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**MULTIPLE SITE FATIGUE DAMAGE
IN FUSELAGE SKIN SPLICES**

by

GRAEME F EASTAUGH, BSc

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 of
Mechanical and Aerospace Engineering**

Carleton University
Ottawa, Ontario

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"MULTIPLE SITE FATIGUE DAMAGE IN FUSELAGE SKIN SPLICES"

submitted by Graeme F Eastaugh, BSc
in partial fulfilment of the requirements for
the degree of Master of Engineering

Thesis Supervisor

Chair, Department of Mechanical and Aerospace Engineering

Carleton University
14 January 1994

ABSTRACT

The results of experimental and theoretical research into multiple site fatigue damage (MSD) in the skin splices of pressurized fuselages are described. Firstly, a coupon-type specimen has been designed, which approximates on-aircraft conditions, and allows realistic MSD to be created and studied in a laboratory environment. Other coupon specimens used in industry and research cannot be used for MSD testing, because the net section stress increases untypically as the cracks progress, and because the specimens tend to fail before MSD has developed. The specimen concept described in this thesis overcomes these problems. Secondly, a cost-effective computer program for predicting MSD crack growth within a frame-bay has been developed. It applies Rooke's superposition and compounding technique in a new way, and agrees well with experimental MSD crack growth curves.

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1 INTRODUCTION

Multiple site fatigue damage (MSD) in a longitudinal skin splice has caused the failure of the fuselage of a passenger aircraft. This type of damage can adversely affect the fail-safe characteristics of a fuselage, and can have a short growth period between detectable length and fuselage failure. It is recognized as a major airworthiness problem, and has prompted major international research efforts.

Longitudinal fuselage splices in transport aircraft are typically formed by overlapping and rivetting together 2 sections of skin. This type of splice is known as a single shear lap splice. Repeated cabin pressurization cycles subject the splices to fatigue loading, and can cause cracks to initiate and grow at several rivet holes at the same time.

The objectives of this thesis are to define this type of MSD problem in practical terms, to analyze its causes and effects as far as possible from available information, and to define and initiate a research project that would be of value to the aircraft industry.

Sections 2 and 4 of this thesis provide a comprehensive definition and analysis of the MSD problem. Based on this analysis, 2 initial research projects are defined - one experimental, and one theoretical. The remainder of the thesis describes the results of these 2 projects.

The experimental project consists of the design of a coupon-type specimen, which approximates on-aircraft conditions, and allows realistic MSD to be created and studied in a laboratory environment. Other coupon specimens used in industry and research cannot be used for MSD testing, because the net section stress increases untypically as the cracks progress, and because the specimens tend to fail before MSD has developed. This specimen concept described in this thesis overcomes these problems, and will have research and industrial applications.

The theoretical project consists of the development of a cost effective computer program that predicts multiple-site crack growth within a frame-bay. This program runs on a PC or workstation, and produces graphical output of crack growth curves for any typical splice size or geometry. It is based on the superposition and compounding technique pioneered by Rooke, rather than on more widely used and costly iterative finite element techniques. Rooke's technique has been applied in a new way, and close agreement has been achieved with experimental crack growth curves. The program has potential research and industrial applications, but requires further validation against test results.

2 MSD PROBLEM DEFINITION

2.1 Definition of Terms

Multiple site damage is the simultaneous development of fatigue cracks at multiple sites in the same structural element, such that fatigue cracks may coalesce to form one large crack [1]. If the linkup of existing adjacent cracks will not degrade the structure below limit load capability, the MSD is referred to as local MSD. Local MSD could progress to failure either by further growth of the single large crack to a critical length, or by growth and linkup with other local MSD.

2.2 Reasons for Concern Over MSD

MSD in the fuselage skin splices of pressurized transport aircraft is currently of concern, because:

- a. It was a major contributor to a catastrophic accident in 1988.
- b. It was not fully taken into account in the original fatigue analyses and structural maintenance policies of existing aircraft. Specifically, it is now recognized that:
 - (1) MSD can invalidate previous fail-safe assumptions concerning the extent of fatigue damage that a fuselage can sustain; in particular, MSD in the frame-bay adjacent to a long crack may prevent the arrest of the crack at the frame.
 - (2) MSD within one frame-bay can have a very short growth period from detectable length to the formation of a lead crack one or more frame-bays in length; the inspection threshold, frequency and technique must take this into account.
- c. Additional analysis tools, test methods and regulatory guidelines are needed for the reassessment of existing and future aircraft.

2.3 Historical Perspective

It is useful to put the problem in a historical perspective. The explosive decompression failure of the fuselage of an Aloha Airlines 737 passenger aircraft in 1988 after almost 90,000 flights resulted from rivet line cracks in a longitudinal skin splice that were not detected and repaired early enough. The failure originated in the splice immediately above the window line on the port side, and ended with the almost complete detachment of the fuselage structure above the cabin floor from the cockpit to the wing leading edge, as can be seen in the photograph in Figure 2-1, which was taken from a paper by Tong [2]. The precise crack configuration at the time of failure was not determined. There was considerable evidence that fatigue cracks had started at many rivet holes in the top row of the splice in question, and had grown together until one or more large cracks reached critical proportions.

Such cracks were not a new phenomenon. At least 10 cases of cracking at lap splices on 737 aircraft had been reported previously, including one instance of a 7.5 inch crack. In 1987 Boeing had completed a fatigue test of the rear fuselage of a retired 737, during which multiple small cracks in a splice joint were monitored until they grew to a linked crack almost 40 inches long. All these cases involved early production 737.

On these 737 aircraft, the splices were formed by cold-bonding and then riveting together the two .036 inch thick faying edges. The bond was found not to be durable, and random disbonding occurred over time. In these locations, the countersink in the outer skin formed a knife edge stress concentration, which promoted early fatigue cracking.

Later 737 aircraft (No. 292 onwards) use a hot-bonded doubler to strengthen the outer skin at the joint, and incorporate a polysulphide sealant instead of glue between the faying surfaces. The skin splices on these aircraft appear to be more durable.

