible to several megahertz are emitted. These

emissions can be detected and correlated with energy release allowing, in principle, the amount of crack face area produced to be calculated.

Application: Crack initiation and propagation.

Advantages: Requires only receiving transducers; can monitor large areas; detection of cracks while actively growing.

Disadvantages: Requires a skilled operator for accurate interpretation; sensitive to noise; difficulties in discriminating flaw-generated signals from other sources, such as fretting.

Capabilities: Acoustic emission has been used in simple coupon specimens and in built-up structures such as wings [88, 89]. The structure has to be loaded in order to propagate the cracks.

Comments: The acoustic events detected during fatigue cycling of an unnotched aluminum alloy specimen are shown in Figure 5-2. The events in the first 10% of fatigue life are thought to be due to localised yielding although this has not been confirmed. A quiet period typically follows. Further acoustic events are then detected which are thought to be due to crack nucleation. Consistent crack growth indications are seen once a dominant crack is established and grows to failure. The final stage corresponds to visible crack growth.

9. *Strain Gauges*

Principle of Operation: Strain gauges have been used to monitor the change in load distribution as cracks propagate from an open hole in a sheet in uniaxial-tension fatigue-loading [90]. It is not known whether this technique has been applied to built up structures, or joints with load transfer through riveted holes but the principle is sound. Further, it should be possible to monitor nucleation and

propagation of cracks hidden under rivet heads. The structure has to be loaded in order to propagate the cracks.

5.4 Summary

Current NDE technologies overcome many of the difficulties associated with visual inspection (e.g. closed cracks, painted surfaces, tunnelling cracks) but remain time consuming and sensitive to individual interpretation. The difficulty in detecting small cracks beneath the rivet head remains. The situation is well summarised in Reference 59:

The most critical issues inhibiting a successful maintenance program include inadequate inspection standards, lack of quantitative defect interpretation and lack of definitive rejection criteria.

The principle of operation should be borne in mind when interpreting results. For example, Eddy-Current and Ultrasound methods respond directly to corrosion damage within the splice in terms of thickness loss; Shadow Moiré and D-Sight respond to the splice distortion caused by the accumulation of corrosion product within the splice.

6. PROJECT DEFINITION

6.1 Problem Statement

Multiple Site Damage can link-up rapidly to form long lead cracks and can compromise fuselage residual strength.

Corrosion is a known problem in the worldwide transport aircraft fleet, especially in joints which are also subject to MSD. The impact of corrosion on MSD development within a joint is not fully understood.

The questions to be answered, then, are: *How does MSD develop? And how is it affected by corrosion?*

The work of Eastaugh [12] and Wakeman [13] has already begun to answer these questions.

6.2 Current Research Needs

Ongoing investigations have shown that further data on MSD alone and in the presence of corrosion is needed (Sections 4.4 and 4.5). The role of corrosion inside joints is not yet well understood. Specifically,

- How does corrosion begin and develop inside a joint?
- How does it interact with fatigue, particularly with respect to crack nucleation and growth?
- What is the effect of changing levels of corrosion during joint life?

To better understand MSD and corrosion two additional factors must be considered:

1. Improved NDE methods are needed for detection, location and quantification of cracks under rivet heads and of corrosion within intact joints (Chapter 5).
2. The possible importance of detail design features on joints subject to MSD has not been adequately treated. Previous work has shown the importance of detail design variations on joint behaviour (Chapter 3). The scale of these effects needs to be assessed and the effect of corrosion included.

6.3 Areas Investigated in this Thesis

In view of the expense of the MSD specimens, the test program was designed to study several complementary and overlapping areas. The principal objectives, in building upon the work of Eastaugh and Wakeman, were to investigate:

- the effects of environmental exposure during the life of a joint instead of prior to it, using alternating corrosion and fatigue;
- the characterisation of corrosion within intact lap joints and means of correlating it with fatigue characteristics;
- the development of cracks under rivet heads, since cracks do not generally become visible until late in joint life;
- the effect of a different rivet type on joint behaviour; and
- the relation between fatigue characteristics and secondary bending in lap joints.

6.4 Specification of Test Program

Tests were performed using the same specimen configuration and test conditions as used by Eastaugh and Wakeman. Fatigue testing was conducted in lab air using a constant amplitude sine wave with a maximum load of 7219 lb_f (32.1 kN) and a load ratio, R , of

0.02. Tests were conducted at 8 Hz except where noted. Corrosion development was achieved in a modified CASS fog, as described in Section 4.3. The specimens were tested in four groups:

1. Baseline (six tests identified as F-6 through F-11 inclusive):
 - Tests F-6 to F-9 were fatigue to failure; test F-7 was conducted at 4 Hz;
 - Tests F-10 and F-11 were fatigue to initiation, conducted at 2 Hz;
 - Tests F-7, F-10 and F-11 included acoustic emission monitoring.
2. Alternate Rivet Comparison (two tests identified as R-1 and R-2):
 - Using a rivet with a smaller fastener head to eliminate the knife-edge;
 - Test R-1 was fatigue to failure;
 - Test R-2 was fatigue to initiation.
3. Pre-Corrosion Fatigue (three tests identified as CF-4, CF-5 and CF-6):
 - Pre-exposure in modified CASS fog for 56 days total;
 - Fatigue to failure.
4. Alternating Exposure And Fatigue (three tests identified as AE-1, AE2 and AE-3):
 - 'Pre-fatigue' for 75 kcycles;
 - Exposure for 21 days;
 - Fatigue for 75 kcycles;
 - Exposure for 35 days;
 - Fatigue to failure.

Several comments may be made:

- The tests in Groups 1 and 3 added to the existing data for fatigue-only and pre-corrosion plus fatigue. The changes in frequency were made to accommodate

acoustic emission testing requirements. They are not expected to have affected the results but the possibility remains (Section 4.4.1).

- Two tests with a smaller rivet (Group 2) were conducted to determine the impact on specimen behaviour.
- The pre-corrosion plus fatigue schedule (Group 3) represented the conservative case of an aircraft beginning its life already corroded. The exposure periods were selected based on the experience of Wakeman.
- The alternate corrosion and fatigue schedule (Group 4) was intended to follow more closely the experience of an aircraft. Each test began with 75 kcycles and no corrosion to represent the flight-cycling fatigue damage expected to occur in a new, well-protected aircraft. The first exposure block of 21 days represented the early environmental exposure seen in an aircraft beginning to experience aging and breakdown of the corrosion prevention measures. A second fatigue block of 75 kcycles accounted for the flight cycles during this portion of life. The second exposure block of 35 days represented the more extensive damage expected to occur in an older aircraft. At this point, with a total of 150 kcycles and 56 days of exposure in total, the specimens were fatigued to failure.
- The six corroded specimens (Groups 3 and 4) were exposed to an accelerated corrosion environment for the same total amount of time in order to assess the effect of different simulation methods of introducing corrosion and the resultant effect on MSD development. The pre-corrosion specimens were in the chamber for the same 21 and 35 day periods as the alternating specimens.
- Concurrently with the tests, a program of investigation, development and application of many existing and emerging NDE techniques was undertaken.

7. EXPERIMENTAL PROCEDURES

7.1 *The Test System*

The fatigue test rig with an installed specimen and the control system are shown in Figures 7-1 and 7-2. A MTS load frame was used with a MTS TestStar II control system and TestWare SX program software [91, 92]. This is the same set-up used by Wakeman and is fully described there [13]. The specimen was clamped in two pairs of 0.50 inch (12.7 mm) thick steel plates with steel bolts. This assembly was mounted in the rig with precision ground 1.250 inch (31.75 mm) diameter steel pins in machined steel grips.

7.2 *The Test Specimen*

The specimen design used for the baseline and corrosion testing is unchanged from that used by Wakeman. Drawings and manufacturing procedure for the MSD coupon specimen are included in Reference 13.

The rivet specification was MS20426AD5-5. It was made of 2117-T4 alloy, had a nominal shank diameter of 5/32 inch (3.97 mm) and an undriven length of 5/16 inch (7.94 mm). The manufactured head had a 100° countersink and a head height of 0.063 inches. The driven-head diameter was 0.240 ± 0.003 inch (6.10 ± 0.08 mm) producing a D/D_0 ratio of 1.54.

The alternate rivet was NAS1097AD5-5. It was identical to the MS20426 except that the manufactured head was slightly shorter at 0.038 inches. The resulting smaller diameter reduced the clamping area by about 45%. The driven-head diameter was 0.235 ± 0.003 inch (5.97 ± 0.08 mm) producing a D/D_0 ratio of 1.50.

Rivet head flushness was $+0.003/-0.000$ ($+0.076/-0.000$ mm). Therefore in the 0.040 inch (1.02 mm) sheet used here the standard rivet produced a knife-edge countersunk hole while the alternate rivet did not.

7.3 Crack Detection and Measurement

Crack initiation at a rivet was usually preceded by visible plastic distortion for as many as 5 kcycles before a crack tip could be discerned. While this indication aided in detection, early crack growth was difficult to measure accurately because of the relatively slow growth and the difficulty in distinguishing the crack tip within the plastic area. A travelling microscope was used for crack growth measurements. Crack lengths were measured horizontally from the rivet centre and so include the ~0.14 inch (3.6 mm) rivet head radius. Cracks beyond ~0.2 inches (5.1 mm) were more easily distinguished as the tip was generally unobscured.

Crack measurements were recorded for all cracks until a single crack spanned all eight rivets. This was the “fully-cracked” condition and marked the end of specimen life.

Crack growth measurements were calculated by the secant method [93]. Briefly, total growth between two successive measurements is divided by the number of cycles in that interval. This average crack growth rate is plotted against the mid-point crack length for the interval. The accuracy depends on the frequency of measurements.

