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**INTERACTION OF MULTIPLE SITE FATIGUE DAMAGE
AND CORROSION IN FUSELAGE LAP SPLICES**

by

R. Brett Wakeman, B.Eng

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
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submitted by R. Brett Wakeman
in partial fulfilment of the requirements
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Thesis Supervisor

Chair, Department of Mechanical and Aerospace Engineering

Carleton University
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Many thanks goes to Professor Paul V. Straznicky, who not only was responsible for my thesis supervision, but whose counsel I came to rely upon when tackling the many problems faced. His knowledge and experience were a constant and dependable source of support.

Graeme F. Eastaugh, the predecessor of this project, successfully developed the design and manufacture of the MSD specimen which was used for this project.

David L. Simpson, Chief, Structures, at the IAR, co-supervised the project, organized IAR support, and provided expert advice. The entire staff of the IAR is sincerely thanked for all their efforts in making this project such a success. Specifically,

- MSD corrosion expertise: GF Eastaugh, JP Komorowski, NC Bellinger;
- Non-destructive inspection: JP Komorowski, RW Gould, A Marincak;
- Fatigue testing: Dr RL Hewitt, BE Shaver, J Heath, I Benak, MTS;
- Microscopy/fractography/SEM: RV Dainty, D Chow, P Gaudert;
- Instrumentation: PA Adams, RA Brackett;
- Manufacturing: Institute for Advanced Mfg. Technology, MI Edkins, A Luteyn.

And last, a special thanks goes to my fiancé Jennifer, whose undying love, support and encouragement throughout this work made everything possible.

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- C. Technique to Quantify Corrosion Damage

NOMENCLATURE

σ_{bend}	Bending stress at the critical row
σ_{local}	Membrane stress at the critical row
σ_{remote}	Membrane stress at the nominal position
AAWG	Airworthiness Assurance Working Group
AGARD	Advisory Group for Aerospace Research and Development
AD	Airworthiness Directive (FAA)
CASS	Copper-Accelerated Acetic Acid-Salt Spray Testing Method
CF-x	Pre-corroded, fatigue in air specimen no. x
CFTCP	Corrosion Fatigue Cooperative Testing Programme
CPCP	Corrosion Prevention and Control Program
CT-1	Corrosion trial specimen
D Sight	Diffraeto Sight
DAIS	D Sight Aircraft Inspection System
EXCO	Exfoliation Corrosion Testing Method
F-x	Baseline fatigue in air specimen no. x
FAA	Federal Airworthiness Administration (U.S.)
FACT	Fatigue in Aircraft Corrosion Testing Programme
FAR	Federal Airworthiness Regulations
IAR	Institute for Aerospace Research (NRC)
MED	Multiple element damage
MSD	Multiple site damage
NDI	Non-destructive inspection
NRC	National Research Council (Canada)
NTSB	National Transportation Safety Board (U.S.)
PoC	Proof-of-concept specimen
PSE	Principal structural element

SB	Service Bulletin (Boeing)
SEM	Scanning electron microscope
SSID, SID	Supplemental structural inspection document
SSIP	Supplemental structural inspection program
USAF	United States Air Force
WFD, WSFD	Widespread fatigue damage

The practice of the aircraft industry and the Institute for Aerospace Research is quantitative representation using the Imperial system of units. This thesis therefore has adopted the same practice. Conversion factors into the S.I. system of units are given below.

1 inch	=	2.54 cm
1 foot	=	3.048×10^{-1} m
1 lbf	=	4.448 N
1 ksi	=	6.89 MPa

1. INTRODUCTION

The phenomena of Multiple Site Damage (MSD) and corrosion in fuselage lap splices are currently of great interest to continuing airworthiness and structural integrity of aging aircraft. This degradation, in which lap faying (mating) surfaces corrode and fatigue cracks emanate from the edges of several adjacent and collinear rivet holes, was found to contribute significantly to the Aloha Airlines accident in April, 1988. In this incident, an explosive failure occurred leading to the loss of a large section of the upper fuselage. It was caused by an "unzipping" effect of many MSD and larger cracks in lap splices, most of which had become disbanded and corroded [1].

Since the Aloha incident, there has been what might be called a "structural integrity revolution" in the aircraft industry. It has spurred a myriad of research projects and several international symposia and conferences (e.g. [2 - 7]) to bring the needed technical and research focus on the issues involved in the life enhancement and safety assurance of aging airplanes. The present work is intended to contribute to these efforts.

It has been suggested by some that the significance of MSD in rivetted fuselage joints has been vastly overemphasized. Although few accidents have been attributed to MSD, it would be ill advised not to pay adequate attention to the problem, knowing that many aircraft will soon reach or exceed their original design lives. It is estimated that by the year 2000, 64% of the worldwide fleet of U.S.-manufactured aircraft will reach 20 or more years of age. The problem becomes more serious with the addition of lap splice corrosion, a common form of damage found in aircraft operating in harsh environments.

Unfortunately, the interaction of corrosion with fatigue in fuselage lap splices is not well understood and as such, has not typically been considered by manufacturers. Given the potential threat the combination poses, there is an urgent need for the analysis tools, test methods and regulatory guidelines to properly address the issues involved. Research

initiatives to provide the required knowledge and understanding have been few. Therefore, in view of the potential benefits that would arise from research in this area, it was decided to initiate an experimental research project to investigate the effects of lap splice corrosion on MSD initiation and growth.

The objectives of this thesis are:

- (a) to introduce the reader to the two major structural problems of aging transport aircraft: that of MSD and corrosion in fuselage lap splices;
- (b) to describe the results of an experimental research project, whose goal was to evaluate the effects of lap splice corrosion on MSD initiation and growth characteristics through comparisons with baseline results for uncorroded splices; and
- (c) to provide the community of researchers in the fields of continuing airworthiness and structural integrity of aircraft with much needed data and insight into the problem of MSD and corrosion in fuselage lap splices of aging aircraft.

Chapters 2 and 3 of this thesis provide the background into the MSD problem and the airframe corrosion problem, respectively. Chapter 4 provides the fundamentals in simulating the interaction of corrosion and fatigue in lap splices, and reviews the available literature written on the subject. Based on these analyses, the relevant research needs are identified and an experimental research project is defined.

The experimental work employs a coupon-type specimen designed by the predecessor of this project, Mr. G.F. Eastaugh [8]. The specimen approximates on-aircraft conditions and is capable of developing and growing MSD in a realistic manner. A series of eight fatigue tests was conducted, five specimens with no corrosion, and three with

pre-corrosion. Pre-corrosion was used for its simplicity and its high likelihood of producing conservative yet useful comparative results in relatively short timeframes.

The trends which emerged as a result of the introduction of corrosion are described and analyzed. Finally, conclusions and recommendations based on the experimental results are presented.

2. OVERVIEW OF THE MSD PROBLEM

2.1 Explanation of Terminology

As a consequence of the Aloha incident the Airworthiness Assurance Working Group (AAWG) was formed in August of 1988 by the Air Transport Association of America and the Aerospace Industries Association of America to promote discussion between the manufacturers, airlines and authorities on issues relating to aging aircraft. The Widespread Fatigue Damage (WFD) sub-committee was formed in 1991 to coordinate research and analytical methods for the interpretation of WFD, resulting in the following definitions [9]:

Widespread Fatigue Damage (WFD, or WSEFD): 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.

MSD and MED are both manifestations, and as such, subsets of WFD. The type of MSD and MED is referred to as 'local' if the resulting crack, after linkage of adjacent finite cracks, does not degrade the structure below limit load capability. Whereas local damage is covered by the existing regulations, widespread damage is not. An example illustrating the differences between local and widespread MSD or MED is shown in Fig. 2.1. "Lap

splice MSD" is referred to here as the occurrence of several cracks emanating from adjacent and collinear rivet holes in a longitudinal joint that connects two sheets of skin in a fuselage. MSD, which from now on will always refer to lap splice MSD, is caused primarily through the repeated pressurization de-pressurization cycles corresponding to each flight. An example of MSD taken from the Aloha accident aircraft is shown in Fig. 2.2.

For proper assessment of the WFD phenomena, several parameters have been defined allowing the determination of necessary maintenance programs. The terms are illustrated in Fig. 2.3 with respect to time (flight cycles) and are defined as:

Initiation of WFD: The first finite crack is expected in a group of details where WFD is likely.

Onset of WFD: The crack growth rate and/or residual strength characteristics start to diverge significantly from those attributable to local cracking.

Critical Damage Extent: The damage has grown such that the residual strength loads can just be sustained.

Monitoring Period: The interval between INITIATION and ONSET.

The upper diagram in Fig. 2.3 depicts the loss in residual strength as a result of the presence of MSD, where residual strength refers to remaining strength, or more accurately, the fracture stress of the cracked structure. The lower diagram illustrates the resulting change in the frequency of inspections required throughout the monitoring period.

Interestingly, the Industry Committee on WFD, formed in January of 1991, arrived at these definitions in March of 1992, almost four years after the Aloha incident. This

illustrates the inherent complexity of the problem and the effort involved in arriving at standard definitions of the key elements.

Finally, note that the terms *lap splice*, *lap joint*, or *skin splice* will be used interchangeably throughout this thesis. A typical fuselage lap splice construction is shown in Fig. 2.4.

2.2 Historical Background

2.2.1 The Aloha Incident

Information concerning the Aloha incident was retrieved primarily from the National Transportation Safety Board's (NTSB) aircraft accident report [1], as well as from other published sources [11,12,13].

As previously mentioned, the MSD problem was thrust into the limelight with the Aloha Airlines accident which occurred on April 28, 1988, in the Hawaiian islands. A Boeing 737-200 passenger aircraft on its 89,681st flight and in its 35,496th hour experienced an explosive decompression shortly after leveling at an altitude of 24,000 feet. In the explosion, an approximately 18 foot section of the upper fuselage just behind the cockpit was completely blown off. Miraculously, the fuselage held together by the remaining floor support beams and seat rails, a testimony to the B-737 residual strength. A photograph of the left side of the forward fuselage after landing is shown in Fig. 2.5.

The aircraft, production number 152, was delivered new to Aloha in April 1969, and from there it averaged 13 flights per day over the next 19 years, operating 24-25 min. sectors throughout the Hawaiian islands. The Aloha Airlines' fleet of 11 B-737s were accumulating cycles at a faster rate than any other operator, in addition to operating in a

severe corrosive environment. Whereas Boeing economic design life projections were based on accumulating $1\frac{1}{2}$ cycles per flight hour, Aloha aircraft were accumulating approximately 3 cycles per hour. Boeing 737s were originally designed for an economic service life of 20 years, 51,000 flight hours and 75,000 cycles. Two other Aloha B-737s, including the fleet leader in cycles, had accumulated 90,051 and 85,409 cycles. Both of these were dismantled and sold for scrap and parts following the accident.

The more pertinent findings from the post-accident investigation by the NTSB were:

- The failure initiated in the lap joint along S-101L (stringer 10 left side; see Fig. 2.5); the failure was a result of lap joint MSD in the upper rivet row and tear strap disbond which negated the fail-safe characteristics of the fuselage.
- The fatigue cracking initiated from the knife edge associated with the countersunk lap joint rivet holes; the knife edge concentrated stresses that were transferred through the rivets because of lap joint disbonding.
- The disbonding of lap joints originated from manufacturing difficulties with surface preparation and/or bond material processing during the construction of the airplane. This resulted in bonds with low environmental durability or a lack of bonding.
- Although a representative fuselage section of a Boeing 737 was tested to 150,000 cycles (two lifetimes) during certification, the test did not reflect the fatigue performance of the actual fleet aircraft. The test parameters did not consider the long term effects of disbonding, corrosion, and fatigue cracking in the lap joints as experienced in airline service.

For early production B-737s through to tail number 291, the 0.036 inch thick fuselage skins were "cold" bonded using an epoxy impregnated woven "scrim" cloth, and rivetted

using three rows of countersunk rivets. Early B-727s (Nos. 1-849) and B-747s (Nos. 1-200) also employed cold-bonded lap joints. In this design, fuselage hoop loads were intended to be carried through the bonded joint, rather than the rivets, which were to provide an alternate load path. This allowed for the use of lighter and thinner skin panels with no degradation in fatigue life. However, as mentioned above, early service history with cold-bonded lap joints revealed that the quality of the bonds was inconsistent leading to occurrences of early disbonding and premature fatigue cracking. The cold bond process was discontinued in 1972, and for B-737s number 292 onwards, a redesigned smooth, close-fitting, and thicker lap joint completely sealed using a chromated polysulphide interfaying compound was introduced. Later B-727s (No. 850 onwards) and B-747s (No. 201 onwards) also included faying surface sealed lap joints. From a recent Boeing report [14], this configuration appears to provide superior durability.

The occurrences of disbonding, corrosion, and multiple fatigue cracks in B-737 lap joints had been addressed by Boeing prior to the Aloha accident. Boeing had released a Service Bulletin (SB), SB 737-53-1039 [15], as early as July 19, 1972 to initially address lap joint corrosion (note that compliance with SBs is voluntary). An update followed in Feb. 1974 after Boeing received 30 reports of lap joint disbonds and corrosion. In 1984, Aloha reported a 7½ inch crack along a lap joint at S-10R, although no subsequent actions were taken. The SB was upgraded to "Alert" status on Aug. 20, 1987 following an operator's report in April of that year, of multiple fatigue cracks on three airplanes all in the range of 41,000 flight hours (47,000 cycles). In response to the alert SB, the U.S. Federal Aviation Administration (FAA) issued an Airworthiness Directive (AD) 87-21-08 effective Nov. 2, 1987, which mandated the inspection for fatigue cracks of B-737 line numbers 1 to 291, but only for No. 4 stringer lap joints, and not all locations referenced in the SB.

The NTSB accident report states that, although there is evidence from the wreckage that Aloha performed the mandated repairs, there is considerable doubt as to whether eddy current inspections required upon the discovery of cracks were ever completed.

The probable cause for the accident was determined by the NTSB to be the failure of the Aloha Airlines maintenance program to detect the presence of significant disbonding, corrosion, and fatigue damage which ultimately led to the failure of the lap joint at S-101 and the separation of the fuselage upper lobe. Other contributing factors were:

- failure of airline management to properly supervise its maintenance force;
- failure of FAA to properly evaluate Aloha's maintenance and inspection programs;
- failure of FAA AD 87-21-08 to require inspection of all the lap joints proposed by Boeing Alert SB 737-53A1039;
- lack of complete terminating action, neither generated by Boeing nor required by the FAA, after the discovery of difficulties in the early production B-737 cold-bond lap joint. (A *terminating action* is an action to overcome potential structural problems before they occur.)

The NTSB investigation generated several priority action recommendations, which included:

- Require that all turbojet transport category airplanes certified in the future, undergo full-scale structural fatigue testing to a minimum of two times the projected economic service life (instead of one); also require that all current aircraft in this category be subjected to such testing. As a result of this testing and subsequent inspection and analysis, require manufacturers to identify structure

susceptible to MSD and adopt inspection programs appropriate for the detection of such damage.

- Discontinue classification of fuselage skin as "malfunction evident" or "damage obvious" on supplemental structural inspection documents. This classification will be explained in the next section.

At the time the B-737 was certified in 1967, regulations required only that the fuselage be capable of sustaining 80% of limit load with any complete or partial failure of any single structural element. Boeing's fail-safe design, which includes longitudinal and circumferential tear straps every 10 inches, had proven that it could withstand a 40 inch crack (two frame bays) without suffering catastrophic failure. The straps or frames redirect a growing crack perpendicularly causing the skin to "flap" open, and allow for a controlled decompression of the fuselage. The problem was that no consideration was given to the widespread development and link-up of MSD cracks in multiple frame bays which might develop during extended service or on aircraft operating in harsh environments.

When considering the Aloha incident, it is reasonable to conclude that the combination of circumstances was unusual: a heavy usage aircraft flying in a severe corrosive environment for an operator with questionable maintenance practices. However, aircraft fatigue history shows that if one specimen fails at 89,000 flights, another could fail at half or even a third of that life. The accident, therefore, is clearly not of remote concern.

Finally, note that there had been one previous in-flight B-737 fuselage failure over Taiwan on Aug. 22, 1981, although in this case, the failure was in the belly section and led to mid-air disintegration. The 737-200 was a 33,000 flight high-timer with a history of belly bilge corrosion, exacerbated by seawater spilling out of fish tanks, a regular

cargo carried by the operator. Boeing and the FAA reacted quickly with alert SBs and ADs.

2.2.2 Supplemental Fatigue Inspections of Aging Aircraft

During the mid- and late 1970s, some airplanes of the jet transport fleet began to approach their original lifetime design objectives. The need was recognized for a structural reassessment and additional directed inspections to supplement existing maintenance actions to monitor high-time airplanes for the potential onset of fatigue damage with increasing age. A collaborative effort between the manufacturers, airlines, and the airworthiness authorities resulted in the formation of supplemental structural inspection programs (SSIP) and documents (SSID). For each airplane model, approximately 50 principal structural elements (PSE) were selected for monitoring. No test or service fatigue damage had been experienced in any of these areas; their selection was based on the potential consequences of undetected cracking.

The SSIDs for B-707, B-727, B-737, and B-747 are mandatory, apply to high-time airplanes at specific thresholds, require specified repeat inspections, and have been in place for approximately 12 years. If any damage is found, the appropriate corrective action is taken through the release of an AD-mandated SB applicable to all airplanes of the affected model, and the PSE is then dropped from the program.

In creating the Boeing SSIP, a complete reassessment of the affected airplanes was performed using the revised 1978 Federal Airworthiness Regulations (FAR) 25.571 [16]. The revised regulations required consideration of damage growth characteristics at multiple sites, and an inspection program to ensure that damage is detected before the airplane residual strength drops below regulatory fail-safe requirements. This is familiarly known as the *damage tolerance* concept. The reassessment involved

determining residual strength in the presence of multiple active cracks, extensive analysis of crack growth rates, and incorporation of these engineering analyses into the airplane's maintenance program.

For the B-737, Boeing had recommended a candidate fleet of about 125 airplanes to be inspected under the SSIP (including the accident aircraft). Unfortunately, thin gauge fuselage skins were not included in the original SSID since Boeing classified them under "damage obvious or malfunction evident". This classification reflected the test and service evidence which indicated that skin cracks would, as previously stated, turn at frame locations and result in safe decompression through flapping failure.

This 'safe' failure did not occur in the Aloha incident, and prompted a review of the inspection programs to ensure adequate protection of an aging fleet. The review was conducted during 1989 and 1990, and by mid-1991, revisions to B-727, B-737, and B-747 SSIDs were issued and included changes to approximately 10 significant structural items for each model, including coverage of thin gauge fuselage structure (0.056 inch or less for models 727/737). Also included were considerations for structure susceptible to widespread MSD with appropriate changes of damage detection periods and inspection intervals. Boeing currently maintains that their FAA-approved inspection program is adequate for detecting lead cracks resulting from MSD before the damage becomes critical.

2.2.3 The Aloha Aftermath

In the post-accident aftermath, in addition to the changes just mentioned in SSIDs, the industry saw a complete overhaul in how aging aircraft were now to be treated. Some of the highlights following the accident were:

- May 4, 1988: all operators of B-737s with more than 30,000 landings are directed to inspect (visual and eddy current) all lap splices at S-4 and -10L and R. There were 49 findings of corrosion or minor cracking from 18 operators, with 14 airplanes exhibiting MSD.
- June, 1988: an airline industry task force recommends to the FAA a modification or replacement program to assure the airworthiness of aging aircraft. The task force is comprised of 150 international experts representing the airlines, airframe manufacturers, regulatory agencies, NASA, and professional aviation mechanics from the U.S. and Asian and European nations.
- Oct. 27, 1988: the mandatory replacement of the rivets along the upper rows of the lap splices is proposed and signed; the final rule is FAA-mandated on May 8, 1989.
- March, 1989: prompted by the industry task force, Boeing releases all of its proposed modifications for early models of 727, 737 and 747. Most will require modification or replacement of structural items such as lap skins, tear straps and fittings where cracking, disbonding or corrosion is found.
- April 17, 1990: FAA issues a directive mandating the proposed Boeing modifications when the affected aircraft reach their economic design goal. A total of 115 U.S. aircraft are immediately affected, with an estimated repair bill of \$142 M.

The requirements of these FAA directives represent a significant change in their aging aircraft policy in that they set a life limit on individual parts considered to be critical structure. This differs considerably from their original approach of repetitive inspections and necessary repairs to assure adequate safety.

Other aircraft manufacturers were also affected. Douglas has since initiated several programs (called "structural audits") since almost half its fleet have exceeded their original design goals [17]. The median age for the DC-8 and DC-9 in 1991 was 24 and 20 years, respectively. In comparison, however, the evidence shows that Douglas lap splices are far more resistant to fatigue cracking and corrosion than early Boeing splices. On the other hand, military aircraft, such as Lockheed's C-5A, C-141, and especially Boeing's KC-135, have experienced both widespread MSD and corrosion in their airframe lap splices [18,19]. As of 1994, the average age of the KC-135 was approximately 34 years, with a current schedule which would keep them in service out to year 2040. The United States Air Force (USAF) has instituted several programs to assess and deal with existing problems and potential aging problems associated with this unprecedented life extension. Unfortunately, specifics on their research programs and results are not readily available in the open literature. Other parties with MSD concerns, with some aircraft undergoing structural audits, are Lockheed (L-1011), Fokker (F28), British Aerospace (BAe 1-11) and Airbus Industrie (A-300; B2, B4-100, B4-200, C4 and F4 models). All are currently members of the sub-committee on WFD, although published information as to their activities was not found.

2.2.4 Lessons Learned

Some crucial lessons learned from this historical perspective are:

- Despite an aircraft's fail-safe design and specialized maintenance and inspection programs, the existence of uncontrollable human factors and unforeseen critical situations can eventually lead to a serious accident.
- Although damage at multiple sites has been addressed in residual strength analyses since the regulatory changes of FAR 25.571 in 1978, the presence of widespread

fatigue damage can significantly reduce these local damage containment areas and negate the fail-safety of the fuselage.

- Current government and industry activities to address aging aircraft issues must be continued to prevent further accidents caused by structural failures. Specifically, inspection, maintenance and repair procedures need to be re-evaluated and their importance bolstered.

Continuing airworthiness and structural integrity of the world's aging fleet will thus be a shared responsibility between the manufacturers, the aviation authorities and the operators with large contributions coming from the supporting research fraternities. With higher and higher repair bills, operators must now weigh the price and availability of new aircraft (around \$55 M) against the operating efficiency and costs to maintain older planes in safe operating condition (around \$2 - \$20 M).

With the problem and some of the issues involved in their historical perspective, the focus will now change to look at some of the specific characteristics of MSD.

2.3 MSD Characteristics

2.3.1 Susceptible Structures

There are a few pre-requisites for an MSD situation to exist. They are:

1. there are numerous closely spaced rivets in the same row;
2. the sheet thickness is low;
3. the stress field is close to uniform;

All of these items can generally be found in longitudinal rivetted lap splices of pressurized fuselage, particularly on commercial transport aircraft. The widespread cracking in the wings of the USAF KC-135 and C-5A suggest that MSD can occur at locations other than the fuselage. It is, however, considerably less common at these locations. Lincoln of the USAF relates MSD in wing structures to high design stress levels and the use of crack sensitive materials [20].

Service experience has shown that MSD usually appears first in lap joints which have become disbonded, corroded or are located in a higher than normal stress area. In a B-737, lap strains can increase from 4 to 6% due to nearby windows and floor beams, and by approximately 30% due to fuselage body bending [21]. Indeed the lap joint which initiated the Aloha failure was located just above the window line. Therefore, it is likely that lap splices at various locations will be at different stages of MSD development, with the splice exhibiting the most advanced MSD being the most critical (i.e. the weakest link).

2.3.2 Crack Initiation Phase

MSD development can generally be divided into two distinct phases: a crack initiation phase and a crack growth phase. For the purposes of this thesis, the crack initiation phase will encompass three events defined in general terms as follows:

1. *Crack Nucleation*: the coalescence of micromechanical damage to form a small crack under the rivet head.
2. *Crack Initiation*: the visible occurrence of the crack tip emerging from under the countersunk rivet head.
3. *Small Crack Growth*: the period of growth between nucleation and initiation.

Two important questions will be answered in this section: where and when does MSD appear in fuselage lap splices?

Where does MSD occur?

MSD tends to initiate at the top row of rivets on the outer skin (the "critical row") for the following reasons:

- this row is a primary load transmission row;
- secondary bending induced by joint eccentricities is a maximum at this row;
- pillowing (bending induced by internal pressurization) is a maximum at this row;
- stress concentrations due to countersunk holes are high.

The situation is illustrated in Fig. 2.6. In view of inspections, it is desirable that cracking occur on the outer skin where it is visible. There is, however, another critical row on the inner sheet, although less critical since there are no countersunk holes and the secondary bending effect is negated to some degree by the pillowing. Evidence of cracking at this location ('B' in Fig. 2.6) has been found in-house and was reported by Schijve [22]. In these cases, knife edges were blunted since the countersinks did not penetrate through the full skin thickness. Schijve further observed that, for failures initiated from the countersunk holes ('A' in Fig. 2.6), some cracks in the other outer row have been observed. Strain measurements on an actual B-737 fuselage conducted by FAA [21] show that in some lap joints, the peak stresses on the inner sheet critical row are similar to those of the outer sheet critical row. The problem remains a concern for the industry.

There is also some evidence that the middle row of countersinks may develop MSD after a replacement of the outer sheet critical row flush-head rivets with Universal (button-head) rivets (a Boeing repair for detected MSD). In the NTSB Aloha inquiry [1], several middle row cracks were found in a repaired lap joint. The FAA has also found similar cracking in full panel and coupon specimens during an evaluation of the effectiveness of

this terminating action repair [23,24]. Inspection procedures need to be modified to account for this potential.

MSD cracks will generally have unequal lengths because of the inhomogeneous hoop (circumferential) stress distribution across each bay caused by skin pillowing. The skin/frame connection restrains the radial expansion causing lower hoop stresses at the frames. An in-service example is shown in Fig. 2.7. This case was reported by Mayville and Warren [25] for an aging B-727 which had multiple cracks in three bays between the frames, but no cracks at rivet holes at the frame locations.

The specific site for crack initiation has already been defined as a point on the outer surface beside the rivet head. However, crack nucleation sites are normally located on the faying surface at the hole edge and/or away from the hole edge. The precise location is influenced primarily by the rivet type, the clamping force, and the local bending. In reviews by Schijve [22, 26], and recent experimental investigations by Wit [27], and Eastaugh et. al [28], the following observations are noted:

1. For countersunk rivets, the clamping force is generally low causing cracks to nucleate at hole edges. High local bending loads will tend to move the nucleation sites away from the rivet centerline towards the top of the hole (left of 'A' in Fig. 2.6). This is offset to some degree by the bearing loads induced through rivet interference fits.
2. For button-head rivets, the clamping force is strong causing cracks to nucleate either away from the hole edge or at the edges of the clamped area.
3. Most nucleation sites can be found within regions of fretting damage, located between the two mating sheets or between the rivet shank and the wall of the rivet hole. (As the fretting products are of an oxide composition, the damage is also referred to as *fretting corrosion* damage.)

4. Numerous crack nuclei may exist before a dominant micro-crack is formed. This can lead to very erratic crack shapes.
5. The first small fatigue crack nuclei initially grow as part-through cracks, stimulated by the bending loads. After penetration of the skin thickness, the fatigue crack becomes a through crack. At this point, the crack tip on the faying surface may be well ahead of the visible crack tip, a phenomenon known as *crack tunnelling*.

Other sites around the rivet holes where cracks may nucleate include initial manufacturing flaws, intermetallic inclusions, and corrosion damage sites such as pits or intergranular cavities (more detail in next chapter). If any of these exist in a high stress concentrated area, their likelihood to start a crack is substantially increased.