Boeing originally designed the 737 for an economic service life of 20 years, 51,000 flight hours and 75,000 cycles. At the time the 737 was certified in 1967, FAA regulations required that the structure be capable of sustaining 80% of limit load with any complete or obvious partial failure of any single structural element. This was referred to as the fail-safe philosophy, which replaced the safe life philosophy for most aircraft structures in the 1950s following the Comet disasters. The fuselage of the 737 was tested against this requirement by driving blades into the skin of adjacent 20 inch frame bays while the

fuselage was at full pressure. A .40 inch crack appeared almost instantaneously, but the tips turned circumferentially to create a flap and then arrested. Explosive decompression was avoided. This test did not deal with the fail safe characteristics of the fuselage in the event of multiple cracks in a skin splice. However, such cracks occurred during the 707 full scale fatigue test, and developed to a safe flapping failure of the type just described.

Flapping failure evidently did not occur in the Aloha incident, which led the National Transportation Safety Board (NTSB) to conclude that sufficient fatigue cracking or tear strap disbond (or a combination of both) existed in the lap joint to negate the design-intended controlled decompression of the structure [3].

The cold-bonding process was not used on the 707, but was used on early 727 and 747 aircraft. Later versions of the 727 (No. 850 onwards) and 747 (No. 201 onwards) use polysulphide interfaying sealant.

Multiple cracks in longitudinal splices have occurred during fatigue testing and/or in service on 707, 727 and 747 aircraft. Advanced cracking has been noted on the 727 at 40,000 pressurization cycles. Cracks appeared beyond rivet heads at several sites on the 747 between 21,500 and 40,000 cycles. Subsequent crack growth to a linked length of around 6 inches appears to take 20,000 to 40,000 pressurization cycles on Boeing aircraft. The growth rate of longer cracks appears to be very high.

There have been references in the literature to occurrences of splice cracks on Douglas and BAe aircraft, but there is less data on these than on Boeing aircraft.

In 1978, the portion of Federal Airworthiness Regulations (FAR) 25.571 dealing with fail safe requirements was revised to reflect advances in fracture mechanics and structural analysis. The new regulation also required consideration of damage growth characteristics at multiple sites, and an inspection program incorporating these analyses to ensure that the damage was detected before residual strength dropped below the regulatory fail-safe requirements.

Prior to the Aloha incident, Boeing had reassessed the 727/737/747, all of which were certified under the pre 1978 criteria, using the revised 'damage tolerance' requirements. This reassessment required determination of residual strength with the presence of multiple active cracks, extensive analysis of crack growth rates, and incorporation of these

determinations into the aircraft's maintenance program. The same methodology was used as for the certification of the 757/767. The FAA subsequently mandated such a program; for the 737, it was to be completed by 1985. Boeing implemented a Supplementary Structural Inspection Program (SSIP) on a sample of high time aircraft. Unfortunately, the fuselage skin was not included in the SSIP, since it was classified as 'damage obvious or malfunction evident'.

The NTSB's recommendations from its Aloha inquiry included the following (in abbreviated form):

- a. Perform a full fatigue test to at least 2 lifetimes.
- b. Identify structure susceptible to MSD and adopt inspection programs.
- c. Discontinue the inspection classification of fuselage skin as 'malfunction evident or damage obvious'.

2.4 Issues to be Addressed

These recommendations imply that all longitudinal splices in the pressure hull must be subjected to a full damage tolerance assessment. A major international research effort is under way to develop the necessary analysis tools, test methods and regulatory guidelines. In this context, 2 important issues are addressed in this project:

- a. It appears that MSD has so far only been realistically simulated in large scale, expensive test specimens. An attempt should be made at overcoming the problem of designing suitable smaller scale, inexpensive specimens.
- b. While several analysis methods have been proposed for predicting multiple crack growth in a splice, most of these are expensive and time consuming. There is a need for inexpensive analysis methods.

3 DAMAGE TOLERANCE ANALYSIS PHILOSOPHY AND THEORY

3.1 General Requirements

An evaluation of the damage tolerance of structure usually includes the following activities:

- a. The identification of components or sites in flight-essential structure which are at greatest risk of cracking as a result of fatigue, corrosion or discrete source damage (eg, accidental damage during maintenance, debris from engine disintegration, or undetected manufacturing flaw).
- b. Prediction of crack growth rate under service loading.
- c. Calculation of the residual strength vs crack length.
- d. Calculation of the critical crack length corresponding to the design limit load.
- e. Assessment of the probability of crack detection prior to component failure using available methods of inspection and economically feasible inspection intervals.

This analysis is performed in order to comply with certification requirements introduced during the 1970s, ie, the Federal Aviation Authority (FAA) requirements in Reference [4]. Aircraft certified to earlier regulations are generally required to undergo some form of retrospective damage tolerance analysis if they are to be kept in service beyond the previously certified airframe fatigue life. Damage tolerance analysis is now also under consideration for proposed structural repair schemes. The purpose of damage tolerance analysis is to avoid catastrophic failure during the operational life of the aircraft, and so it must result in an adequate inspection program. It is also likely to result in design changes or mandatory modifications, as applicable.

Guidance on FAA philosophy underlying current regulations is given by Swift [5]. The following subsections identify current practice as regards activities a to d above.

3.2 Identification of Structure at Risk

For the purpose of identifying potentially dangerous crack sites, the principal structural elements are first identified. These are defined by the FAA as those elements that contribute significantly to carrying flight, ground and pressurization loads, and whose failure, if undetected, could eventually lead to loss of the aircraft. The selection of sites in these elements for crack growth analysis is made on the basis of several criteria, including: low static strength margin; high stress concentration; high spectrum severity; secondary member subject to high stress after failure of a primary member; adverse material characteristics (eg, high crack propagation rates and low fracture toughness); leading crack site during fatigue tests; and potential for multiple-site damage.

3.3 Crack Growth Rate

3.3.1 Basic Theory

Several models of crack growth rate have been proposed. The simplest is the Paris equation:

$$\frac{da}{dN} = C \Delta K^n \quad (1)$$

where a is the half-crack length, N is the number of load cycles, ΔK is the stress intensity range for constant amplitude loading, and C and n are material constants.

For aluminum alloys 7075-T6 and 2024-T3, Paris obtained good correlation with experimental data with $n=4$. However, other researchers have found that with these and other alloys the Paris equation does not always provide adequate accuracy: growth rate is higher than predicted at very high ΔK , and tends to zero at a threshold value of ΔK ; a higher value of n at the lower values of ΔK is sometimes apparent; growth rate tends to increase with increase in mean stress; and at high growth rates the rate is sensitive to

material properties and stress state. The Forman equation allows the effects of mean stress and high ΔK to be taken into account, and is widely used:

$$\frac{da}{dN} = \frac{C \Delta K^n}{(1-R) K_c - \Delta K} \quad (2)$$

where R is the stress ratio and K_c is the critical stress intensity factor.