Several methods of crack detection were explored with the goal of significantly improving upon visual inspection (Chapter 5). The equipment used is listed in Appendix A. Ultrasound was used successfully to detect cracks which had tunneled beyond the rivet head but could not unambiguously detect cracks still under the rivet head. Eddy-current could only detect cracks which had already visually initiated. Two other methods were much more successful:

1. Strain gauges were used to monitor the load transfer expected to occur as a result of crack growth beneath the rivet head. These gauges are labelled ‘C’ in Figure 7-3. Gauges were placed as close to the rivets as possible.

2. Acoustic emission was explored using three specimens. Two pairs of sensors were used to record acoustic events emitted during fatigue cycling. The difference in arrival time at a pair was used to locate the event on the specimen. For specimens F-7 and F-10, a vertical and horizontal pair (labelled 'V' and 'H' in Figure 7-3) were used. Because these tests confirmed that all acoustic activity was coming from the upper critical row, two horizontal pairs (labelled 'H1' and 'H2') were used on F-11. All of this work was performed by Acoustic Emission Monitoring Services (AEMS) Inc. of Kingston, Ontario.

7.4 Strain Gauge Methods

Strain gauge measurements were made at several positions (Figure 7-3) using both 350 Ω and 120 Ω MicroMeasurements gauges. Not all positions were used in each test. In order to monitor both membrane and bending stress at a location, a pair of gauges was used with one gauge on either side of the specimen in the position shown. The gauges were grouped as follows:

1. The positions labelled 'N' were called the "nominal row" positions and were used in most specimens to monitor the stress distribution across the width of the joint. These stresses were representative of the applied stress.
2. The positions labelled 'S' were used to estimate secondary bending at the critical row. They lie on the tangent horizontal to the rivet holes. The gauges on the faying surfaces were accommodated by milling 0.020 inch (0.51 mm) deep grooves in the opposite sheet. This groove was at the free edge of the sheet and so in an essentially unloaded region.
3. Positions labelled 'B' were used to augment the bending data.

4. Three positions labelled 'P' were used in an attempt to estimate the stress change induced by corrosion pillowing (i.e. corrosion product accumulation between the skins).
5. Several gauges were installed before the specimens were exposed to the corrosion environment. They were protected with polyurethane varnish M Coat A provided by the strain gauge manufacturer, followed by a silicone sealant. In some cases these barriers were penetrated and the gauges disbonded. Pooling of the corrosion solution on the silicone sealant appears to have been responsible.

The gauges and corresponding bridge circuits were powered by a 5.0 V precision power supply and connected to a Hewlett Packard 3497A Data Acquisition/Control Unit. On early tests gauge output voltages were recorded using a Hewlett Packard 85 computer and custom data acquisition software developed by Wakeman. For later tests this set-up was replaced by a Hewlett Packard Interface Bus (HPIB) and a PC running custom software written in-house in C++. In both cases the software calculated the stress in real-time for each gauge and the membrane and bending stress for corresponding pairs. It also allowed plotting of these stresses and gauge re-zeroing as necessary.

7.5 Corrosion Procedures

Specimen corrosion was achieved with the modified CASS solution developed by Komorowski et al [69]. The equipment and procedure were the same as that used by Wakeman [13]. Specimens were protected using Tremclad™ white enamel paint applied in several coats with a spray gun. The paint coating was opened up at both lap edges with a fine blade to simulate aged or damaged paint but without exposing the sheet surfaces or edges. The specimens were suspended vertically in a Singleton Salt Corrosion Cabinet (SCCH Model #22) (Figure 7-4) in the same orientation as on an aircraft (outer lap edge at bottom). This allowed the fog solution to collect on the inner lap edge as moisture

does in aircraft. The solution could also enter the lower lap edge by capillary action, again as on aircraft.

After each exposure block, the paint was stripped from the joint area on both sides using Polystrippa™ to allow close inspection of the rivets for crack initiation during the subsequent fatigue block. Specimens were dried for four to six hours at 50°C in a dry oven to prevent any further corrosion from developing during fatigue cycling (Section 4.4.2).

7.6 Evaluation of Corrosion Development

Several non-destructive methods were used to evaluate corrosion development during specimen exposure. Complete inspections were made before and after each exposure block, i.e. at 0, 21, and 56 days. The following techniques, most of which were described in Chapter 5, were employed:

- Visual inspections were made approximately once a week while specimens were in the chamber to look for external corrosion due to paint deterioration and to make an initial qualitative assessment of pillowing. Blistering was common at the lap edges and driven head corners where paint adherence was poor. Moisture within the blisters led to surface staining and occasional pitting.
- D-Sight inspections were made of both sides of all splices. For the final inspection, the specimens were vacuum-packed in black plastic which gave a fixed reflectivity. This coating could only be used on the countersunk surface.
- Shadow Moiré inspections were made of the outer sheet of three specimens. The height of the driven head and the presence of strain gauges with lead wires precluded use on the inner sheets and other specimens.

- Digitally Enhanced X-Ray (DXR) images were made of the joint in both wet and dry conditions. The x-ray film was scanned as a 10-bit grayscale image and then artificially coloured.
- Eddy-current thickness-loss inspections were made using a Zetec MIZ 40 Eddy Current Instrument and two different sets of interpretive software. The Portascan software with a 0.50 inch (12.7 mm) diameter probe employed by Wakeman was initially used but has a threshold of detection beginning at 5% thickness loss. Subsequently, Winspect software was used with a robotic scanning system to scan the area of interest. The Winspect software combined with the robot scanning apparatus and a 0.25 inch (6.35 mm) diameter probe resulted in improved resolution. Calibration was performed using two 0.040 inch (1.02 mm) thick sheets of 2024-T3 riveted together. The upper sheet had milled grooves of different depths. The calibration joint did not include spacing between the sheets to represent accumulated corrosion product which is expected to affect the results. The degree of sensitivity to the second (inner) sheet is unknown.
- Point thickness loss measurements for all specimens were made using ultrasound. Due to the constraints on the availability of the equipment, these inspections could only be performed after the completion of fatigue testing. An accuracy of ± 0.0003 inches (0.008 mm) was achieved. Measurements were taken at the points indicated in Figure 7-5 and interpolated at the points between to generate a grid of data with a pitch of 0.25 inches (6.35 mm) horizontally and vertically. This data was plotted using Microsoft Excel™ which itself interpolates between points to produce a full-field colour surface plot.
- Overall joint thickness measurements were made using a sheet metal caliper-gauge. The same points were used as for the ultrasound measurements and the data was plotted in the same way. Caliper thickness-loss can be interpreted in two ways, as follows:

- a. The overall thickness measurement less the known uncorroded joint thickness yields the degree of pillowing.
- b. The thickness loss can be estimated for each point if the volume ratio of corrosion product produced to alloy consumed is known. This method presumes that corrosion product forms at the site of alloy consumption, which is a reasonable assumption given the product's insolubility, and that complete conversion of the alloy occurs, which is a less certain assumption.

Either way, the method cannot distinguish in which sheet the corrosion damage exists.

- The condition of the joint's outer surface was recorded using a 35 mm SLR camera before fatigue testing began. A Digital Camera System (DCS) was used after fatigue testing and therefore shows the joints in the fully-cracked condition.

7.7 Interpretation of Corrosion Inspection Results

A quantitative comparison of thickness loss indicated by the caliper-gauge measurements was made by defining areas of interest in the joints and counting the pixels for each thickness-loss range in the image. This technique was developed by Wakeman and has been extended here. Thickness-loss intervals of 0.000 6 inches (0.015 mm) were used.

The pixel-counting technique indicates those portions of the area of interest which fall into each of the defined ranges. The images were resized and cropped so that identical areas in terms of size and pixel density were being compared for each specimen. The primary sources of error were as follows:

- Information could not be obtained close to the rivets. The area corresponding to the rivet shanks was subtracted. The annular area beneath the rivet head was assigned to the lowest corrosion level based on previous experience which showed that corrosion is lightest in the clamped area (Section 4.1.3).

- Caliper-gauge point measurements rely on interpolation to produce full-field images. Use of a finer grid would improve accuracy.

For the purposes of general comparison the terminology shown in Table 7-1 was adopted. The joints used in this thesis consist of two 0.040 inch (1.0 mm) thick skins.

Table 7-1: *Corrosion Classification Levels*

Classification Level	Thickness Loss in 0.080 inch (2.03 mm) Joint		
	%	(inches)	(mm)
Light	< 2.5	< 0.002	< 0.051
Light-Moderate	2.5 - 5.0	0.002 - 0.004	0.051 - 0.102
Moderate	5.0 - 7.5	0.004 - 0.006	0.102 - 0.152
Moderate-Severe	7.5 - 10.0	0.006 - 0.008	0.152 - 0.203
Severe	> 10.0	> 0.008	> 0.203

7.8 Joint Teardown and Examination

Three of the corroded specimens were opened after the fatigue tests were completed. using the following procedure:

1. The doubler and sidestrap regions above and below the joint area were removed with a bandsaw to leave a section approximately 10 inches (254 mm) wide (including remaining sidestraps) and 4 inches (102 mm) high.
2. The driven heads were drilled out to a depth roughly equal to their height using a 1/8 inch (3.18 mm) diameter drill in a hand-held power drill. This is slightly smaller than the 5/32 inch (4.0 mm) nominal rivet diameter and allowed for easy

removal of the driven head while reducing the risk of damaging the hole in the event of overdrilling or drill misalignment.

3. The remainder of the driven heads was snapped off with a hammer and small chisel, except for the four rivets in the corners. The rivets were pressed out using a centre-punch. This was done with the countersunk sheet lying on a rubber sheet on a level surface in order to avoid disturbing any loose corrosion product when the driven sheet was lifted off.
4. The remaining sidestraps were removed with the bandsaw.
5. The final four rivets were removed as in Step 3.
6. The non-countersunk sheet was removed taking care not to disturb the corrosion product.

The DCS was used to record the condition of the faying surfaces after opening.