When does MSD develop?

Since most lap joints were and continue to be primarily designed for strength, and therefore, have a limited fatigue life, it is just a matter of time before the onset of MSD. Unfortunately, very few investigations focusing on crack initiation life are found in the literature. The most useful data has come from full-scale fatigue tests of new and aged airframe articles.

Boeing has conducted several full-scale tests [29,30,31], some of which have produced visible cracking as early as half the test article's life. For the most part, however, cracks remain hidden for around 65-80% of test life. With respect to the original design lives, and considering normal fatigue scatter, MSD can be expected to develop within these periods.

One question, however, remains: how long, then, do the cracks grow before they become visible? The only evidence found in the literature which can be used to calculate this

value is reported in the NTSB Aloha accident report [1]. In a fractographic examination conducted on some post-accident samples, evidence was found to indicate that crack growth underneath rivet heads was responsible for between 9,500 and 18,000 cycles, or 13 to 24% of the predicted airframe fatigue life (75,000 cycles for B-737). This suggests that crack nucleation may begin at an early stage of life. A further implication is that if MSD is first detected visually, there are likely other cracks present, although masked by the rivet heads. Unfortunately, reliably detecting cracks under rivet heads remains today a difficult task.

Crack nucleation and small crack growth under the rivet head are complicated phenomena whose lives are influenced by the presence of many factors [8,27,28]:

1. *Global stress state:* higher far-field (nominal) stress leads to earlier and more uniform MSD across the critical row.
2. *Local stress state:* influenced by sheet thicknesses, local bond quality, stringer design, joint dimensions. Disbonding and high bending promote earlier crack nucleation. Peak stresses are highly influenced by rivet type, clamping force/area, bearing loads, and surface finish and treatment.
3. *Local load transfer:* influenced by rivetting details (rivet pitch, diameter, stiffness, clamping force/area), and bond integrity. Different configurations will alter (2).
4. *Fretting:* a combination of high relative motion and contact forces will lead to earlier crack nucleation [32]. The amount of fretting is influenced primarily by (2) and (3). The situation may be improved with the use of an anti-fretting sealant.

5. *Environment*: ingress of corrosive media to local faying surfaces and rivet/hole interfaces may promote earlier crack nucleation by creating corrosion damage sites. The extent and magnitude of the corrosion may also alter (2), (3), and (4).
6. *Other damage*: the presence of an approaching crack can alter (2) and (3) causing an acceleration in crack nucleation. Initial manufacturing flaws or material defects can also act as stress risers from which a crack may nucleate.

MSD crack nucleation is almost always influenced to some degree by fretting and corrosion or a combination of the two. Fretting usually shortens the time to initiate a crack while corrosion can both shorten the initiation time and accelerate the crack growth. Further details on corrosion in lap splices will be presented in the next chapter.

2.3.3 Crack Growth Phase

The crack growth phase generally encompasses two distinct stages of growth: from initiation to first link-up, and from first link-up to failure. The primary features of these stages can be extracted from various investigations [e.g. 8,10,23,24,26,27,29-31,33-35] and are summarized as follows:

Initiation to first link-up

- (a) This period of growth is slow and accounts for the majority of MSD initiations. It represents around 20-30% of total fatigue life, and approximately 85-95% of the entire crack growth phase.
- (b) MSD may form as a single cluster contained within one frame bay, or may appear in several frame bays, either adjacent or non-adjacent. Clusters may be limited to only a few rivet holes, or may include most of the frame bay.

- (c) Initial crack paths can be somewhat erratic in this stage due to local effects (e.g. rivet/skin interactions, stress state, out-of-plane bending, local geometry).
- (d) With continued MSD development, two cracks eventually link-up to form a small lead crack. The link-up may appear in the form of a direct joining of the crack tips, or as two cracks overlapping.

First link-up to failure

- (e) This period of growth is extremely fast and can represent as little as 2% of total fatigue life. Most tests suggest that, after first link-up, there is less than 5% of life remaining.
- (f) The small lead crack dominates all further cracking, and gradually engulfs all of the surrounding cracks. Quite often, in the early stages of lead crack growth, hidden cracks will quickly emerge to join the lead crack as it nears the rivet head.
- (g) The more MSD present within the frame bay, the faster the lead crack will grow until it spans most of the frame bay. Conversely, small MSD clusters promote slower growth.
- (h) With uniform crack lengths surrounding the lead crack, subsequent link-ups will be the most rapid. However, in-service evidence shows that MSD appears more often in varying crack lengths, and thus has a correspondingly slower link-up frequency.
- (i) As the lead crack grows in length, the influence of larger structural details, as opposed to local effects, becomes more pronounced.
- (j) Intact frames and tear straps serve to slow lead crack growth and promote fail-safe flapping failure through crack curving at the frames. Frames, however, may fail due to overstressing precipitated by surrounding MSD and resultant load redistribution.

- (k) The presence of MSD in adjacent frame bays increases the lead crack growth rate and diminishes its critical length. It might also prevent formation of a flap, leading to unstable cracking and an uncontrolled decompression. Small undetectable cracks under 0.020 inch at several holes in front of a large crack can significantly reduce the local residual strength characteristics.
- (l) If several adjacent bays have MSD (i.e. WFD) and the lead crack reaches its critical length, unstable cracking in the form of rapid link-ups of successive holes along the critical row ("unzipping" effect) may occur across the entire sheet (around 18 feet between circumferential joints) as seen in the Aloha incident. MSD essentially forces the lead crack to remain in the upper fastener row for a longer period of time. The event is accompanied by an explosive decompression of the fuselage structure.

It should be mentioned that most of the above observations regarding crack growth have come from full-scale and stiffened panel testing (curved and flat). These tests accurately reflect the load redistribution to the surrounding structure throughout the crack growth phase. This is a critical element which, in many experimental MSD tests, has not been incorporated. Consequently, the crack growth is uncharacteristic since the net section stress increases atypically as the cracks progress. Many of these tests also tend to fail before MSD has developed. Valid studies of MSD growth should therefore include these concepts in the specimen design.

2.4 Primary Concerns and Effects

Based on the characteristics of MSD, it is evident that its presence in fuselage lap splices can have serious consequences on the overall structural integrity of the fuselage structure. The threats of primary concern which certain local and widespread MSD scenarios pose are summarized as follows:

1. Subcritical cracks can link up very rapidly before detection and quickly develop into a long lead crack (typically one to two frame bays). This occurs since fatigue lifetimes of MSD cracks are shorter than that of equivalent single cracks due to crack interactions.
2. Small undetectable MSD ahead of a long lead crack (i.e. in adjacent bays) can substantially compromise the fuselage residual strength characteristics and lead to diminished critical crack lengths. The result is illustrated schematically in Fig. 2.8.
3. The presence of MSD can seriously degrade the fuselage's ability to sustain a certain degree of discrete source damage due to engine disintegration and other possible sources. The size of MSD that would reduce the residual strength capability below certification limits depends on the structural configuration. The situation is shown in Fig. 2.9.

An example of the rapid and destructive nature of MSD cracking was recently shown in a fatigue test of a full-scale fuselage panel with no adhesive bonding conducted by Foster-Miller for the FAA Aging Aircraft Program [23]. The first damage was detected at 37,000 cycles, and by 73,700 cycles, there were only two cracks more than an inch long. However, over the next 1600 cycles, another crack grew from under one inch to over 50 inches long, resulting in a complete loss of pressurization.

Swift of the FAA [36] relates a case in 1993 in Europe where an explosive decompression occurred in the fuselage of a full-scale test specimen (specific model not mentioned). Apparently, structural engineers were completely surprised, even though they were aware that MSD was present. The thinking was, and possibly still is, that small MSD can never be a problem because much larger cracks have always been tolerable in the past.

Boeing has also witnessed the effects of MSD in full-scale fatigue and residual strength tests of both narrow-body and wide-body fuselage structure [29,30,31]. These tests have revealed most importantly that:

- longitudinal fatigue cracks in wide-body fuselage tend to grow in a longitudinal direction, and **do not** result in a controlled decompression;
- similar cracks in narrow-body structures only curve to grow circumferentially in about 50% of the cases.

These results were shocking, and have served to shed some light on the seriousness of the MSD problem. These discoveries, along with other results, have prompted several industry responses.

2.5 Industry Initiatives

In an effort to deal with the primary MSD threats, an international task group was chartered in 1990 comprising manufacturers and operators. By 1991, they proposed conducting model-specific audits on aircraft that have exceeded or are approaching their original design service objectives. The audits involve the completion on the following tasks [31], many of which were originally proposed in the NTSB Aloha accident report:

- determine areas potentially susceptible to MSD/MED
- assess each suspect area's level of safety with current and augmented maintenance programs
- select areas requiring additional monitoring to establish the required level of safety
- determine additional non-specific actions to achieve the required level of safety.

In conjunction with the audits, six options listed below were identified as possible candidates singly or in combination to achieve the required level of safety. The format of prudent fleet implementation of the audit recommendations is still under review.

1. Selected limited non-destructive disassembly, inspection, and refurbishment of high-time airplanes continuing in service;
2. Continuing assessment of the fleet-demonstrated capability through diligent monitoring of service experience;
3. Fleet examination of high-time airplanes with improved state-of-the-art non-destructive inspection techniques;
4. Testing of new or used structure on a smaller scale than full component tests (i.e. subcomponent and/or panel tests);
5. Fatigue test of high-time airplane or full-scale major component followed by detailed teardown of test article; and
6. Teardown of high-time airplane.

Based on this information, guidelines to estimate the onset of widespread MSD can be more accurately defined. From there, the monitoring period and associated inspection regime can be established (see Fig. 2.3). Both Douglas [17] and Boeing [31], however, point out that, although all of the options adopted are valid to one degree or another in predicting the onset and location of MSD/MED, none of them can be considered foolproof. Ultimately, conscientious and reliable inspections and/or modifications are the key to confidence in ensuring continuing airworthiness.

3. THE AIRFRAME CORROSION PROBLEM

3.1 Introduction

In the previous chapter, it was mentioned that one of the major initiators and possible accelerators of MSD development and growth in fuselage lap splices is corrosion. In this chapter, a brief look at the more common manifestations of corrosion within lap splices is presented, although the material is directly applicable to other airframe components as well. Unless otherwise noted, the information is based largely on works by De Luccia [37], Fontana [38], Godard et al. [39], Bauccio [40], and from other specialty handbooks [41-44].

The susceptibility of high strength aluminum alloys to environmental degradation has been known for decades. Corrosion observed on aging airframes falls, according to De Luccia, into three generalized time categories reproduced in Fig. 3.1. The speed with which these corrosion failures occur is also indicated.

The most common forms of corrosion are the events labeled "time dependent", and are continually active providing the four principal elements for the corrosion process are present (see Fig. 3.2):

1. a metal that is susceptible to electrochemical corrosion (anode);
2. a dissimilar conductive material (cathode) which has less tendency to corrode;
3. a conductive liquid (electrolyte); and
4. electrical contact between the anode and the cathode (usually metal-to-metal contact, or a fastener).

Time dependent corrosion, such as pitting, exfoliation and crevice corrosion will, if not prevented or controlled, accumulate to potentially unsafe proportions and may eventually compromise airworthiness. However, these mechanisms often act as precursors to the more damaging time related event of corrosion and fatigue when enduring cyclic loads, or the "time independent" event of stress corrosion cracking when enduring static loads. The interaction of corrosion and fatigue cracking will be addressed in later chapters. The scope of this chapter will be limited to a review of the more common time dependent forms of corrosion.

3.2 Principal Causes

Before beginning an analysis of corrosion characteristics and mechanisms, it is first important to understand why the process occurs. In airframes, there are several causes which contribute, whether singly or in combination, to corrosion damage [45,46,47]:

1. *Poor design:* Lack of attention to the basic principles of corrosion-conscious design, which include:
 - (a) selection of materials resistant to corrosion – generally the higher the degree of purity of the aluminum or its alloys, the greater the corrosion resistance; cladding both sides with near pure aluminum greatly enhances corrosion resistance; 2000 series alloys provide better environmental durability than 7000 series alloys;
 - (b) protection of all faying surfaces – accomplished using proper sealing materials on all faying surfaces, and the use of corrosion-inhibiting primers;
 - (c) selection of proper finishing systems – choice of chemical treatments and paints on the basis of service requirements to minimize corrosion;

- (d) avoidance or elimination of crevices – accomplished using a smaller joint overlap, fewer rivet holes, close tolerances, higher clamping, and (b) and (c) to fill and cover existing crevices;
 - (e) avoidance of dissimilar metal contact – most airframes use similar aluminum alloys (2024-T3 for skins, 2117-T4 for rivets, and 7075-T6 for stiffeners); when unavoidable, a lay surface sealant should be applied;
 - (f) allowing for proper drainage (particularly in bilge areas); and
 - (g) avoidance of high stress areas which may compromise protection schemes and create avenues for ingress of corrosive agents – achieved in part by bonding lap joints, avoiding knife-edge countersinks, and using high bearing loads (i.e. increasing interference between rivet shank and hole after installation).
2. *Faulty manufacturing:* The assembly and finishing processes can determine whether a specific component will be subject to premature corrosion. For example, of particular importance is the proper surface pretreatment and application of protective coatings or sealants, which must offer long term durability in order to provide adequate corrosion protection. Additionally, insufficient attention to detail during component manufacture can cause many problems (e.g. drilling through previously applied protective coatings).
3. *Harsh operating conditions:* Operational environments such as marine, tropical, and industrial can accelerate corrosion markedly; this is due primarily to their abundant salt content, high temperatures and humidity, and excessive concentrations of industrial pollutants. During operation, the protective surface finish can deteriorate by chipping, scratching, breaking around fasteners, abrasions and aging. Environmental conditions inside an airplane can even be more damaging. For example, lavatory or galley spillage, condensation, oils and

hydraulic fluids, microbial growth, fire residue, and corrosive cargo such as fish, can all contribute to airframe corrosion.

4. *Poor maintenance:* As was revealed from the Aloha investigation, poor maintenance practices resulting from neglect, lack of corrosion control, and poor training can lead to the rapid development of detrimental corrosion levels. Another factor which is one of the major causes of recurring corrosion is poor repairs. Doubler patches (also called "band-aid" or "scab" patches) used in the repair of corrosion and/or fatigue damage, are well known to develop crevice corrosion. The damage progresses quickly when the patch is improperly bonded or sealed. In addition, corroded areas which have been repaired through reworking or blending out the damage have, on many occasions, exhibited subsequent signs of corrosion due to an incomplete removal of the initial corrosion.

In-service evidence, particularly for early generation aircraft, shows that airframe corrosion seems to concentrate within lap splices, both longitudinal and circumferential. Certain design features may act as catalysts to corrosion, making lap splices particularly vulnerable when compared to other structure (see Fig. 3.3):

1. *Disbonding.* Improperly bonded lap joints are particularly prone to disbonding and subsequent corrosion problems, with the cold bond procedure having the worst results. The degradation begins once the adhesive layer comes in contact with the corrosive medium. The bonding resins deteriorate and become brittle and hygroscopic (i.e. subject to moisture accumulation). The moisture becomes trapped inside the joint, and corrosion begins on the wetted surfaces. The situation is worsened with the presence of scrim cloth, which transports solution into the joint like a wick.

2. *Crevices:* A lap joint, by its very design, contains many potential crevices where moisture can condense or a corrosive solution may enter. The most common entry points are at lap edges. When the protective mechanisms in these areas fail, corrosion will likely develop on faying surfaces.
3. *Loading:* Highly localized cyclic strain around rivet holes in combination with fretting action, are very efficient in breaking down protective measures and opening up crevices. In addition, high out-of-plane bending can induce sheet separations at lap edges, the inner sheet being most susceptible. This leaves critical surfaces exposed and vulnerable to attack.

The most common forms of corrosion found in fuselage lap splices are pitting, crevice corrosion, and intergranular/exfoliation corrosion. In the following sections, these basic forms of time dependent corrosion are briefly discussed in terms of their characteristics and mechanisms.

3.3 Primary Forms of Corrosion in Fuselage Lap Splices

3.3.1 Pitting Corrosion

Pitting corrosion is defined as localized corrosive attack which leads to the formation of cavities on metal surfaces. It is quite often described as one of the most destructive and insidious forms of corrosion. Unfortunately, most common aerospace metals are susceptible to this type of attack. Age hardenable, high strength aluminum alloys with naturally forming protective oxide films, such as those found in fuselage lap splices, are particularly vulnerable to pitting corrosion.

Pitting generally occurs in two stages, initiation and growth (see Fig. 3.4):

1. *Pitting initiation*: This stage involves the period before pits have formed and may extend over periods of several months or years, even for nominally constant environmental conditions. Factors responsible for pitting initiation are:

- local breakdown in a protective film, accelerated by exposure to environments containing high concentrations of chloride ions or with high acidity; the chloride ion is the major environmental contaminant for pitting in most metals;
- the presence of localized impurities in the alloys which are either anodic or cathodic to the base metal;
- deposits of heavy metals (from water) on airframe surfaces;
- localized damage in applied protective coatings;
- the accumulation of deposits (dirt, dust, grease) on bare aluminum surfaces.

2. *Pitting growth*: Once a pit is initiated, it penetrates the metal at an ever-increasing rate due to a self-sustaining, or autocatalytic, process; that is, the corrosion processes within a pit produce conditions that are both stimulating and necessary for the continuing activity of the pit. Propagation is thought to involve the dissolution of metal and the maintenance of a high degree of acidity at the bottom of the pit by the hydrolysis of dissolved metal ions. However, if the pit cavity fills completely with insoluble aluminum oxide corrosion product, subsequent growth may be arrested.

Pits in lap splices are difficult to detect because of their small size and because they are often covered with corrosion products or coatings, or masked by structure. In addition, it is difficult to measure quantitatively and compare extent of pitting because of the varying depths and numbers of pits that may occur under identical conditions.

The electrochemical conditions at the base of a pit can be such that other forms of corrosion, such as intergranular attack will occur, leading to widespread subsurface damage. The negative geometrical aspect of pits acting as notches, can trigger more damage when stresses are superimposed on the joint.

An example of an early stage of lap joint pitting is shown in Fig. 3.5. This sample comes from an aged Boeing 727 airframe. Note that the pits tend to be hemi-spherical, with many containing other pits around their bases. There is also a tendency for pits to coalesce and form clusters. Clusters may also coalesce to form large pit colonies. The mechanisms involved in these processes are beyond the scope of this review.

3.3.2 Crevice Corrosion

Localized corrosion which occurs on faying metal surfaces on entry of corrosive media is termed *crevice corrosion*. This form of attack is usually associated with small volumes of stagnant solution which become trapped in individual parts or between different components of a structural assembly. As previously stated, in lap splices, the most susceptible crevices are between the mating sheets, and at rivet/skin interfaces (see Fig. 3.3).

Within crevices, there are differences in concentrations of dissolved salts or dissolved oxygen in the entrapped fluid. These differences produce concentration cells and promote the development of anodic sites where corrosion occurs. On lap faying surfaces, the damage generally takes the form of pitting. The corrosive action at the base of the pit accentuates the concentration differences, and corrosive attack progresses more rapidly as the depth of the pit increases. The stagnant solution also becomes more acidic with time; fluid within crevices exposed to neutral dilute sodium chloride solutions has been

observed to contain 3 to 10 times as much chloride as the bulk solution and to have a pH of 2 to 3.

It is apparent that crevice corrosion and pitting corrosion share many of the same characteristics. However, pitting deserves special consideration since it is a self-initiation form of crevice corrosion. Simply, it does not require a crevice - it creates its own.

An example of crevice corrosion on an actual aircraft fuselage lap joint is shown in Fig. 3.6. The sample comes from a Boeing 727-200, line number 590, which had accumulated 48674 cycles and 56259 hours. The specimen makes up part of an extensive library of corroded lap joints compiled by the Institute for Aerospace Research at the National Research Council of Canada. In the full-joint view, the upper half is the outer skin, and the lower half is the inner skin. For the most part, the joint is very badly damaged, although the area between rivets 11 and 16 on both sheets reveals almost no evidence of corrosive attack. It appears that in this case, the joint had become only partially disbanded.

Another interesting feature of the 727 sample is that the surfaces immediately outside the joint are relatively undamaged. The reason is that, with crevice corrosion, the surrounding external surfaces are cathodically protected, and as such, suffer little or no damage. This leaves no significant evidence outside lap edges to indicate the presence of internal damage. However, a phenomenon known as *corrosion pillowing* occurs, where increased internal pressure, due to the corrosion products occupying more volume than the consumed metal, causes the skin to bulge around the rivets in the shape of a pillow. The effect is visible with proper illumination, even for small amounts of corrosion. Fig. 3.7 shows the result for the 727 sample before it was disassembled. Note the pillowed areas indicative of corrosion, as well as the flat, undamaged areas.

Another visible effect, although far less common, comes from a special form of crevice corrosion known as *filiform corrosion*. It often starts from a corrosion pit at a rivet hole underneath the paint layer, but instead of penetrating into the metal, it spreads out sideways to form worm-like traces of corrosion on the surface. The initiating cause has been suggested to be the permeation of moisture through the painted surfaces under conditions of high humidity and high ambient temperatures. This form of corrosion, therefore, is more prevalent on aircraft operated in the tropics where hot and wet conditions exist. Fig. 3.8 shows an example of filiform corrosion before and after paint removal.

3.3.3 Intergranular/Exfoliation Corrosion

Localized subsurface attack in which a narrow path is corroded preferentially along the grain boundaries of a metal is known as *intergranular corrosion*. The corrosive attack can produce a network of surface corrosion, occasionally dislodging whole grains, or it may penetrate deeply leaving behind little evidence of the damage.

In airframe skins, where the grains are flattened and elongated in the direction of working, the presence of intergranular corrosion can lead to flaking and leaflike bulging, producing a delamination effect with surface grains being pushed out by the underlying corrosion products. The phenomenon is known as *exfoliation corrosion*, and is essentially a severe form of intergranular corrosion occurring in the direction of grain flow.

Intergranular and exfoliation corrosion at a rivet hole are illustrated in Fig. 3.9. Rivet holes have a high vulnerability to exfoliation corrosion because they provide both a pathway for corrosive electrolytes to enter, and a region with exposed grain boundaries. An example of intergranular corrosion in the B-727 sample is shown in Fig. 3.10. In this

case, the intergranular attack started from a pit on the countersink. Note the dark corrosion products in the intergranular fissures.

With intergranular corrosion, the material in the grain boundary areas behaves anodically with respect to the grain interiors. In corrosive environments, dissolution of the anodic grain boundaries occurs at a very rapid rate while the bulk of the alloy is little affected. The small area of the anode with respect to the cathode is an important factor influencing the corrosion rate. The anodic nature of the grain boundary may be due to the presence of local impurities, or either the enrichment or depletion of one of the alloying elements. These effects may be associated with the precipitation of grain boundary phases, which may themselves behave anodically with respect to the adjacent alloy.

The degree of susceptibility of an alloy to intergranular attack can thus vary appreciably depending on its microstructure, which is the result of its metallurgical history and thermal treatment. Heat treatments that cause precipitation throughout the grain tend to diminish the tendency to intergranular attack. However, it has been shown for the 2000 series alloys (e.g. 2024-T3) that, if the subsequent quenching following heat treatment is not rapid enough, the metal may be sensitive to intergranular corrosion.

Another factor known to have marked effects on intergranular corrosion is the degree of aging after quenching, which is affected by the aging temperature and the aging time. Overaging treatments such as T8 for 2000 series alloys, or T76 or T73 for 7000 series alloys (e.g. 7075) generally produce increased resistance to intergranular corrosion.

3.4 Prevention and Control Measures

Corrosion prevention and control in airframe structures is of the utmost concern because of its potential impact on human safety and on expensive aircraft. With widebody aircraft

reaching \$100 million per unit and their full-scale inspections exceeding \$2.5 million, the possibility of an in-flight mishap or excessive down time for structural repairs necessitates an active Corrosion Prevention and Control Program (CPCP).

Since corrosion is inevitable in the long term, some allowance should be made for its effects in the initial design. The elements of the so-called "corrosion-conscious" design were briefly described at the beginning of this chapter. Careful attention to these factors will yield significant benefits over the life of the aircraft.

Once an aircraft is delivered, however, corrosion prevention and control becomes the responsibility of the operator. To aid operators in defining an individual program, manufacturers generally provide them with extensive training programs, corrosion control handbooks and customer service documents. The type and aggressiveness of the program depend on the operational environment of the aircraft. For example, aircraft exposed to the harsh operating conditions mentioned earlier will require a more stringent CPCP than an aircraft that is operated in a dry environment.

To prevent corrosion, a constant cycle of cleaning, inspection, and operational preservation must be followed. Prompt detection and removal of corrosion will limit the extent of airframe damage. The basic philosophy of a CPCP should consist of the following:

1. Adequately trained personnel in the recognition of corrosion including conditions, detection and identification, cleaning, treating, and preservation;
2. Inspection for corrosion on a scheduled basis;
3. Prompt corrosion treatment after detection;
4. Airframe washing at regularly scheduled intervals;

5. Routine cleaning or wipe down of all exposed unpainted surfaces:
6. Inspection, removal, and re-application of preservation compounds on a scheduled basis;
7. Early detection and repair of damaged protective coatings:
8. Accurate record keeping and reporting of material or design deficiencies; and
9. Use of appropriate materials, equipment, and technical publications.

In the late 1970s, a basic assumption was that existing approved maintenance programs were controlling corrosion below levels that could affect airworthiness. Since then, it has become apparent that without effective corrosion control programs, the frequency and severity of corrosion can increase dramatically. However, a fleet survey conducted by Boeing from 1986 to 1992 revealed that many operators were using unproven or ineffective prevention measures, and in some cases, delaying repair actions upon the discovery of damage [31,47]. Evidently, some airplanes were continually operating with significant structural corrosion which, in turn, could substantially influence fatigue cracking and the damage tolerance capability of the airframe.

To combat the airframe corrosion problem, the AAWG has established some mandatory standards in the form of baseline CPCPs. The programs define an airplane age at which the CPCP should begin and an interval at which the CPCP task should be repeated. This departs from the usual basis of flight hours or flight cycles since it was recognized that the initiation and progression of corrosion is basically a function of calendar time. Consequentially, the newly developed CPCPs have had major influences on many operator's maintenance programs, some involving costly re-structuring. The CPCPs were implemented by operators from 1991 to 1993 and, judging from a 1993 Boeing report, have not yet been proven effective [47]. Their evaluation is on-going.

Development of the CPCPs has required a major effort by the aircraft industry, and is a positive step to ensuring that corrosion is maintained at acceptable levels. However, there remains today much confusion over exactly what these "acceptable levels" are or should be, especially with regard to hidden corrosion, such as that found in lap splices.

3.5 Characterization of Damage

Unfortunately, few guidelines exist to aid in assessing corrosion severity of airframe structure. Lincoln of the USAF recently stated that, "the standards for corrosion damage are, at best, poorly defined such that characterization of damage found is not well understood" [18]. Understanding how to properly characterize and quantify corrosion damage is the first step to determining what levels of corrosion are acceptable within a particular structure.

In industry, it appears to be common practice to quote lap joint corrosion damage in terms of percentage thickness loss or percentage corrosion. The reference is to the amount of material loss at the faying surfaces of the mating sheets, where the majority of damage occurs. The concept is illustrated in Fig. 3.11. A threshold value which appears often is 10% thickness loss, a value which can be reliably detected by non-destructive inspection. In general, if either sheet contains an area of at least 10% thickness loss, repairs are considered to be necessary [48,49]. Remarkably, this criterion gives no consideration to the distribution of corrosion throughout the joint. For example, a joint with 80% of its area above 10% thickness loss is obviously worse than a joint with only 1% of its area above 10% thickness loss. Additionally, a joint with 10% thickness loss in a non-critical fatigue area is not as serious as if the same damage was located at the critical rivet row.