Nevertheless, where adequate test data exists, it may be more accurate to interpolate on this data, or to use a polynomial best fit, than to use any of the numerous published models for crack growth rate.

3.3.2 Retardation and Other Interaction

During variable amplitude cycling and random loading, crack growth models based on constant amplitude loading have been found to give excessively conservative predictions. The inaccuracy occurs because high amplitude loads reduce the amount of growth during subsequent low amplitude load cycles. The high amplitude loads induce higher compressive residual stresses at the crack tip than the low amplitude loads, such that crack growth must subsequently take place at lower effective values of ΔK . This situation persists until the crack grows through the farthest extending plastic zone of previous high amplitude loads.

3.4 Residual Strength and Critical Crack Length

A cracked component may fail either by unstable crack growth (fast fracture), or by general yielding across the uncracked section of material (plastic collapse). In the former case, the nominal stress at failure, σ_c , is given by:

$$\sigma_c = \frac{K_c}{\beta \sqrt{\pi a}} \quad (3)$$

and the critical crack length, $2a_c$, is given by:

$$2a_c = \frac{K_c^2}{\pi \beta^2 \sigma_L^2} \quad (4)$$

where σ_L is the nominal stress corresponding to design limit load, and β is a factor which depends on component geometry and the crack length.

In the case of plastic collapse, failure occurs when the average stress across the uncracked section of material (ie, the net section stress) has reached a collapse value, σ_{col} , and before K has reached a critical value. This collapse stress lies between the yield strength and the ultimate tensile strength. The associated gross section stress is the collapse strength of the component.

The residual strength, σ_{res} , is the lesser of σ_c and the collapse strength. In the case of a panel of width W with a central crack of length $2a$, the residual strength is either:

$$\sigma_{res} = \sigma_c = \frac{1}{\sqrt{\sec\left(\frac{\pi a}{W}\right)}} \frac{K_c}{\sqrt{\pi a}} \quad (5)$$

or

$$\sigma_{res} = \sigma_{col} \frac{(W - 2a)}{W} \quad (6)$$

From these equations, it is apparent that a collapse failure is more likely if the fracture toughness is high, and/or the crack is short, and/or the panel is narrow. The situation is illustrated by the graph of σ_{res} vs $2a$ in Figure 3-1. On this type of graph, a complete lower bound curve is usually approximated by drawing a tangent to the σ_c curve from the material tensile yield stress, F_{ty} , as shown on the figure.

It is possible to calculate analytically the fatigue crack behaviour, and in particular the residual strength of fuselage skin structure to a reasonable degree of accuracy, provided the behaviour of an unstiffened sheet of similar size is known. However, finite element and mixed finite element and boundary element methods are also used.

A graph of residual strength vs crack length for a stiffened panel is shown diagrammatically in Figure 3-2, which is reproduced from a paper by Vlieger and Broek [6]. This graph illustrates the mutual interaction between the skin and the stringer, and also the tendency of thin aluminum alloy sheet to tear in a stable manner at stresses slightly below the critical stress for fast fracture.

The shaded area represents the area of stable crack growth (tearing). For example, at a crack length of $2a_1$, stable crack growth starts at a nominal stress corresponding to point L. A small amount of growth occurs at constant stress. If the stress is increased slightly, further stable crack growth will occur. Above the critical stress, corresponding to point M on the graph, growth is no longer arrested, and fast fracture occurs.

If stiffened panels with different crack lengths are tested to failure, the stress at the onset of stable crack growth will be given as per the lower bound of the shaded area on the graph, while the failure stress will be as per the upper bound, provided the stringer does not fail first.

The proportion of applied load carried by the stringers depends on the crack length in the skin. It is possible to calculate the hypothetical nominal panel stress at which the stiffener would fail, assuming that the skin remained intact and elastic. This stress is given for different crack lengths by curve g. In typical fuselage structures, this curve might intersect the residual strength curve of the stiffened skin at, say, point H.

If, for example, the panel developed a fatigue crack of length $2a_1$, and the maximum fatigue load or a temporary overload exceeded the stress at point M, fast fracture would occur. It would be arrested at a length corresponding to point N. Further fatigue crack growth could then occur, until the stiffener failed and precipitated total panel failure.

For the purposes of damage tolerance analysis, the residual strength curve is usually plotted by taking the stress at instability with the crack length at the start of stable crack growth (ie, the initial crack length). Calculations of K_{Ic} are also based on these values. The reason for this approach is that there are practical difficulties in assessing the exact crack length at the onset of unstable crack propagation.

4 MSD PROBLEM ANALYSIS

4.1 Scope of Analysis

A comprehensive analysis of all issues relevant to MSD in longitudinal skin splices was undertaken as part of this project. Such an analysis is not available in the literature. The analysis comprised:

- a review of literature on the design and testing of skin splices, fracture mechanics for stiffened sheet structure, fuselage construction, damage tolerance analysis and MSD; and
- tests on several different types of specimen, in order to explore the nature of fatigue cracking in skin splices, to gain experience in the manufacture and testing of splices, and to try out different ways of achieving realistic MSD in a small specimen.

The analysis formed the basis of the project definition in the next section.

In this section, the issues relevant to MSD are developed in a logical manner. Typical splice construction and loading are described. The causes and effects of MSD are analyzed, using actual MSD scenarios as the starting point. Finally, the current state of the art in modelling MSD is reviewed.

4.2 Description of Fuselage Structure

The fuselage of a modern aircraft is a stiffened shell, known as a semi-monocoque structure. Semi-monocoque structure is very efficient, and can withstand limited local failure through load redistribution. A typical example is shown in Figure 4-1. Light frames serve primarily to maintain the shape of the fuselage and to reduce the column length of the stringers for stability in compression. Most commercial transports have a 20 in frame pitch. Heavier frames or bulkheads are used to distribute concentrated loads from wing attachments, etc. The stringers carry axial tensile and compressive loads, and

support the skin. Stringer pitch is typically between 6 and 10 inches. The skin carries tensile, compressive and shear loads due to fuselage bending, torsion and pressurization. The action of frames and stringers typically reduces the actual hoop stress carried by the skin to about 85% of pR/t , which is the theoretical hoop stress in a thin-walled cylinder of radius R and thickness t subjected to a differential pressure p . The operating pR/t stress in commercial transport aircraft ranges between 10.1 ksi (DC-9) and 16.2 ksi (Boeing 747) [7].