8. RESULTS AND DISCUSSION

The results are presented in four sections corresponding to the test groups defined in Section 6.4. Each section contains results and general analysis. A final fifth section incorporates the principal results in an analysis of the effect of corrosion on MSD cracking. Previous results are included for portions of the analysis. *Note that all data is presented with the specimen oriented in the on-aircraft position and the rivets numbered from left to right.*

A summary of all fatigue test results to date is presented in Table 8-1. The original proof-of-concept (PoC) test, performed by Eastaugh [12], is included as are the five fatigue-only tests (F-1 to F-5 inclusive) and three pre-corrosion plus fatigue tests (CF-1, CF-2 and CF-3) performed by Wakeman [13]. Some published Boeing data, not directly comparable but indicating the orders of magnitude and relative differences seen on aircraft, is included.

8.1 Baseline Fatigue-Only Tests

Six fatigue-only tests, identified as F-6 to F-11 inclusive, were performed. The first four were tested until fully-cracked and the last two to initiation only.

8.1.1 Fatigue Testing Results

Crack-growth and the rates of crack-growth for the four, fully-cracked specimens are plotted in Figures 8-1a, b, c and d. The crack locations are defined relative to the position and side of the rivet at which they appeared. Thus a crack appearing from the left side of rivet 4 is denoted 4L.

The upper plot shows crack length against life in full-load cycles for each crack location. Thus, at a given number of cycles, the crack length at either side of each rivet can be seen. The curves begin at visible initiation which is typically at a length of about 0.14

Table 8-1: Crack Initiation and Growth Results for Fatigue Testing

Specimen	Initiation		First Link-Up		Fully Cracked		Total Cracking		Total Life
	Cycles	% of Life	Cycles	% of Life	Cycles	% of Life	Cycles	% of Life	
Previous Fatigue-Only Results [12, 13]									
PoC	336 563	81.8%	+ 69 437	+ 16.9%	+ 5 500	+ 1.3%	74 937	18.2%	411 500
F-1	291 955	79.3%	+ 70 545	+ 19.2%	+ 5 835	+ 1.6%	76 380	20.7%	368 335
F-2	48 722	60.0%	+ 28 028	+ 34.5%	+ 4 400	+ 5.4%	32 428	40.0%	81 150
F-3	175 000	75.7%	+ 52 500	+ 22.7%	+ 3 550	+ 1.5%	56 050	24.3%	231 050
F-4	195 000	74.9%	+ 60 500	+ 23.2%	+ 4 900	+ 1.9%	65 400	25.1%	260 400
F-5	133 000	75.2%	+ 40 452	+ 22.9%	+ 3 505	+ 2.0%	43 957	24.8%	176 957
New Fatigue-Only Results (8.1)									
F-6	385 000	87.2%	+ 51 940	+ 11.8%	+ 4 393	+ 1.0%	56 333	12.8%	441 333
F-7	201 000	77.6%	+ 49 791	+ 19.2%	+ 8 359	+ 3.2%	58 150	22.4%	259 150
F-8	225 000	75.4%	+ 67 660	+ 22.7%	+ 5 890	+ 2.0%	73 550	24.6%	298 550
F-9	130 000	64.0%	+ 69 700	+ 34.3%	+ 3 375	+ 1.7%	73 075	36.0%	203 075
F-10	90 000	-	-	-	-	-	-	-	-
F-11	130 000	-	-	-	-	-	-	-	-
Alternate Rivet Fatigue-Only Results (8.2)									
R-1	437 000	77.9%	+ 112 003	+ 20.0%	+ 12 067	+ 2.2%	124 070	22.1%	561 070
R-2	400 000	-	-	-	-	-	-	-	-
Previous Pre-Corrosion + Fatigue Results [13]									
CF-1	168 850	76.2%	+ 41 720	+ 18.8%	+ 11 030	+ 5.0%	52 750	23.8%	221 600
CF-2	92 000	62.0%	+ 49 500	+ 33.4%	+ 6 810	+ 4.6%	56 310	38.0%	148 310
CF-3	130 000	66.0%	+ 61 250	+ 31.1%	+ 5 600	+ 2.8%	66 850	34.0%	196 850
New Pre-Corrosion + Fatigue Results (8.3)									
CF-4	145 000	64.3%	+ 76 800	+ 34.0%	+ 3 800	+ 1.7%	80 600	35.7%	225 600
CF-5	130 000	70.3%	+ 43 500	+ 23.5%	+ 11 400	+ 6.2%	54 900	29.7%	184 900
CF-6	230 000	88.8%	+ 16 050	+ 6.2%	+ 12 850	+ 5.0%	28 900	11.2%	258 900
Alternating Exposure & Fatigue Results (8.4)									
AE-1	325 000	79.7%	+ 75 500	+ 18.5%	+ 7 200	+ 1.8%	82 700	20.3%	407 700
AE-2	200 000	75.1%	+ 58 000	+ 21.8%	+ 8 300	+ 3.1%	66 300	24.9%	266 300
AE-3	151 000	77.4%	+ 39 890	+ 20.4%	+ 4 185	+ 2.1%	44 075	22.6%	195 075
Boeing Test Data [53, 54, 80]									
737/S-4R	79 000	79.0%	+ 17 500	+ 17.5%	+ 3 500	+ 3.5%	21 000	21.0%	100 000
747/S-14R	21 500	53.8%	+ 10 500	+ 26.3%	+ 8 000	+ 20.0%	18 500	46.3%	40 000
747/S-44L	38 333	63.9%	+ 19 867	+ 33.1%	+ 1 800	+ 3.0%	21 667	36.1%	60 000
747/S-44R	40 000	80.0%	+ 9 000	+ 18.0%	+ 1 000	+ 2.0%	10 000	20.0%	50 000

Note: A specimen was fully-cracked when a single crack linked all eight rivets.

F-10, F-11 and R-2 were fatigued to initiation only (see text).

inches (3.6 mm) due to the size of the rivet-head. The relative growth of cracks and any interaction with adjacent cracks can be discerned by examining the slopes of the curves. For instance, a kink or 'knee'-shape occurs in a curve when an adjacent pair of cracks link-up. This 'knee' indicates an acceleration of crack-growth after the link-up due to the added shed load. When two cracks link-up, their corresponding curves meet and end. If the cracks overlapped in the test then the plotted curves cross and continue but slow significantly.

These features can also be seen in the lower plot of each figure which shows the instantaneous crack-growth rate for each crack-growth interval of the principal MSD cracks.

Several observations were made based on both Table 8-1 and Figures 8-1a to d:

- a. Crack-initiation (Section 2.2) occurred after about 75% of total life on average but showed a wide variation: from 64% (F-9) to 87% (F-6). This variation is thought to be partially a sensitivity to manufacturing details and will be discussed later.
- b. Cracks tended to initiate just above the horizontal centreline of the critical rivet row which corresponds to the position of maximum local bending on the faying surface. This location is consistent with the D/D_0 ratio of 1.5 (Section 3.3.5) used here (Section 7.2).
- c. First cracks always initiated at the middle two rivets (numbers 4 and 5) and were followed by cracks at rivets 3 and 6, which corresponds with the stress distribution across the splice and matches published results for aircraft.
- d. Cracks occasionally initiated under the influence of an approaching crack tip; e.g. in F-7 (Figure 8-1b) 3R emerges under the influence of 4L. It is likely that a small crack already existed beneath the head of rivet 3.

- e. Crack-growth, which accounts for the remainder of life, varied as a fraction of total life but was relatively constant in terms of actual cycles, ranging from 56 to 74 kcycles after first visible initiation.
- f. The uniformity of MSD development varied in the four specimens. Taking the number of cracks at first link-up as a rough measure of MSD uniformity, F-9 had the most uniform MSD (8 cracks at first link-up) and F-7 and F-8 the least uniform (5 cracks).
- g. Cracks tended to curve towards the rivet centreline during growth, following the maximum bending stress contour between rivets. This tendency was less common later in the tests (i.e. for crack-growth near the outer rivets) when the stress fields were altered by the growing lead crack.
- h. Crack-growth rates increased steadily from initiation but typically started at about 2×10^{-6} in/cycle ($0.05 \mu\text{m}/\text{cycle}$). Peak rates of 10^{-3} in/cycle ($25 \mu\text{m}/\text{cycle}$) were reached just prior to link-ups and overlaps. These rates are consistent with those in aircraft joints reported by Pelloux et al. [56] (Section 3.7.3).
- i. Cracks which overlapped quickly slowed and often halted. The crack tips always turned towards one another. Subsequent link-up was rare, usually only occurring late in the test if crack opening was high enough to deform the remaining ligament.
- j. Final growth rates of the long lead crack often approached 2×10^{-3} in/cycle ($51 \mu\text{m}/\text{cycle}$). It is likely that instantaneous growth rates were even higher but reliable measurements were difficult to make at this point.
- k. Fretting product generally appeared around a rivet once cracks had initiated on both sides. This corresponds to the increased movement which occurs once the clamping is reduced.

Figure 8-2 compares the crack-growth portions of specimen life. In the left-hand graph, the sum of all visible crack lengths is plotted against cycles, from a datum of 0.5 inches (13 mm). The right-hand graph illustrates the overall life of individual specimens with respect to the same half-inch datum. The point at which a half-inch of crack existed was found by interpolation. A datum was used because it is somewhat more precise than using visible initiation (there was some scatter in the length of the first detected crack length); 0.5 inches was used because it is still early in crack life. The number of cycles before and after the half-inch datum are tabulated below.