Whichever case is examined, a more precise characterization of corrosion severity seems to be warranted. Involved in this characterization would inevitably be a series of tests to identify corrosion configurations critical to fatigue, allowing for a more informed response to the problem. It may indeed turn out that some repairs are being conducted prematurely. That is, certain damage scenarios above 10% thickness loss might not pose an immediate threat and could be repaired during a later inspection. Conversely, the tests may reveal that non-critical concentration of corrosion damage around key rivets is in fact, a critical situation which requires immediate attention. Guidelines are needed to ensure that actions upon corrosion detection are based on assessments which take into consideration the potential for combined forms of damage, such as corrosion and fatigue.

4. INTERACTION OF CORROSION AND FATIGUE

4.1 Introduction

With the MSD and corrosion problems separately discussed and placed in their proper perspective, some consideration will now be given to the potential effects of their interaction. Although both fatigue cracking and corrosion have been recognized as significant individual detriments to lap splices, their presence in combination is a far worse situation. Corrosion can have a more insidious effect by interacting with and promoting fatigue crack nucleation and propagation. Indeed the evidence suggests that these mechanisms were partly the cause of the Aloha in-flight decompression discussed in Ch. 2.

In general, corrosion damage increases with time, and when added to the regular accumulation of fatigue damage, the life of the aircraft can be markedly reduced. The concept is shown schematically in Fig. 4.1 for both short-life and long-life aircraft, by superimposing the corrosion and fatigue damage. As shown for short-life aircraft, corrosion damage may only represent a small part of the total damage. However, for long-life aircraft, the cumulative effects of corrosion damage may eventually be responsible for a major reduction in life. The effect on both short and long-life aircraft is worsened if corrosion begins earlier than anticipated. As stated in the figure, one cause of this is lap joint disbonding. Other causes were discussed in the previous chapter.

The concept that corrosion reduces fatigue life is not a new revelation, nor is it surprising. It is, however, not well understood and in many cases, not even considered. As mentioned at the 1995 USAF Aging Aircraft Conference, "the reduction in life [due to the existence of corrosion] cannot be accurately determined at this time due to a shortage

of analysis tools and data" [50]. There is a recognized need in this area for some fundamental understanding, in what some might call a "return to basics".

In order to build this understanding, and define the framework for research into corrosion and fatigue interaction within lap splices, two key items need to be considered. They are:

1. *Developing a corrosion and fatigue simulation model*

It is of critical importance to first understand how corrosion and fatigue might occur on actual airframes under various circumstances, then decide on how best to simulate their interactions. The development involves the following steps:

- establishing a set of desired criteria by which a model may be assessed
- identification of primary factors contributing to corrosion and fatigue
- understanding the influences these factors have on corrosion and fatigue
- identification of potential threat scenarios
- proposition of simulation models
- assessment of models based on the criteria set

2. *Review of relevant research*

A review of research incorporating the approaches identified in Pt. 1 will potentially bring some further understanding with regard to environmental fatigue effects, and aid in further evaluating the models. The review will summarize some key results and conclusions which have contributed to the overall understanding of this problem.

The first step is necessary since research into corrosion and fatigue interaction in lap splices is still in its infancy, and there are few proven models on which to base an approach. The second step is therefore primarily limited to a review of corrosion and

fatigue investigations which employ specimens made of similar materials to a fuselage lap splice, but are structurally simpler.

The remaining sections of this chapter are devoted to the two tasks listed above. Based on this knowledge, the various research avenues will be identified, from which a thesis project will be defined.

4.2 Corrosion and Fatigue Simulation Methodologies for Airframes

The development of a model which successfully integrates corrosion concepts with fatigue concepts in an MSD context is a non-trivial exercise because of the many factors involved.

The objective is a method which satisfies the following basic criteria:

1. models as closely as possible the major aspects of the airframe service conditions and life while remaining on the conservative side;
2. can be simply and swiftly applied in a laboratory setting; maximizes use of existing technologies/knowledge and available resources; and
3. is relatively inexpensive to setup and perform (a trade-off between the previous two criteria).

The principal considerations in developing a simulation model capable of meeting the above criteria are shown in Fig. 4.2. The illustration depicts the driving chemical and mechanical processes of corrosion and fatigue, as well as the major manifestations of these elemental damage processes. Also shown are the major causes and promoters of corrosion and fatigue, the airframe service conditions.

Swift of the FAA believes that "great care should be exercised in making sure that all realistic effects experienced by the actual aircraft are properly simulated" [51]. It is therefore of fundamental importance to establish the airframe service conditions, and the degree to which each variable affects the airframe damage mechanisms. With this understanding, various simulation models can be postulated and evaluated on their ability to meet the prescribed criteria.

4.2.1 Airframe Service Conditions

In assessing the influences on airframe damage, the conditions in which the aircraft operates must be established. To accomplish this, the atmospheric variables are identified and characterized throughout the various stages of a typical flight profile.

The flight profile and schedule considered for the purposes of this analysis will be a worst case scenario involving many short pressurized flights per day in a severe corrosive environment. The Aloha operation of their earlier B-737s seems an appropriate example. Highlights of the operation are reproduced below. Aloha's current practices are unknown to the author.

Aloha Boeing 737 Operational Data [1]

Flight Type	Short-haul (island hopper)
Avg. Flight Time	24 min.
Avg. Flights per Day	13 (most pressurized to 8000 ft.)
Max. Cruising Altitude	35,000 ft.
<hr/>	
Hours per Day in Flight	5.2 (21.7%)
Hours per Day on Ground	18.8 (78.3%)

4.2.1.1 Atmospheric Conditions

The atmosphere considered in this analysis is that of a marine environment, although many of the conclusions are applicable to other environments as well. Geographically, this encompasses all open sea areas as well as coastal locations, where the corrosion potential is a maximum. The content described herein is largely derived from literature in the disciplines of atmospheric science, climatology and meteorology, namely [52 - 59].

The composition of the marine environment to which an aircraft is exposed varies most dramatically throughout the ascent and descent stages of the flight cycle. The atmospheric components of interest, which also have a direct effect on airframe damage, are:

- temperature
- humidity
- airborne corrodents

The variation of temperature with altitude is shown in Fig. 4.3 for different seasons and geographical locations. As shown, temperature drops rapidly with increasing height until it reaches a point where it stops decreasing, commonly referred to as the tropopause. Note that the tropopause altitude is much higher in the tropics where the temperature reaches much lower values (-81°C at 53,500 ft.). The actual temperature distribution in the airframe skins is likely to be close to the ambient temperature. In lap splices, there are small thermal gradients set up by the outer skin exposure to ambient conditions and the inner skin exposed to a warmer cabin interior. Quantitative information on these temperature distributions is not available in the literature. However, for thin skins, the gradient is likely negligible.

Humidity is the term used to describe the water vapour content of the atmosphere, also known as the moisture content. The amount of moisture in the air, like temperature, also changes rapidly with altitude as shown in Fig. 4.4. The humidity parameter depicted is the mass mixing ratio (symbol ' r '), defined as the mass of water vapour per unit mass of dry air. The curves and figures are those for middle latitudes and must be taken as illustrative only since the atmosphere is very dynamic. In the tropics, the magnitude of the mass mixing ratio at sea-level can be up to seven times greater than that of the middle latitudes. With increasing altitude, however, the capacity for the air to hold moisture decreases with falling temperature, even continuing to drop past the tropopause (due to the inflow of dry cold air from the tropical tropopause). It is estimated that about 90% of all atmospheric moisture resides below 20,000 feet, and probably less than 1% above the tropopause. The atmosphere thus exhibits a rapid transition from a moist air environment at low altitudes into one that is extremely dry.

In addition to water vapour, the level of corrosivity of the atmosphere is also related to the presence and concentrations of airborne corrodents. In marine environments, this is primarily due to the presence of sea-salt aerosol, formed when air bubbles in breaking waves burst at the ocean surface. The most damaging feature of the aerosol is an abundant presence of chlorides, primarily sodium chloride derived from sea-salt particles. Although no direct information was found on the levels of sodium chloride at various altitudes, a study by the U.S. National Center for Atmospheric Research [58] does provide some data. In the study, particles were collected at various altitudes and locations over the continents and oceans, then x-rayed to determine their elemental composition (i.e. nature of the compounds was unknown). Over the ocean, the concentrations of chlorine (predominantly the chloride ion) and sodium are reproduced in Fig. 4.5. The concentrations are normalized to the maximum concentration of chlorine at sea-level. As shown, from sea-level to about 3000 feet, the concentration of chlorine drops by some 84%. Sodium concentrations, not large to begin with, fall more gradually with altitude. Oore & Russell [60] have also determined that the salty atmosphere above

sea-level is predominant within 3000 feet of altitude. It follows that the damaging effects of the sea salt aerosol on the airframe will be greatest only during periods of down-time. As listed earlier for Aloha, the average amount of down-time accounted for almost 80% of its daily scheduling. Therefore the aircraft spent most of their time in a very corrosive environment.

The most severe environment from an airframe damage context is one that combines high temperatures with high humidity levels and high concentrations of airborne corrosives. This deleterious mix is found mostly in tropical coastland locations, such as those of the Hawaiian islands. Furthermore, in many parts of the tropics, the relative humidity can rise nightly to almost 100% (highest during rainy season). This finding is significant because sodium chloride particles are hygroscopic and form droplets with water when the relative humidity of the air is about 78%. The moisture condenses on suitable surfaces such as airframe skins and provides the electrolyte required for the corrosion process. Corrosion severity is worsened by the presence of prevailing on-shore winds, which transport the highly chlorinated sea air inland (e.g. sea breeze). An example of levels of marine atmosphere corrosivity is shown in Fig. 4.6 which depicts the island of Barbados in the West Indies.

To summarize, the environmental conditions to which an aircraft is exposed during a typical flight cycle are as follows:

- *Temperature:* while relatively hot temperatures exist near the ground, the ambient temperature drops quickly as the aircraft gains altitude to magnitudes substantially below freezing.
- *Humidity:* high levels of water vapour can be found in ground-level air, but drop to extremely low levels as altitude is gained. Therefore during flight, aircraft generally fly in dry conditions.

- *Airborne Corrodents:* in marine atmospheres, there are high concentrations of chlorides at sea-level but in just 3000 feet of altitude, the levels drop by over 80%. Therefore the aircraft will see the maximum effect of this damaging influence throughout all periods of time at or near ground-level.

As an illustrative summary, the worst case scenario, as described above, is shown in Fig. 4.7. It depicts the tropical marine environmental data as a function of altitude along with the flight ascent for a typical island-hopping aircraft. The pressurization loading scale is also shown and represents the actual loading of the Aloha B-737s. This loading will be discussed in sections to follow. From the figure, it is evident that the tropical coastland location provides all the necessary ingredients for a very concentrated and destructive atmosphere. However, these conditions change rapidly with altitude as the aircraft moves from hot to cold, from moist to dry, and from salty to virtually salt-free conditions.

4.2.1.2 Environmental Influences on Airframe Damage

Now that the principal environmental factors have been established, a consideration of how each one contributes to damage within the airframe lap splices is in order. The available literature provides some answers but is considerably lacking in many of these areas. Influences on corrosion will be considered first, followed by fatigue. Wherever possible, examples using the typical airframe aluminum alloy 2024-T3 will be used, although most of the trends observed hold for other alloys in other tempers as well.

The influence of temperature on corrosion rates in aluminum has been investigated by Mears [61], and Mears and Brown [62], although not for the full aircraft temperature spectrum. Mears conducted a study on the effect of temperature on the corrosion rate of an Al-Mg alloy in 10% hydrochloric acid. The study shows that from 30°C down to 0°C

the corrosion rate drops almost 100% from 0.041 inch/day to around 0.0003 inch/day. This result is significant since in Ch. 3, it was shown that hydrochloric acid tends to form at the base of pits as a by-product of corrosion. Mears and Brown found a similar result in their study on the influence of temperature on pitting corrosion of aluminum in chloride solutions. Their results are reproduced in Fig. 4.8 and clearly show that as the temperature falls (i.e. as the aircraft climbs), both the density and the probability of pitting decrease substantially. From 27°C down to 4°C the drop in the pit density is almost 40%. From these results, it is clear that the probability of any further corrosion development is severely curbed soon after the aircraft takes off and its skins cool down. Conversely, the longer the aircraft remains in hotter regions, the higher the corrosion potential and subsequent growth will be.

The effect of humidity and moisture content of the air on corrosion is directly related to the corrosion process itself. As described in Ch. 3, for corrosion to occur by an electrochemical process, an electrolyte must be present for current to flow. Since water that contains ions provides a very good electrolyte, the amount and availability of moisture is therefore an important factor in atmospheric corrosion. Indeed, without the presence of water, there is no corrosion of aluminum. Furthermore, when the relative humidity rises above a certain critical level (~78% for aluminum), there will be an acceleration in the rate of corrosion [59]. It follows that the majority, if not all, of the corrosion which develops inside a lap joint, does so while the aircraft is at ground-level and primarily overnight when the relative humidity is highest and moisture freely condenses on metallic surfaces. The corrosion products, being hygroscopic, absorb and retain much of this moisture. As the atmosphere becomes very dry and cold during the ascent stage of the flight cycle, any moisture or moisture-laden particles trapped within the joint become temporarily frozen, or rather, freeze-dried. The corrosion process thus stops as the aircraft climbs to its cruising altitude, and re-starts during descent.

Another important factor which affects corrosion rates is the contaminant content of the air. For marine environments, there is a direct relationship between atmospheric salt content (i.e. salinity) and measured corrosion rates. Griffin [59] details a study along the coast of Nigeria which observed that a hundredfold increase in the salinity resulted in a hundredfold increase in the corrosion rate. The maximum corrosion rates were found to be related to the maximum chloride concentrations in the atmosphere. As this has been determined in the previous section to be within roughly the first 3000 feet above sea-level, it follows that the airframe is at greatest risk within that region, accentuated whilst closest to the sea. With altitude, chloride levels quickly diminish and thus provide a negligible contribution to any further corrosion development.

Changing the focus to fatigue damage, the influence of temperature has not been considered by many researchers. This is quite surprising since, as Fig. 4.7 shows, the majority of the airframe loading due to pressurization occurs at temperatures below freezing (shown by the scale of differential pressure). However, there was a study conducted by Broek [63] which examined the fatigue crack growth and residual strength characteristics of 2024-T3 and 7075-T6 aluminum alloys at room temperature and 0, -25, -50, and -75°C. The results for the fatigue crack growth in 2024-T3 are reproduced in Fig. 4.9. Broek concluded that:

- (a) Temperatures in the high layers of the troposphere and in the stratosphere have a beneficial effect on fatigue crack growth. Crack propagation rates are reduced by a factor between 1 and 10 (depending upon crack length) in 2024-T3 sheet and by a factor between 1 and 4 in 7075-T6 sheet. In general, temperatures of -25°C and below will lower the crack growth rates as compared to room temperature; and
- (b) The reduction in the fatigue crack propagation rates at low temperatures is a result of the virtual absence of water vapour in cold air.

Fig. 4.9 suggests that the temperature effect is restricted to lower growth rates. This implies that at higher growth rates, the effect of temperature, and thus moisture, in promoting crack extension is negligible. The overall results of this study suggest that it may be sufficient to model the effect of temperature simply by maintaining low humidity throughout a test.

The sensitivity of fatigue damage in aircraft aluminum alloys to variations in humidity has been investigated in great detail by Hartman [64-66]. Hartman monitored the growth of a centre-cracked specimen under various conditions of water vapour content. The results for 2024-T3 alclad sheet are shown in Fig. 4.10 in terms of the number of cycles necessary to grow a crack from 0.080 inch to 1 inch, termed by Hartman as the "endurance". From that result, he concluded that in air, the endurance increased continuously with a decrease of the water vapour content of the air. For dry air and in the early stages of growth, the crack rate was an order of magnitude smaller than in wet saturated air. Finally, Hartman noted that near the end of the fatigue test, the crack rate was independent of the humidity of the air. Indeed this is the same trend which Brock obtained in his study of temperature effects. It appears that humidity effects are most significant in the earlier stages of crack growth.

The specific influence of airborne corrodents in varying concentrations on the development of fatigue damage is not reviewed. As previously stated and shown in Fig. 4.7, the primary fatigue loading on the airframe occurs only at altitudes where the atmospheric salinity is extremely low. Therefore, information in this area was not required.

One final consideration of the environment's influence on airframe damage concerns bonded joints, specifically the integrity of the bond itself. A crucial part in the overall damage development cycle occurring within certain earlier-designed joints centers around the quality of this bond. As was mentioned in Ch. 2, there have been many reported cases

of lap joint disbonding, and although primarily attributable to initial manufacturing defects and cyclic mechanical stressing, the environmental conditions claim some responsibility. A severe corrosive atmosphere can be detrimental to an adhesive layer, causing a gradual deterioration of the bond quality. In addition, thermal fatigue through successive ascents/descents may also contribute to disbonding. Ultimately, it is a combination of all these forces acting over time which gradually render the bond ineffective, and allow for ingress of the environment.

4.2.2 Airframe Service History

The airframe service history is closely linked to how and where the aircraft is used throughout its life. This is based on a few key factors: the amount of usage in terms of the frequency of duty cycles (one duty cycle = takeoff, climb, cruise, descent, and landing), the types of these flight profiles, and the varying conditions to which the airframe is exposed.

Considering the worst case scenario and the Aloha B-737 profile described earlier, it is very likely that aircraft operating under these conditions will develop lap joint damage starting at an early stage of life. Unfortunately, no information was found in the open literature concerning damage development profiles in lap splices incorporating both corrosion and fatigue. The only data of this nature from actual aircraft seems to come only after accidents have occurred, and provides only a small glimpse of life. It is extremely difficult, perhaps impossible, to retrace the steps in a splice's damage history from observations conducted after the part has failed.

This apparent lack of information regarding airframe damage profiles is not surprising. Documenting the damage development within a lap splice (or any other part) over the course of one lifetime, would be a time-consuming and expensive venture. Traditionally,

full-scale testing of airplane structure has not incorporated the effects of corrosion [67], and aircraft operators are generally not interested in compiling research data in conjunction with their maintenance and inspection schedules.

For the purposes of modelling, available data and observations will be used to provide a more complete picture of a possible airframe service history.

4.2.3 Observations Based on Airframe Service Conditions and History

With regard to the onset and development of corrosion damage within a fuselage lap splice, the following observations are noted:

1. As the ground-level atmosphere has been shown to be the most corrosive and aggressive, the time periods where the corrosion process is most active is thus throughout all periods of down-time. The data also suggests that the majority of moisture is generated overnight when the air is closest to saturation.
2. The majority of the corrosion damage likely occurs once the joint sealant has been compromised on one or both of the faying surfaces. A disbond near either faying edge opens up a pathway to allow the ingress of moisture. In marine atmospheres, the airborne moisture is laden with sodium chloride and provides the corrosive medium which then attacks the metallic surfaces.
3. The corrosion process is likely stopped shortly after take-off due to the rapid emergence of cold and dry conditions which exist for the majority of the flight. Trapped moisture in the form of droplets or absorbed into corrosion products becomes temporarily frozen or is removed through aerodynamic suction at certain external areas (e.g. cracked paint around a faying edge, through opened cracks, or around loose-fitting rivets).

4. The worst corrosion conditions are those combining high temperatures, high humidities, and high salinity. This situation exists at most tropical coastland locations, and is aggravated by the presence of on-shore winds. The worst airframe corrosion damage scenario is one in which the aircraft spends most of its time in this harsh environment.

In relation to the development of fatigue damage in fuselage lap splices, the following observations are noted:

5. Crack propagation rates in 2024-T3 are reduced by low temperature and humidity levels, but only during stages of early crack growth. Tests conducted in dry air at room temperature should adequately model these aspects.
6. Other environmental conditions which may directly influence fatigue crack initiation and growth are ignored since their severity is sharply reduced shortly after take-off.

Finally, when considering the potential interaction of corrosion and fatigue, the following observations are noted:

7. There is considerable evidence that the processes of corrosion damage and fatigue damage in fuselage lap splices **DO NOT OCCUR CONCURRENTLY**, but rather **SEQUENTIALLY**. That is, at the point where the airframe is pressurized and any existing cracks have opened, the environments both externally and within the joint are inconducive to any further corrosion development.
8. Although ingress of the environment to the crack tip may become possible due to incomplete crack closure, this likely occurs only for larger cracks. At this point, growth is dominated by fatigue, and therefore crack propagation through environmentally-assisted means is negligible.

4.2.4 Simulation Scenarios

Open literature does not describe the sequence of events in the development of lap splice damage. In this context, several important questions are left unanswered:

1. *Initiation of disbond:* Should the joint be initially disbanded? Should it fail early on in life, later in life, or at some intermediate point?
2. *Type and duration of disbond:* Is it a rapid occurrence throughout the joint, or does the disbond grow with the progression of either corrosion or crack growth?
3. *Crack-initiating damage:* Do cracks typically initiate from fatigue or corrosion damage?
4. *Growth-dominated periods:* What are the periods where the damage is either dominated by corrosion or fatigue crack growth?

Certain simulation approaches will be more representative of certain scenarios. The lack of information makes it difficult to know which case would be the most appropriate to model with some assurance that the results will remain conservative.

For the purposes of this thesis, a typical sequence of events representing a worst case scenario will be considered: that of an initially disbanded joint. There is some validity in choosing this scenario based on a Boeing report which states that, for the cold bond procedure, there were reports of complete joint delamination on airplanes with as low as 3000 flight hours [15]. Using this scenario permits both corrosion and fatigue to act sequentially on the joint from the beginning. The sequence of events is then postulated as follows:

- because of immediate disbond, there is an initial period of considerable time over which the overall damage is dominated by the development of corrosion; the rate of development is relatively slow;
- with time, the rate of corrosion gradually increases creating large areas of internal attack; fatigue cycling continues until the first fatigue cracks become visible;
- more cracks at other sites emerge and MSD develops; crack growth is initially slow; the rate of corrosion is high yet contributes less to the overall damage development since it has now less time in which to act (i.e. further damage begins to be fatigue-dominated);
- MSD cracks grow at faster rates until eventually, two cracks link-up; corrosion growth is minimal during this period;
- crack growth accelerates with successive link-ups until the lead crack causes failure; period is dominated by fatigue (i.e. no further significant corrosion development).

4.2.5 Simulation Approaches

Three experimental testing approaches will be discussed in this section. Each method will be critically analyzed with respect to its ability to model the case of an initially disbanded joint, while adhering to the prescribed criteria. A brief description of each is given below with some implementation details and obvious experimental pitfalls.

1. Constant Immersion: Specimens are fatigue tested while in continuous contact with the corrosive medium. This is normally accomplished using an insulated perspex box enclosure around the critical area with either a continuous presence of saline solution or alternatively a salt spray using a pump and fluid storage assembly. For this test, all exposed surfaces and the crack tip region, are continually under corrosive attack. Therefore, the test is highly sensitive to test frequency. Generally, the more time

available for reaction with newly exposed metal, the faster the crack nucleation and growth will be.

2. Alternate Fatigue/Immersion: Specimens undergo a repetitive cycle of fatigue testing in air followed by exposure to the corrosive medium (or vice-versa). Fatigue cycling and corrosion development are done in pre-determined blocks of cycles and time, respectively. The number, lengths, and organization of the blocks can be optimized in attempting to model a specific situation. The test is normally conducted by using a similar rigging to the constant immersion test but with the incorporation of a purge and drying cycle. Alternatively, the specimen can be removed from the rig and placed in a separate corrosion chamber, then re-mounted after exposure. For this test, evidence suggests that improper purging and drying will result in dramatic and unrealistic reductions in fatigue life [68].
3. Pre-Immersion: This is essentially the simplest form of the alternate fatigue/immersion test which uses one large block of exposure to the corrosive medium followed by one large block of fatigue cycles. This is normally accomplished by exposing the specimen inside a corrosion chamber until a desired amount of corrosive attack has developed. The specimen is then fully dried and fatigued in air to failure. The same caution concerning drying as with the alternate approach applies for this method as well.

Since it already has been determined that corrosion damage and fatigue damage in lap joints occur sequentially, the use of the constant immersion test can be eliminated. Although conservative results are almost guaranteed, they are likely to be unduly so and thus, far from reality. This leaves the alternate fatigue/immersion and pre-immersion formats.

The operational scenario is probably best matched using the alternate fatigue/immersion test. The problem lies in determining the number of blocks, the duration of each block, and the sequencing of blocks. Logically, this approach would start with larger blocks to represent the period up to crack initiation, then smaller blocks throughout the MSD development. Finally, one fatigue block to failure would be used once the cracking begun to accelerate (e.g. after first link-up).

Unfortunately, the alternate fatigue/immersion approach may not meet all of the established criteria. Although it is adaptable to model various real situations, it is difficult to assess its conservatism, which would change under different arrangements. Furthermore, the format might be too complex and time-consuming to perform.

There may, however, be a few simple combinations of alternate fatigue/immersion which overcome the potential pitfalls. The simplest format is a three-block combination: (1) one large fatigue block to simulate a number of years in service, away from the point where fatigue alone is known to begin cracking; (2) one large block of corrosion to a desired severity; (3) one final fatigue block in which all cracking occurs. The three-block format is simpler because it only involves determining the length of two blocks. It models reality a little closer than pre-immersion yet is also likely to give conservative results. The three-block alternate fatigue/immersion test is therefore a possible solution. An extension beyond three blocks, say to five or seven blocks, could be considered, but at added risk since it involves more than just an incremental jump in complexity.

The simplest approach is the pre-immersion test. The only decision lies in determining the required amount of exposure to develop a desired amount of corrosion. Unfortunately, this approach lacks in modelling the real situation. A two-block cycle removes any significant sequential interactions which may exist at various stages in the overall damage development within a splice. The results, however, will tend to be conservative seeing that the onset of fatigue cycling only starts once the joint has been

corroded. This is comparable in reality to a new aircraft entering service with already a significant degree of lap splice corrosion.

The most significant benefits offered by pre-immersion over other methods are that the tests do not require any special rigging onto the specimen or load frame, nor do they require any other apparatus except for a corrosion chamber. With only one corrosion and one fatigue block, the tests can be performed in the comparatively shortest time frame with a minimum requirement for setup. It then follows that this type of test is the least expensive of all considered. The pre-immersion test thus provides another possible solution.

Note that with all of these approaches, it is assumed that the corrosion development and damage are similar to that occurring within actual airframes. Without accurately modelling the damage occurring naturally in aging lap splices, it does not matter how well a test simulates the true corrosion/fatigue interactions. This is a key simulation parameter which will be discussed in more detail in later sections.

4.2.6 Summary

This section attempted to provide a few avenues of experimental research into the simulation of corrosion and fatigue interactions in fuselage lap splices. To this end, the airframe service conditions and history for a worst case scenario were established.

The evaluation revealed most importantly that when considering the atmospheric profile against typical commercial aircraft missions, the airframe damage processes of corrosion and fatigue occur sequentially, and not concurrently. Therefore crack propagation through environmentally-assisted means does not occur under these circumstances.

The potential simulation models which emerged from the analysis were a three-block alternate fatigue/immersion test format, and the pre-immersion test format. Both are very similar and the choice of one over the other simply involves a trade-off between either achieving more realism or opting for more simplicity.

The following sections will review the status of research into simulating corrosion and fatigue using the proposed simulation models.

4.3 Review of Research Efforts

4.3.1 Overview

It was mentioned earlier that research into corrosion and fatigue interactions in lap splices is still in the preliminary stages. Lack of published data in the available literature confirms that this is true. The majority of investigations which are reported apply the processes of corrosion and fatigue concurrently, an approach shown to be inapplicable to airframe structure for the scenario under consideration. Therefore no review of these investigations is presented in this section.