In many aircraft, frames are attached to the skin and to the stringers, and have cut-outs through which the stringers pass. This arrangement is sometimes achieved by attaching the main body of the frame to the stringers, and joining the frame to the skin by means of shear ties. The frames are used as a means of slowing or arresting longitudinal cracks. To reinforce this purpose, a tear strap is sometimes bonded or rivetted to the skin under each frame.

Some aircraft, such as the 737, have a 'floating' frame, which is attached only to the stringers, as shown in Figure 4-1. The method of attachment is by a shear clip at each stringer. A crack arrest capability is added by means of circumferential doublers bonded to the skin every 10 inches. These doublers are manufactured by hot-bonding 2 pre-formed sheets of 0.036 inch 2024-T3, and then etching away unwanted material from the inner sheet. Doublers to 'pad-up' the skin thickness are also formed at the stringer locations, resulting in a 'waffle doubler' arrangement. This can also be seen in Figure 4-1. On early 737, the doublers at the skin splice locations were etched away, leaving only a single sheet thickness at the joint. On later 737, the doubler on the outer sheet was retained.

Skin assemblies are made as large as available mill sizes permit, in order to minimize splices. They are assembled to frames and stringers using accurate jigs.

4.3 Review of Splice Construction

Circumferential skin splices are usually single shear butt splices, and are made at bulkheads or slightly heavier frames. Longitudinal skin splices are usually single shear

lap or butt splices, but double shear splices are also used. These categories of joint are shown schematically in Figure 4-2.

For longitudinal application, the lap joint is possibly the most efficient design, and is used widely by Boeing and other companies. Douglas used lap splices on the DC-8 and DC-9, but used a special single shear butt splice with a fingered doubler for the DC-10 and later aircraft. One purpose of this design was to ensure that any fatigue cracking would start in the outer skin, rather than in the hidden doubler.

Some aircraft, including the L-1011 and early versions of the 727, 737 and 747, have longitudinal skin joints that are bonded as well as rivetted. In these cases, the rivets provide a backup in the event of bond failure, but do not necessarily provide an adequate fatigue endurance. The potential advantages of a bonded joint are a reduction in skin thickness without loss in fatigue life, and the exclusion of moisture from the joint. Unfortunately, the reliability of bonded joints has become suspect, and there are not yet efficient and effective means of inspecting them. Consequently, most aircraft built in the last 20 years have used plain rivetted joints, usually with an interfaying sealant. There is further discussion on this topic later.

Longitudinal skin splices are invariably located over a stringer, and are attached to it by one of the rows of rivets in the joint.

The latest production models of most commercial transport aircraft make use of doublers to step up the skin thickness of lap and butt splices. An interfaying sealant is often used to exclude moisture and reduce corrosion.

Skin minimum thicknesses vary between 0.036 inches and 0.071 inches. Wide body aircraft tend to use thicker skin, because hoop stress is proportional to fuselage radius.

The material specification of the skin of almost all commercial transport aircraft is 2024-T3 clad aluminum alloy. A common rivet material in skin splices is 2117-T4 aluminum alloy. This type of rivet can be driven in the as received condition. Rivets in 2024-T4 aluminum alloy are used in some skin splices, where extra strength is required. These must be heat treated to soften them immediately prior to use.

Most lap splices have 3 rows of rivets, but the rivet pitch in each row may vary. In the critical (outer) rows, a pitch of 1 inch is common, but larger pitches of up to 2 inches are also used. Pitches of less than 1 inch may be used in the centre row where a stringer is located. Countersunk rivets are invariably used in medium and long range aircraft, except for repairs. Rivet diameter varies, but $5/32$ inch is common in narrow body aircraft.

4.4 Loading and Stress Distribution of a Longitudinal Splice

4.4.1 External Loads on a Splice

The external loading environment of a longitudinal splice is complex. Under pressurization, a splice is subjected to a combination of hoop tension, longitudinal tension, and out-of-plane bending due to pillowing of the skin. The eccentricity of a single shear lap or butt splice causes additional out of plane bending, known as secondary bending. Secondary bending stresses affect the whole splice and also part of the skin panel, but are greatest along a line roughly tangential to the edges of the outer rivet holes. In some locations, skin panels may also be subjected to significant in-plane shear loads, because of bending and shear loads applied to the fuselage.

For a visualization of the external loads, a circumferential cross-section through 2 skin panels can be represented by the sequence of beams shown in Figure 4-3. The portion of splice between the outer rivet rows is assumed to behave like a beam with twice the depth of the skin. It is joined to the skin by rigid links of length e representing the eccentricity between the neutral axes of the skin and the splice. This eccentricity causes the splice to rotate under tension, and the rotation is resisted by the underlying stringer.

4.4.2 Stress Distribution in a Splice

Within a splice, load is transferred from one sheet to the other primarily through rivet shear, but also through friction between the sheets.

Uneven load transfer, the stress concentrations at the edges of holes, and secondary bending give rise to a complex stress distribution throughout the splice. This is clearly

illustrated by the contours of maximum shear strain for a 2 row lap splice in Figure 4-4. This image was obtained using photoelastic techniques, and is discussed further in connection with specimen design.

The critical location for cracking in the single shear lap splice in Figure 4-3 is at the edges of rivets at point (a), which is at the first row of rivets in the outer sheet of the splice. At this point, there is only a single sheet thickness, and the secondary bending moment and the pillowing bending moment tend to reinforce each other. Also, in most splices, the rivet holes in the outer sheet have a higher stress concentration, because they are countersunk.

4.4.3 Out of Plane Bending

An analytical solution is available [8] for the out-of-plane bending stress in a lap splice under tensile load only. This uses Bernoulli-Euler bending theory to develop soluble differential equations for the representation in Figure 4-3. The normalized bending stress at (a) is given by:

$$\text{Bending Factor } \frac{\sigma_{\text{bend}}}{\sigma_0} = \frac{6e}{t_1} \left(1 + \frac{\alpha_1}{\alpha_2} \frac{T_2}{T_1} \right)^{-1}$$

$$\text{where } T_j = \tanh(\alpha_j m_j) \quad (7)$$

$$\alpha_j = \left(\frac{12 \sigma_0 t_1}{E t_j^3} \right)^{\frac{1}{2}} \quad (j = 1, 2)$$

and where: σ_0 is the applied remote tensile stress
 e is the eccentricity in the lap splice
 t_1 is the thickness of the sheet of interest
 t_2 is the thickness of the other sheet
 m_1 is the free length of the sheets
 m_2 is 1/2 the distance between the outer rivet rows

These variables are also marked on Figure 4-3.