Table 8-2: *Life in Cycles with Respect to Half-Inch Datum*

Specimen	Initiation	Life to Half-Inch	Life from Half-Inch	Total Life	Batch No.
F-6	385 000	396 650	44683	441 333	4
F-7	201 000	212 649	46501	259 150	2
F-8	225 000	253 733	44817	298 550	2
F-9	130 000	145 726	57349	203 075	3
R-1	437 000	489 205	71865	561 070	-
CF-4	145 000	185 353	40247	225 600	2
CF-5	130 000	143 232	41668	184 900	2
CF-6	230 000	230 657	28243	258 900	3
AE-1	325 000	339 258	68442	407 700	1
AE-2	200 000	227 601	38699	266 300	2
AE-3	151 000	159 245	35830	195 075	2

A longer datum, of say 1 inch (25 mm), masks the effects of early crack-growth. This effect already exists for the half-inch datum. For instance, in F-9, 0.5 inches of crack was reached at 146 kcycles at which point two cracks were visible (4L and 5R, as seen in Figure 8-1d). Shortly after this, two more cracks initiated (3R at 148 kcycles and 5L at

151 kcycles) which suggests that more than 0.5 inches of crack actually existed at 146 kcycles. (Examination of strain gauge data confirms that cracking was occurring at these two locations before 146 kcycles: see Section 8.1.3.) Compensating for this error would result in lowering the aggregate crack-growth curve, bringing it closer to the other results. Applying the same argument to the other tests might have a similar effect but examination of the corresponding crack-growth plots shows that crack-initiations after the half-inch mark came later than in F-9, suggesting that any hidden cracks were shorter.

With this in mind, inspection of Figure 8-2 reinforces the conclusion that specimen behaviour from a point slightly beyond first initiation is relatively consistent, regardless of the crack pattern. The variation in overall specimen life is thus largely due to the period up to and including initiation, as expected (Section 3.6). Whether this behaviour occurs in fuselage joints should be investigated. Overlapping cracks were considered equivalent to linked-up cracks for the purposes of these graphs.

8.1.2 Strain Gauge Results

8.1.2.1 Secondary Bending

Membrane and bending stresses plus secondary bending are presented in Figure 8-3a for the upper critical row of F-6 at 40 kcycles and in 8-3b and c for the two critical rows of F-10 at 30 kcycles. The upper graph shows the membrane and bending stress at both the nominal (1 inch) and critical rows. Bending stresses are calculated as $(\sigma_{\text{inside}} - \sigma_{\text{outside}})/2$. Note that measurements for the critical row were made at the three locations indicated by the symbols: symmetry was assumed. The experimental values at the centre of these joints are compared to other data in Table 8-3.

Schütz and Lowak found the Schijve model overestimated secondary bending by about 100% (Section 3.5). Their result comes from simple lap joints without stiffening elements such as sidestraps and so probably forms an upper limit for secondary bending in

aircraft lap joints. On the other hand, the Boeing 737 data is for a point on the upper skin just above the free edge and so will underestimate secondary bending, forming a lower limit. The experimental data for specimens F-3, F-6 and F-10 varies, but lies between these bounds. Note that the bending in the lower critical row of F-10 is lower than at the upper critical row, a feature which will be discussed in the following section.

Table 8-3: Secondary Bending in a Lap Splice

Specimen	SBF	SBR
F-6 (upper critical row)	0.27	0.33
F-10 (upper critical row)	0.38	0.47
F-10 (lower critical row)	0.11	0.14
F-3 (Wakeman [13])	0.45	0.48
Schijve (Section 3.5) [47, 48]	0.914	-
Schütz & Lowak estimate [49]	~0.46	-
Boeing 737 [32]	-	0.23

8.1.2.2 Surface, Membrane and Bending Stresses Across the Joint

Data from specimens F-9, F-10 and F-11 was used to analyse the stress distribution across the joint on the centreline. Figure 8-4 shows the stresses at several locations. The nominal row is labeled N with the upper critical, middle and lower critical rows located at A, B and C respectively.

The critical row measurements for F-10 are from the secondary bending gauges installed on either side of the same sheet and previously presented in Figures 8-3b and c. The critical row gauges for F-9 and F-11 were located on the outside of the joint; that is one gauge of each pair is located in the essentially unloaded location on the lap edge. Some experimental data for a Boeing 737 lap joint is included for comparison [32]. It was

scaled to the 14 ksi nominal stress experienced by these specimens. The gauges at the upper critical row were placed on the outside of the joint, as with F-9 and F-11. The average line was calculated without using data from these lap edge gauges. Several observations were made:

- a. Bending is higher at the upper critical row (A) than at the lower critical row (C).
- b. Bending at the middle row (B) is approximately zero.
- c. Use of gauges on the lap edge (F-9, F-11 and Reference 31) instead of on the loaded sheet (F-10) changes the apparent sign of the bending indication at the critical rows.
- d. The lap edge critical row gauges indicate less than 2 ksi (13.8 MPa) of stress which suggests that, as expected, little load transfer has yet occurred. These gauges are, however, measuring a surface stress on the side opposite to the surface at which load transmission is occurring.
- e. The membrane stresses for F-10 at the critical row locations are less than at the nominal row location, suggesting either that some load transfer has already occurred or that the load is shifted towards one side of the sheet. Since the gauges are located slightly ahead of the rivets, the latter possibility is more likely.
- f. Membrane stress outside the joint area is even, as expected, at about 14 ksi (96.5 MPa).

Two significant details emerge from this limited analysis:

1. There is a difference in the degree of bending at the upper and lower critical rows (i.e. at the countersunk- and driven-head locations) which implies that the driven-head clamps the sheets more tightly, leading to a lower peak bending stress.

2. Readings from gauges on the lap edge change the apparent sign of the bending indication at the critical rows. Readings from such gauges should, therefore, be interpreted with care.

8.1.3 Crack Detection Techniques

Previous work such as that by Boeing [54, 55] and Wakeman [13] showed that cracks often only emerge after 75% of specimen life. Because of this, the state of crack development is unknown during the bulk of the life. Given the potential threat of MSD, any means of detecting cracks significantly before visible initiation would be valuable.

An attempt was made to detect crack-growth beneath the rivet-head using strain gauges at the four central rivets of specimens F-7 to F-11 inclusive (Section 7.3). The readings were monitored throughout the test. The calculated stresses for F-9 are plotted in Figure 8-5; plots for the remaining specimens are included in Appendix B. The data was interpreted as follows: a divergence in the stresses at either side of a rivet indicates that load was being shifted from one side to the other which in turn implies that a crack was growing on the side of the rivet from which load was being shed. Examination of the figure shows that abrupt stress changes occurred up to 100 kcycles prior to visible initiation at that rivet. Two general indications were noted: 1) a steady stress decrease for a gauge followed by visible initiation at that location; and 2) a steady increase followed by visible initiation at the other side of that rivet. These indications are consistent with the load shedding which accompanies crack-growth. In the case of F-9, cracking is indicated at 3R, 4L and 5R from 75 kcycles. The length at which cracks were detected is unknown and should be investigated.

At the start of a test the stresses at the eight gauges on each specimen varied by as much as 3 ksi (21 MPa); however the stresses at each pair of gauges often varied by less. There was no correlation between initial stress at a gauge and initiation at that location, nor

between the average stress over all eight gauges and the first crack initiation in the specimen.

The first crack was detected before visible initiation in all cases. Subsequent initiations were predicted in about half of cases. Later in the specimen life, with several small cracks growing, the gauge readings at the (visually) uncracked locations increased due to the load being shed elsewhere. This made it difficult to identify crack-growth indications. Further analysis may yield better results, as would improvements in the strain gauge placement, perhaps with reference to a suitable FE model.

Acoustic emission equipment was also used to monitor crack-growth, for specimens F-7, F-10 and F-11. Technical difficulties with F-10 meant that little meaningful data was obtained. The data for F-7 and F-11 suggests that crack activity was occurring up to 140 kcycles before visible initiation.

Eddy-current and ultrasound techniques were also used but did not noticeably improve upon visual detection.

Figure 8-6 compares crack detection techniques for specimens F-7 to F-11. Visible detection corresponds to the first point at which an emerging crack was seen. Strain gauge detection was based on the stress plots such as Figure 8-4. Acoustic detection was taken as the point at which consistent crack-growth indications were first recorded (Section 5.3). Study of the five graphs in Figure 8-6 illustrates that the strain gauge and acoustic emission techniques were often able to detect cracks beneath rivet heads 100 kcycles before their visual initiation. The graphs also show quite clearly that cracks appeared at either side of a rivet within less than 50 kcycles of one another. This suggests that once one side of a rivet begins to crack the load shed to the other side plus the reduced clamping and relaxed rivet-hole interference promote crack-growth at the other side, if it is not already underway.

8.1.4 Specimen Teardown

Specimen F-7 was opened using the procedure detailed in Section 7.6. Fretting was evident at all twenty-four rivets but was heavier at crack-initiation locations in the critical row than elsewhere. Additional damage was evident on the inner sheet at these same locations caused by gouging from the cracked outer sheet. There was no obvious evidence of multiple crack nucleation sites on the hole countersinks of this specimen: however a detailed fractographic examination was beyond the scope of this project.

8.1.5 Discussion

These results, following those of Wakeman, confirm that MSD occurs at the middle four rivets of this eight-rivet wide specimen. The results also confirm that the life of the specimen is largely determined by initiation: life after the 0.5 inch (13 mm) aggregate crack length is consistent, regardless of the crack-growth pattern, and hence of MSD uniformity.

The high uniformity of crack-growth in F-9 illustrates the potential hazards of MSD. The large number of active crack tips appears to have delayed first link-up relative to the other specimens. Just prior to link-up, the maximum tip-to-tip crack length was 0.9 inches (23 mm). Within about 300 cycles, three link-ups occurred creating a 3.3 inch lead crack. If the uniformity occurring in F-9 (eight cracks at the eight most vulnerable locations) is extended to a typical 20 inch (508 mm) frame-bay, the results of the dangers posed by MSD are apparent. Based on the ranges of uniformity seen here, there is reason to expect similar crack patterns of, say, many 0.5 inch (13 mm) cracks linking up to form a 12 to 15 inch (300 to 380 mm) lead crack relatively quickly.