However, there has been some progress in the research of pre-corrosion fatigue and alternate corrosion/fatigue, albeit with simpler specimens than a lap splice. Key results and conclusions from a few significant investigations in these areas are reviewed.

4.3.2 Corrosion and Fatigue Investigations

By far, the largest scale investigations into the environmental effects on fatigue life were the *Corrosion Fatigue Cooperative Testing Programme* (CFTCP) [68] from 1977 to 1981

and the *Fatigue in Aircraft Corrosion Testing Programme* (FACT) [69] from 1981 to 1985. Both programs were organized by the Advisory Group for Aerospace Research and Development (AGARD), a NATO organization, and involved the participation of ten laboratories (six completed both programs).

Details for the CFTCP core program are shown in Fig. 4.11 (a). As illustrated, the specimen used in all tests was a 1½ dogbone made of 1/8 inch thick 7075-T76 incorporating two countersunk holes (100°, no knife edge), intended to be representative of a structural joint. The compiled results from all ten participants are shown in Fig. 4.11 (b) for the four test cases. Note that for pre-exposure and fatigue in air, there are two data points; this is as a result of some specimens being incompletely dried after exposure, resulting in shorter fatigue lives and corroded fracture surfaces. The successful pre-corrosion results show a 10% and 20% reduction in fatigue life for the low and high stress, respectively. The authors point out the fact that the trend to a greater environmental effect at a higher stress is contradictory to data in the literature which shows that the reverse trend should be expected. This revelation, as well as the general results, led to the following conclusions:

- The overall effects of the environment on fatigue lives were significant and consistent at both stress levels; and
- Realistic specimens are necessary for correct assessment of environmental effects.

The CFTCP was succeeded by the FACT program which recommended the use of the same specimen and identical mechanical/environmental testing conditions. From there, the participants defined their own individual programs, which generally amounted to comparisons of environmental effects on fatigue life between different materials, protection systems, and fasteners. Of all the participants, however, only one performed identical tests using alclad 2024-T3. In these tests, conducted by NLR in the

Netherlands, the result of pre-corroding the specimens resulted in an average 17.5% reduction in fatigue life at the lower stress level. For another common aerospace alloy tested, 7075-T6, the life reduction due to pre-corrosion was a more dramatic 36%. Compared to CFTCP results, it is evident that the choice of material and heat treatment can significantly alter the results. Based on this and other results, the FACT program concluded that in addition to realistic specimen geometries, the evaluation of environmental effects on fatigue life should include realistic specimen materials, load histories, stress levels and environments. This re-iterates comments stated earlier by Swift [51].

Both the CFTCP and FACT programs generated an immense amount of data and knowledge with regard to the overall environmental effects on fatigue life. The authors, however, are quick to point out that extrapolation of the results to other structural configurations should not be attempted. Nevertheless, with respect to pre-corrosion, the trend that pre-existing corrosion can significantly reduce fatigue life was thoroughly established.

Building upon the results of the CFTCP and FACT programs, Oore and Russell [60] initiated a program for generating relevant airframe material fatigue data in support of the Canadian Air Force operating in the Atlantic region. The program involved fatigue testing dogbone specimens with a central hole, made of 0.08 inch thick alclad 2024-T3, at 4 different stress levels and 4 different pre-exposure times. In total, 96 specimens were tested (6 per testing configuration). Pre-corrosion was developed in 5% NaCl salt spray in compliance with the widely-used ASTM standard B-117 [70]. A plot of the average life values for each stress level and pre-exposure time is shown in Fig. 4.12. For the most part, the fatigue lives for 0, 3, and 7 days exposure are quite close, whereas for 14 days exposure, they are consistently lower at all stress levels. This revealed an interesting trend in that life reduction only became significant for the highest amount of pre-exposure. Compared to the uncorroded specimen results, the 14-day pre-exposure

resulted in fatigue life reductions in the order of 10 to 20%. This agrees well with the NLR results.

Similar tests were conducted by Fadrakas et al. [71] on alclad 2024-T3 but, instead of dogbone specimens, $1\frac{1}{2} \times 7$ inch panels with a central countersunk hole were used. Three batches of 11 specimens were tested: one uncorroded batch, one pre-corroded batch (ASTM B-117), and one where corrosion was introduced after 100,000 cycles of pre-fatigue. The last batch corresponds with the three-block alternating fatigue/corrosion option discussed earlier. The results show that on average, pre-corrosion reduced life by around 8% whereas corrosion introduced at 100,000 cycles accounted for only a 1% life reduction. The authors attribute the pre-corrosion life reduction to an increase in the severity of defects responsible for crack nucleation. However, no explanation for the apparent ineffectiveness of late corrosion introduction is offered. It appears that since most of the cracking began shortly after 100,000 cycles, the introduction of corrosion at that late stage had virtually no effect. Based on all results, the authors concluded that pre-existing corrosion increased the probability of early crack formation, but had little effect on subsequent crack growth.

Regarding Oore and Russell's three-block test, a similar result is reported by Du et al. [72]. Using alclad 2024-T3 dogbone specimens, their goal was to determine the effect of varying the amount of initial fatigue cycles before a fixed exposure (3.5% NaCl, 10% H_2O_2 , 48 hrs). The results in Fig. 4.13 reveal a 5% increase in fatigue life due to initial fatigue cycling, when compared to pre-corrosion alone (labelled "fatigue-alone" i.e. no subsequent corrosion exposure). It was shown that for low initial cycles (50k and 100k), the corrosion would attack the surface defects created by fatigue, then act as stress raisers in subsequent cycling causing earlier crack nucleation. However, for higher initial cycles (150k+), where substantial micro-cracking had already developed, the introduction of

corrosion tended to blunt the tips of micro-cracks and the sharp corners of voids. This reduced stress concentrations and led to the observed longer fatigue lives.

No other alternate corrosion/fatigue test results of relevance were found in the literature. Therefore, based on these two results, it appears that the three-block approach may not be as simple or more realistic as previously suggested. In the first case, the results did not differ significantly from uncorroded tests, and in the second case, the results were similar to the simpler pre-corrosion approach. Although neither test involved lap splices, it nevertheless seems that the pre-corrosion format is the logical starting point for analyzing MSD and corrosion.

One such investigation, involving MSD and pre-corrosion, is reported by Smith et al. [73]. Their project first considered several approaches to achieve accelerated pre-corrosion, then examined the chosen method's effect on fatigue behaviour in aluminum alloys. In assessing several ASTM accelerated corrosion procedures, they decided that an acidified salt fog procedure (ASTM B-368-85 [74], known as the CASS test) works best at reproducing pitting corrosion, while the EXCO test (ASTM G-34-90 [75]) best reproduces exfoliation corrosion. Their basis of comparison was from 7-year seacoast exposure tests on flat panels, and thus no attempt was made to correlate the damage with that from actual corroded aircraft structure. The fatigue results in this report only refer to EXCO-corroded specimens made of 7075-T651 and 2024-T351 bare aluminum. The specimens were 10 inch wide, 3/8 inch thick plates with three collinear open holes as shown in Fig. 4.14. Before initiating fatigue cycling, a strip of corrosion was developed over the row of holes during a 5 day exposure in the EXCO solution. For the 7075-T651 alloy, pre-corrosion resulted in a 74% reduction in crack initiation and a 43% reduction in crack growth lives. For the 2024-T351 alloy, the effect of pre-corrosion was a 78% reduction in crack initiation and a 36% reduction in crack growth lives. The dramatic life reductions are due to the extremely aggressive nature of the EXCO solution and the use of unclad materials. The authors concluded that the pre-corrosion had the effect of

greatly reducing crack initiation and growth lives. However, they fail to mention that some of the growth reduction is attributable to more MSD achieved in the corroded specimen tests. No report on their tests using the CASS solution was subsequently found in the literature.

Work by Chubb et al. [76] and Koch [77] dispute Smith's findings for 2024 (in T3 and T351 tempers), both revealing that pre-corrosion has no effect on fatigue crack growth behaviour. However, for the aluminum zinc alloys tested (e.g. 7075-T6/T651, 7178-T6), both confirmed that pre-corrosion in the EXCO solution has a significant accelerating effect on fatigue crack growth.

Specifically with regard to fuselage lap splices, Alcoa, one of the primary suppliers of aluminum to the aerospace industry, as well as the sole supplier for the CFTCP and FACT programs, has begun some preliminary research into simulating corrosion and fatigue. Most recently reported was a series of pre-corrosion fatigue tests on KC-135 lap joint specimens shown in Fig. 4.15 [78]. Pre-corrosion development involved the use of an artificial acid rain formulated to provide the same chemical species found in naturally corroded lap joints. Their results revealed that the fatigue lives for the pre-corroded specimens were similar to those of uncorroded specimens. In a subsequent breakdown, it was found that the corrosion damage was limited to the fillet shoulder away from the critical rivet row (no quantification given). Apparently the solution was not sufficiently aggressive to penetrate into the full faying region. The experience highlighted the fact that replicating natural lap splice corrosion is difficult and requires multi-disciplinary understanding. Nevertheless, the results reveal what might be an important trend that corrosion damage away from the critical row does not affect fatigue life. Whether this trend is evident in other configurations is unknown since no further data of this nature was found in the literature.

No other reports of investigations into corrosion and fatigue in fuselage lap splices were found. This is surprising since with the final recommendations of the CFTCP/FACT programs in 1989 to use realistic specimens and environments, it was expected that more investigations adhering to these suggestions would have been carried out. Although there are many reports of investigations into MSD in lap splices, few have considered its interaction with other damage processes, such as corrosion.

In summary, it appears clear from the reviewed material that pre-corrosion of aerospace aluminum alloys can lead to significant reductions in fatigue life. Although most tests reveal the life reduction as being primarily a reduction in initiation life, some tests show small reductions in crack growth lives as well. For aluminum zinc alloys (e.g. 7075-T6), there appears to be a consensus that pre-corrosion accelerates crack growth. However, for aluminum copper alloys (e.g. 2024-T3), there remains some debate.

4.3.3 Accelerated Corrosion Procedures

It is evident from the foregoing that there has not been much research into the development of accelerated corrosion procedures which simulate naturally occurring damage within lap splices. Although Alcoa has developed an artificial acid rain which simulates natural corrosion products, the solution has not yet been successfully introduced into realistic lap splice specimens.

However, the Institute for Aerospace Research at the National Research Council of Canada has achieved considerable progress in this area. Krishnakumar et al. [79,80] have successfully developed a lap splice corrosion method which simulates the chemical species of natural corrosion products and gradually builds up within the joint causing skin deformation (so-called corrosion pillowing).

After an evaluation of all available ASTM standards for accelerated corrosion, Krishnakumar settled on a modified version of the B368 Copper-Accelerated Acetic Acid-Salt Spray (CASS) Testing Method. To verify the nature of the corrosion products, samples from specimens subjected to CASS were chemically analyzed and compared with samples extracted from lap splices in aged Boeing 727 transport aircraft. The artificial corrosion products were found to be similar both in composition and phase to those occurring naturally in the 727. However, no microscopic or metallographic studies were conducted to verify the similitude between physical damage developed from artificial corrosion to that developed from natural corrosion. Nevertheless, the acidified salt fog procedure seems to be a logical starting point for developing accelerated corrosion in lap joint specimens.

5. PROJECT DEFINITION

5.1 Current Research Needs

This project definition follows from the definition and analysis of the MSD and corrosion problems in fuselage lap splices discussed in previous chapters. Within these sections, several areas of research with potential benefits to improve the current understanding of these problems, were identified. They include:

1. Experimental research to investigate the effects of pre-existing corrosion on fatigue cracking in fuselage lap splices.
2. Involved with (1), more detailed investigations to assess and quantify the effect of various parameters (e.g. stress level, skin thickness, rivet type, rivet pitch, clamping force) on the initiation patterns and growth characteristics of MSD, both with and without the presence of corrosion; this is needed to support current fleet activities and to improve the design of new fuselage lap splices.
3. Proven methods of developing accelerated lap splice corrosion similar to that occurring naturally; this involves verification that the artificial corrosion products are of the same chemical species as natural corrosion products, and that the physical damage characteristics are also comparable.
4. The identification of corrosion configurations which are known to produce significant reductions in fatigue life.
5. The development of quantitative methods to characterize corrosion damage.
6. The compilation of corrosion and fatigue data from naturally aged aircraft lap splices; the creation of damage development profiles.

7. Estimates of crack growth periods under the rivet heads.
8. The development of improved non-destructive inspection techniques for detecting early fatigue cracking and corrosion damage.
9. The incorporation of the foregoing research into regulatory guidelines, service bulletins, design handbooks, aircraft maintenance and inspection documents, structural repair manuals, and other related material.

5.2 Chosen Research Areas

Industry initiatives into research on the interaction of corrosion and fatigue in fuselage lap splices have been few. Consequently, a considerable lack of understanding and data currently exists. In view of this and the potential benefits that would arise from research in this area, it was therefore decided to initiate an experimental research project to investigate the effects of pre-corrosion on MSD initiation and growth characteristics in fuselage lap splices.

Several other worthwhile tasks were attempted as part of the experimental research project

- (a) an initial set of fatigue tests without corrosion to confirm the validity of the MSD specimen, and to generate the required baseline data (e.g. crack initiation and growth lives, characteristics of MSD growth and link-up, estimates of crack growth rates);
- (b) verification that the modified CASS test developed by Krishnakumar et al. [79,80] produces physical damage similar to that which occurs in naturally aged aircraft lap splices;

- (c) development of new techniques to aid in the characterization and quantification of lap splice corrosion damage:
- (d) application of the former techniques in identifying correlations between corrosion location and severity with the observed fatigue results:
- (e) post-failure teardown and analysis of uncorroded and corroded MSD specimens, as well as samples from an aged Boeing 727 lap splice: microscopic and fractographic studies to:
 - generally assess the condition of the faying surfaces
 - document various degrees of attack due to artificial and natural corrosion: necessary to accomplish (a)
 - examine and document characteristic features of fatigue fractures, corrosion products and fretting damage
 - determine crack origins: assess effect of pre-corrosion
 - estimate crack growth periods and rates under the rivet heads: assess effect of pre-corrosion.

5.3 Research Program Specification

The research program was divided into the five phases of work shown below, each with its own breakdown of tasks. Fatigue testing was completed in conjunction with a strain monitoring program which determined membrane and bending stresses at a nominal position (1 inch from critical row away from splice) and at the critical row (specific details to follow in next chapter).

Phase 1 - MSD Specimen Manufacturing

- manufacture of 8 MSD specimens (see Appendix A, and note below)

Phase 2 - Baseline Fatigue Testing

- fatigue in air of 5 MSD specimens (labelled F-1, F-2, ..., F-5; F = fatigue)
- nominal strain readings from gauges on 4 specimens
- critical row strain readings from gauges on 2 specimens
- analysis of results

Phase 3 - Pre-Corrosion Development

- evaluation of modified CASS test: trial MSD specimen and a 42-day exposure
- preparation of 3 MSD specimens for pre-corrosion
- pre-corrosion of 3 MSD specimens in a salt fog chamber for 52 days
- periodic inspection throughout exposure using available NDI equipment

Phase 4 - Pre-Corroded Specimen Fatigue Testing

- fatigue in air of 3 pre-corroded MSD specimens (labelled CF-1, CF-2, and CF-3; CF = corrosion + fatigue)
- nominal strain readings from gauges on all 3 specimens
- no critical row strain readings
- analysis of results

Phase 5 - Teardown and Analysis

- teardown and analysis of F-2, F-3, CF-1, CF-2, CF-3, and a B-727 splice
- visual, microscopic and fractographic studies on select samples

The basic parameters used for all fatigue tests were as follows:

- Loading: Constant amplitude, uniaxial
- Load Ratio: 0.02 (Min = 144 lbs., Max = 7219 lbs.)
- Frequency: 8 Hz, sine wave
- Environment: Laboratory air (22-24°C, dry)

Note that specimen F-1 was manufactured using the original procedure detailed in reference [8]. The main difference was in the rivet installation, which employed a straight method of driving the rivet, whereas subsequent specimens used an orbital driving method.

5.4 MSD Test Specimen

The original concept and design of the MSD specimen was developed by G.F. Eastaugh [9], the predecessor of this project. An illustration of the specimen is shown in Fig. 5.1. It was designed to approximate on-aircraft conditions and in a single proof-of-concept fatigue test conducted by Eastaugh, demonstrated a capability to develop and grow MSD in a realistic manner.

The materials, splice geometry and stress distribution were chosen to be roughly representative of current generation commercial transport aircraft. The bonded side straps provide a simulation of the load transfer which occurs as cracking progresses, allowing for more realistic crack growth. The doublers provide a nominal stress distribution across the width of the splice which is close to the hoop stress gradients on Boeing aircraft. This generally amounts to a fairly flat distribution, falling to around 80% of the maximum value at the stiffeners.

The MSD specimen's unique simulation of actual aircraft splice parameters provides a high level of confidence that the trends which develop are likely to be representative of real aircraft structural fatigue. With the accurate simulation of lap splice corrosion, it is reasonable to believe that the specimen will also behave with a similar realism. The pre-corrosion fatigue tests will assess the validity of the specimen for use in corrosion studies in combination with MSD research.

6. EXPERIMENTAL METHODS AND EQUIPMENT

6.1 Overview of the Test System

The experimental test setup is shown in Fig. 6.1. The specimen, including clamping plates, was mounted in an MTS load frame containing a single, servo-controlled hydraulic jack. Using constant amplitude sine wave fatigue testing with a maximum load of 7,220 lbs. and a load ratio of 0.02, the system tuning was optimized resulting in a test frequency of 8 Hz. This setup allowed for 100,000 consecutive cycles in only 3 $\frac{1}{2}$ hours.

Strain data was measured using a 20-channel strain gauge multiplexer and display system. Data acquisition software was written to scan and post-process the data. Plots of gauge, hoop and bending stresses, as well as the raw data were then immediately available.

Crack initiation was detected using a combination of visual and microscopic methods. Crack growth was measured using a travelling microscope. Lengths were measured horizontally from the rivet centre out to the crack tip. Crack tips were located with the aid of a fibre optic light source.

Crack configurations starting with specimen F-2 (except F-5) were digitally recorded using a small CCD camera connected to a video processing card inside a PC. For the corrosion tests, standard 35mm black and white photography was also employed.

More specific details on the test system are provided in sub-sections to follow.

6.1.1 Load Frame

All fatigue tests were conducted on an MTS three-poster uniaxial load frame, shown in Fig. 6.1, with a maximum capacity of 100,000 lbs. The frame contains an upper screw which can be set to accommodate various specimen sizes. Large caps through which the screw is threaded can be loosened for alignment purposes.

The hydraulic jack used had a 55,000 lb. capacity and six inch stroke. It was also equipped with an LVDT position sensor. The load cell was located above the specimen and had an 11,000 lb. maximum capacity which provided good discrimination for the loading range used. It was re-calibrated by IAR personnel before testing began.

The specimen was held firmly in place by a set of fork-ends and close tolerance pins. Once the rig was properly aligned, the upper fork was fixed. The lower fork was raised to the correct height once the specimen was pinned and hanging from the upper fork.

The various components were bolted together by a set of high strength studs. These were pre-tensioned at 10% above the maximum load using spiral washers, to prevent fatigue failure.

Several servo-valves were initially evaluated before it was decided that a 2½ GPM (gallons per minute) valve would provide the necessary control yet still be able to handle test frequencies up to 10 Hz. With this initial system, an optimized frequency of 8 Hz was achieved, substantially away from any instability. The valve was later changed to a 10 GPM flow capacity for use with a new control system, although the test frequency was not changed. At 8 Hz, tests were normally completed within 3 to 5 days.

The jack was supplied with oil at 3000 psi via the servo-valve from a hydraulic service manifold (HSM) which contained high and low pressure supply valves, as well as an

accumulator to minimize supply pressure fluctuations. The HSM received oil from a large-capacity hydraulic pump.

A special platform was manufactured from 1 inch thick aluminum alloy to provide a flat and rigid surface on which to place the travelling microscope. It was placed at a height which allowed for easy crack measurements.

6.1.2 Control System

The control system automatically regulates the applied load to provide a feedback signal (from the load cell, or position sensor) that is as close as possible to the command signal sent to the servo-valve. Other basic functions of the control system include shutting down the hydraulic pressure, and stopping a test or providing warnings when certain pre-defined interlocks are breached.

Two different control systems were used as a part of this project. Specimens F-1 to F-4 were tested using the MTS FlexTest Digital Servo Controller (DSC). FlexTest DSC is a multi-channel system which provides complete system servo control for up to four test stations. The operator interacts with the system through specially-designed Windows-based software on a separate PC. A photograph of the setup is shown in Fig. 6.2.

Extensive system calibration and manual tuning was necessary with FlexTest and was completed with several 'dummy' specimens in accordance with procedures in the User's Manual [81]. Shunt calibration was used for the load-feedback circuits. Loading programs were created using MTS 1/RAC Windows-based software [82] and downloaded into a real time active controller before beginning a test. Although these programs set the appropriate upper and lower loads, it was still necessary to make slight

adjustments to the set point and span controls throughout a test in order to achieve the desired load range.

For specimen F-5 and all 3 pre-corroded specimens, a much improved control system was used. It was a new MTS TestStar II digital servohydraulic control system as shown in Fig. 6.1. This system is controlled by the operator using OS/2-based software on a PC [83], and/or through the use of an external pod which displays real-time system parameters and allows for full test and actuator control. Several interactive loading programs were created using MTS TestWareSX, also an OS/2-based software package [84] designed for use with TestStar II. After MTS performed an initial system calibration, the test system was manually tuned to the specimen. Once a cycling program was started, no further adjustments needed to be made: through the use of dynamic compensation, the control system automatically brings the loading range to the one desired. It also employs a feed-forward loop to bring the command and feedback signals into phase.

6.1.3 Strain Gauge System

All specimens with the exception of F-5 contained seven gauges (350 Ω) at the nominal position (one inch from critical row) on each side of the countersunk sheet to provide the nominal stress distribution. For specimens F-1 and F-3, three gauges (120 Ω) were placed on a line tangential to the outer hole edge of the critical row on each side of the countersunk sheet to provide a rough estimate of the secondary bending effect. An example of the installation is shown in Fig. 6.3. Note that three of the critical row gauges are hidden within the splice (not shown), and therefore the non-countersunk sheet must have grooves milled to compensate (see Appendix A for further details). All gauges were installed using Micro-Measurements equipment and procedures [85,86].

The strain gauges were wired to locally-manufactured bridge circuits connected to a Hewlett Packard 3497A Data Acquisition Control Unit. Connected to this unit was a Hewlett Packard 85 computer. Gauge excitation was provided by a precision voltage source. Excitation levels were optimized in accordance with established procedures [87]. The complete setup was shown in Fig. 6.1.

As part of this project, a generic data acquisition software package called "G-Scan" was written in HP-85 Basic (listing in Appendix B) to perform several data acquisition tasks:

- scan gauge output on demand
- record the data (including real-time calculation of stress and change in stress since last scan) in hard copy format
- produce auto-scaled plots of gauge, membrane and bending stresses for nominal and critical row positions
- allowance for re-zeroing gauges or checking existing zero values.

The objective of G-Scan was to provide immediate pictures of the various stress distributions without having to manually generate the required information. This allowed for a more immediate response to the results, and also translated into shorter test times.

An assessment of the strain gauges' ability to reveal evidence of crack growth under the rivet head was also performed. This involved frequent scans and close examination of the data throughout a test.

6.1.4 Crack Detection, Measurement and Recording

The initial detection of cracks was done fairly easily since before a crack would emerge from beneath the rivet head, it was preceded by a small but noticeable area of surface deformation. The perturbations for initial MSD cracks would generally form over 5000

cycles before producing a discernible crack tip. A combination of viewing the suspect location at a sharp angle in diffuse daylight and through a microscope with backlighting provided the best evidence of the onset of these surface deformations. Some difficulty arose when there were either machining marks or corrosion damage (due to blistering) close to the rivet edge.

All cracks were measured using a 60× travelling microscope with a digital readout accurate to within ± 0.0005 inch. Measurements were made horizontally from the rivet centre (small dimple left from manufacturing) out to the crack tip. A modified fibre optic light source aided in finding the crack tip. Determining the precise location of the crack tip was sometimes difficult, especially in the early stages of growth when cracks were tight, or when a crack grew into an abraded area (e.g. from strain gauge installation). With experience, most crack tips could generally be located within ± 0.001 inch.

A set of crack measurements was generally made when the fastest crack had grown another 0.020 to 0.030 inch. This also coincided with a recording of the strain gauge data. Crack measurements were stopped when a single crack spanned all rivets on the critical row. This is referred to as the "fully cracked" condition.

Careful attention was given to ensuring that the travel of the microscope was both parallel to the rivet row centreline and the plane of the crack currently being measured. The latter was especially important in the later stages of cracking when the splice area would start to deform due to the redistribution of load. Following these methods led to more precise crack measurements.

Rivet centres along the critical row were also measured relative to the left-most rivet. This data along with the crack lengths provided information from which all crack co-

ordinates were calculated. These were later used to produce a single plot of all crack growth curves as well as crack growth rate curves.

Crack growth rates were calculated in accordance with ASTM E-647-93 "Standard Test Method for Measurement of Fatigue Crack Growth Rates" [88], which specifies the *secant method* as one of the recommended data reduction techniques for calculation of crack growth rates. This method calculates the average growth rate between two adjacent data points, which is then plotted against the average crack length. The accuracy of the calculated growth rates is linked to the frequency of measurements that have been recorded for a particular crack. The method lends itself more to the initial MSD cracks which grow for longer periods and have more recorded measurements. Once the data points were plotted, a weighted least squares algorithm was applied to fit the data. The experimental crack growth rate curves provided additional perspective to the interpretation of test events.

Crack configurations starting with specimen F-2 (except F-5) were digitally recorded using a crack imaging system. The system consisted of a small CCD camera and associated parts (35mm lens, 70mm lens, extender tubes, video cables and switches, power supply, tripod, small stand), connected to a video signal processing card inside a 386 computer. A variable intensity ring light was also used to provide illumination for close-up images. The system provided excellent quality close-up images down as small as a 1/4 inch view width. An example of the camera setup is shown in Fig. 6.4. The quality of images was good when viewing up to around 5 rivets, then degraded the more the view width was increased. For the corrosion tests, this prompted the addition of standard 35mm black and white photography. Photos were later digitized but nevertheless provided superior quality and detail compared to what was achieved with the CCD camera for the wider shots.

2

PM-1 3 1/2" x 4" PHOTOGRAPHIC MICROCOPY TARGET
NBS 1010a ANSI/ISO #2 EQUIVALENT

PRECISIONSM RESOLUTION TARGETS

6.1.5 Alignment of the Load Frame and Specimen

The load frame was levelled and the upper screw thread was aligned with the lower assembly within ± 0.002 inch to minimize loading asymmetries. The position of the screw remained fixed throughout all tests.

When attaching the clamping plates to the specimen, the vertical axis of the clamping plates was aligned to be parallel with the vertical axis of the specimen.

Before installing a specimen, an alignment check was done on the plane of the fork ends to make sure that they were parallel. This allowed for easier pin fitting and ensured that there would be no twist introduced into the specimen. Once installed, the same check was done on the clamping plates to further confirm the alignment.

An initial alignment of the specimen and clamping plates on the pins was also performed. This could be done visually. If some misalignment remained, normally it could be heard once cycling began and the test was stopped to make the required adjustments.

Based on the test results, the alignment procedures were successful as there were no premature failures due to nuisance cracking.

6.2 Pre-Corrosion Development

6.2.1 Testing Methods and Apparatus

Pre-corrosion was developed using a modified version of the ASTM Standard B368 Copper-Accelerated Acetic Acid-Salt Spray (CASS) Testing Method [74]. The

modifications involved using a slightly more acidic salt fog (pH of 3.0 instead of 3.1-3.3), a lower fog collection rate, and suspending the specimen vertically instead of at the specified 15° to the vertical inclination.