This equation can give only a rough estimate of bending stress in an aircraft splice, since it does not include the effects of pillowing and of stringer torsion.

Its accuracy in the case of a laboratory specimen under tensile load only is also uncertain. Some test data by Schutz [9] suggests that it tends to overestimate the secondary bending stress by 60% to 100%. Schutz found evidence that yielding at the rivet/sheet interface is responsible for the discrepancy. He observed that a permanent extension in the splice developed during the first load application.

It is possible that the equation is more accurate than Schutz claims. Schutz's measurements were made between the rivets, where a reduced clamping effect could be expected to result in a lower value of secondary bending stress than at the rivets.

For the moment, the equation should only be used for rough estimates - preferably with empirical adjustment. It has been used in this manner in the computer model of MSD crack growth discussed in a later section.

Secondary bending stress varies considerably with the geometrical design of the joint. Measurements of secondary bending ratio ($\sigma_{\text{bend}}/\sigma_{\text{local}}$) made by Schutz on fuselage splices range from 0.2 to 1.8. The calculated bending factor, $\sigma_{\text{bend}}/\sigma_{\text{remote}}$ per equation (7), for a 3 row splice with 1 inch row separation in 0.040 inch thick sheet is 0.9.

The combination of secondary bending and the stress concentration at a hole could result in a maximum skin stress close to or above the yield strength under normal operating conditions. For example, using a typical skin stress value of 13.5 ksi, a secondary bending factor of 0.5, and a bi-axial hole stress concentration factor of 2.5, the hole edge stress would be 51 ksi. This stress exceeds the yield strength of 2024-T3 aluminum alloy sheet, which is between 42 and 48 ksi.

Therefore, as a result of secondary bending, it appears likely that crack initiation in a lap splice is in many cases taking place in a high stress, low-cycle fatigue environment.

4.4.4 Effect of Longitudinal Stress

Even though hoop stress is the dominant fatigue loading in a longitudinal splice, longitudinal stress might affect initiation life and crack growth characteristics. The effects of longitudinal stress can only be estimated, since there appears to be no comparative studies in the literature of splice crack growth under uniaxial and biaxial stress. In the presence of longitudinal stress, crack initiation life is likely to be longer, and early crack growth (under the rivet head) is likely to be slower. In the case of cracks that have appeared beyond the rivet head, longitudinal stress is unlikely to exercise much influence on growth rate. These estimates are based on a review by Broek [10] of stress intensity factor solutions for cracks at a hole under biaxial stress.

A relatively inexpensive coupon-type bi-axial test system for splices is currently under development at the NLR in Holland [11], and so some comparative data may become available soon. NLR's method involves applying the loads to a splice through uniaxial composite material. The orthotropy of the material is exploited to avoid the nuisance fatigue failures that can occur in biaxial coupons.

4.5 Actual MSD Scenarios

Several examples of MSD are described well enough in the literature to permit a useful analysis. These are catalogued in this sub-section. While the examples described are on Boeing aircraft, there have been reports of MSD in the longitudinal splices of aircraft by other manufacturers.

4.5.1 Aloha Airlines 737

Some aspects of the Aloha 737 incident were discussed in the context of MSD problem definition.

The failed splice was lost during the explosive decompression, but MSD cracks were found at other locations. The National Transportation and Safety Board (NTSB) reported multiple cracking in several different longitudinal splices on the damaged aircraft. In 2

cases, 16 and 18 consecutive rivet holes had small cracks. The largest crack length measured from the edge of the countersink was 0.27 inches. The longest tip to tip measurement was 0.53 inches. The larger cracks tended to be located centrally between 2 adjacent tear straps (10 inch pitch). It appears from the NTSB report that the joints were fully disbonded in the cracked areas. Photographs of several rivet details are shown in Figure 4-5.

Striation counts provided by the NTSB on a total of 7 cracks at 2 different splices suggest that the net growth rate was fairly uniform, and that cracks would penetrate beyond rivet heads 20,000 to 45,000 cycles after initiation at the edge of the countersink.

At the time of the accident, the aircraft had accumulated 35,960 flight hours and 89,680 flight cycles in 19 years. The maximum pressure differential on the aircraft was 7.5 psi, but this was not reached on every flight.

4.5.2 737 - Unidentified

The 737 photograph in Figure 4-6 was supplied by Swift of the FAA [12]. It shows uniform MSD in 10 adjacent rivets of a longitudinal splice. The cracks are inclined at about 20 degrees to the splice. They are all well advanced, and overlaps have occurred in 7 ligaments. Swift attributes the inclination of the cracks to in-plane shear stress, and highlights the importance of testing splices under realistic loading, including shear.

4.5.3 727 - Unidentified

A bar chart showing the distribution of crack lengths in a 727 splice after 43,400 flights is in Figure 4-7. This was included in a paper by Mayville and Warren [13]. It shows multiple cracks in 3 adjacent frame-bays. There are 5, 9, and 8 cracked rivets respectively in each bay. Most have cracks on both sides, and none are linked. In all 3 bays, the length of the cracks tends to be greater towards the centre of the panel. Cracking in the centre bay is more advanced than in the others.

4.5.4 727 Line #342

The detailed sketch of a splice at stringer 4 (upper fuselage aft of the cockpit) on 727 Line #342 in 1981 is in Figure 4-8. It was included in a paper by Bobo [14]. The cracks are linked almost across the whole frame-bay; nevertheless, there is evidence that the cracks in the central half of the bay were fairly uniform prior to first link-up. The cracks at 5 of the end rivets of the sequence are inclined to the splice. Many of the cracks overlapped before linking.

4.5.5 Boeing Test of a 707

Maclin [15] describes full scale fatigue testing on the fuselages of several Boeing aircraft. During the full scale fatigue tests of the 707, multiple cracks developed in adjacent frame-bays in lap splices. These grew until they eventually linked up across one bay. They subsequently formed flaps that resulted in a controlled and safe decompression.

4.5.6 Boeing Test of 737 Aft Fuselage

Goranson [16] describes Boeing's test on the rear fuselage of a retired 737 with 59,000 service flights. Fatigue cracking started in the splice at stringer S14R in the upper fuselage at 79,000 cycles. A 2-bay crack developed and caused flapping failure at 100,000 cycles. MSD was present in adjacent holes and in adjacent frame bays with a non-uniform distribution of crack sizes. The early distribution of cracks is shown in Figure 4-9. Most of the cracks are in one frame bay, and these are all located to one side of the mid-bay strap. Seven out of 8 rivets between the strap and the frame have one or more cracks. One rivet in the adjacent frame-bay has 2 cracks.