Results of these further baseline tests for crack-initiation patterns and crack-growth behaviour agree well with the previous results (Table 8-1). The scatter in the life-to-

initiation, however, was variable. The potential sources of scatter in this testing were as follows:

1. Testing: Variation in the applied load, according to oscilloscope readings, was less than 0.1% of maximum load. The load cell was calibrated prior to the beginning of the test program. The grips were aligned at the same time.
2. Specifications: The drawing tolerances for the coupon specimen are smaller than in typical aging aircraft.
3. Materials: All specimens were made from the same batch of Alclad 2024-T3 aluminum alloy. The components for all specimens were laser-cut to size in one batch. Any effect of position within the sheet will, however, remain.
4. Machining: The specimens were manufactured in small batches. Although the same CNC equipment was used for drilling and countersinking, the operator changed for each batch. The drawings do not specify machining parameters such as tool rotation speed, feed rate and cooling rate. Tools were dedicated and periodically replaced but the effect of tool wear-rates on the specimens within batches is unknown. Differences in these parameters may have changed the surface quality and residual stress state of the holes before riveting.
5. Riveting: Each specimen was riveted in a single operation using the same automatic riveting machine. Rivets were installed in a random pattern. Some rivets were out of tolerance by up to 0.005 inches (0.13 mm) but no correlation was found between rivet flushness and initiation. Neither was any correlation found between driven-head diameter and initiation. Given that all rivets were from the same batch, this is not surprising. It is possible that residual stress differences were induced during riveting due to hole differences produced during machining. These could affect nucleation characteristics and hence visible initiation.

6. Assembly: Specimens were laid-up with adhesive by hand. Autoclave curing of each specimen was performed individually.

Machining, riveting and assembly emerge as the primary potential sources of scatter. Since specimens were riveted and assembled individually, any resultant scatter should be evenly distributed amongst the specimens while scatter due to machining should be small within batches. Table 8-2 (pg. 88) included the 'Batch Number' of the tested specimens. The lives-to-initiation for the two specimens in 'Batch 2' (F-7 and F-8) differed by less than 12%; the lives-to-initiation for the two in 'Batch 3' (F-9 and F-11) were equal. The 'Batch 4' specimens (F-6 and F-10) however differed by a factor of 4.3. Because of the small sample size, statistical analysis of the few results is uninformative. Nevertheless, batch sensitivity is felt to be a possible explanation for the scatter seen and will be considered for the other test results.

8.2 Alternate Rivet Fatigue-Only Tests

Two tests, denoted R-1 and R-2, were performed using a rivet with a smaller manufactured head that eliminated the countersink knife-edge and had a slightly smaller driven-head (Section 7.2). The specimens were otherwise identical in all respects.

8.2.1 Fatigue Testing Results

Both specimens demonstrated greater fatigue resistance, initiating cracks in the upper critical row later than in all baseline tests. Unlike the baseline tests, they also cracked at the lower critical row; i.e. in the inner sheet.

For test R-1 (Figure 8-7), in which cracks initiated at 437 kcycles, inner sheet cracks were first noticed at 505 kcycles but are thought to have been growing for some time. The cracks had not penetrated the sheet but were visible due to the plastic distortion produced as they tunneled from beneath the driven-head (Figure 8-8a). Link-up in the upper critical row occurred at 549 kcycles and the specimen was fully cracked at 561 kcycles. At this point, eight cracks were discernible at four rivets in the lower critical row. None exceeded 0.25 inches (6.4 mm) in length; nor did any penetrate the sheet surface. Several observations were made:

- a. As noted, initiation life was longer than for any other test. Crack-growth from initiation through link-ups was slower, on average, than for previous specimens in terms of cycles. The aggregate crack length was included with the baseline results in Figure 8-2. Growth from the half-inch datum (Section 8.1.1) to fully-cracked took 72 kcycles versus 48 kcycles on average for F-6 to F-9. The percentage proportions of life were comparable to other tests (Table 8-1).
- b. Several cracks in the upper critical row emerged beyond the rivet-head to form ‘fissures’ with two tips, one pointing away from the rivet, as usual, and the other

back towards it (Figure 8-8b). The 'backward' facing tip quickly grew under the rivet to leave a crack whose appearance was indistinguishable from other cracks on this and other specimens. These fissuring cracks appeared both early and late in the cracking period. They were most noticeable in the case where the crack emerged under the influence of another approaching crack. This form of initiation may be quite common: it was difficult to detect initiation before the backward facing tip grew under its rivet. Fissuring suggests that the initiation site was somewhat away from the hole edge and that a semi-elliptical crack grew through the thickness, emerging as the visible fissure. The remaining ligament between such a semi-elliptical crack and the countersunk face would quickly crack.

- c. At several rivets, after one crack had emerged at a rivet location, one or more further cracks would emerge and grow some distance before either stopping or growing into the lead crack (Figure 8-8c). None of these secondary cracks reached 0.20 inches (5.1 mm) in length. It is known that several small cracks can form during nucleation but these typically stop or link-up to form a single dominant crack while still under the rivet-head, i.e. before visible initiation occurs [13]. The multiple cracks seen here are probably extreme cases of this behaviour.

In test R-2, initiation occurred in the upper critical row after 400 kcycles, slightly earlier than for R-1. Distortion had, however, been found on the inner surface at 360 kcycles. Due to the cracking with R-1, closer attention was paid to the inner sheet and distortion was detected at a much earlier stage. Nevertheless, these cracks grew much more quickly, resulting in through-cracks at 435 kcycles. Because of the rapid advance of these cracks the test was halted at 460 kcycles at which time only two visible cracks existed in the upper critical row, although there may have been subsurface cracks at other locations. Eight cracks were discernible in the inner sheet at the four central rivets in the lower critical row. At one rivet there were two through-cracks of about 0.4 and 0.45 inches (10.2 and 11.4 mm) in length. Of the other six cracks, none was a through-crack

and none exceeded 0.20 inches (5.1 mm) in length. It was thought that lower critical row cracking was unduly influencing the upper critical row and that further testing would only yield suspect results. Both specimens were subsequently used for NDE testing by IAR.

8.2.2 *Strain Gauge Results*

Strain gauges were used to determine hoop stress variation and secondary bending in both specimens. This data is presented in Figures 8-9a and b in the same format as for the baseline tests (Figure 8-3 and Section 8.1.2.1). The membrane stresses at the nominal position match those of the baseline tests in which the larger rivet was used. The bending stresses at the nominal position are higher than in the baseline tests while those at the critical row are lower, implying that the curvature in the skin is more gradual but extends further from the critical row. The SBF's are ~0.15 and ~0.25 for R-1 and R-2 respectively, somewhat less than those measured in the baseline specimens.

The critical row data shows more variation than in the baseline cases. The cause of this is unknown but may be a sensitivity to gauge installation. The strain gauge readings were consistent through out the tests.

Strain gauges were not used to monitor crack-growth on either of these specimens.

8.2.3 *Discussion*

The change of rivet had a significant effect on fatigue behaviour. Three distinct trends were seen with respect to the baseline results, namely: longer initiation life; initiation in the inner (driven-head) sheet; and reduced bending, possibly due to the following:

- Longer Initiation Life: The lower bending stresses in these specimens (Section 8.2.2) reduced the stresses on the faying surface at the site of crack nucleation which is consistent with the longer initiation lives and slower crack-growth. The smaller rivet-head led to a blunted knife-edge which in an open hole would be

expected to lower the stress concentration and increase initiation life. For a riveted hole, as noted in Section 3.6, the difference between a knife-edge and blunted hole may be minimal.

- Initiation in the Inner Sheet: The driven-head on these specimens is slightly smaller than that on the baseline specimens. This is indicative of a smaller squeeze force during riveting which would reduce the initiation life in the lower critical row (Section 3.4.5). This seems reasonable since no initiation was seen in the driven row of any baseline test, three of which lasted beyond 400 kcycles.
- Reduction in Bending: The value of the SBF adjacent to the alternate rivet lies between those for the manufactured- and driven-heads of the standard rivet (Figure 8-3, Table 8-3 and Section 8.1.2 for F-10). The difference should be attributable to the differences between the two rivets. The smaller manufactured head reduces the clamped area which would be expected to reduce stiffness, leading to an increase in bending and a consequent reduction in initiation life. In fact the opposite trends were seen which suggests another factor must be involved: the smaller head also leaves a cylindrical portion of hole beneath the countersunk hole. It is suggested that this cylindrical portion leads to a residual stress state which is stiffer than the knife-edge case. The exact mechanism is unknown but such a possibility should be investigated.

Based on these two tests it appears that the upper and lower critical rows with the alternate rivet are about equally sensitive to crack-initiation: R-1 first initiated in the upper critical row and R-2 in the lower. The criticality of the countersunk upper row has been reduced and that of the driven-head lower row possibly increased. The results suggest that an aircraft using the alternate rivet is at risk of inner skin cracking, which is difficult to detect. Further investigation is therefore warranted, and should consider the effect of pressurisation pillowing to reduce the stresses at the lower critical row (Section 3.1.3).

8.3 Pre-Corrosion + Fatigue Tests

Three specimens, identified as CF-4, CF-5 and CF-6, were fatigued to failure after having been exposed in a corrosive environment for 56 days. The data are included in Table 8-1.