Specimens were corroded in a Singleton Salt Corrosion Cabinet (SCCH Model #22). A schematic of the setup is shown in Fig. 6.5. The cabinet is equipped with a humidifying (bubble) tower with a micro-bubble aerator for generating humidified air. Air is fed into a fog tower reservoir inside the cabinet where it is mixed with the salt solution before being dissipated through an atomizer into the dispersion tower. The latter is fitted with internal baffles to produce a homogeneous mist of free falling spray. Distilled water (for the bubble tower) and salt solution are contained in large reservoirs. The temperatures inside the bubble tower (35°C) and the cabinet (48°C) are monitored by an automatic electronic control system. The bubble tower pressure was lowered from the specified 18 psi to 13 psi to avoid the lid from lifting and venting salt fog into the lab. This resulted in the observed lower fog collection rate.

As per ASTM B368, the salt solution is prepared with 5 parts by weight of sodium chloride dissolved in 95 parts of distilled or reagent grade water and reagent grade copper chloride ($\text{CuCl}_2 \cdot 2\text{H}_2\text{O}$). The pH of the salt solution was lowered to a value of 2.8-2.9 by the addition of glacial acetic acid, which provided the atomized salt spray with a pH value of 3.0.

A corrosion trial specimen, CT-1, which had been damaged, was devoted to evaluating the modified CASS test. It was also intended as a means of developing experimental techniques and procedures for non-destructive inspection (NDI), application of protective coatings, and teardown and analysis. These methods would be later applied with specimens CF-1 to CF-3.

Before exposing CT-1, all external surfaces were first spray painted using a white enamel based Tremclad product. Prior efforts by LAR had proven that the moderately thick and uniform film left by the Tremclad provided good environmental durability, preventing corrosion from developing in two to three weeks trials of continuous exposure [80]. Tremclad also provided a smooth glossy finish which made it ideal for an enhanced visual technique used for NDI purposes (discussed in next section). Any paint filleting around the lap edges was removed to provide the salt spray with an unobstructed path into the splice. In addition, the specimen was oriented in the chamber with the critical row at the top. This was done to maximize the flaying surface corrosion around the critical row.

In 42 days (6 weeks) of continuous exposure, specimen CT-1 remained intact, although there were signs of external corrosion. These areas seemed to be confined to where the paint was thinner and salt product would build up (e.g. at most edges and corners). Some rivets also had corrosion blisters. Consequently, future specimens were given a thicker coating of paint. This added protection was also used since, based on NDI and teardown results for specimen CT-1, specimens CF-1 to CF-3 would undergo a longer exposure of 52 days (7½ weeks). Nevertheless, specimens edges still needed to be repainted after 42 days of exposure due to areas of significant deterioration. Once a specimen reached its final exposure and was extracted from the chamber, the Tremclad was easily removed from splice and strain gauge areas using an environmentally acceptable paint stripper.

Throughout an exposure period, the build-up of solid salt was periodically removed using warm water and a soft bristle brush, being careful not to let water into the splice.

After the last exposure period, the specimens were immediately cleaned and dried in an unpressurized autoclave at 50°C for a period of 4 hours. This drying procedure was adapted from one specified in the FACT program [69] and is intended to stop any further corrosion development. Further precautions involved individually wrapping and storing

specimens in a sealed plastic bag in a cool 21 °C temperature-controlled, low-humidity room. Specimens remained in this environment until they were tested, and were returned to this location upon test completion. As previously mentioned, fatigue testing was completed within a three to five day period in a relatively cool and dry lab environment, again to minimize the possibility of further corrosion growth which could affect the test results. With these precautions taken, it is unlikely that any further significant corrosion developed beyond that generated through accelerated exposure.

6.2.2 Non-Destructive Inspection Techniques

Specimens were examined using four NDI methods: visual, enhanced visual, shadow moiré, and eddy current. For specimens CF-1 to CF-3, this was performed after accumulating 21 days, 42 days, and 52 days of exposure.

Visual inspection was made immediately after the specimen was extracted from the corrosion chamber. By holding the specimen at certain angles in diffuse lighting, it was possible to see some of the pillowing around the rivet heads which, as Ch. 3 points out, would indicate the presence of faying surface corrosion. However, this determination was only possible at the later stages of exposure, when significant volumes of corrosion products had accumulated. Skin deformation was difficult to see because of the salt fog's ability to remove the paint's original gloss leaving a dull, matte finish.

An enhanced visual technique called Diffracto Sight (D Sight) was able to overcome the difficulties with visual inspection. D Sight has been developed by Komorowski [80] of IAR for application to corrosion detection in fuselage splices. It is based on the determination that corrosion products occupy a significantly greater volume than the consumed metal (around 6× greater [79]). D Sight can produce a full field visually

enhanced view of corrosion pillowing in real time. The process uses a two pass illumination of a reflective surface using a light source and low grazing angle, and a retroreflective screen. When the local curvatures on the surface are viewed or imaged from the principal light source, the light returned by the retroreflective screen and reflected off the surface creates distinct grey scale signatures of the surface in proportion to the amount of local surface curvature. This optical arrangement is extremely sensitive to subtle surface curvatures caused by pillowing and reveals the presence of sub-surface corrosion even at very low corrosion levels. For this project, the D Sight Aircraft Inspection System (DAIS) model 250C corrosion sensor was used. A schematic of the setup is shown in Fig. 6.6, with the components contained within a portable hood. The D Sight hood is operated and controlled by the DAIS computer system. Because of its ease of use, both sides of a splice were examined for signs of pillowing. For better imaging, the splice areas were first coated with a clear, brightening solution to maximize the reflected light. Unfortunately, D Sight does not provide any direct quantitative information and can often be subject to misinterpretation.

A method which overcomes some of the problems with D Sight is shadow moiré. Shadow moiré is a simple and efficient means of obtaining whole-field contour maps of out of plane deformations. The specimen is first placed horizontally on the ground with the countersunk sheet facing upwards. It is illuminated using a collimated light source from an angle of 45° and viewed through a high resolution CCD camera from vertically above. A moiré grid is then placed over the splice in a preset location. A grid with 200 lines per inch was used for all specimens which provided a sensitivity of 0.005 inch per fringe. However, since a fringe of the opposite colour lies halfway between two fringes of the same colour (black or white), estimates of changes in lateral heights equal to half the fringe sensitivity (i.e. 0.002-0.003 inch) could reliably be made. Unfortunately, the moiré grid used did not provide enough sensitivity for accurate measurements of the small out of plane displacements achieved through 52 days of accelerated corrosion. A grid providing a sensitivity of 0.001 inch per fringe would have produced better results.

Nevertheless, the method made the interpretation of D Sight images much simpler, as both focus on corrosion pillowing.

The most widely used method for detailed inspection of lap splices is eddy current inspection. It is used to obtain the distribution of material loss as a result of the faying corrosion. For this project, eddy current inspection was conducted to obtain data for the countersunk sheet only. The procedure was performed in accordance with Douglas specifications [89] using a Zetec MIZ-40 multi-frequency eddy current test instrument attached to a low frequency eddy current reflection probe (3/8 inch dia., 10 KHz). An 18 inch manual XY positioner was used to locate the probe over the specimen. Positioner data and MIZ-40 information was fed into a Dupont PortaScan PS1 portable colour scan imaging system, equipped with the PSIS-MIZ40 Eddy Current Software. The PortaScan system provides an instant two-dimensional colour mapping of the information from the eddy current probe filtered through gates which can be set to any desired values to display flaws.

The eddy current setup was calibrated using a reference specimen which consisted of two sheets of 0.040 inch thick 2024-T3, one of which had three rectangular grooves of 5%, 7½%, and 10% thickness loss cut into its inner surface. These values represent edges of detectability and appear in the colour mapping as three distinct colours representing three thickness loss ranges from 5 to 7½%, 7½ to 10%, and greater than 10%. The information remaining encompasses the final range from 0 to 5% thickness loss, as well as some missed data (near and under the rivet heads), the rivet hole areas, and a few spots where the software had difficulty resolving the probe data. The latter, however, amounted to only around 2% of the total scanned information. Given the probe diameter and the restriction of the straps, the scans were successful in covering approximately 83% of the splice area.

Eddy current inspections were performed on specimens CI-1 to CI-3 as part of their 42-day and 52-day inspections. Overall, the process provided a good estimation of the material loss distribution on the countersunk sheet.

6.2.3 Technique to Quantify Corrosion Damage

Although the NDI methods provided some measure of quantification, there remained no clear and purely quantitative basis for comparing the damage in one specimen relative to another. Neither D Sight nor shadow moire offered a suitable parameter, and although eddy current provided quantitative information, it could only offer pictorial comparisons of this data between specimens, leaving the assessments primarily qualitative.

It was therefore decided to develop an approach to further quantify the corrosion damage, and allow for more accurate and precise comparisons between specimens. The result was a technique which takes the scanned material loss information from the eddy current inspection and determines the percentages of that information corresponding to certain corrosion levels. The classification of corrosion levels used for this purpose corresponds with the thickness loss ranges from the eddy scans as follows.

<u>Corrosion Level Classification</u>	<u>Thickness Loss Range</u>
Light to Moderate	0 to 5 %
Moderate	5 to 7½ %
Moderate to Severe	7½ to 10 %
Severe	10 %

This corrosion severity index is in good agreement with one established in 1991 by the FAA [44] which specifies 'Light' corrosion as 0-1% thickness loss, 'Moderate' as 1-10% thickness loss, and 'Severe' as above.

The procedural details with an example of its application within this thesis are given in Appendix C. To summarize, the procedure essentially totals the individual 'pixels' (picture elements which make up a computer image) of each colour so that an exact calculation can be made as to each colour's percentage of the scanned area. In that way, the total splice corrosion damage can be broken down into percentages of each corrosion level described above. For example, from an eddy current scan, it might be desired to know exactly how much of the splice is above 10% thickness loss. The method is able to provide this kind of information, in a simple and swift manner.

The main technique used is actually independent of the NDI method used to obtain the original thickness loss information. As it simply works from the basis of a colour image, it does not matter how that image was generated. Therefore, it can be applied with other methods as well, such as DXR (digitally-enhanced x-ray).

The primary attraction of the technique is that it does not introduce any error in the summation of pixels: the totals are exact numbers. It might even be possible to improve on the error of the method used to generate the original data. This could be done through interpolation in areas which are known to be corroded but returned no information. This, however, was not attempted in this thesis.

One of the drawbacks of using the method on the full splice data is that it does not convey any information as to location. However, this is easily overcome by first converting the image into separate pieces, then applying the method on each individual piece. In this manner, the location of critical areas will become evident. For this thesis, the image data for the critical row was separated into individual rivet segments and put through the procedure. This then revealed the location and magnitude of the corrosion concentrations along the critical rivet row. An attempt was made to correlate these findings with the observed crack initiation and growth data for each specimen. The results are presented in the next chapter.

6.3 Specimen Teardown and Analysis

6.3.1 Teardown Procedure

The ultimate goal in the teardown of a specimen was not to disturb the contents of the splice. It was desired to keep the corrosion product distributions intact for later observation. Therefore, specimens were kept flat with the countersunk sheet down throughout the entire teardown procedure.

The first step was to cut the splice portion from the surrounding straps and doublers. This was done on a high powered band saw, keeping close to the straps in order to maximize the amount of splice removed. Sharp edges were then filed for safer handling.

The driven heads of rivets were centre punched and drilled to the depth of the head using a drill press equipped with a 5/32 inch drill bit. Once all were drilled, the remaining portions of the driven heads were chipped off using a chisel and hammer. The countersink heads were popped out using a nail and hammer with the splice resting on soft padding material. Finally, the driven sheet was removed and the contents of the splice were revealed. All specimens were opened successfully with little to no disturbance of the internal products.

6.3.2 Visual Examination

Once a specimen was opened, a visual examination was performed on both sheets. This generally involved the following:

- observing the distribution of the corrosion products, as well as their colour, size, and shape
- assessing the damage around rivet holes due to corrosion and fretting

- checking for and recording any abnormalities
- determining areas for future viewing using optical microscopy

Note that throughout the fatigue testing of the pre-corroded specimens, some of the corrosion products escaped either through the lap edge or through cracks. The amount of escaped product was small compared to the amount remaining, and thus the observed distributions are likely representative of those from a pre-fatigue state.

In order to have a permanent record of the observed corrosion product distributions, several 35mm colour photos were taken of each surface. Various lighting conditions had to be employed since the products were all of similar colour.

6.3.3 Microscopic Examination

A low magnification stereoscopic optical microscope was first used to explore the faying surfaces and examine the various topographical features. Although the examinations provided useful information, the varying depths of most features made it inconducive to micro-photography. Optical microscopy was therefore used only to identify areas or surface features which would later be examined using a NovaScan 30 Scanning Electron Microscope (SEM), shown in Fig. 6.7.

The SEM specimens were made by sectioning the splice into roughly 1 inch² pieces (size restricted by SEM). For the non-cracked rows, this was done so that the rivet hole would be at the centre of the sample. All SEM samples, except for ones where corrosion products were to be viewed, were then chemically cleaned.

The ASTM Standard G1 "Preparing, Cleaning, and Evaluating Corrosion Test Specimens" [90] suggests two methods to chemically remove corrosion products from

aluminum alloys. Some initial trials using specimen C1-1 were conducted to evaluate both procedures. The first method involves immersion in nitric acid (HNO_3) at 20 to 25°C for between 1 to 5 minutes. This procedure was able to remove all evidence of corrosion products, but it also damaged the metal surface by etching away most of the features. The second method involves immersion in phosphoric acid (H_3PO_4) and chromium trioxide (CrO_3) at 90°C to boiling for a duration of 5 to 10 minutes. This initially produced similar results to the nitric acid but, after several trials using different temperatures and exposure durations, a solution was found. Using a mixture temperature of 92°C and a duration of 6 minutes produced enough un-etched areas from which the desired information could be extracted. This optimized corrosion removal procedure was used for all subsequent cleaning.

Finally, before being placed in the SEM, specimens were mounted, and gold-coated to prevent particle charging. An inclination of 15 to 20 degrees to the horizontal was used for optimum viewing.

6.3.4 Fractographic Examination

The fractographic examinations were also conducted in the NovaScan 30 SEM. Before viewing, specimens were cleaned using a procedure from the Fracture Handbook [91] for removal of oxidation products from aluminum alloy fracture surfaces. The method uses the same elements as the chosen solution for cleaning corrosion products, but in different proportions. It also specifies that the mixture can be either cold or warm. After a few trials, it was decided that a temperature around 40°C would suffice. Specimens were watched throughout the exposure and removed when there was no further evidence of black oxide deposits remaining on the fracture surfaces. The procedure worked well and did not seem to impart any further damage to the fracture surface.

Once a batch of specimens was cleaned, the time before viewing them in the SEM was minimized in order to minimize the formation of oxide layers. If a sample was not going to be viewed immediately, it was placed in a desiccator. Six samples separated by spacers could be viewed in one SEM session. Inclination of the samples was varied throughout a session in accordance with the inclination of the particular fracture surface being viewed.

7. RESULTS AND DISCUSSION

7.1 Overview

This chapter presents and assesses the results of the experimental project defined in Ch. 5, which identified four areas of investigation:

1. *Baseline Fatigue Testing:* fatigue in air of 5 MSD specimens.
2. *Pre-Corrosion Development:* 3 MSD specimens exposed to the modified CASS solution inside a salt fog chamber for a duration of 52 days.
3. *Pre-Corroded Specimen Fatigue Testing:* fatigue in air of 3 pre-corroded MSD specimens.
4. *Specimen Teardown and Analysis:* disassembly of specimens, followed by visual, microscopic and fractographic examinations.

A summary of the fatigue test results including the original proof-of-concept (PoC) test performed by Eastaugh [8] and some published Boeing data [29-31] is shown below in Table 1. The results are given in terms of the number of cycles between first detection of a crack, first link-up, and the fully cracked condition. Cycle counts have also been converted into their respective percentages of the test article's total fatigue life. Boeing data is for published cases of MSD on narrow-body and wide-body aircraft having differences in fuselage and splice design, stress levels and predicted life. It is included only to convey the order of magnitude of crack initiation and growth periods.

The results from the four areas of investigations are discussed in detail in subsequent sections. The chapter concludes with an overall summary and brief discussion of the project results.

specimen F-2. At first link-up, Fig. 7.1 (b) shows that there were a total of 9 cracks from rivets 2 to 6 (note: crack at 3L appeared just before link-up). F-2's large amount of MSD partly accounts for its relatively short crack growth period. The least uniform MSD was observed in the test of specimen F-1. By first link-up, only 4 cracks were present.

- (j) In many cases, initial crack growth exhibited a tendency to move in a slightly upward direction (i.e. toward rivet centreline). An example is shown in Fig. 7.3 (b). The process was governed by local effects, which gradually decreased with distance away from the rivet head. The actual crack path was indicative of the direction of maximum principal stress.
- (k) Early crack growth, like initiation life, was also sensitive to the amount of frictional load transfer between the faying surfaces. Whereas crack growth rates for the majority of initial MSD cracks were on the order of 10^{-6} inch/cycle, rates for specimens F-2 and F-5 were between 2 to 5 times greater. This is likely attributable to their relatively higher local stresses, and partly accounts for their shorter crack growth lives.
- (l) An example of the ability of small MSD to force the lead crack to remain in the critical row of rivets was observed in testing F-2. Fig. 7.2 (i) depicts the lead crack tip at 6R growing in an upward direction, appearing as if it was going to grow above rivet 7. However, as Fig. 7.2 (j) reveals, cracks at 7L/R quickly emerged and changed the trajectory of the crack.
- (m) The plots of crack growth rate reflect a sharp rise and fall as two opposing cracks overlap. It appears that the cracks first accelerate toward each other until they are near the same horizontal position. For first overlaps, peak growth rates were consistently between 0.0002 and 0.0004 inch/cycle. As the cracks overlap, there is a rapid deceleration, and the growth rate falls by over one order of magnitude.

growth rates were best captured for specimen F-5 using the new MTS TestStar control system which provided finer control through the link-up stage.

The main observations from the tests are listed and discussed below. The cracking sequences captured for specimens F-3, F-4, and F-5 are shown in Figs. 7.2, 7.3, and 7.4, respectively. References to location refer to the specimen in the tested orientation (i.e. critical row as the lower row). Also recall that crack initiation refers to the point at which the crack first becomes visible.

Crack Initiation

- (a) Crack initiation life represented the major part of the fatigue life, accounting for approximately 75% of the total.
- (b) Initiation life was quite sensitive to the amount of frictional load transfer between the faying surfaces. Load transfer through friction was partially removed in specimens F-2 and F-5 by the presence of oil contamination which led to shorter crack initiation lives. Shortly after beginning the tests, dark oily fluid was observed emerging from beyond several rivet heads in F-2, and at a few rivets in F-5. The contamination occurred during manufacture, and likely resulted in a significant increase in load transfer at the rivets. F-2's considerably shorter initiation life than F-5 is believed to have resulted from widespread contamination.
- (c) Initiation life was also found to be sensitive to the method of rivet installation. It was mentioned in Section 5.3 that PoC and F-1 were manufactured using a different rivet installation method than the remainder of the specimens. Apparently, the process imparted a higher clamping force, which led to longer initiation lives. According to some tests by Schijve [26], doubling the rivet squeezing force can more than quadruple the fatigue life of a lap joint. This relationship implies that the rivets of PoC and F-1 had approximately 25% more clamping force.

- (d) Crack initiation lives for specimens F-3 and F-4 showed little scatter, differing by only 20,000 cycles.
- (e) Crack initiation sites were predominantly below the rivet centreline. In most cases, cracks emerged near the line of maximum bending. An example is shown in Fig. 7.3 (b). The initiation sites provided an indication of the location of the local peak stresses. This result agrees well with published cases of in-service and full-scale test MSD. A notable exception were cracks in F-2 which, as shown in Fig. 7.2 (c), initiated closer to the rivet centreline.
- (f) Crack initiation patterns were also in agreement with published cases of MSD. In 4 out of 5 tests, at least one of the first cracks detected was located at one of the two central rivets (Nos. 4 and 5).
- (g) In many cases, crack initiation was stimulated when an opposing crack drew near. It is likely that these cracks were already present, hidden under the rivet heads. An example is shown in Fig. 7.3 (i) which depicts a crack which emerged and subsequently overlapped with one of the lead crack tips. A slightly different example is shown in Fig. 7.2 (j) in which two cracks initiated at the same time the lead crack grew into the rivet.

Crack Growth

- (h) On average, crack growth life accounted for 25% of the total fatigue life. After first link-up, there was generally less than 2% of life remaining. This is illustrated by the asymptotic nature of the crack growth curves beyond first link-up, and confirms the potential for MSD to link-up and form a long lead crack in a relatively short time period. The result heightens awareness of the critical importance of detecting these cracks before they reach the first link-up.
- (i) The tests revealed a mixture of relatively uniform and non-uniform MSD typical of published cases. The most uniform and widespread MSD was achieved with

specimen F-2. At first link-up, Fig. 7.1 (b) shows that there were a total of 9 cracks from rivets 2 to 6 (note: crack at 3L appeared just before link-up). F-2's large amount of MSD partly accounts for its relatively short crack growth period. The least uniform MSD was observed in the test of specimen F-1. By first link-up, only 4 cracks were present.

- (j) In many cases, initial crack growth exhibited a tendency to move in a slightly upward direction (i.e. toward rivet centreline). An example is shown in Fig. 7.3 (b). The process was governed by local effects, which gradually decreased with distance away from the rivet head. The actual crack path was indicative of the direction of maximum principal stress.
- (k) Early crack growth, like initiation life, was also sensitive to the amount of frictional load transfer between the faying surfaces. Whereas crack growth rates for the majority of initial MSD cracks were on the order of 10^{-6} inch/cycle, rates for specimens F-2 and F-5 were between 2 to 5 times greater. This is likely attributable to their relatively higher local stresses, and partly accounts for their shorter crack growth lives.
- (l) An example of the ability of small MSD to force the lead crack to remain in the critical row of rivets was observed in testing F-2. Fig. 7.2 (i) depicts the lead crack tip at 6R growing in an upward direction, appearing as if it was going to grow above rivet 7. However, as Fig. 7.2 (j) reveals, cracks at 7L/R quickly emerged and changed the trajectory of the crack.
- (m) The plots of crack growth rate reflect a sharp rise and fall as two opposing cracks overlap. It appears that the cracks first accelerate toward each other until they are near the same horizontal position. For first overlaps, peak growth rates were consistently between 0.0002 and 0.0004 inch/cycle. As the cracks overlap, there is a rapid deceleration, and the growth rate falls by over one order of magnitude.

- (n) Peak growth rates for a direct joining of the crack tips were between 4 to 10 times greater than for overlaps. Direct link-ups therefore occurred in shorter time periods than for cracks overlapping. Only F-2 and F-5 had all direct link-ups (i.e. no overlapped cracks); they also had the shortest crack growth periods.
- (o) Once the lead crack had spanned most of the rivets, its growth rate was on the order of 0.0025 inch/cycle. Continuing at this rate on an actual aircraft splice, the crack would span almost an entire frame bay in the next 4,400 cycles, and could occur between inspection intervals.

A method of directly comparing crack growth periods is an *aggregate crack growth plot*. This graph plots a sum of all crack lengths at a particular cycle count, against the number of cycles beyond the point where cracks were first detected. The aggregate crack growth plot for the baseline fatigue tests is shown in Fig. 7.5, with the first link-ups marked. Horizontally, the curves span at least 7 inches which represents the lead crack length in the fully cracked condition.

The scatter in the crack growth periods is evident in Fig. 7.5. A few other observations are noted, to be later compared with the corrosion test results:

- (a) The average aggregate crack growth rate up to a length of $2\frac{1}{4}$ inches was on the order of 3.8×10^{-5} inch/cycle (mainly for F-3 and F-4). After first link-up, it was approximately 0.0017 inch/cycle, almost 50 times faster. After 5 inches, growth rates reached almost 0.003 inch/cycle.
- (b) First link-ups occurred consistently at an aggregate crack length of 2.5 ± 0.5 inch. F-3 and F-4 both had link-ups after accumulating 2.6 inches of cracking.

7.2.2 Strain Gauge Results

The membrane and bending stress distributions measured for specimen F-3 in an uncracked state are shown in Fig. 7.6. Nominal stress distributions in other specimens did not differ by more than 2%. Critical row distributions were only successfully obtained for F-3, and only while uncracked.

The magnitude of the nominal membrane stress ranged from a low of 13.2 ksi near the straps, to a peak of 14.3 ksi at the middle. Peak stresses were in the order of those found on Boeing narrow-body aircraft [29]. Comparable values for the critical row were 12% and 6% lower, respectively. The magnitude of the nominal bending stress ranged from 0.7 to 2.5 ksi. At the critical row, secondary bending pushed the range to between 3.8 to 6.4 ksi.

The secondary bending ratio ($\sigma_{\text{bend}}/\sigma_{\text{local}}$) and secondary bending factor ($\sigma_{\text{bend}}/\sigma_{\text{remote}}$) are shown in Fig. 7.7. The predicted value for the secondary bending factor for the MSD specimen based on previous research by Hartman and Schijve [92] and Schütz and Lowak [93] is approximately 0.4 at the centre. The measured value of 0.45 compares well with this prediction and suggests that the secondary bending in the specimen is generally representative of that found in real aircraft splices.

The redistribution of stress during the cracking period is shown in Fig. 7.8 in the form of carpet plots for each gauged specimen. The figures depict the variation of hoop stress at the nominal position as a function of the number of test cycles. In all cases except for F-2, the last 3 to 4% of the test is shown; F-2 has the last 8% shown due its overall shorter fatigue life. The plots reveal how rapidly the hoop stress dropped as the cracks advanced. Sharp peaks are also evident and represent the shed load being picked up by adjacent material. Peak stresses around 25 ksi were common before final link-up. Once fully

cracked, the final distributions suggests that the load was being transferred through the straps in all cases.

The strain gauges revealed no evidence of cracking underneath the rivet head before visible detection.

7.2.3 Summary

The baseline fatigue tests were all successful in developing and growing MSD in a realistic manner under laboratory conditions. The crack patterns revealed a mixture of relatively uniform and non-uniform MSD, with most MSD clustered centrally in the higher stress areas. All specimens exhibited stable and consistent crack growth patterns until cracks had linked up spanning the whole critical row. These results agree well with published cases of MSD.

Nominal stress distributions were fairly consistent between specimens, and preliminary results from critical row strain gauges reveal that secondary bending in the specimen is generally representative of that found in real aircraft splices.

The results generally revealed that both on-aircraft conditions and in-service cracking were well simulated, and confirm the validity of the coupon specimen for MSD research.

Qualitatively, the patterns of MSD initiation, growth and link-up observed in all specimens are directly comparable to the corrosion test results. Quantitatively, the results of F-3 and F-4 will form the basis of comparison. F-3 and F-4 were manufactured using the same procedures as the pre-corroded specimens, and neither contained any contamination.

7.3 Pre-Corrosion Development

7.3.1 Non-Destructive Inspection Results

The three pre-corrosion specimens, CF-1, CF-2, and CF-3, underwent non-destructive evaluations after accumulating 21, 42, and 52 days of accelerated exposure in a salt fog chamber using the modified CASS solution. Inspections were performed using D-Sight, shadow moiré, and eddy current procedures described in the previous chapter. Their purpose was to track corrosion development within the splice.