Goranson provides data indicating that early growth was relatively slow, while growth from a cumulative length of about 3.5 inches to a crack across the full 20 inch frame-bay took only 3,500 cycles.

The crack growth curve for a crack at one rivet is shown as the curve on the right of the graph in Figure 4-10. In this case, the splice was assessed as still mostly bonded. A curve to the left shows a faster growth rate for a crack in a disbanded splice in a different aircraft. In both cases, the starting length (tip to tip) is recorded as just larger than the diameter of the rivet. Growth to a length of 0.7 inches (tip to tip) took a further 42,000 cycles and 24,000 cycles respectively. It appears that a link-up occurred at a length of between 0.70 and 0.75 inches in each case. Maclin assesses the damage detection periods prior to link-up as 6 years and 12 years for the disbanded and mostly bonded cases respectively, based on usage of 3,000 flights per year.

4.5.7 Boeing Test of 747-100SR

A 747-100SR with 20,000 service flights was tested by Boeing for a further 20,000 pressurization cycles. The locations of the minimum gauge splices on this aircraft are shown in Figure 4-11. Maclin reports that an initial crack was detected at 21,500 cycles in the splice at stringer S14R, in the front fuselage below the right side window of the cockpit. This crack eventually linked up with other small cracks and grew to a length of approximately 6 inches by the time the test was concluded.

A sketch of the lead crack and other nearby cracks at the end of the test is in Figure 4-12. The cracks are clustered in the central third of a frame-bay. All are linked except for cracks at 2 rivets at one end, and an overlapped crack at the other end.

Figure 4-12 includes a plot of the cumulative crack length against total aircraft pressure cycles. The cumulative growth rate at the end of the test appears to be about 0.5 inch per 1000 cycles.

Maclin states that the inspection program ensures detection before or just after link-up. He assesses the damage detection period before link-up as 5 years, based on usage of 2,000 flights per year.

4.5.8 Boeing Test of 747-400 Front Fuselage

A front fuselage taken from the production line was pressure cycled by Boeing to determine some of the fatigue and damage tolerance characteristics of the latest production configuration. Test results are summarized by Maclin for the period up to 60,000 cycles.

Cracks initiated in at least 3 locations between 38,000 and 40,000 cycles. They appeared later and grew more slowly than on the 747-100SR. Three locations are discussed below.

At stringer S44R, just aft of the gear opening, 4 successive bays initiated cracks between 38,300 and 53,000 cycles. The crack configuration and cumulative crack growth curve for the first bay after 60,000 cycles are in Figure 4-13. In the first bay, cracks grew at several locations, and the first link-up occurred at about 57,500 cycles. It seems to have been followed by rapid growth and several other link-ups, which took the cumulative crack length from 6 inches to 17 inches in less than 1,000 cycles.

At stringer S44L, well aft of the gear opening, cracks appeared in adjacent bays at 40,000 and 42,500 cycles respectively. In the forward bay, crack growth appears to have followed a generally similar pattern to that at S44R: multiple initiation followed by rapid cumulative growth after the first link-up. In this case, the average growth rate was faster, than at S44R, and a single crack formed across the whole bay after 50,000 cycles. In the adjacent bay, cracks developed at 5 rivets near the closest end of the bay, but the area was repaired at 49,500 cycles. The forward bay was repaired shortly afterwards at 50,000 cycles. The crack configurations just prior to the repairs and the cumulative crack growth curve are in Figure 4-14. It is unfortunate that the repairs had to be done, since it would have been very instructive from a damage tolerance standpoint to see the effect of unlinked MSD ahead of a large lead crack in a full scale test.

The final MSD configuration described by Maclin, at stringer S6R, consists of cracks at 10 rivets in one bay and 15 rivets in an adjacent bay. A sketch is in Figure 4-15. The cracks are biased towards a severed tear strap. It is not clear when or how the strap was severed, or whether the frame at the same location was also severed. The cracks at 8 adjacent rivets in the aft bay overlap each other and appear to have grown directly into rivet holes instead of linking with the oncoming crack. The reasons for this unusual

growth pattern are unclear, because the cracks at each end of this sequence follow a typical, more longitudinal path. The cumulative crack growth curve for the aft bay is also in Figure 4-15, and indicates that cracks initiated at 40,000 cycles. There is no evidence of the rapid acceleration after first link-up seen in the other curves described earlier.

4.6 Discussion of MSD Scenarios

The above case studies allow several general observations to be made about fatigue cracking in longitudinal fuselage skin splices of Boeing aircraft.

MSD can occur in isolated frame-bays and also in several adjacent frame-bays.

In lap splices, the most highly stressed row of holes in the outer skin is usually the only location of cracking.

The initiation periods and growth periods for the earliest cracks are approximately as follows:

a. Appearance of first cracks beyond the rivet heads:

727	less than 40,000 cycles
737	between 60,000 (estimated) and 79,000 cycles
747-100	21,500 cycles
747-400	38,000 cycles

b. Additional period to first link-up:

737	between 17,500 (estimated from Goranson) and 42,000 depending on the state of the bond
747-100	10,000
747-400	between 9,000 and 20,000

c. Additional period to crack across one frame-bay:

737	3,500 cycles after 2 link-ups to a 2-bay crack
747-400	1,000 (relatively uniform cracks)
	10,000 (non-uniform cracks)

Taking normal fatigue scatter into account, these figures indicate that MSD can be expected to occur within the original design lives of all the aircraft types listed: 60,000 cycles for the 727; 75,000 cycles for the 737; and 20,000 cycles for the 747.

They also indicate that the period of detectable growth to failure can be less than 21,000 cycles (approximately 7 years) on the 737 and less than 10,000 cycles (approximately 10 years) on the 747. The probability of detection during this period will depend on the frequency of inspection and the inspection technique.

If the cracks are not detected prior to the first link-up, it appears that failure in the 737 and 747 could occur in the next year of normal operations. Consequently, cracks should be detected earlier than the first link-up, if at all possible. This requirement poses economic and practical problems in view of the large number of splices that are involved (Figure 4-11), and the small size of the cracks.

On the 737, the hidden growth period appears to be roughly similar in magnitude to the visible growth period to first link-up: between 20,000 and 40,000 cycles. The above figures suggest that a similar relationship exists on other Boeing aircraft.