8.3.1 NDE Results

Several NDE techniques were employed to record and evaluate corrosion development within the three splices (Section 7.6). Figures 8-10, 8-11, and 8-12 compare these results. Several comments may be made to aid their interpretation:

- a. 'Total Corroded Joint Thickness by Caliper-gauge' shows total joint thickness due to pilling. The nominal joint thickness was 0.080 inches (2.03 mm). The colour-coded key indicates the thickness of the corroded joint in intervals of 0.003 inches (0.076 mm).
- b. 'Thickness-loss Off Both Sheets by Ultrasound' is the sum of ultrasound measurements made on both skins. A rudimentary comparison with the caliper-gauge plot may be made as follows: taking a volume ratio of 6:1 for the corrosion product to the consumed alloy (Section 4.1.3), each 0.003 inch (0.076 mm) interval of overall thickness is equivalent to 0.0006 inches (0.015 mm) of thickness-loss: i.e. -0.0006 (alloy lost) + 6×0.0006 (product gained) = 0.003 (net change). This is the same interval used in the ultrasound plot, which allows direct comparison. This method is very approximate and assumes that all the alloy is converted to the final product. While qualitatively similar, the indicated magnitudes in the two scans differ by a factor of between 2 and 3.
- c. 'Enhanced X-Ray Image of Corroded Joint' includes a scale indicating increasing absorption. Since absorption is a function of thickness and density it is difficult to interpret the results. However there does appear to be some similarity between the indicated high absorption areas (blue to white) and the peaks indicated by the

other techniques. Note that in these images strain gauges appear as white rectangular areas which should not be confused with corrosion peaks. In some scans the strain gauge leads are also visible.

- d. 'Eddy-Current Scan with Portascan' shows the indicated thickness-loss in inches. The key is calibrated from two grooved riveted sheets (Section 7.6). These plots indicate damage which is consistent with but higher than those given by the caliper-gauge and ultrasound measurements. The grey areas were obstructed by the presence of strain gauges.
- e. 'Eddy-Current Scan with Winspect' provided much better resolution than the Portascan system due to the controlled step size but no change in absolute measurements. Because these scans were made after fatigue testing, the final crack configuration is present. It appears as a black band where no data was recorded due to probe lift-off from the uneven crack surface. The same effect occurs at the rivets. The strain gauges were, however, removed. The colour bar indicates increasing corrosion.
- f. 'Shadow Moiré Image of Countersunk Surface' provides a contour-like plot of the surface (CF-4 and CF-5 only). Consecutive dark (or light) bands indicate deflection changes of 0.005 inches (0.13 mm).
- g. 'Faying Surface of Outer and Inner Sheets' shows the condition of the two surfaces immediately after opening of the splice (CF-4 only). The image of the outer sheet has been reversed to allow direct comparison with the other images. Although difficult to see in the images, fretting damage was evident at the central holes in the upper row on the inner sheet where gouging by the cracked outer sheet occurred later in life. It is difficult to discern any direct similarity between these images and the NDE scans since the effect of height is lost. However

examination at the time of opening showed that the corrosion product was concentrated in the areas indicated as damaged by the other NDE techniques.

- h. 'D-Sight Image' shows the final surface deflection of the countersunk surface. The strain gauges and portions of the final crack configuration are also visible.

Comparing all the NDE images shows that good qualitative agreement was achieved in terms of position, extent and relative magnitudes of features, particularly between the caliper gauge, ultrasound and eddy-current images. The absolute degree of damage indicated differed. Nevertheless, these plots form a library of results which can be used to directly compare scans on future specimens taken with two different NDE methods.

Figure 8-13 shows correlation between thickness-loss and life-to-initiation, for each corroded specimen. The thickness-loss data is taken from the caliper-gauge measurements following the procedure described in Section 7.7. The coloured bands in each bar in the upper graph show the fraction of total joint area corresponding to the thickness-loss levels defined in the key at right; the colours match those used in the comparison plots (Figures 8-10 to 8-12). For each specimen, an average thickness-loss value was calculated and compared with the life-to-initiation, as shown in the lower graph.

From this data it can be qualitatively deduced that CF-5 and CF-6 had similar average damage while CF-4 was more heavily damaged. The average thickness loss in all three specimens was less than 0.0020 inches (0.051 mm), or 2.5% of the total joint thickness; the peak thickness loss, which was reached only in small areas, was less than 0.0050 inches (0.127 mm). The corrosion damage to the joint was classified as light (Table 7-1). An inverse relationship between average loss and initiation is intuitively expected but was not evident here, probably because the averaging process does not account for corrosion concentrations which encourage nucleation at individual rivets.

Figures 8-14a, b and c are analogous to Figure 8-13, but for individual specimens. The bars in the upper graphs describe the damage in a reference area around each of the six

central critical row rivets. The areas near the centre rivets tended to be more heavily damaged. The average thickness loss around the rivets in all three specimens was less than 0.0020 inches (0.051 mm), except at rivet 4 in CF-5 where 0.0023 inches (0.058 mm) was reached; the peak thickness loss was less than 0.0036 inches (0.091 mm), except at rivet 4 in CF-5 where the peak loss was still less than 0.0042 inches (0.107 mm).

Despite these light corrosion levels, an inverse relationship between average thickness-loss and life-to-initiation at the rivets within each specimen is apparent in the figures. From the results of the previous sections, it will be recalled that the cracks tend to initiate at the central rivets, due to the higher hoop stress at the centre of the specimen. Thus any correlation between thickness-loss and initiation cannot be wholly attributed to the presence of corrosion.

8.3.2 Fatigue Testing Results

Crack-growth and rates of crack-growth for CF-4, CF-5 and CF-6 are plotted in Figures 8-15a, b and c. Cracks initiated at 145, 130 and 230 kcycles respectively. Initiation and growth characteristics were consistent with the fatigue-only tests (Section 8.1.1) with the following differences:

- a. The presence of corrosion reduced life-to-initiation. Visible initiation tended to correlate with damage around that rivet; first initiation tended to occur at rivets in the more heavily corroded locations.
- b. The first cracks always initiated at rivets 4 or 5 but were not followed by cracks at rivets 3 and 6 until shortly before or after the lead crack reached them. This led to much less uniform MSD development, particularly in the case of CF-5. In this specimen, initiation at both sides of rivet 4 created a short lead crack which dominated the test, consuming the outer rivets in turn. Two cracks initiated at rivet 5

shortly before the lead crack reached it, although strain gauge readings (Section 8.3.3) had indicated their presence before the appearance of the cracks at 4. The pattern of cracking was consistent with the levels of corrosion damage at these two rivets. MSD in specimens CF-4 and CF-6 was more uniform than that in CF-5 but less uniform than in any of the fatigue-only tests.

- c. In the early stages of cracking, crack-growth rates in these tests were up to 10 times faster than for cracks of similar length in the baseline tests. A thickness-loss in the crack-growth path is expected to increase the crack-growth rate due to thinning and the consequent increase in stress. For example, a thickness-loss of 0.001 inches (0.025 mm) will raise the stress by 2.5%; 0.003 inches (0.076 mm) leads to an 8.1% increase. In light of these small changes, the higher crack growth rates in the pre-corroded specimens were probably largely due to the smaller number of active crack tips and to increased stresses induced by corrosion-pillowing.

Figure 8-16 compares the crack-growth portions of specimen life. The results are similar to one another and to the fatigue-only tests. Crack-growth after the half-inch aggregate crack length is therefore relatively unchanged, which is consistent with the low levels of corrosion found.

8.3.3 Strain Gauge Results

Membrane and bending stress distributions were similar for all three specimens. Crack gauges successfully detected crack-growth beneath the rivet-head in specimens CF-4 and CF-5. They were not installed on CF-6 due to the presence of other gauges. The stress plots for these two tests are included in Appendix B. No correlation was found when the difference in cycles between indicated strain gauge and visible initiation was compared for corroded and non-corroded specimens.

The attempt to monitor stress changes induced by corrosion pillowing using gauges installed prior to exposure in the chamber was unsuccessful (Section 7.4). In most cases moisture penetrated the silicone sealant and attacked the adhesive, disbonding the gauge. Pooling of the corrosion solution on the horizontal surfaces of the silicone is thought to have been responsible. No useful data was obtained.

8.3.4 Discussion

Pre-corrosion noticeably reduced MSD uniformity which matches the pattern seen in Wakeman's results. At present, too little information is available to determine whether reduced uniformity is also seen in corroded aircraft. It appears that even for these low levels of pre-corrosion, enough damage is done to hasten nucleation at the crack-sensitive sites (Section 4.1.2). Early cracking at the most heavily corroded holes tended to dominate, before additional MSD could develop. MSD was more uniform in CF-6 which proved to be the least corroded specimen.

Corrosion introduces an additional source of scatter that may mask other sources and is expected to increase the total scatter. The lives-to-initiation for specimens CF-4 and CF-5, both from Batch 2, differed by less than 11%, while cracks initiated in CF-6 (Batch 3) almost 100 kcycles later. While cracks in CF-4 and CF-5 both initiated earlier than in the corresponding baseline Batch 2 specimens (F-7 and F-8), the opposite is true for CF-6 with respect to the baseline Batch 3 specimens (F-9 and F-11). This implies that a sensitivity to manufacturing batch may not fully explain the scatter in the results. Given that there were only three specimens, it is difficult to draw conclusions.

8.4 Alternating Corrosion and Fatigue Tests

Three specimens, identified as AE-1, AE-2 and AE-3, were fatigued to failure following the alternating corrosion and fatigue schedule defined in Section 6.4. The data are included in Table 8-1.

8.4.1 NDE Results

Figures 8-17, 8-18 and 8-19 compare the NDE results. Additional images are included for AE-1 and AE-3 because these splices were opened. As with the pre-corrosion specimens, each set of images provides good qualitative similarity. The AE specimens exhibited higher average and peak levels of corrosion than did the pre-corrosion specimens despite being subjected to environmental corrosion for the same periods.

The thickness-loss indicated by the caliper-gauge measurements for these specimens was included in Figure 8-13. The average thickness-loss was less than 0.0024 inches (0.061 mm) or 3.0% of the total joint thickness; the peak thickness-loss was less than 0.0060 inches (0.152 mm). Although AE-3 was the least heavily corroded on average, it showed the highest peak values and the lowest life-to-initiation of the three AE specimens. Conversely, AE-1 was the most heavily corroded, on average, but the damage was much more even; life-to-initiation was the longest.