NDI results at each inspection interval are shown in Figs. 7.9 to 7.12. Note that eddy current inspections were only performed during the 42- and 52-day evaluations. The images revealed several interesting features for each interval:

- 0 days:
 - all specimens have relatively flat surfaces
 - some distortion is evident at rivet 2 of the middle and lower rows of CF-1 (Fig. 7.9 a); a result of dimpling while rivetting;
- 21 days:
 - faint signs of pillowing beginning in all cases; evidence of light corrosion damage
 - definite signs of a corrosion concentration between and above rivets 6 and 7 on CF-1 (Fig. 7.9 a, Fig. 7.11 a)
 - some indications of corrosion development around the middle row in CF-3 (Fig. 7.9 c, Fig. 7.11 c);
- 42 days:
 - definite signs of corrosion throughout all specimens, ranging from light to heavy damage; all methods show primary attack around the middle rows;
 - D-Sight and shadow moiré reveal that significant corrosion pillowing has begun

- maximum out-of-plane displacements from moiré images are 0.007, 0.005, and 0.007 inch for CF-1 through CF-3, respectively
- eddy current reveals areas of greater than 10% thickness loss, especially in CF-3; areas below critical row show significant damage due to solution entry at the lap edge on the reverse side
- on the critical row, corrosion is concentrated around rivet 6 in CF-1, between rivets 3 and 5 in CF-2, and between rivets 3 and 4 in CF-3;

- 52 days:
- results are similar to 42 days: corrosion development primarily around the middle rows
 - D Sight reveals evidence of widespread corrosion pillowing, including on the driven sheet (Fig. 7.10);
 - maximum out-of-plane displacements are 0.007 inch for all specimens;
 - with additional 10 days exposure, thickness loss distributions changed moderately for CF-1 and CF-3, but more dramatically for CF-2; areas in the 7½-10% range for CF-2 at 42 days were likely closest to the 10% mark
 - corrosion concentrations on the critical row identified at 42 days have developed slightly
 - faying surfaces have developed sufficient amounts of corrosion damage, which, according to current guidelines, would warrant repair.

It is unclear why the corrosion had a tendency to concentrate around the middle row. It might simply be that the corrosive solution was most stagnant in this area, and therefore had the highest acidity. Another explanation based on research by Fontana [38] relates to this location being the most oxygen-depleted area since it is furthest from the lap edges. As suggested by Fontana, oxygen depletion produces an excess of positive charge (Al^+ ions) which is necessarily balanced by the migration of chloride ions (Cl^-) into this

area. This forms aluminum chloride, which disassociates in water into an insoluble hydroxide and a free acid. The acid then accelerates the dissolution rate of the metal.

Unfortunately, no similar data for real aircraft splices was available for comparison. The splices in the IAR library were at more advanced stages of corrosion development, thus precluding meaningful comparisons. With no evidence to the contrary, it therefore is possible that a similar distribution of corrosive attack might occur in actual aircraft splices.

7.3.2 Evaluation of Global Corrosion Damage

The material loss information was quantified using a technique discussed in the previous chapter and outlined in Appendix C. The breakdown of the thickness loss ranges as a percentage of the corrosion damage for the 42- and 52-day intervals is shown in Fig. 7.13.

At 42 days exposure, the plot reveals fairly consistent portions of the four ranges amongst all three specimens, with all levels within 8% of each other. Overall, it appears that approximately half the damage was under 5% thickness loss (i.e. in the light to moderate regime).

At 52 days exposure, the degree of damage among the specimens is still fairly consistent, although with an overall damage increase. The most dramatic change was for CF-2 which saw almost a 10% increase in the amount of area over 10% thickness loss compared to the 42-day results. Above 7½% thickness loss, CF-1 through CF-3 show 21%, 29% and 24% of their respective corrosion damage within this range. This result suggests that within the 8% of variability, CF-2 could be considered to be the most corroded and CF-1 the least corroded.

Notwithstanding the small amount of variability, the pre-corrosion procedure was successful at corroding almost half of the upper faying surface over 5% thickness loss during the 52 days of accelerated exposure.

7.3.3 Evaluation of Critical Row Corrosion Damage

The global damage results do not provide any information regarding location. Since the critical row was of most interest, the material loss information for this row was assessed on a rivet by rivet basis. The results for all three specimens are shown in Fig. 7.14, where only levels above 5% thickness loss have been included. Rivets 1 and 8 are not shown since the straps prevented the eddy current probe from obtaining all the information in these areas.

The plots reveal that, unlike the global picture, there are significant differences between specimens in both the magnitude and location of the maximum corrosion concentrations. For CF-1, the corrosion peaks around rivet 6 with 37% of its local damage above 5% thickness loss. CF-2 exhibits much worse damage at rivet 4 where damage above 5% thickness loss accounts for just over 60%. CF-3 is similar to CF-2 in location (i.e. rivet 4), yet the portion of its local damage above 5% thickness loss is less, at 48%.

Considering the whole rivet row, it appears that the assessment of specimens is identical to the one stated earlier, with CF-2 being the most corroded and CF-1 the least corroded. The effect of these concentrations on crack initiation and growth characteristics was revealed during the fatigue testing, and will be discussed in a later section.

7.3.4 Comparison of NDI with Corrosion Product Distributions

In order to further evaluate the NDI methods, the results were compared with the actual corrosion product distributions once the splices were opened. The results are shown in Fig. 7.15 in the form of comparisons between photographs of the countersunk sheet faying surfaces, with the associated eddy current and shadow moiré images. The latter images have been mirrored horizontally to correspond with the photograph (i.e. view from inside the joint). Rivet numbers have been labelled accordingly on the photograph to avoid confusion. Note that most of the corrosion products along the crack line fell out through the lap edge or through cracks during a test.

The comparisons reveal that the eddy current and the shadow moiré methods correlate quite well with the actual corrosion product distributions. In areas of large clusters of corrosion, there are similarly-sized clusters of thickness loss, and noticeable corrosion pillowing shown by separate moiré fringes. Areas containing little evidence of corrosion deposits appear as white areas in the eddy current image, and as areas with few fringes in the shadow moiré image. The corresponding D-Sight images shown in Fig. 7.9 were in general agreement with these results.

The positive correlation of the NDI methods with one another and with the actual corrosion product distributions prove that they were successful in accurately tracking corrosion development within a splice.

7.3.5 Comparison of Artificial and Natural Corrosion Damage

Although the modified CASS solution had already been shown to produce corrosion products of the same chemical species as those found in naturally aged aircraft lap splices

[79,80], there was no evidence that the physical damage was also similar. Using a lap splice from an aged Boeing 727, an investigation was conducted into this matter.

The aircraft splice, previously shown in Fig. 3.6, was removed from a Boeing 727-200, line number 590, which had accumulated 48674 cycles and 56259 hours. The joint was originally cold-bonded, but like many, had become disbanded and subsequently corroded. For the most part, the 727 splice is more heavily corroded than any of the three pre-corrosion specimens. However, there were some transitional regions between the heavy and uncorroded areas which appeared similar to ones achieved artificially. Several samples from these areas were chosen for examination.

Samples from the three pre-corrosion specimens were chosen on the basis that the specimen contained areas which included all four thickness loss ranges. An image library was then compiled of the various features within these regions.

A comparison of the physical damage achieved artificially to that developed naturally is shown in Fig. 7.16. The images are representative of the average appearance of the corrosion damage for a particular level of severity. The damage comparisons at all levels indicate a high degree of similitude, and suggest that the modified CASS procedure was successful at simulating corrosion damage occurring naturally within fuselage lap splices of aging aircraft.

7.4 Pre-Corroded Specimen Fatigue Tests

7.4.1 Crack Initiation and Growth Characteristics

The overall fatigue life data was shown in Table 1. The crack growth and crack growth rate curves for each test are shown in Fig. 7.17. A description of these graphs was given

earlier. The cracking sequences for the three specimens are shown in Figs. 7.18, 7.19, and 7.20, respectively.

Although all the tests were successful at developing and growing MSD until fully cracked, there was, however, an early problem with specimen CF-2 which warrants an explanation. Disbonding between CF-2's main sheets under the left strap eventually led to a failure of the upper strap (severed across its width) at approximately 57,000 cycles into the test. A repair, shown in Fig. 7.21, was designed, implemented, and held for the duration of the test. Stress distributions before and after repair are shown in Fig. 7.22. As shown, the disbonding initially caused a slight rise in the stress compared to CF-1. The repair lowered the stress on the left side slightly below normal since it was designed to provide added stiffness to avoid another failure. The mean stress distribution before initiation was less than 1% lower than CF-1, and after initiation, the initial stresses were only around 4% lower. On these bases, the results for CF-2 will be included in both qualitative and quantitative assessments.

The main observations and trends which the tests produced are listed and discussed below. Each point is explained in terms of similarities or differences to the baseline fatigue results. Life reduction calculations were performed with reference to the average results for specimens F-3 and F-4.

Crack Initiation

- (a) Crack initiation lives for CF-1, CF-2, and CF-3 were all shorter than either F-3 or F-4. The average reduction in initiation life with the addition of lap splice corrosion was 30%. The initiation period represented from 62 to 76% of the total fatigue life, slightly lower than the baseline results.
- (b) Cracks first initiated at rivets whose local faying surfaces revealed a maximum concentration of corrosion. With reference to Fig. 7.14, cracks at CF-1 first

initiated at rivet 6, CF-2 initiated at rivets 3 and 4, and CF-3 started at rivet 4. These rivet areas all contained the highest levels of critical row corrosion.

- (c) Crack initiation lives were inversely proportional to the severity of damage around the rivets where cracks were first detected. CF-2 initiated at 92,000 cycles where 60% of the local damage was above 5% thickness loss. CF-3 initiated 42,000 cycles later than CF-2, with 48% of its local damage above 5% thickness loss. Finally, CF-1 initiated 38,850 cycles later than CF-3, with 37% of its local damage above 5% thickness loss. The results show a clear trend to shorter initiation lives based on higher local corrosion levels. The differences in local damage created more scatter in initiation lives compared to F-3 and F-4.
- (d) Crack initiation sites were similar to the baseline results, with a slight trend for initiations more toward the rivet centreline.

Crack Growth

- (e) Early crack growth produced fairly non-uniform MSD, typically with only 2 to 4 cracks for the majority of the period up to first link-up. The only similar baseline result in this context was F-1; most produced around 6 MSD cracks.
- (f) Crack growth lives for CF-1, CF-2, and CF-3 were similar to those of F-3 and F-4; on average, they were only 3½% lower. Scatter in the crack growth lives was small, and of the same magnitude as that between F-3 and F-4 (scatter factor ~ 1.2).
- (g) The small amounts of MSD in the corroded splices would normally imply longer growth periods. This suggests the existence of other mechanisms which effectively increased the crack growth rates. The most probable mechanisms are local and global material thinning, and corrosion pillowing causing a general stress increase in the splice skins. The effect of local thinning might be limited as the eddy current scans revealed a majority of light corrosion along the typical crack path. There were, however, significant regions of corrosive attack in nearby locations.

- (h) CF-1 exhibited the least amount of MSD, yet it had the shortest crack growth life. Conversely, CF-3 had the most MSD before link-up, yet it had the longest crack growth life. The results were counter-intuitive but are likely attributable to fatigue scatter. A portion of the difference might be related to the presence of worse damage along the crack path in CF-1, or possibly the absence of overlapped cracks in CF-1. As was revealed in the baseline tests, peak growth rates for a direct link-up of the crack tips can reach an order of magnitude higher than with an overlap. This trend was further revealed in the growth rates for the 5R/6L link-up in CF-1 and the 4R/5L overlap in CF-3.
- (i) The average growth rate for the lead crack once it had spanned most of the rivets reached a maximum of 0.0036 inch/cycle, 40% higher than the corresponding baseline result.
- (j) Link-up close to the rivet head produced quite erratic crack shapes. Evidence of this is shown in Fig. 7.18 (j) to (m), and Fig. 7.20 (i). The wave-like crack path at link-up suggests the existence of a complex stress state, due in part to the close proximity of local rivet effects. This occurred only once during the baseline tests, in specimen F-5.

The aggregate crack growth plot for this series of tests is shown in Fig. 7.23. Results for F-3 and F-4 have been added for comparison. The plot reveals some differences:

- (a) There appears to be an acceleration in the growth rates beginning at an aggregate crack length of only 1½ inches, just before the first link-ups. The growth rate in each case jumped over 6× then remained fairly constant until an aggregate crack length of around 2½ inches. The average growth rate for F-3 and F-4 in this range was almost 5× slower. The change to faster growth rates enabled CF-2 and CF-3 to have similar crack growth periods to F-4 and F-3, respectively. Possible reasons for the faster growth rates were discussed earlier.

- (b) All first link-ups occurred between aggregate crack lengths of $1\frac{1}{2}$ inches and $2\frac{1}{4}$ inches. This range is earlier than the baseline result due to the relatively smaller MSD development within the corroded splices.

The reduction in total fatigue life achieved through pre-corrosion was on average 23%. This is of the same order as many of the investigations reviewed in Ch. 4. Unfortunately, no other similar tests on lap splices were found to verify the magnitude of any of the life reduction percentages.

7.4.2 Strain Gauge Results

Some of the strain gauge results for the uncracked state were shown for CF-1 and CF-2 in Fig. 7.22. The average distribution for CF-1 and CF-3 is shown in Fig. 7.24, compared with the distribution for F-3. As shown, hoop stress at the nominal position for the corroded specimen were marginally higher than that of the uncorroded specimen. At the middle position, the stress was approximately 14.8 ksi, which is 3.2% higher than F-3. As all other test variables were identical, the increase is likely attributable to the overall material loss on the primary faying surfaces as shown in Fig. 7.12. Bending stress at the nominal position did not reveal any significant differences.

An interesting result was obtained with the gauges on specimen CF-1. At approximately 120,000 cycles, the stress level around rivet 6 began to drop. This continued until cracks became visible on either side of rivet 6 at 168,850 cycles. The situation is illustrated in Fig. 7.25. As shown, the decrease in stress is accompanied by a similar increase around rivet 4. The event suggested that cracks were growing sub-surface, potentially with a large degree of tunnelling since no visible signs on the outer surface were evident. For CF-2 and CF-3, small decreases in the gauges near initiation were also noticed, although only 12,000 cycles before visible detection. The results suggest that with gauges closer

to the rivets, it should be possible to get a more accurate estimation of the growth period under the rivet head. It is unknown if this method would be effective on uncorroded splices.

The redistribution of stress during the cracking period is shown in Fig. 7.26 for each specimen. For CF-1 and CF-2, the last 5% of the test is shown; CF-3 has the last 3½% shown. No significant differences to the baseline results for stress redistribution were evident. Both show the characteristic fallout in stress in a very short period of time. Note that for CF-1 and CF-2, the redistribution is uneven due to MSD development in non-central locations.

7.4.3 Summary

Like the baseline tests, the pre-corrosion fatigue tests were also able to develop and grow MSD with similar realism under laboratory conditions. MSD development, however, was less uniform, limited to one or two rivet hole areas with high concentrations of corrosion damage. Higher local corrosion levels led to shorter initiation lives, and ultimately, shorter fatigue lives. On average, initiation life was reduced by 30%, crack growth life was reduced by 3½%, and overall, total fatigue life was reduced by 23%.

The nominal stress distribution was slightly higher than in an uncorroded splice, a result of material thinning on the splice faying surfaces. Stress increases caused higher crack growth rates, and enabled the corroded splices with little MSD to have similar crack growth periods to the uncorroded splices which had more MSD.

Evidence of crack growth under the rivet head was revealed by the strain gauges for the first MSD cracks. This occurred over 50,000 cycles before initiation in CF-1, and over 12,000 cycles before initiation in CF-2 and CF-3. No similar signs were evident during

the baseline tests. Possible explanations for these occurrences were investigated during the teardown and analysis.

The overall results provide confidence that the coupon specimen is adaptable to corrosion studies in combination with MSD research.

7.5 Specimen Teardown and Analysis

Following the conclusion of the testing, specimens F-2, F-3, and all three pre-corroded specimens were disassembled and examined using visual, microscopic and fractographic techniques. Some of the results for the visual and microscopic investigations were presented and discussed earlier. The remainder of the findings are presented in this section. Objectives and procedures for the investigations were outlined in previous chapters.

7.5.1 Visual and Microscopic Analyses

Both faying surfaces of F-2 revealed distinct evidence of a spread of oil throughout the joint. It appeared that the contamination originated in the upper row of rivets and, through gravity, flowed to the critical row. No oil contamination was found in any of the other opened joints.

Fretting damage was evident in all specimens as areas of black oxide deposits. These areas would typically be located around the hole edges extending to a distance around 0.040 inch. The uncorroded splices revealed quite distinct regions of fretting severity, ranging from light to heavy. At the critical row, a typical representation of these regions is shown in Fig. 7.27. The cracks generally nucleated close to 45° from the horizontal.

which coincides with the areas of maximum fretting damage. This indicates that the peak stresses were located within these areas. Finite element analysis of lap joint structure conducted by IAR [94] also reveals this location as containing the peak stresses. Evidence of light fretting damage was also noticed on many of the countersunk surfaces.

The corrosion products were generally white to light grey in colour with a crystalline form. The deposits comprised a mixture of loose and fixed product masses. A close-up view of a splice before and after removal of the loose products is shown in Fig. 7.28. A micrograph showing the typical appearance of the corrosion products is shown in Fig. 7.29. The mud-cracked pattern of the products is similar to crystalline formations which have been observed on naturally corroded specimens. The products shown are the monohydrate and trihydrate forms of aluminum oxide [79].

A metallographic study to determine pit depth distributions, or to investigate intergranular attack was beyond the scope of this project. It nevertheless should be completed to provide further verification that the damage developed artificially is representative of that developed naturally.

7.5.2 Fractographic Analysis

7.5.2.1 Crack Nucleation Sites

In examining over 50 fracture surfaces, it was determined that cracks predominantly nucleate on the faying surface within 0.005 to 0.015 inch of the countersunk knife edge. This was evident for both corroded and uncorroded splices (Fig. 7.30). The precise location of crack nucleation could not be determined due to significant fretting damage which removed much of the detail at the faying surface, such as corrosion pits. However,

the existence of tear ridges and other secondary markings on all of the surfaces generally could be followed back to the area where cracks first nucleated.

A notable exception was the crack at 4R on CF-3 (Fig. 7.31) which started 0.010 inch from the knife edge, but on the countersunk surface. No similar instances were found.

7.5.2.2 Crack Growth Under the Rivet Head

Crack propagation under the rivet head produced several interesting features:

- (a) In most cases, there was evidence of a coalescence of two to three crack fronts, creating large ridges and plateaus (Fig. 7.32). The sub-surface link-ups occurred under the rivet head, and eventually produced a dominant crack with a single front before penetrating the upper surface. In some cases, small portions of the knife edge would become completely detached due to the crack front link-ups (Fig. 7.33 c).
- (b) In many cases, however, other small cracks along the countersink did not link-up with other cracks, and were arrested. Most countersinks revealed between 1 to 3 arrested cracks on either side of the primary fracture, with a variety of lengths (Fig. 7.33). The longest arrested crack was observed at 5R on F-3; it had grown nearly the full length of the countersink to the point where it almost became visible. No arrested cracks were found on any of F-2's countersinks since insufficient fatigue life had accumulated to develop such cracks.
- (c) The plane of most cracks tended to proceed in a direction similar to the behaviour of early visible growth, i.e., toward the rivet centreline. This is again indicative of the direction of the maximum principal stress.

- (d) Crack fronts generally progressed radially outward from the sites of nucleation. Fig. 7.34 shows the typical markings on the fracture surfaces which are generally normal to the crack front. From this, it is estimated that the average fracture progressed from the knife edge region to the top of the countersink, a distance which varied between 0.070 and 0.110 inch.
- (e) Fatigue striations existed, but were not continuous, and could not be resolved within 0.015 inch of the nucleation site. Instead, they were limited to only the faces of favourably-oriented grains, with a typical appearance as shown in Fig. 7.35. A striation spacing of 0.2μ (7.9×10^{-6} inch) was common under most rivet heads. Close to the nucleation site, spacings between 0.08μ (3.1×10^{-6} inch) and 0.13μ (5.1×10^{-6} inch) were found for a few cases of initial MSD cracks. The SEM could not resolve striations any finer than a spacing of 0.08μ , leaving an area around the nucleation site where no growth rates could be estimated. Assuming a spacing of 0.1μ (3.9×10^{-6} inch) throughout the area under the rivet head, a crack would need to grow between 18,000 and 28,000 cycles before penetrating the upper surface, for crack lengths of 0.070 and 0.110 inch, respectively. This provides a rough estimate of the crack growth period under the rivet head. Perhaps the most accurate estimation was provided by the strain gauges on the corroded splices which suggested minimum growth periods between 12,000 and 50,000 cycles. More accurate estimations could be achieved if fracture surfaces were examined immediately after cracks first became visible. This would minimize further fretting damage and might reveal more information than was evident after fully cracking the specimen.

No evidence was found to suggest that the addition of corrosion produced any change in the fracture surfaces beneath the rivet head. It therefore could not be determined why the strain gauges in the corroded specimens revealed early signs of cracking.

7.5.2.3 Crack Growth Beyond the Rivet Head

Once a crack became visible at the upper surface, the fracture surface revealed some significant changes:

- (a) Fatigue striations became clearly defined in the soft, ductile clad layers (Fig. 7.36), beginning at a distance of approximately 0.030 inch from the site of initiation. Notice the extensive secondary cracking parallel to the striations, a feature common with fatigue of near pure aluminum. Striation spacings in the clad layer ranged from 0.5μ (2.0×10^{-5} inch) to almost 25μ (9.8×10^{-4} inch) near link-up. Compared to experimental crack growth rates, it appears that striations in the cladding overestimate the growth rates for small cracks, and underestimate the growth rates at link-up. Striations on core alloy grains were too varied to provide a good estimation of the growth rates.
- (b) The transition from tensile cracking to shear cracking is shown in Fig. 7.37, and is characterized by the formation of shear lips as the fracture plane rotates to form a 45° angle with the plane of the sheet. For the initial MSD cracks, the transition region extended to approximately 0.18 inch from the initiation site. For subsequent cracks, the transition region was halved in size due to the increased stress amplitude.
- (c) Differences existed in the general appearance of the fracture surface in the tensile and shear regions (Fig. 7.38). Although both reveal typical transgranular crack growth in aluminum alloys (e.g. voids around inclusions or second phase particles, small local ridges and plateaus), in the shear region, the crack appears to propagate on multiple plateaus that are at different elevations with respect to one another. This is characteristic of fracture at 45° to the loading direction.

- (d) With a direct link-up, the surface changes to a combination of large ridges, plateaus and smooth walls (Fig. 7.39). The plateaus were quite flat and featureless, indicative of fast fracture (Fig. 7.40). Most of these regions were developed in only a few cycles, and were preceded by significant ligament yielding. The same features were not evident for overlapped cracks since the same high growth rates were not attained.

Similar to growth under the rivet head, no evidence was found to suggest that corrosion produced any changes in the fracture surfaces beyond the rivet head.

7.6 Summary Review and Discussion

This chapter presented the results of an experimental program to investigate the effects of pre-corrosion on MSD crack initiation and growth characteristics in an unbonded and rivetted lap splice specimen.

A series of fatigue tests was first performed on uncorroded splices to confirm the validity of the specimen and establish a set of baseline results. The tests were successful at developing and growing MSD in a realistic manner until a lead crack had spanned all the rivets on the critical row. The crack patterns revealed a mixture of relatively uniform and non-uniform MSD, typical of published cases. Nominal stress distributions were fairly consistent between specimens, and preliminary results from critical row strain gauges reveal that secondary bending in the specimen is generally representative of that found in real aircraft splices. Overall, the results confirmed the validity of the specimen for use in MSD research.

Qualitatively, all baseline results were directly compared to the corrosion test results. Quantitative assessments, however, were limited to the results of F-3 and F-4 since they

were manufactured using the same procedures as the pre-corroded specimens, and neither contained any contamination. F-1 employed a different riveting procedure than other specimens, and oil contamination was discovered during the tests of F-2 and F-5. These occurrences are believed to have affected crack initiation and growth lives, and thus were not included in numerical comparisons.

The 52 days of accelerated corrosion using the modified CASS test were successful at generating a variety of corrosion levels, ranging from light to severe. Overall, almost half of the upper faying surface on all specimens had damage above 5% thickness loss. At the critical row, the damage was primarily light to moderate corrosion, although there were heavier concentrations around some rivets. The damage was compared microscopically with damage from a corroded Boeing 727 splice and found to have a high degree of similitude. NDI methods correlated well relative to one another and to the actual corrosion product distributions.

The pre-corroded specimen fatigue tests were also successful in generating and growing MSD until failure. However, compared to baseline data, initiation life was reduced by an average 30%, growth life was reduced by an average 3½%, and overall, total fatigue life was reduced by an average 23%. MSD was less uniform, and occurred primarily at rivet holes which exhibited high concentrations of local corrosion damage. Initiation life was found to be quite sensitive to the magnitude of the maximum corrosion concentration on the critical row. Crack growth rates were generally higher due, in part, to an increase of the hoop stress caused by material thinning. This allowed the corroded splices with little MSD to have similar crack growth periods to the uncorroded splices which had more MSD. The strain gauges revealed signs of cracking before visible detection for the first MSD cracks, beginning at 50,000 cycles before initiation in CF-1, and 12,000 cycles before initiation in CF-2 and CF-3.

Subsequent teardown and analysis of several specimens revealed no significant differences in the fretting damage or the features of fracture between corroded and uncorroded splices. Most cracks nucleated on the faying surface between 0.005 and 0.015 inch from the countersunk knife edge, and propagated between 0.070 and 0.110 inch before penetrating the upper surface. Due to fretting damage, it could not be determined if cracks in the corroded splices began at the bases of corrosion pits. Fatigue striations provided a rough estimate of 18,000 to 28,000 cycles for the growth period under the rivet head.

The trend to a large reduction in fatigue life due to lap splice corrosion could have serious implications for the aircraft industry. As a minimum, estimation of the potential onset of MSD in lap splices should consider the effects of corrosion. The problem should also be reflected in other areas, such as design and damage tolerance analyses, and scheduling of maintenance and inspection intervals. A maximum response would be to amend current airframe certification requirements to include consideration of the interaction of corrosion and MSD in lap splices as experienced in airline service. However, before any serious recommendations can be made, the observed trends need to be verified through more testing. Nevertheless, the experimental program was successful in establishing a basis which provided some much needed insight into the interaction of MSD and corrosion in fuselage lap splices.

8. CONCLUSIONS

8.1 Conclusions

The objectives of this thesis were to provide an overview of the MSD and corrosion problems in fuselage lap splices, and to develop and initiate a research project to investigate the interaction of the two damage mechanisms.

A multi-disciplinary review was conducted to determine how corrosion and fatigue might interact under various circumstances in actual airframes, and what methods are best suited to simulate these interactions. The review concluded the following:

1. When considering the atmospheric profile against typical commercial aircraft missions, the processes of corrosion and fatigue in fuselage lap splices occur sequentially. Corrosion and fatigue acting concurrently thus does not exist under these circumstances.
2. The interaction of corrosion and fatigue in fuselage lap splices is best simulated using either a pre-corrosion test or an alternating fatigue and corrosion test. No evidence of successful implementation of these approaches in lap splices was found. A review of research using structurally simpler specimens concluded that the pre-corrosion approach was the logical starting point.

An initial set of fatigue tests was conducted on five uncorroded splices to confirm the validity of the specimen and establish a set of baseline results. The results of these tests concluded the following:

3. The baseline fatigue tests were successful at developing and growing MSD in a realistic manner until a lead crack had spanned all the rivets on the critical row. The crack patterns revealed a mixture of relatively uniform and non-uniform MSD, typical of published cases. Stress distributions approximated on-aircraft conditions well, and preliminary results for secondary bending indicate that it is generally representative of that found in real aircraft splices. Overall, the results confirmed the validity of the specimen for use in MSD research.
4. Fatigue life was found to be quite sensitive to small changes in the variables. Oil in one of the splices acted to lower fatigue life, whereas a different rivetting procedure used with one specimen increased fatigue life. Scatter of the total fatigue life predominantly reflected the variability in the crack initiation period.