There is a tendency for cracks within a particular frame-bay to appear at clusters of rivets away from frames, rather than at isolated rivets spread across the full width of a frame-bay.

Clusters tend to expand as new cracks appear at rivets near each end. The expansion continues until the first link-up, after which the cluster is likely to develop quickly into a continuous lead crack.

The relative uniformity of crack lengths in a particular cluster varies. Figure 4-12 shows a cluster with non-uniform crack lengths, while Figure 4-7 shows a cluster with more uniformity.

It seems likely that the size of a cluster and the uniformity of the crack lengths prior to first link-up are functions of the hoop stress distribution and the scatter in fatigue crack initiation life. This hypothesis implies that, for example, a flat stress distribution across a frame-bay together with low scatter in initiation life would promote the development of a large cluster of relatively uniform cracks.

Under normal cyclic loading, link-up of cracks within the same frame-bay appears usually to occur in one ligament at a time. A 'zipper' failure of several rivets during one load cycle has not been observed, but could conceivably occur in the event of an overload.

Several link-ups can occur in quick succession, ie, within 1,000 cycles, as indicated in Figure 4-13 and Figure 4-14. In these 2 cases the cumulative crack length grew from about 6 inches to 18 inches in less than 1000 cycles after the first link-up.

A rapid series of link-ups like these is potentially more serious in a large cluster of cracked rivet holes, than in a small cluster. In the latter case a full bay crack might not develop for perhaps 10,000 cycles after the first link-up (Figure 4-12).

Cracks can develop at an angle to the splice (Figure 4-6, Figure 4-8 and Figure 4-15). In many cases, the cracks tend to curve back to a longitudinal trajectory, rather than continuing at an angle. The oblique growth might be due to an applied in-plane shear load, and indicates that such conditions might complicate the prediction of crack growth rates, crack trajectories and residual strength.

4.7 Failure Modes

Two categories of fuselage failure can be defined:

a. Flapping failure

Flapping failure typically occurs when a crack curves through 90° and arrests to form a large flap. Controlled decompression then relieves the stress that was driving the crack without immediate further damage to the fuselage. Such failures have reportedly occurred in service, and Maclin reports several such failures during fatigue testing of the 707 and 737. A photograph of a 2-bay flapping failure that occurred during full scale testing of a 737 is in Figure 4-16.

b. Uncontrolled Decompression

Uncontrolled (explosive) decompression failure occurs when the growth of a lead crack becomes unstable and continues longitudinally. There is an accompanying loss of rigidity in the fuselage structure, and a rapid release of strain energy. Aerodynamic forces are then likely to promote further, catastrophic damage. The in-flight failure of the Aloha Airlines 737 took this form.

Both types of failure can occur at normal operating pressures when the lead crack in a longitudinal splice spans one or more frame-bays (20 inches), but flapping failure is less likely to occur in wide body fuselages, for reasons given below. The critical crack length varies with the situation and the aircraft, but is unlikely to be much longer than 2 frame-bays. The most damage tolerant aircraft are able to withstand a 2-bay crack with a broken central frame. MSD in adjacent frames can reduce this performance.

In the case of flapping failure, the crack deviates from the longitudinal direction because the reaction to bulging of the crack under internal pressure changes the stress state at the tip. Kosai and Kobayashi [17] theorize that these changes bias the coalescence of micro-cracks at the tip in an off-axis direction. The predominant changes at the tip are:

- an increase in the longitudinal stress;

- the creation of a mode II stress state which promotes crack growth in the circumferential direction.

Circumferential frames and straps promote crack curving by reinforcing these effects, and also by decreasing the local hoop stress.

Experimental data cited by Swift [18] indicates that fuselage radius and skin thickness also determine whether a lead crack will curve and cause flapping failure rather than continue longitudinally until dynamic failure occurs. An increase in fuselage radius elevates the hoop stress, and so a longer crack is needed to generate enough longitudinal stress at the crack tip to cause flapping. An increase in skin thickness inhibits crack bulging and a longer crack is again needed to cause flapping. Swift states that flapping within a length equivalent to 2 frame-bays tends not to occur in unstiffened shells of a similar radius to wide body transports if the wall thickness is greater than 0.063 inches.

Narrow body aircraft require relatively thin skin to withstand the hoop stress, and so most have been designed to promote flapping in the event of failure.

Wide body aircraft typically have a radius of 120 inches compared to 75 inches for narrow body aircraft, and require thicker skin; consequently, controlled flapping failure is less likely to occur.

MSD ahead of the lead crack can prevent a flap from forming. It is believed that this is the reason why the Aloha 737 experienced explosive decompression instead of a flapping failure. Also, Maclin says that Boeing have demonstrated on a full scale test fixture that dynamic failure of a 737 fuselage will result from a one bay crack if there is a 'generous amount of MSD' in the adjacent bay. The effect of MSD on flapping failure is discussed further in the next subsection.

If flapping does not occur, the lead crack will continue along the splice, and MSD ahead of a lead crack can be expected to have some effect on residual strength; from a practical standpoint, the important question is whether or not the MSD will significantly reduce the critical length of the lead crack.

Research on this question is proceeding, but only a limited amount of useful data has so far been published. The basic requirement is for experimental and theoretical residual strength curves for a variety of crack configurations in real structure. No test data for representative structure has yet been published. Coupon test data are of limited value, because residual strength is only of concern when lead cracks are long. The issue is explored further in the next section.

The issues of flapping failure and arrest of longitudinal cracks is pursued in more detail in the next subsection. On a general note, if the earlier information on MSD scenarios is viewed in the context of the 2 potential failure modes just discussed, the following conclusions can be drawn:

- a. Fuselage failure precipitated by MSD is unlikely to occur without the prior development of a long lead crack.
- b. MSD ahead of a lead crack will only significantly reduce the critical length of a lead crack if either the MSD is substantial (ie, aggregate crack length is large), or it is very close to the lead crack.
- c. In view of a and b, the state of cracking in the most advanced frame-bay is the primary determinant of the remaining life to failure.
- d. Therefore, the potential role of MSD in promoting the rapid development of a lead crack within a frame-bay is a key issue, and should be investigated in detail.

4.8 The Effect of MSD on Crack Arrest and Flapping Failure

Since the effect of MSD on crack arrest and flapping failure appears to have been one of the key issues in the failure of the fuselage of the Aloha 737, it is worth exploring it in more detail.