The breakdown for the critical row rivet reference areas is presented in Figures 8-20a, b and c. The average thickness-loss around the rivets in all three specimens was less than 0.0024 inches (0.061 mm); the peak thickness-loss was less than 0.0042 inches (0.107 mm). As with the pre-corrosion results, the results suggest an inverse relationship between average thickness-loss and life-to-initiation at the rivets within each specimen.

8.4.2 Fatigue Testing Results

Crack-growth and rates of crack-growth for AE-1, AE-2 and AE-3 are plotted in Figures 8-21a, b and c. Cracks initiated at 325, 200 and 150 kcycles respectively. Initiation and growth characteristics were consistent with the fatigue-only and pre-corrosion tests (Sections 8.1.1 and 8.3.2) with the following differences:

- a. The presence of corrosion in this case had no discernible effect on initiation life when the average of all AE specimens was compared to the average fatigue-only values. When only Batch 2 was considered, a difference emerged. This will be treated in Section 8.5.
- b. The crack-initiation portion of life showed less scatter in percentage terms.
- c. The first crack initiated within the four centre rivets (3 to 6) unlike the pre-corrosion tests, in which cracks always initiated at rivets 4 or 5. Uniform MSD developed, however, in all three cases, producing patterns similar to the fatigue-only tests. Test AE-2 had the least uniform corrosion with five cracks existing at first link-up. The initial crack pattern in this specimen, beginning at rivets 3 and 6 with nothing evident at 4 and 5 for 30 kcycles, suggests that corrosion may have played a role here, causing cracks to initiate at rivets 3 and 6 earlier than would otherwise have occurred. In all other specimens, cracks initiated at 4 or 5.

The crack-growth portions of specimen life were included in Figure 8-15. The results are similar to one another and to those for the pre-corrosion specimens. Crack-growth after the half-inch aggregate crack length is relatively unchanged and there is no obvious evidence of higher crack-growth rates. Specimen AE-1 took significantly longer to crack from the half-inch mark. Inspection of the crack-growth plot reveals that several cracks emerged shortly after the half-inch point, producing a similar delay to that seen in F-9 (Section 8.1.1).

8.4.3 Strain Gauge Results

Membrane and bending stress distributions were similar for all three specimens.

As with the pre-corrosion specimens, the attempt to monitor stress changes induced by corrosion pillowing using gauges installed prior to exposure in the chamber was unsuccessful (Section 7.4).

On AE-3 (Figure 8-22) however, gauges installed prior to the first fatigue block (0 to 75 kcycles) to monitor crack-growth survived the first exposure block and indicated crack-growth towards the end of the second fatigue block (75 to 150 kcycles). After the second exposure block, cracks immediately initiated at 3R and subsequently at 5R. The cracks at both sites were probably present during the second exposure block but were small and therefore tightly closed. No visible evidence of corrosion was found on these crack faces on opening but any such damage would likely have been destroyed by rubbing of the crack faces during the test. The jumps in the readings at 75 and 150 kcycles are due to disconnecting and reconnecting the gauges for each exposure block which changed the resistance in the gauge leg of the bridge, requiring a recalibration. Nevertheless, strong crack indications between 75 and 150 kcycles for 3R (3R falling steadily) and for 5R (5L increasing steadily) preceded initiation at both locations. The difference in the indications is explained as follows: the decrease in 3R with constant 3L is attributed to a developing crack at 3L which became the third visible crack. This hidden growth apparently prevented the stress from increasing at the 3L gauge. Instead, the load at this rivet must have been transferred to adjacent rivets. On the other hand, cracking at 5L occurred much later than at 5R; in this cases, load was shifted to the left-hand side, perhaps helping to initiate 5L. Readings from the gauges for the final fatigue block were unreliable: during the second exposure period at 150 kcycles the gauges at rivet 4 disbonded completely; those at 3L and 5L had suffered visible attack, making the data unreliable.

The gauges on AE-1 and AE-2 were only installed after both exposure blocks and were successful in indicating cracks (The stress plots are included in Appendix B). In AE-1 visible initiation occurred at 200 kcycles. It is therefore possible that cracks existed during the final exposure block. AE-1 initiated at 325 kcycles and, given the indications typical of other specimens, is therefore almost certain not to have had substantial cracks during exposure. This does not, however, imply that it can be compared with the pre-exposure specimens. Fatigue damage created before exposure may have interacted with the subsequent corrosion, as seen in Section 4.4.2.

Figure 8-23 compares the crack detection techniques used in the corroded specimens (except CF-6), in the same format as for Figure 8-6 (Section 8.1.3). Study of the five graphs shows that strain gages indicated the first crack in all cases and often indicated the existence of cracks 100 kcycles before visible initiation. The graphs also again show that cracks on either side of a rivet generally initiate within 50 kcycles of one another and frequently much less.

8.4.4 Discussion

As noted previously, the presence of corrosion is expected to increase scatter. Here, too, Batch Number appears to correlate with the initiation life results. The lives-to-initiation for specimens AE-2 and AE-3, both from Batch 2, differed by less than 28%, while cracks initiated in AE-1 (Batch 1) about 150 kcycles later. Lives-to-initiation in AE-2 and AE-3 lie between the Batch 2 baseline (F-7 and F-8) and pre-corrosion (CF-4 and CF-5) results. This agrees with the hypothesis that corrosion applied during life should be less severe than beginning life fully corroded. Because AE-1 is the only specimen from Batch 1, it cannot be compared.

8.5 Final Analysis

This chapter presented the results of four groups of specimens tested. Including the previous results of Eastaugh and Wakeman, 23 specimens have now been fatigue tested, of which 20 have been to failure (Table 8-1). All have developed MSD, of varying uniformity, in the fatigue-critical upper rivet row. Crack patterns, stress distributions and secondary bending results are similar to results from aircraft lap joints. The specimen design therefore produces realistic MSD development, from nucleation to the fully-cracked state.

Table 8-4 and Figure 8-24 show the average life, in cycles and in percent, for baseline fatigue (F), pre-corrosion (CF) and alternating corrosion and fatigue (AE) specimens. Specimens F-2 and F-5 were not included because oil-contamination and glue migration during manufacture affected them [13]. Specimens F-10 and F-11 were not included since they were tested only to first visible initiation. Because of the possible correlation between initiation life and the manufacturing batch, the results for the six specimens in Batch 2 are also treated separately.

Table 8-4: *Comparison of Life in Cycles to Fatigue-Only for All Specimens*

Test Type	No. of Tests	Cycles to: (Percent of Fatigue-Only:)			
		Initiation	Half-Inch	First Link-Up	Total Life
All Specimens To Date					
Fatigue-Only	8	242 445	n/a	303 949	309 164
Pre-Corrosion + Fatigue	6	149 308 (61.6%)	n/a	197 445 (65.0%)	206 026 (66.6%)
Pre-Corrosion (CF-1..3)	3	130 283 (53.7%)	n/a	181 107 (59.6%)	188 920 (61.1%)
Pre-Corrosion (CF-4..6)	3	168 333 (69.4%)	n/a	213 783 (70.3%)	223 133 (72.2%)
Alternating Exposure	3	225 333 (92.9%)	242 035	283 130 (93.2%)	289 692 (93.7%)
Batch 2 Only					
Fatigue-Only (F-7 & 8)	2	213 000	233 191	271 726	278 850
Pre-Corrosion (CF-4 & 5)	2	137 500 (64.6%)	164 293 (70.5%)	197 650 (72.7%)	205 250 (73.6%)
Alternating (AE-2 & 3)	2	175 500 (82.4%)	193 423 (82.9%)	224 445 (82.6%)	230 688 (82.7%)

Examination of the figure and table reveals the following when all tests are considered:

1. Pre-corrosion reduces total life in cycles by about 38% over fatigue alone.
2. Pre-corrosion reduces life-to-initiation as a percentage of total life from 88% to 73%, reinforcing the idea that pre-corrosion leads to corrosion-dominated initiation and hence less uniform MSD.
3. Alternating corrosion and fatigue has little effect on total life or on the fractions of life spent in each stage, as compared to fatigue alone.

When the specimens in Batch 2 are treated separately, the following features are noted:

1. The average life values in Batch 2 for fatigue-only and pre-corrosion do not differ greatly from the overall data.
2. Alternating corrosion and fatigue has an apparent linear effect which decreases the total life in cycles but does not change the fractions of life spent in each stage. For the Batch 2 specimens the life in cycles at all stages is reduced to about 83% of the fatigue-only value. Note that this is due to an estimated average thickness-loss of about 3%.

Comparison of results suggests that alternating corrosion and fatigue may have an effect on life which is masked when all the data is taken together. Because there were only three AE specimens, the data is easily influenced by one high or low value.

Consideration of possible manufacturing differences reduces scatter in the results. The life-to-initiation of the shortest- and longest-lived specimens differ by a factor of 4.3. The life-to-initiation of each pair of specimens in Batch 2 varies by less than 1.4.

The degree of corrosion in the six corroded specimens varied. Examination of CF-4, CF-5 and CF-6 using the Portascan system suggests that heavier corrosion was achieved in

these three specimens than in those tested by Wakeman. This is partially due to the longer total exposure time (56 days versus 52) but is in greater measure probably due to the breakdown of this time. Wakeman used three blocks of 21, 21 and 10 days with considerable time in between during which some drying was unavoidable. The present program used only two blocks, of 21 and 35 days, with a single break of 42 days. Using fewer longer blocks was probably more severe.

The higher corrosion in the alternating exposure specimens suggests that fatigue damage interacts with corrosion in some way, despite occurring sequentially. This is consistent with the results of Du et al. (Section 4.4.2). Most likely, the mechanism is that the fatigue-damaged sites are sensitive to attack leading to more corrosion around the rivet holes and a consequent reduction in initiation life. This reduction is seen when life is plotted from the point at which the corrosion is introduced (Figure 8-25); the alternating exposure specimens, with higher average corrosion, failed more quickly than the pre-corrosion specimens.