Three specimens were submitted to the process of pre-corrosion development. With regard to this process, the following conclusions are noted:

5. The 52 days of accelerated corrosion using the modified CASS test were successful at reproducing damage representative of that occurring within naturally corroded lap splices. Compared with damage in a Boeing 727 lap splice, the artificial damage was found to have a high degree of similitude at various levels of severity. Previous work had shown that the corrosion products developed with the modified CASS solution were of the same chemical species as those found in naturally aged lap splices.
6. D-Sight, shadow moiré and eddy current scans provided an accurate status of corrosion development within each splice. All methods correlated well relative to one another and to the actual corrosion product distributions.
7. An image processing technique to quantify material loss information was developed and successfully used in the evaluation of both the global and critical row corrosion

damage. It proved to be an effective method of representing lap splice corrosion damage, with potential applicability to identifying configurations critical to fatigue.

The three pre-corroded specimens were fatigue tested until failure. The tests were successful in identifying some initial trends which occurred with the addition of corrosion into the splice. The major conclusions were:

8. The pre-corrosion fatigue tests were successful in generating and growing MSD of a non-uniform nature until fully cracked. This provided evidence that the specimen is adaptable to corrosion studies in combination with MSD research.
9. The addition of pre-corrosion within the lap splices revealed a marked effect on crack initiation life and a small effect on crack growth life. On average, when compared with the two non-corroded splices of identical manufacturing, pre-corrosion produced a reduction in initiation life of 30%, a reduction in growth life of 3½%, and an overall reduction in total fatigue life of 23%.
10. Crack initiation life was found to be quite sensitive to the magnitude of the maximum corrosion concentration in the critical row. A clear trend to shorter initiation lives based on higher local corrosion levels was present. All first cracks initiated from the rivet holes which exhibited the highest concentrations of local corrosion damage regardless of location.
11. Pre-corrosion increased the hoop stress within the splice (primarily due to material thinning) and caused less MSD. The resulting higher crack growth rates produced crack growth lives in the corroded specimens comparable to those in the uncorroded splices with more MSD.

Following the fatigue testing, several specimens were broken down and examined. In addition to some conclusions stated earlier, the exercise produced a few other significant results:

12. Crack nucleation sites for both corroded and uncorroded splices occurred predominantly on the faying surface within 0.005 to 0.015 inch of the countersunk knife edge. These sites generally coincided with regions of maximum fretting damage.
13. Crack growth under the rivet head initially involved a complex coalescence of crack fronts from multiple cracks to eventually form a single dominant crack. When this occurred, other active cracks were arrested.
14. The crack growth period under the rivet head could not be accurately measured using fatigue striation counting. However, a rough estimate of between 18,000 and 28,000 cycles was obtained. Some strain gauges revealed signs of cracking for the first cracks between 12,000 and 50,000 cycles before visible detection.

8.2 Recommendations for Future Work

The present thesis documents the initial effort in a much larger project to understand the interaction of MSD and corrosion in fuselage lap splices. Its intent was to develop viable simulation techniques, and to establish a basis for further research. To this end, the following recommendations are noted:

1. Further baseline and pre-corrosion fatigue testing needs to be completed in order to verify the trends established within this thesis. It would also be beneficial to know how the situation changes with shorter and longer exposures to accelerated corrosion. This work is already in progress.

2. Once the effects of pre-corrosion are well known and documented, it would be prudent to investigate the effects of alternating fatigue and corrosion on MSD initiation and growth characteristics. Preliminary work with alternate fatigue/corrosion testing is in progress.
3. To more accurately model actual aircraft lap splices, a small doubler should be added at the middle row position on the driven sheet to represent a stringer. The presence of a stringer in corroded splices prevents much of the corrosion pillowing on the driven sheet, causing increased pillowing on the countersunk sheet.
4. A metallographic study to determine pit depth distributions and the degree of intergranular attack would provide further verification that the accelerated corrosion damage is representative of damage occurring in naturally aged aircraft splices.
5. Investigation into more accurate NDI methods to determine material loss information in lap splices would be beneficial, and could be used to confirm the eddy current data. This work is in progress.
6. With the early signs of cracking revealed by some strain gauges, it may be worth further investigating their usefulness in detecting crack growth under the rivet head. This could be accomplished by placing gauges closer to the rivet heads. This technique is currently under evaluation.

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(cont'd)

(a) 48,922 cycles: First cracks at 4L, 4R and 5L.

(b) 62,500 cycles: New cracks at 2L/R, 3R and 5R.

(c) 65,500 cycles: Close-up view of MSD cracks at rivets 2 through 5.

(d) 74,500 cycles: Link-up of 4R and 5L is near. New crack at 6L.

(e) 76,750 cycles: 4R and 5L have linked. New crack starting at 3L.

Figure 7.2 Images of MSD initiation and growth for specimen F-2.

(f) 78,500 cycles: 3R and 4L have linked.

(g) 79,250 cycles: 2R/3L and 5R/6L have linked. New crack at 6L.

(h) 80,050 cycles: Lead crack tips at 2L and 6R are growing very fast.

(i) 80,250 cycles: 2L grows into rivet 1. Small crack emerges at 7L.

(j) 80,450 cycles:
6R grows into rivet 7 before linking
with 7L. New crack at 7R emerges.

(k) 81,150 cycles: Lead crack spans all rivets.

Figure 7.2 *Images of MSD initiation and growth for specimen F-2.*
(cont'd)

(a) 197,000 cycles: Cracks at 4L/R, 5L and 6L/R.

(b) 217,000 cycles: Cracks on all sides of rivets 4, 5, and 6.

(c) 228,750 cycles: 4R and 5L have overlapped within the strain gauge. 5R and 6L have linked. Small cracks at 3R/L.

(d) 229,850 cycles: 3R and 4L have overlapped. 6R and 7L have linked. Lead crack tip at 3L growing rapidly.

Figure 7.3 Images of MSD initiation and growth for specimen F-3.

(e) 230,250 cycles: 2R and 3L have linked. Slow growth at 7R.

(f) 230,650 cycles: 2L has grown into gauge. 7R almost at rivet 8.

(g) 231,050 cycles: Lead crack spans all rivets.

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(i) Image of overlap of 3R and 4L at full load after test completion.

Figure 7.3 *Images of MSD initiation and growth for specimen F-3.*
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(a) 220,000 cycles: Cracks at 3L and 4L/R.

(b) 224,000 cycles: Close-up view of MSD cracks at rivets 3, 4, and 5.

(c) 254,000 cycles: 4R and 5L headed for an overlap. 3R and 4L headed for a direct link-up. Some indication of a crack at 2L.

(d) 257,000 cycles: Definite crack at 2L. 3R and 4L and linked. 4R and 5L have overlapped and their growth has slowed considerably. New cracks at 6L/R.

Figure 7.4 Images of MSD initiation and growth for specimen F-4.

(e) 258,600 cycles: 5R and 6L are starting to overlap. New crack at 2R.

(f) 259,200 cycles: 2R and 3L have linked. 5R and 6L have overlapped.

(g) 259,800 cycles: Lead crack tip at 6R grew rapidly into rivet 7 and emerged at 7R.

(h) 260,200 cycles: Lead crack tip at 2L grew rapidly into rivet 1 and stopped.

(i) 260,400 cycles: Lead crack spans all rivets.

(j) Note that eventually 5R grew into rivet 6 causing the original crack at 6L to close. Note also that this did not occur with 4R.

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(cont'd)

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(b) Specimen CF-2 **0 days** **21 days** **42 days** **52 days**

Figure 7.9 *D-Sight images showing corrosion development in (b) Specimen CF-2.*
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(cont'd) *Orientation as tested (critical row at bottom).*

Figure 7.10 *D-Sight images showing corrosion development on driven sheet of (a) Specimen CF-1 (b) Specimen CF-2.*

(a) Specimen CF-1

0 days

21 days

42 days

52 days

Figure 7.11 *Shadow moiré images showing corrosion development in (a) Specimen CF-1. Orientation as tested (critical row at bottom)*

Figure 7.11 *Shadow moiré images showing corrosion development in (b) Specimen CF-2. Orientation as tested (critical row at bottom).*

(c) Specimen CF-3

0 days

21 days

42 days

52 days

Figure 7.11 *Shadow moiré images showing corrosion development in (c) Specimen CF-3. Orientation as tested (critical row at bottom).*
(cont'd)

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Figure 7.12 *Eddy current images showing material loss distributions after (b) 52 days of accelerated corrosion. Dotted lines indicate rivet row centerlines.*
(cont'd)

Figure 7.13 Breakdown of material loss information as a percentage of the corrosion damage at (a) 42 days and (b) 52 days of accelerated corrosion

Figure 7.14 Breakdown of material loss information along the critical row as a percentage of the local corrosion damage

(a) Photograph

(b) Eddy Current

Thickness Loss: Drk(5-7½%), Lt.(7½-10%), Med(>10%)

(c) Shadow Moiré

Figure 7.15 Comparison between (a) photograph of appearance and distribution of corrosion products on the countersunk sheet of specimen CF-1, with (b) eddy current scan, and (c) shadow moiré image of the same area.

(a) Photograph

(b) Eddy Current

Thickness Loss: Drk(5-7½%), Lt.(7½-10%), Med(>10%)

(c) Shadow Moiré

Figure 7.15 *Comparison between (a) photograph of appearance and distribution of corrosion products on the countersunk sheet of specimen C'F-2, with (b) eddy current scan, and (c) shadow moiré image of the same area.*
(cont'd)

(a) Photograph

(b) Eddy Current

Thickness Loss: Drk(5-7½%), Lt.(7½-10%), Med(>10%)

(c) Shadow Moiré

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Figure 7.16 Comparison of natural and artificial corrosion. (e) Severe corrosion - close-up view (f) Severe corrosion - wider view.

Figure 7.17 Plots of crack growth and crack growth rates for (a) Specimen CF-1.

Figure 7.17 Plots of crack growth and crack growth rates for (b) Specimen CF-2. (cont'd)

Figure 7.17 Plots of crack growth and crack growth rates for (c) Specimen CF-3. (cont'd)

(a) 168,850 cycles: First cracks at both sides of rivet 6.

(b) 202,000 cycles: Cracks at 6L/R continue to grow. No new cracks.

(c) 209,700 cycles: Crack initiating at 5R to meet 6L. Nothing at rivet 7.

(d) 218,000 cycles: 5R and 6L have linked and started a crack at 5L. Crack at 7L appeared just before 6R grew into rivet 7. This initiated a crack at 7R. Image before 5R/6L link-up shown on next page.

(e) 219,100 cycles: Crack at 4R appeared just before 5L grew into rivet 4. This initiated a crack at 4L. Lead crack now spans over 3".

(f) 220,200 cycles: Lead crack approaches rivets 3 and 8. No signs of cracks at 3R or 8L.

Figure 7.18 Images of MSD initiation and growth for specimen CF-1.

(g) 220,650 cycles: Lead crack has grown through rivet 3 and into rivet 8.

(h) 221,200 cycles: Lead crack has grown through rivet 2. Some distortion at 8R indicating that a crack is about to emerge.

(i) 221,650 cycles: Lead crack spans all rivets, including through rivet 8.

(j) 210,450 cycles: Approaching link-up of 5R/6L.

(k) + 112 cycles: Plastic distortion evident between crack tips.

(l) + 8 cycles: Crack tips begin curling toward each other.

(m) + 4 cycles: Crack tips have linked up.

Figure 7.18 *Images of MSD initiation and growth for specimen Cr-1.*
(cont'd)

(a) 104,000 cycles: First cracks at 3L and 4L.

(b) 128,000 cycles: New cracks at 3R and 4R.

(c) 141,000 cycles: 3R and 4L close to link-up. New crack emerges at 2R.

(d) 143,500 cycles: 3R and 4L have linked. Signs of a crack starting at 2L.

(e) 144,550 cycles: 2R and 3L have linked. 2L has now fully emerged.

(f) 145,600 cycles: 4R grew into rivet 5 but not before a small crack emerged at 5L causing a small overlap. It then emerged at 5R.

(g) 146,600 cycles: Small crack emerged at 1R to link with lead crack.

Figure 7.19 Images of MSD initiation and growth for specimen CF-2.

(h) 147,400 cycles: 5R linked with 6L and emerged at 6R. New crack at 1L.

(i) 147,800 cycles: Lead crack has just grown up to the left strap, and through rivet 7.

(j) 148,310 cycles: Lead crack spans all rivets.

Overlapping crack sequence for 3R and 4L.

141,250 cycles

+ 85 cycles

+ 20 cycles

+ 15 cycles

+ 20 cycles

+ 15 cycles

+ 95 cycles

+ 500 cycles

+ 2000 cycles

Figure 7.19 Images of MSD initiation and growth for specimen CF-2.
(cont'd)

Figure 7.20 Images of MSD initiation and growth for specimen CF-3.

(d) 195,120 cycles: 3R and 4L have linked. New crack at 6L (masked by corrosion damage).

(e) 196,080 cycles: 3L has grown into rivet 2. 5R and 6L have overlapped.

(f) 196,350 cycles: Lead crack has grown through rivet 7.

(g) 196,750 cycles: Lead crack has grown into rivet 1 and is approaching rivet 8.

(h) 196,850 cycles: Lead crack spans all rivets.

(i) Features of a link-up close to the rivet head

196,080 cycles

194,300 cycles

Figure 7.20 Images of MSD initiation and growth for specimen CF-3.
(cont'd)

Figure 7.21 Repair for broken strap in specimen CF-2.

Figure 7.22 Nominal stress distribution in specimen CF-2 at various stages. Stress distribution for CF-1 added for comparison.

Figure 7.23 Aggregate crack growth plot for pre-corroded specimen fatigue tests.

Figure 7.24 Nominal stress distributions in CF-1 and CF-3 compared to F-3

Figure 7.25 Early sign of stress decrease before crack initiation in specimen CF-1

Figure 7.26 Variation of measured hoop stress across the specimen as cracks advanced in (a) Specimen CF-1 (b) Specimen CF-2.

Figure 7.26 Variation of measured hoop stress across the specimen as cracks advanced in (c) Specimen CF-3.
(cont'd)

Figure 7.27 Regions of fretting damage observed at a typical hole on the critical row of an uncorroded splice. Crack paths shown are also typical showing initiation at the hole edge near 45° from horizontal positions.

(a) with loose corrosion products

(b) without loose corrosion products

Figure 7.28 Photographs of corrosion products (a) before removal of loose deposits, and (b) after removal of loose deposits on specimen CF-3.

Figure 7.29 *Micrograph showing the typical appearance of the corrosion products.*

(a) Crack at 4L-a, Specimen F-2

(b) Crack at 6L-a, Specimen F-3

Figure 7.30 *Fractographs showing typical site of crack nucleation. (4L-a signifies the crack to the left of rivet 4 - upper half)*

Figure 7.30 *Fractographs showing typical site of crack nucleation.*
(cont'd)

Figure 7.31 *Fractograph showing crack nucleation on the countersunk surface.*

Figure 7.32 *Coalescence of crack fronts under the rivet head.*

Figure 7.33 *Examples of arrested cracks along the countersinks.*

Figure 7.34 *Determination of crack front shape from tear ridges on the fracture surface.*

Figure 7.35 *Typical appearance of striations under the rivet head.*

Figure 7.36 *Striations in the ductile clad layer. Note the secondary cracking in (b). (cont'd)*

Figure 7.37 *Transition from tensile to shear mode of crack propagation.*

Figure 7.38 *View of fracture surface within different regions of crack propagation.*

Figure 7.39 *Appearance of fracture surface at link-up of 3R/4L in specimen CF-3.*

Figure 7.40 *Close-up view of fracture surface just before link-up.*

APPENDIX A**MANUFACTURING PROCEDURE FOR MSD SPECIMEN**

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WARNING

The preparation and application of isopropyl alcohol, methyl ethyl ketone (MEK), Pasa-Jell, BR127 and PR1431 must be performed in a fume hood or other well-ventilated area, and only while wearing impervious protective clothing, including rubber gloves and splash-proof goggles. A face mask is also required when working with Pasa-Jell, since it is highly corrosive. Depending on the airborne concentrations of the chemicals, respirators may also be required. Check the appropriate safety regulations.

Notes: (1) All dimensions are in inches unless otherwise stated.

(2) The following special equipment is required:

- an accurate sheet metal guillotine or laser cutting tool
- a drilling and countersinking jig
- an oil-free grit-blasting machine and fine aluminum oxide grit
- compressed nitrogen supply (approx. 3 bottles per specimen)
- stainless steel racks for holding components during surface preparation
- a spray gun with agitator
- an adjustable, orbital riveting machine
- vacuum pump and hoses
- vacuum-bagging supplies as used for bonding and curing of metal and composites
- an autoclave

(3) A set of reference drawings for the MSD specimen is given in Fig. A.1

1. Procure sheets (standard size is 8ft x 4ft) of clad 2024-T3 aluminum alloy QQ-A-250/5, complete with certificate of compliance. Cut components as specified in Table A1 for one specimen from the sheets.

Item	Drawing	Description	Qty.
1	GE306-1	Skin Pieces Matched Pair	2
2	GE307-1	Short Strap	4
3	GE308-1	Long Strap	4
4	GE309-1	Doubler	4

Table A1 - Components required for one specimen.

The required tolerance can be achieved with an accurate, powered guillotine, but it is acceptable to use other techniques, such as laser cutting or milling (with several sheets clamped together).

Note: To avoid marking the aluminum surface, place Kraft paper underneath the sheet and clean rubber pads on the guillotine clamping fixtures.

2. Inspect the components according to the drawings listed in Table A1. If there are any discrepancies, record them and notify the customer immediately.
3. Drill and countersink the matched pair of skin pieces being careful to choose the most scratch-free area of the two sheets to be the countersunk area. The drilling/countersinking tasks should be completed using an NC machine and the appropriate program but alternatively can be completed using a manual method described as follows:
 - a. Taking care not to scuff the surface of the clad aluminum sheet, mark the position of rivet hole centres on one skin piece using a pencil with a very fine point.
 - b. Clamp together the 2 skin pieces (Item 1) with an overlap of 3.00. While setting the clamps, hold one edge of the subassembly against a straight edge (± 0.001). Check that the clamped subassembly is straight within 0.005. A

drilling/countersinking jig incorporating a straight edge and clamps is available to facilitate this process.

- c. Centre-punch the rivet positions. While punching, place the sheet on a smooth, flat metal surface to avoid dimpling it. Using a drill press, centre-drill the holes, and complete them using a #21 drill (required hole diameter 0.160 min, 0.162 max). Remove the clamps, deburr all holes, and remove all swarf.
 - d. Clamp the skin pieces again as described above, and countersink (100°) the holes such that the rivet heads will be slightly proud of the skin (0.005 max). Use a drill press and an adjustable countersinking tool; first check the setting of the tool by drilling and countersinking a test coupon. Bring the reference ring of the tool into contact with the surface of the specimen before engaging the drive, in order to avoid scuffing the specimen. Remove all swarf from the tool in between each countersink operation by blowing with compressed air. Remove the clamps and clean away any swarf.
4. If hidden strain gauges are to be installed, refer to the strain gauge installation drawing GE310-1 and machine the required grooves in the hidden face of the non-countersunk sheet.
 5. Inspect all holes and grooves to drawings GE304-1 and GE310-1 respectively. If there are any discrepancies, record them and notify the customer immediately.
 6. Prepare all items in Table A1 for bonding with FM73 as follows:
 - a. Clean with isopropyl alcohol, until a KimWipe tissue lightly dampened with alcohol remains clean after vigorous rubbing.

Notes: (1) Steps a. and b. may be replaced by vapour degreasing using trichloroethylene.

(2) Hereafter, components must be handled so as to avoid any possibility of contamination of the surfaces to be bonded. To aid in this effort, replace rubber gloves regularly.

(3) *Use the stainless steel racks for holding components as necessary during steps a. to g*

- b. Repeat step (a) using Methyl Ethyl Ketone (MEK). Use a single stroke motion - i.e. no back and forth rubbing which will tend to grind in any remaining surface contaminants.
- c. Apply masking tape to each side of the two main sheets over the area defined in Fig. A.2. This area will not be bonded.
- d. Grit-blast both sides of all components to a matte finish using fresh fine (#220) aluminum oxide grit and a nitrogen propellant. Purge the machine of used grit before starting and refresh the grit when its effect diminishes. Also, clean the base and back of the grit-blasting chamber using isopropyl alcohol. Note that grit-blasting may result in distortion of the thin sheet and it may be advisable to place a flat metal plate behind the part to minimize the warping. If necessary, restore the flatness by further grit-blasting of the concave surface. Guides should be used for the short and long straps so that they remain in place during grit-blasting. After grit-blasting, remove all residue by blasting component with compressed nitrogen.

Note: *Do not subsequently solvent-clean, wipe or touch the grit-blasted surfaces.*

- e. Immediately after the abrasive treatment, apply a layer of PRC Corporation Pasa-Jell 105 to the *grit-blasted areas only*. Use a polyethylene, polypropylene or fluorocarbon bristle brush. Allow the Pasa-Jell to stand for about 15 minutes, and then rinse it off with cool tap water. Check that water fully wets all surfaces - lack of proper wetting indicates that there are still contaminants on the surface. If necessary, repeat steps a to c. Blast most of the water from the surfaces with compressed nitrogen, and then dry the components at room temperature. **The next step (priming) must be performed within 1 hour.**
- f. Spray (or brush if necessary) the *grit-blasted areas only* with corrosion-inhibiting primer Cyanamid BR127 to a thickness of .0001 to .0003. Use a spray gun with an agitator to ensure proper mixing as well as a nitrogen propellant and

dedicated nitrogen hoses. Application techniques should be practiced so as to achieve the correct thickness of primer. For spraying, position the gun 6 to 8" from the article and use a one-pass approach with a 20-30% overlap making sure to finish the stroke beyond the end of the article. With brush application, the correct thickness is usually obtained with one or 2 (max) strokes with a loaded but not dripping brush. Minimize the overlap of adjacent brush strokes.

Note: *BR127 primer must be stored in a freezer. Before using the primer, it must be warmed to room temperature (allow 4 hours for a 1 gallon container) and must be thoroughly mixed. To avoid moisture contamination, do not open the container until it is at room temperature. During use it must be kept thoroughly mixed. The primer may be omitted if FM73 is applied within 1 hour of the Pasa-Jell treatment, and if subsequent curing is done within 24 hours.*

- g. Air dry the primer for 30 minutes, and then oven cure it for 30 minutes at $120 \pm 6^{\circ}\text{C}$ ($250 \pm 10^{\circ}\text{F}$). Wrap the primed components in Kraft paper. In this condition, they may be stored for 6 months at room temperature. The skin pieces should first be wrapped such that only the splice area is left exposed; this approach will allow the pieces to be riveted together without exposing other treated areas to contamination.
7. If hidden strain gauges are required, install the gauges, wire leads and terminal strips per drawing GE310-1 on the hidden surface of the countersunk sheet. Check that the finished gauge installations have sufficient clearance when the two sheets are brought together.
8. Cut two pieces $2\text{-}7/8 \times 7/8$ of FM73 (nominal thickness 0.010) supported film adhesive. Place these at each end of the mating face of one of the skin pieces, leaving $1/16$ clearance from the free edges of the skin piece. If necessary, warm the FM73 pieces slightly (43°C max) using a heat gun to help tack them in place.

Note: *FM73 is normally stored in a freezer in a sealed plastic bag. In order to avoid contamination with condensed moisture, ensure that the FM73 has*

fully warmed to room temperature before removing it from this bag (around 2 hrs).

9. Place a thin strip (between 1/64 and 1/32 diameter) of Schnee-Morehead (S-M) "Tacky Tape" close to the edges of the FM73, to form a barrier to the migration of the glue into the riveted area during curing. Extend both ends of each strip 1/4" beyond the mating surfaces. One end of each strip will remain unattached until the two skin pieces are brought together. To form the strip, the tacky tape must be cut and stretched until an even strip of the correct diameter is formed. When placing the strip in position apply firm pressure to keep it in place using a piece of non-stick Tacky Tape backing paper.
10. If specified in the order, apply a thin (0.005-0.015) layer of polysulphide sealant Courtaulds (formerly PRC) PR1431 (MIL-S-81733) over the remainder of the mating surfaces.
11. Align the mating surfaces of the splice joint, and clamp the specimen loosely using plastic countersunk bolts in the four corners. Tape rivets in place in the remaining holes with the exception of the second rivet holes from the edges of the middle row (these holes will be used later for locating pins during the hole-cutting procedure). Tighten the bolts to clamp the specimen firmly. Keep the primed surfaces covered with Kraft paper to avoid surface contamination. Set the rivets in place by gently tapping their heads.
12. Rivet the skin pieces together in random sequence taking care to center each rivet under the riveting spindle before engaging. Using a powered, adjustable riveting machine, configure the machine to produce the required driven head dimensions (0.24 ± 0.01).

Note: (a) If PR1431 interfaying sealant has been used, dip each rivet in PR1431 before driving it. When riveting has been completed, clean off all excess PR1431 using a chlorinated solvent (sparingly). Allow the PR1431 to cure for 14 days at room temperature, before proceeding further with this MP.

(b) The uncured FM73 in the riveted specimen will have a shop life of up to 30 days at 24°C, and so the remainder of this MP should be completed within this time.

13. Cut pieces of FM73 undersize by between 1/16 and 1/8 on all dimensions corresponding to Items 2,3 and 4 of Table A1. Aluminum templates are available for this purpose. It is strongly recommended that a piece of clean backing paper be placed between the FM73 and template to facilitate removal of the template after cutting; it also produces more accurate pieces. If these pieces of FM73 will not be used within 8 hours, seal one specimen's worth in its own plastic bag, and store all bags in a freezer.
14. Stick the FM73 pieces to the straps and doublers leaving the backing paper on and centering the piece on the component. Working with one side of the riveted specimen at a time, first mark the required distance of the doublers and the long straps from the specimen centre (4.00 ± 0.01 and 3.00 ± 0.01 , respectively). Then stick the short straps and a doubler followed by the long straps and another doubler to the main sheets. Ensure that the straps are flush (± 0.005) with the edges of the specimen. Where the short straps and skin pieces butt together, the gap must be 0.04 ± 0.02 . Note that at the two extreme ends of the specimen, the edges do not necessarily have to be flush. With the components properly positioned, use a heat gun to warm the required area slightly (43°C max), and then apply direct pressure to tack the components in place. Carefully flip the specimen over and repeat for other side.
15. Place an even fillet of Tacky Tape about $\frac{3}{4}$ " from the edges of the lap joint (avoiding the hidden gauges on one side) coming in at the ends to join up with the exposed ends of the hidden Tacky Tape as shown in Fig. A.3. Also shown is two more strips between the bucked rivet ends and the straps. The overall objective here is to prevent glue migration into the joint from any source.
16. Vacuum bag the specimen using the components listed in Table A2 and following the procedure outlined below.

Component	Dimensions	Quantity
Strong plastic outer bag	48 × 52	1
Impermeable release film	36 × 14	2
Breather material	See Fig. A.4	
Mushroom	---	1

Table A2 - Components required for vacuum bagging.

- a. With the outer bag fully open on a flat surface, sandwich the specimen between the two layers of impermeable release film (if using a one-side-release film, ensure that the correct surface is in contact with the specimen). Do not wrap the film closely around the edges of the specimen.
- b. Place the breather material as shown in Fig. A.4. The vacuum hose mushroom attachment must be placed at least 6" from one end of the specimen and its centre should be at least 4 to 5" from both bag edges. Note that the thin breather material is to be placed above the top layer of release film and below the bottom layer of release film such that it does not come in contact with the specimen. These strips are positioned over the channels between the straps and the doublers and provide a passageway by which trapped air can be suctioned out efficiently. The stepped layers in the area of the hose attachment provide enough airway to properly evacuate the bag, while avoiding excessive wrinkling of the bag near the specimen.
- c. Fold the bag over and cut a hole (~ 1" dia. - use the mushroom washer to do this) centred over the 4 × 4 piece of thick breather material such that, as previously mentioned, the hole centre is around 4 to 5" from both bag edges. If necessary, re-position the breather material and specimen to achieve these clearances.
- d. Clean the mushroom's rubber seal and washers using isopropyl alcohol. This is done to maintain a good seal and avoid any possible leakage during initially suctioning and throughout the cure cycle. Now with the circular nut and washer removed, place the mushroom through the hole.

- e. When everything is positioned satisfactorily, place three strips of Tacky Tape (with the backing paper on) along the three open lower bag edges leaving a $\frac{1}{2}$ " clearance from each edge and a $\frac{1}{4}$ " overlap at the corners. Remove the backing paper from one strip at a time, then holding the upper bag taught, lower it onto the tape trying to create as few wrinkles as possible. Continue until bag is fully sealed, then apply strong manual pressure along the tape lines to create a strong seal.
- f. With the vacuum pump active and the mushroom properly tightened down, apply suction by connecting the vacuum hose to the mushroom attachment. Check that the bag has been sucked down tightly in all locations, and that it forms wrinkle-free fillets at all internal edges and steps. Smooth out any wrinkles, creases or fillets anywhere that might allow glue to migrate in an unwanted manner. Maintain suction until the specimen has been cured.

17. Cure the FM73 in an autoclave at 40 ± 5 psig. Set the vessel pressure such that
 $(\text{vacuum in psig}) + (\text{vessel pressure in psig}) = 40 \pm 5$ psig

Apply heat according to the following schedule:

- (1) Heat up in 30 min to $120 \pm 3^\circ\text{C}$ ($250 \pm 5^\circ\text{F}$).

Note: autoclave at NRC-IAMT heats to 250°F in 15 min but large aluminum base upon which the specimen lies takes an extra 15 min to heat up, therefore 30 min heat up time is achieved

- (2) Hold for 60 min.

Before beginning the cure, the specimen should have a shim under the bag at the elevated end of the specimen and two weights placed on the visible doublers. The bag should be restrained using weights or clamps on top of heat-resistant padding material to minimize bag movement and specimen warping during the cure. Maintain the restraint until the specimen has cooled.

18. Remove the glue flashing around the outer edges of the specimen. Remove large amounts initially using a medium file, and then switch to a fine one when nearing the aluminum edges. If any aluminum must be removed, ensure that the required tolerances are not exceeded.
19. Using the appropriate NC machine and program, machine the 2×1.251 holes and the $44 \times 9/16$ holes taking the necessary precautions to avoid scuffing the surfaces.
20. Inspect the finished specimen according to reference drawing GE304-1. If there are any discrepancies, record them and notify the customer immediately.
21. Install external strain gauges as per drawing GE310-1, following the manufacturers recommended procedures (MM Instruction Bulletins B-127-12 and B-129-7). *Do not prepare a larger surface area for each gauge than is necessary.*

List of Figures

- A.1 Drawing set for the MSD specimen.
- A.2 Areas of sheets to be masked during grit-blasting.
- A.3 Application of S-M Tacky Tape as barriers to prevent glue migration into the rivetted area.
- A.4 Breather material arrangement for vacuum bagging.

Figure A.1 *Drawing set for the MSD specimen.*

Figure A.1 (cont'd).

Figure A.1 (cont'd).

Figure A.1 (cont'd)

Figure A.1 (cont'd).

Figure A.2 Areas of sheets to be masked during grit-blasting.

Figure A.3 Application of S-M Tacky Tape barriers to prevent glue migration into rivetted area.

Figure A.4 Breather material arrangement for vacuum bagging.

APPENDIX B

DATA ACQUISITION SOFTWARE PROGRAM "G-SCAN"

The strain gauge setup described in Ch. 6 needed some method to perform necessary data acquisition tasks. It would need to retrieve the signals from the Hewlett Packard 3497A Data Acquisition/Control Unit and convert them into a useable format. Immediate pictures of the various stress distributions were desired without having to manually generate the information. This would then allow for a more immediate response to the results, and also translate into shorter test times.

To solve these problems, a generic data acquisition software package called "G-Scan" was written in HP-85 Basic on an HP-85 computer to perform the following data acquisition tasks:

- scan gauge output on demand
- record the data (including real-time calculation of stress and change in stress since last scan) in hard copy format
- produce auto-scaled plots of gauge, membrane and bending stresses for nominal and critical row positions
- allowance for re-zeroing gauges or checking existing zero values
- setup of gauge system information

The program was designed to be easily modifiable to suit a particular application. This normally involves just changing a few generic parameters at the beginning of the program.

It was also designed to be user-friendly. This was accomplished through the use of menus with single push-button controls. A copy of the program listing is shown below.

```

10 ! *****
20 !   PROGRAM NAME:   "G-Scan"
30 !   WRITTEN:       21 JUL 93   BRETT WAKEMAN
40 !   REVISIONS:     30 JUL 93   BW   ADDED PLOTTING ROUTINES
50 !                 16 SEP 94   BW   ADDED SUPPORT FOR 6 MORE GAUGES
60 !                 28 SEP 94   BW   ADDED QUICK SCAN OPTION
70 !                 18 OCT 94   BW   IMPROVED OUTPUT FORMAT
80 !                 19 DEC 94   BW   FIXED BUG WITH QUICK SCAN
90 !
100 !   FUNCTION:      SCANS A NUMBER OF STRAIN GAUGES, COMPUTES STRESSES,
110 !                  AND PRODUCES PLOTS OF GAUGE, MEMBRANE AND BENDING
120 !                  STRESSES.
130 !
140 !   CONTROL:        MENU-OPERATED
150 ! *****
160 CLEAR 709
170 ENABLE KBD 33
180 REAL A(20),B(20),G,G1,G2,M(20),R(20),S(20),T(20),Z(20)
190 INTEGER C1,C2,D,E,I,J,N,P,Q,V,X,Y1,Y2
200 DIM LS[32]
210 C1=40 ! FIRST CHANNEL #
220 D=0 ! DEBUGGING ON/OFF (1/0)
230 E=10500 ! YOUNG'S MODULUS FOR 2024-T3 (ksi)
240 G1=2.165 ! GAUGE FACTOR FOR 350 OHM GAUGES
250 G2=2.11 ! GAUGE FACTOR FOR 120 OHM GAUGES
260 N=20 ! NUMBER OF CHANNELS
270 P=1 ! PRINT TO SCREEN ON/OFF (1/0)
280 Q=0 ! QUICK SCAN ON/OFF (1/0)
290 V=7 ! EXCITATION VOLTAGE
320 P$="PRINTING="
330 O$="AF"&VAL$(C1)&"AL"&VAL$(C1+N-1)
340 OUTPUT 709 ;O$
350 FOR I=0 TO N-1
360 S(I)=0 ! CLEAR THE STRESSES
370 NEXT I
380 PRINTER IS P
390 IF D=1 THE GOSUB 3350 ! GET SAMPLE ZERO DATA
400 CLEAR
410 !
420 ! *****
430 ! * MAIN MENU *
440 ! *****
450 !
460 DISP "   STRAIN GAUGE MEASUREMENTS"
470 DISP "-----"
480 IF P=1 THEN DISP P$;"OFF" ELSE DISP P$;"ON"
490 DISP
500 DISP ">> MENU OPTIONS <<-----"
510 DISP "SCAN   - TAKE NEW READINGS"
520 DISP "QSCAN  - QUICK GAUGE SCAN"
530 DISP "ZERO   - TAKE ZERO READINGS"
540 DISP "0-CHK  - CHECK ZERO READINGS"
550 DISP "PRINT  - TURN PRINTING ON/OFF"
560 DISP "PLOT   - GOTO PLOT MENU"
570 DISP "NUM    - SET NUMBER OF CHANNELS"
580 DISP "STOP   - END MEASUREMENTS"
590 ON KEY# 1,"SCAN" GOTO 690

```

```

600 ON KEY# 2,"ZERO" GOTO 1150
610 ON KEY# 3,"PRINT" GOTO 1130
620 ON KEY# 4,"PLOT" GOTO 1650
630 ON KEY# 5,"QSCAN" GOTO 680
640 ON KEY# 6,"O-CHK" GOTO 1330
650 ON KEY# 7,"NUM" GOTO 1550
660 ON KEY# 8,"STOP" GOTO 1460
670 KEY LABEL @ GOTO 670
680 Q=1 ! TURN QUICK SCAN ON
690 !
700 ! *****
710 ! * SCAN ROUTINE *
720 ! *****
730 !
745 IF D=1 THEN GOSUB 3620 ! GET SAMPLE GAUGE DATA
750 CLEAR
760 OUTPUT 709 ;O$
770 PRINT
780 IF Q=0 THEN PRINT USING 790 ELSE PRINT USING 800
790 IMAGE "SET:",7X,"CYCLES:",2/
800 IMAGE "QUICK SCAN",2/
810 PRINT "SG CH SIG(V) STRESS(ksi) CHANGE"
820 PRINT "-----"
840 !
850 FOR I=0 TO N-1
860 IF D=1 THEN 900
870 PRINTER IS 1
880 OUTPUT 709 ;"ASVR1"
890 ENTER 709 ;B(I) ! GET GAUGE READINGS
900 PRINTER IS P
910 !
920 ! COMPUTE AND OUTPUT STRESSES
930 !
940 IF I<14 THEN G=G1 ELSE G=G2
950 Z(I)=B(I)-A(I) ! READINGS MINUS ZERO VALUES
960 IF Q=0 THEN GOSUB 1010 ELSE GOSUB 1030
965 IMAGE 2D,X,2D,X,.DDDDDD,2X,2D.DDEDD,2X,S2D.DDD
970 NEXT I
980 IF P=1 THEN GOSUB 1060
990 IF Q=1 THEN Q=0 ! TURN QUICK SCAN OFF
1000 GOTO 400
1010 T(I)=S(I)
1015 S(I)=4*E*Z(I)/G/(V-2*Z(I)) ! CALCULATION OF STRESS
1020 PRINT USING 965 ; I+1,I+C1,B(I),S(I),S(I)-T(I)
1025 RETURN
1030 R(I)=4*E*Z(I)/G/(V-2*Z(I)) ! CALCULATION OF STRESS
1040 PRINT USING 965 ; I+1,I+C1,B(I),R(I),R(I)-T(I)
1050 RETURN
1060 !
1065 ! *****
1070 ! * CONTINUE SUBROUTINE *
1075 ! *****
1080 !
1090 DISP
1100 DISP "PRINT [CONT] TO CONTINUE"
1110 PAUSE
1120 RETURN
1130 IF P=1 THEN P=2

```

```

1140 GOTO 380
1150 !
1160 ! *****
1170 ! * ZERO READINGS *
1180 ! *****
1190 !
1200 CLEAR
1210 PRINT "ZERO READINGS:"
1220 PRINT
1230 PRINT "SG# CH#  SIGNAL(V) "
1240 PRINT "-----"
1250 FOR I=0 TO N-1
1260 OUTPUT 709 ; "ASVR1"
1270 ENTER 709 ; A(I) ! SCAN GAUGES
1280 PRINT USING 1290 ; I+1,I+C1,A(I)
1290 IMAGE 2D,2X,2D,3X,.DDEDDD
1300 NEXT I
1310 IF P=1 THEN GOSUB 1060
1320 GOTO 400
1330 !
1340 ! PRINT CURRENT ZERO READINGS
1350 !
1360 CLEAR
1370 PRINT "CURRENT ZERO READINGS:"
1380 PRINT
1390 PRINT "SG# CH#  SIGNAL(V) "
1400 PRINT "-----"
1410 FOR I=0 TO N-1
1420 PRINT USING 1290 ; I+1,I+C1,A(I)
1430 NEXT I
1440 IF P=1 THEN GOSUB 1060
1450 GOTO 400
1460 !
1470 ! *****
1480 ! * END ROUTINE *
1490 ! *****
1500 !
1510 CLEAR
1520 PRINTER IS 1
1530 PRINT "DONE."
1540 END
1550 !
1560 ! *****
1570 ! * SET NUMBER OF CHANNELS *
1580 ! *****
1590 !
1600 CLEAR
1610 DISP "ENTER NUMBER OF CHANNELS"
1620 INPUT N
1640 GOTO 400
1650 !
1660 ! *****
1670 ! * PLOT MENU *
1680 ! *****
1690 !
1700 CLEAR
1710 DISP "      STRAIN GAUGE MEASUREMENTS"
1720 DISP "-----"

```

```

1730 IF P=1 THEN DISP P$;"OFF" ELSE DISP P$;"ON"
1740 DISP
1750 DISP "----- >> PLOT OPTIONS <<-----"
1760 DISP
1770 DISP "GAUGE - PLOT GAUGE STRESSES"
1780 DISP "MEMBR - PLOT MEMBRANE STRESSES"
1790 DISP "BEND - PLOT BENDING STRESSES"
1800 DISP "BACK BACK TO MAIN MENU"
1810 OFF KEY# 5 @ OFF KEY# 6
1820 OFF KEY# 7 @ OFF KEY# 8
1830 ON KEY# 1,"GAUGE" GOTO 1880
1840 ON KEY# 2,"MEMBR" GOTO 2330
1850 ON KEY# 3,"BEND" GOTO 2250
1860 ON KEY# 4,"BACK" GOTO 400
1870 KEY LABEL @ GOTO 1870
1880 !
1890 ! *****
1900 ! * GAUGE STRESS PLOT *
1910 ! *****
1920 !
1930 ! CALCULATE MAX/MIN FOR S(I) FOR AUTO-SCALING
1940 !
1950 !Y1=INT(S(0)+1) ! YMAX
1960 Y2=INT(S(0)) ! YMIN
1970 FOR I=1 TO N-1
1980 IF Y1<INT(S(I)+1) THEN Y1=INT(S(I)+1)
1990 IF Y2>INT(S(I)) THEN Y2=INT(S(I))
2000 NEXT I
2010 !
2020 ! PLOT GRAPH
2030 !
2040 L$="GAUGE STRESSES"
2050 C2=C1+N-1 ! XMAX
2060 GOSUB 3090
2070 FOR I=0 TO 6
2080 PLOT I+C1,S(I)
2090 NEXT I
2100 PENUP
2110 FOR I=7 TO 13
2120 PLOT I+C1,S(I)
2130 NEXT I
2140 PENUP
2150 FOR I=14 TO 16
2160 PLOT I+C1,S(I)
2170 NEXT I
2180 PENUP
2190 FOR I=17 TO 19 ! HIDDEN GAUGES
2200 PLOT I+C1,S(I)
2210 NEXT I
2220 PAUSE
2230 ALPHA
2240 GOTO 1650
2250 !
2260 ! *****
2270 ! * BENDING STRESS PLOT *
2280 ! *****
2290 !
2300 J=1

```

```

2310 L$="BENDING STRESSES"
2320 GOTO 2400
2330 !
2340 ! *****
2350 ! * MEMBRANE STRESS PLOT *
2360 ! *****
2370 !
2380 J=0
2390 L$="BENDING STRESSES"
2400 !
2410 ! J=0: CALCULATE MEMBRANE STRESSES
2420 ! J=1: CALCULATE BENDING STRESSES
2430 !
2440 FOR I=0 TO 6
2450 IF J=0 THEN M(I)=(S(I)+S(I+7))*0.5 ELSE M(I)=(S(I+7)-S(I))*0.5
2460 NEXT I
2470 !
2480 ! FIND MAX/MIN FOR M(I) FOR AUTO-SCALING
2490 !
2500 Y1=INT(M(0)+1) ! YMAX
2510 Y2=INT(M(0)) ! YMIN
2520 FOR I=1 TO 6
2530 IF Y1<INT(M(I)+1) THEN Y1=INT(M(I)+1)
2540 IF Y2>INT(M(I)) THEN Y2=INT(M(I))
2550 NEXT I
2560 !
2570 ! PLOT GRAPH
2580 !
2590 C2=C1+6 ! XMAX
2600 GOSUB 3090
2610 FOR I=0 TO 6
2620 PLOT I+C1,M(I)
2630 NEXT I
2640 PAUSE
2650 IF N<15 THEN 1650
2660 !
2670 ! STRESS PLOTS INVOLVING HIDDEN GAUGES
2680 !
2690 IF J=1 THEN GOTO 2790
2700 M(7)=(S(14)+S(17))*0.5
2710 M(10)=(S(15)+S(18))*0.5
2720 M(12)=(S(16)+S(19))*0.5
2730 M(8)=M(12)
2740 M(9)=(M(8)+M(10))*0.5 ! AN APPROXIMATION
2750 M(11)=(M(10)+M(12))*0.5 ! AN APPROXIMATION
2760 M(13)=M(7)
2770 L$="MEMBRANE - CRIT ROW"
2780 GOTO 2870
2790 M(7)=(S(17)-S(14))*0.5
2800 M(10)=(S(18)-S(15))*0.5
2810 M(12)=(S(19)-S(16))*0.5
2820 M(8)=M(12)
2830 M(9)=(M(8)+M(10))*0.5 ! AN APPROXIMATION
2840 M(11)=(M(10)+M(12))*0.5 ! AN APPROXIMATION
2850 M(13)=M(7)
2860 L$="BENDING - CRIT ROW"
2870 !
2880 ! FIND MAX/MIN FOR M(I) FOR AUTO-SCALING

```

```

2890 !
2900 Y1=INT(M(7)+1) ! YMAX
2910 Y2=INT(M(7)) ! YMIN
2920 FOR I=8 TO 13
2930 IF Y1<INT(M(I)+1) THEN Y1=INT(M(I)+1)
2940 IF Y2>INT(M(I)) THEN Y2=INT(M(I))
2950 NEXT I
2960 !
2970 ! PLOT GRAPH
2980 !
2990 C1=54 ! XMIN
3000 C2=60 ! XMAX
3010 GOSUB 3090
3020 FOR I=0 TO 6
3030 PLOT I+C1,M(I+7)
3040 NEXT I
3050 PAUSE
3060 C1=40
3070 ALPHA
3080 GOTO 1650
3090 !
3100 ! *****
3110 ! * GRAPH SETUP ROUTINE *
3120 ! *****
3130 !
3140 GRAPH ! PUT IN GRAPHICS MODE
3150 GCLEAR ! CLEAR GRAPHICS SCREEN
3160 FRAME ! BOX GRAPH AREA
3170 !
3180 ! SETUP X&Y AXES
3190 !
3200 SCALE C1,C2,Y2-1,Y1+(Y1-Y2+1)/75
3210 IF Y2-1>0 THEN XAXIS Y2 1,1,C1,C2 ELSE XAXIS 0,1,C1,C2
3220 YAXIS C1,1,Y2-1,Y1
3230 !
3240 ! CREATE LABELS
3250 !
3260 PEN 1 ! CHOOSE WHITE DOTS
3270 PENUP
3280 MOVE C1+(32-LEN(L$))/64*(C2-C1),Y1-(Y1-Y2+1)/20 ! CENTRE LABEL
3290 LABEL L$
3300 MOVE C1+(C2-C1+1)/60,Y2-.95 ! POSITION YMAX LABEL
3310 LABEL VAL$(Y2-1)
3320 MOVE C1+(C2-C1+1)/60,Y1-(Y1-Y2+1)/30 ! POSITION YMIN LABEL
3330 LABEL VAL$(Y1)&"ksi"
3340 RETURN
3350 !
3360 ! *****
3370 ! * SAMPLE ZERO DATA *
3380 ! * FOR DEBUGGING *
3390 ! *****
3400 !
3410 A(0)=.005575
3420 A(1)=.006303
3430 A(2)=.007805
3440 A(3)=.007083
3450 A(4)=.007109
3460 A(5)=.006254

```

```
3470 A(6)=.00484
3480 A(7)=.005683
3490 A(8)=.005918
3500 A(9)=.005149
3510 A(10)=.005839
3520 A(11)=.01079
3530 A(12)=.006414
3540 A(13)=.006152
3550 A(14)=.005239
3560 A(15)=.008578
3570 A(16)=.009578
3580 A(17)=.010587
3590 A(18)=.009245
3600 A(19)=.010254
3610 RETURN
3620 !
3630 ! *****
3640 ! * SAMPLE GAUGE DATA *
3650 ! * FOR DEBUGGING      *
3660 ! *****
3670 !
3680 B(0)=.009068
3690 B(1)=.006638
3700 B(2)=.006742
3710 B(3)=.005971
3720 B(4)=.006826
3730 B(5)=.008342
3740 B(6)=.012873
3750 B(7)=.007527
3760 B(8)=.007047
3770 B(9)=.00777
3780 B(10)=.008316
3790 B(11)=.012214
3800 B(12)=.007684
3810 B(13)=.012834
3820 B(14)=.008411
3830 B(15)=.007459
3840 B(16)=.007550
3850 B(17)=.009976
3860 B(18)=.011158
3870 B(19)=.010728
3880 RETURN
```


APPENDIX C

TECHNIQUE TO QUANTIFY CORROSION DAMAGE

The procedure begins with the colour image file of the material loss distribution in a standard graphic format (e.g. Tagged Image Format, TIF) and requires the use of image processing software. From there, the steps are as follows:

1. Crop the splice areas from the image files as necessary. For proper comparison between specimens, these areas should be identical not only in the physical dimensions which they represent (e.g. 20 inch² of splice area) but also in the number of 'pixels' (picture elements) used for the height and width. It is also desirable to maintain an aspect ratio of unity so that, for example, one physical inch horizontally represents the same number of pixels as it does vertically.
2. Convert each colour image file into its greyscale equivalent. This is done to separate each colour into a shade of grey with equal luminance, or brightness. If two colours happen to have equal luminance (i.e. they are equal in the greyscale image), it will be necessary to first change one of the colours before converting to greyscale.
3. Determine the exact percentage of grey each colour has been converted into. The corresponding luminance level is then equal to 100 minus this value (e.g. 7% grey = 93% luminance).
4. Plot the histogram for a greyscale image. This will reveal the relative luminance amongst the image's grey shades, and should appear as a series of solid lines of different heights, similar to a bar chart with variably spaced bars. An example is shown in Fig. C.1. Each line corresponds with a particular brightness ranging from

0% luminance (pure black) to 100% luminance (pure white). Its height is determined by the portion of the image that matches that particular point on the luminance spectrum.

5. From the histogram, record the relative portions for each level of luminance (i.e. for each bar). Using the image processing software (e.g. Picture Publisher), this can be accomplished by determining the number of pixels corresponding to a each luminance level (shown in Fig. C.1).
6. Divide the number of pixels for each bar by the total number of pixels (image height in pixels $\times$ image width in pixels) to determine the respective percentages of the scanned information. Depending on the application, division by other denominators might be desired (see example below).
7. Repeat steps 4 through 6 for each greyscale image.

Following these few simple steps can quickly convert the data into a quantitative characterization of the extent of the various thickness loss ranges within the scanned area. There is no estimation involved as to the amount of each colour; the procedure totals the exact number of pixels to provide an exact calculation of each colour's percentage of the total pixel count. An example of how the technique was applied is shown below.

EXAMPLE OF APPLICATION OF TECHNIQUE

This example will consider the scan of specimen CF-1 after having accumulated 52 days of accelerated corrosion. As previously shown in Ch. 7, the eddy current procedure used in this thesis produced images composed of white, blue, light blue, red, and grey pixels. The blue, light blue and red pixels correspond with the thickness loss ranges of 5-7½%, 7½-10%, and greater than 10%, respectively. The grey pixels represent some bad data

acquired through the eddy current probe. The white pixels represent a combination of the remaining thickness loss range from 0-5%, as well as the areas of the rivet holes and some missed data (close to edges of rivet holes and underneath countersinks).

The histogram of the greyscale-converted image is shown in Fig. C.1. As shown, five bars appear corresponding to each of the five colours and luminance levels. Also shown are the figures for the total number of pixels (68,160) and the number of pixels for a 79% luminance level (8,458). The complete breakdown of pixels was as follows:

<u>Luminance Level</u>	<u>Colour</u>	<u>Number of Pixels</u>
7%	Blue	15,042
19%	Red	5,362
32%	Grey	1,158
79%	Light Blue	8,458
98%	White	38,140
TOTAL		68,160

Next it must be determined what portion of the white pixels represent the remaining thickness loss range from 0-5%. This is calculated as

$$\begin{aligned} \# \text{ of pixels for } 0\text{-}5\% t_{\text{loss}} &= \# \text{ of white pixels} - \# \text{ of pixels for rivet holes} \\ &\quad - \# \text{ of pixels of missed data} \end{aligned}$$

The number of rivet hole pixels can be estimated knowing that the hole diameter after rivetting is roughly 3/16 inch, the scan covers almost 22 rivets (7.2 rivets $\times$ 3 rows), and that approximately 60 pixels represents an inch. The calculation is thus

$$\begin{aligned} \# \text{ of rivet hole pixels} &= \text{rivet area} \times \text{number of rivets} \times \text{pixel density} \\ &= \pi/4(3/16)^2 \text{ inch}^2 \times 21.6 \text{ rivets} \times 3600 \text{ pixels/inch}^2 \\ &= 2,147 \\ &\approx 2,150 \text{ pixels} \end{aligned}$$

The number of missed data pixels can be estimated by totalling the pixels representing the areas under the countersinks with the pixels representing the areas close to the rivet heads. The latter resulted from the eddy current probe being gently lifted off the sheet by the rivet heads which were slightly proud. Through careful examination of the eddy scans, it is estimated that the probe could not process information within 3/64 inch of the rivet head. As the rivet head diameter was typically 9/32 inch, this left a hole of white pixels with a diameter of roughly 3/8 inch encompassing the missed data as well as the rivet holes. The number of missed data pixels is therefore estimated as

$$\begin{aligned}
 \text{\# of missed data pixels} &= \text{missed data/rivet} \times \text{number of rivets} \times \text{pixel density} \\
 &= \pi/4[(3/8)^2 - (3/16)^2] \text{ inch}^2 \times 21.6 \text{ rivets} \\
 &\quad \times 3600 \text{ pixels/inch}^2 \\
 &= 6,441 \\
 &\approx 6,450 \text{ pixels}
 \end{aligned}$$

The number of pixels for the thickness loss range from 0-5% is then

$$\begin{aligned}
 \text{\# of pixels for 0-5\% } t_{\text{loss}} &= \text{\# of white pixels} - \text{\# of pixels for rivet holes} \\
 &\quad - \text{\# of pixels of missed data} \\
 &= 38,140 - 2,150 - 6,450 \\
 &= 29,540 \text{ pixels}
 \end{aligned}$$

The last step remaining is to divide each of the pixel counts by a suitable parameter in order to determine the percentage breakdown. For the purposes of this thesis, it was desirable to know what percentages each thickness loss range represented of the total damaged area. Thus, each value was divided by the total pixel count minus the rivet hole pixels (i.e. $68,160 - 2,150 = 66,010$), since these areas contained no sheet material. The resulting percentages were as follows:

<u>Data Category</u>	<u>% of Damageable Area</u>
0 to 5 % t_{loss}	44.7
5 to 7½ % t_{loss}	22.8
7½ to 10 % t_{loss}	12.8
> 10 % t_{loss}	8.1
Missed Data	9.8
Bad Data	1.8
<hr/>	
TOTAL	100

The above figures suggest that the eddy current procedure was capable of revealing 88% of the material loss information within the damageable surfaces of the scanned area. This is a very respectable result given the simplicity of the method.

Figure C.1 *Histogram of a greyscale eddy current scan for specimen CF-1 after 52 days of accelerated corrosion. [generated using Picture Publisher]*

END

1 9 1 0 6 1 9 7

FIN