9. CONCLUSIONS

This thesis forms part of an ongoing study of MSD and corrosion as it occurs in aircraft lap joints. An existing coupon-type specimen designed to produce MSD was used through-out. The thesis began by studying the factors in lap joint design which affect fatigue behaviour. Combined with subsequent fatigue testing using two different countersunk rivet types, this study identified the following two features as the most significant:

1. The residual stress distribution created during riveting determines the resultant rivet-and-hole interaction and the likely crack nucleation locations. Rivet squeeze force has been previously identified as an important variable.
2. Secondary bending plays a key role in the initiation life of a lap joint. It is affected by the rivet dimensions and the riveting process, in addition to the overall splice geometry. Peak bending stresses adjacent to a typical countersunk head are higher than those near a driven-head. Further, for countersunk heads, the peak at a knife-edge is higher than that for a shallower countersink. This result has two conflicting implications for aircraft: a) knife-edge countersinks lead to earlier crack initiation and should therefore be avoided, but b) shallower countersinks may delay initiation enough that the joint will be at risk of cracking in the (hidden) inner skin before the (visible) outer skin.

Fourteen specimens in four groups were fatigue tested. The following conclusions can be made with respect to fatigue of the MSD specimen:

3. Scatter of total fatigue life was primarily confined to differences in initiation life. Test results revealed a possible correlation between life-to-initiation and manufacturing batch.

4. Aggregate crack-growth after initiation was consistent and repeatable despite variation in MSD uniformity; that is, when fewer cracks were present, the crack-ing life remained unchanged, implying that the fewer cracks grew faster due to the higher load per crack tip.
5. Strain gauges were successfully used to detect load-shedding caused by crack-growth beneath the rivet-heads. Positive indications were obtained for the first visible crack in every specimen and for about 50% of subsequent cracks. The results suggest that load redistribution begins up to 100 kcycles before visible initiation. The state of crack development at this point is unknown.
6. Acoustic emission successfully detected crack-growth beneath the rivet-head. Consistent crack-growth indications were detected up to 140 kcycles before visible initiation. The state of crack development at this point is unknown. Acoustic events were also detected early in the test which may be due to localised yielding during initial cycling.

Six specimens were corroded using a previously-developed, accelerated process. Several NDE techniques were used to examine the corrosion in these joints. The following conclusions can be made:

7. It is generally accepted that the means and importance of characterising corrosion are poorly defined and acknowledged, respectively. For characterising corrosion in lap joints, two parameters which were found useful are thickness-loss of material, and overall joint thickness due to corrosion pillowing between two sheets.
8. Diffracto-Sight, Shadow Moiré, Eddy-Current and Ultrasonic techniques, along with caliper-gauge measurements of the overall thickness, provided consistent qualitative pictures of the corrosion development within the joints and correlated well with each other and with the actual corrosion product distribution.

9. Analysis of the thickness measurements produced by ultrasound and caliper thickness measurements showed differences in the absolute degree of corrosion in terms of thickness-loss. Because the techniques respond to different parameters this was not surprising. Nevertheless, the results should allow the techniques to be calibrated with respect to one another.

Both a pre-corrosion and an alternating corrosion and fatigue schedule were used. The following conclusions can be made:

10. Interaction between corrosion and fatigue was evident despite the sequential application.
11. Alternate exposure specimens were on average more heavily corroded despite undergoing the same periods of accelerated corrosion development and had shorter lives-to-initiation when measured from the time of corrosion.
12. Pre-corrosion produced less uniform MSD than alternating corrosion and fatigue. At present, too little information is available to determine what degree of uniformity is seen in corroded aircraft. Alternating corrosion and fatigue should more closely match the experience of aircraft because it provides for some interaction between fatigue and subsequent corrosion, as occurs in aircraft.
13. Nevertheless, pre-corrosion results are important because they simulate the case of an aircraft whose protective systems have failed very early in life and/or one which operates in a severe environment. Pre-corrosion thus defines one end of the spectrum of aircraft usage.
14. Pre-corrosion reduced average life-to-initiation by 38% and overall life by 33%. The MSD produced was less uniform; on average, only half as many visible cracks existed at first link-up as did in the fatigue-only specimens. When the

possibility of manufacturing differences was considered, pre-corrosion reduced average life-to-initiation by 35% and overall life by 26%.

15. Alternate corrosion and fatigue reduced the life-to-initiation by 7% and the overall life by 6%. Accounting for the possibility of manufacturing differences produced reductions of 18 and 17% respectively.
16. Corrosion tended to concentrate between rivets as expected and was heavier in the centre of the joint near the upper critical row. This may have been due to the orientation of the lap joint during corrosion, with the critical row highest. This is the same orientation as occurs on aircraft.
17. A simple means of revealing any correlation between average thickness-loss near a rivet and life-to-initiation at the rivet was presented. Life-to-initiation was somewhat sensitive to maximum corrosion levels within the joint. First visible cracks tended to initiate at the rivets in the most heavily corroded regions.
18. Early crack-growth (i.e. immediately after visible initiation) was up to 10 times faster in the pre-corroded specimens. This reflects a) higher loads per crack tip at the smaller number of active cracks seen in the less uniform MSD; b) possible increases in stress induced by corrosion pillowing; and c) higher stresses due to corrosion thinning and damage-induced stress concentrations in the crack path. The early crack-growth rates of the alternating corrosion tests lay between the baseline and pre-corroded results.

9.1 Further Work

Several suggestions for further work were identified:

1. Any further testing, with and without corrosion, must consider the sources of scatter. To this end, specimens should be manufactured in larger batches; tool and machine variables, such as tool rotation rate and feed rate, should be recorded. Consideration of scatter due to corrosion variation would be helpful: use of pre-exposure before specimen assembly might be considered to condition the faying surfaces with the goal of promoting more uniform corrosion at all rivets, but in any such attempt the interaction between fatigue and corrosion should be considered. At the same time the causes of the differences between the lives-to-initiation of small-scale test coupons and full-scale fuselage structures need to be identified.
2. Fatigue-only testing can now proceed to the task of identifying the effect of other joint variables on both initiation and aggregate crack-growth. Of initial interest is whether the consistent aggregate crack-growth seen here is peculiar to this specimen design or is characteristic of lap joints. The potential may, for instance, exist to predict life of a given frame bay and lap joint configuration from a point shortly after initiation, regardless of crack pattern.
3. A first step in this direction could be to increase, from four, the number of MSD sensitive rivets, either by widening the existing specimen or by reducing rivet spacing. Ten rivets 0.8 inches (20.3 mm) apart might yield six such locations, a 50% increase. It might also be of interest to test a specimen with starter cracks beneath the rivet-head at some or all rivets. This could be used to investigate the aggregate crack length concept and to determine how sensitive the outer rivets are to cracking.

4. The differences between the specimen design and an aircraft lap joint also need to be considered. Consideration should be given to including a stiffener strap to simulate a stringer, and the use of faying surface sealants. The effect of pressurisation pillowing needs to be quantified to compare it with the present results. Finite element modelling may be appropriate for this study.
5. The role of secondary bending needs to be further investigated. The first step should be to determine the rivet tilt in the existing specimen configuration to see if the Schijve model modified to account for tilt is sufficiently accurate to avoid complex 3D finite element analyses. If necessary, a degree of insight could probably also be achieved using fairly simple FE models based upon the Schijve model.
6. Crack detection using strain gauges can no doubt be improved with detailed analysis of the stress distribution and careful selection of gauge type and positioning.
7. Crack detection using acoustic emission is promising. The next step is to calibrate the energy output of the acoustic signals with the crack face area. This should improve understanding of the growth process of cracks while still beneath the rivet-heads, the largest portion of joint life.
8. Thickness-loss needs to be accurately quantified in order to determine error in the various NDE techniques and to allow their calibration with respect to one another. The data presented here could form the basis of a catalogue of NDE scans which would fully characterise corroded joints.
9. It is of interest to determine whether the apparent linear effect of corrosion on specimen life seen in the small sample of alternating exposure tests was a coincidence. The potential for a safety factor to account for corrosion in design is intriguing.

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- 92 MTS TestStar II Application Manual, Version 3.0a, MTS Systems Corporation, 1995.
- 93 ASTM E-647-93; Standard Test Method for Measurement of Fatigue Crack Growth Rates; American Society for Testing and Materials, April 1993.

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457 keycycles: Two small cracks emerging at 4L.

521 keycycles: Upper is dominating. Lower appears to have emerged as 4R.

549 keycycles: Third crack emerging at 4L between first two.

551 keycycles: Third crack growing up into lead crack.

555 keycycles: Third crack has joined lead crack.

559 keycycles: Third crack dominates. Original two cracks have closed up.

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Faying Surface of Outer Sheet
(reversed to match other images)

Faying Surface of Inner Sheet

D-Sight Image

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Faying Surface of Outer Sheet
(reversed to match other images)

Faying Surface of Inner Sheet

D-Sight Image

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Faying Surface of Outer Sheet
(reversed to match other images)

Faying Surface of Inner Sheet

D-Sight Image

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APPENDIX A

NDE EQUIPMENT

Caliper Gauge:	Mitutoyo Model 118-107 0-1" Sheet Metal Calipre Gauge.
D-Sight:	DiffRACTO Aircraft Inspection System (DAIS) Model 250C.
Eddy-Current:	<p>Zetec MIZ-40A Eddy Current Instrument for thickness-loss measurements with NDT ENG CORP RSNG-500-2.0L and RSNG-250-2.0L probes plus:</p> <ol style="list-style-type: none">1. DuPont Portascan MS1-15x18 X-Y Frame; or2. UTEX Winspect Data Acquisition Software with Automated Scanning Frame. <p>Nortel 30 for crack detection.</p>
Ultrasound:	<p>Panametrics Epoch III Model 2300 in pulse echo mode for crack detection.</p> <p>Novascope 3000 for thickness-loss measurements.</p>
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