The circumferential stiffeners in a transport aircraft fuselage slow down or temporarily stop the growth of longitudinal cracks in one or both of 2 ways: by reducing the hoop stress at the crack tip; and by enhancing the natural tendency for longitudinal cracks to

turn and create a flap in the skin. These mechanisms apply to cracks which are propagating sub-critically under fatigue loading, and also to cracks which are propagating dynamically after reaching critical length.

In the case of a crack that continues longitudinally without flapping, the presence of linked MSD in the next frame-bay will depress the residual strength curves of both the skin and the frame.

As a lead crack approaches a frame under normal operating fatigue loads, events will be dictated by the strength of the frame. If the frame holds, the lead crack will approach close to the MSD before linking up. If the MSD has contributed to severe overstress in the frame, the frame will fail, and the lead crack will probably grow in an unstable manner even if there is still a large separation from the MSD. In either case, the link-up between the lead crack and the MSD will produce a much longer lead crack and will probably precipitate failure of the frame. Subsequent events will be dictated by the strength of the next frame, and the scope of any additional MSD.

This type of scenario is illustrated by Figure 4-17, which shows the residual strength diagram calculated by Tong et al [19] for a symmetrical 2-bay crack in stiffened sheet approaching linked MSD which is immediately beyond the next stiffener (ie, frame). The central stiffener is broken. Curves applicable to 2 different lengths of linked MSD are shown (1 and 1.5 inches), together with the residual strength curves for the same situation without MSD. At typical skin stresses of 10 to 16 ksi, link-up between the lead crack and the MSD in this scenario would take place in a stable manner when the lead crack was close to the frame. On the graph, link-up is indicated by a (hypothetical) reduction to zero in the residual strength, and a discontinuity in the curve. The discontinuity results from the instantaneous extension of the lead crack to point B (or point C in the case of the longer MSD crack). The stability of the structure at this point would depend on the ultimate strength of the frame.

Tong's data is for hypothetical structure, but it is interesting to note that the net effect of the MSD would be to reduce the residual strength of the structure with a 2-bay crack by about 30% from a value of 25 ksi at point A on the graph to 17 ksi at point C.

In the case of flapping failure, as discussed previously, the crack deviates from the longitudinal direction partly because the reaction to bulging of the crack under internal pressure increases the longitudinal stress at the tip relative to the hoop stress.

In longitudinal splices, rivet holes and MSD ahead of the lead crack increase the local hoop stress, and thereby promote continued growth along the rivet row. Changes in trajectory typically occur only in long cracks which are close to a frame or strap, and take the form of a fairly tight curve away from the splice, as shown in the photograph in Figure 4-16.

There are other factors in a splice which help to inhibit crack curving:

- a. The stringer and the other rivet rows tend to clamp one side of the crack, and prevent it from bulging. The potential increase in longitudinal stress at the crack tip is thereby reduced.
- b. The dynamic pressure of the airstream on the protruding edge of the crack also tends to prevent full bulging of the crack.
- c. As mentioned in the last subsection, an increase in fuselage radius or skin thickness will tend to increase the crack length required to initiate a curved trajectory and flapping failure. A larger radius will elevate the hoop stress, and require more crack bulging to cause curving. A thicker skin will be less prone to bulging.

MSD and these other factors probably all played a part in preventing flapping failure in the Aloha incident.

4.9 Crack Initiation Characteristics

4.9.1 Definition of Important Initiation Characteristics

The observations and conclusions on crack initiation characteristics in this section are based on a series of preliminary coupon tests, and on data in the literature, notably Jarfall [20], Schijve [21], Hartman and Schijve [8], Schutz [9], Lee [22], Ozleto and Coyle [23], Speakman [24], and Pelloux et al [25].

For the purposes of this thesis, crack initiation is defined as the development of a dominant micro-crack.

The period of growth until this crack becomes economically detectable will vary with the initiation site, the geometry of the splice, and the inspection technique. Cracks typically initiate in hidden locations: at the edge of the rivet hole, or at the faying surface of the sheets. The techniques currently used for inspecting fuselage skin splices are visual and high frequency eddy current. These cannot be relied on to detect cracks until they are through the sheet and clear of the rivet head.

The initiation characteristics of interest in the context of MSD in longitudinal splices are the initiation site, the initiation life, and the distribution in initiation life:

- a. The initiation site is of interest because it influences the choice of inspection method and frequency.
- b. The initiation life, together with the undetectable growth period determine the inspection threshold.
- c. The distribution in initiation life determines the size and uniformity of the cluster of MSD, and hence the detectable period prior to fuselage failure.

Each of these characteristics is analyzed in the following subsections.

For practical purposes, as the MSD scenarios described earlier have shown, the initiation and life characteristics can usefully be assessed within the confines of one frame-bay at

a time. Every splice and frame-bay must be considered, in view of the potential consequences of MSD in any one frame-bay. However, many frame-bays are probably equivalent in this context, since the detailed design of the splice, the dominant stresses and the corrosion protection are usually similar throughout large portions of a fuselage.

4.9.2 Initiation Site

Cracking will generally start at the faying surface at one of the outer rows of a splice, but the exact location of the initiation site depends on: the extent of the clamped area around the rivet, on the magnitude of the bending stress, and on the type of rivet:

- a. If there is firm clamping over a finite area, such as would occur with a well driven dome-head rivet, cracks will tend to initiate at the edge of this area ahead of the rivet. Fretting plays a major role, and so several initiation sites may be seen before a dominant crack develops.
- b. If there is weak clamping, such as occurs with most countersunk rivets, cracks will initiate at the edge of the rivet hole. Bending will promote initiation forward of the centre-line of the rivet row (the term forward implies further away from the splice). The stress concentration around the (loaded) hole will counteract this tendency by promoting initiation in line with or slightly behind the centre-line of the rivet row.
- c. If a countersunk rivet is used, the critical row will probably be in the outer sheet of the splice, because this is where the stress concentration caused by the countersink occurs. The term 'outer', used here and elsewhere in the thesis, signifies the outer sheet with respect to the aircraft.
- d. If a dome-head rivet is used, the critical row might be in either sheet, depending on the net bending stress realized at the rivet:
 - (1) In the outer sheet, the bending moments due to eccentricity and pillowing are additive, but the maximum bending stress is attenuated by the compliance of the manufactured head.

- (2) In the inner sheet, the effects of eccentricity and pillowing oppose each other, but, if the rivet is rigidly installed, the stiffness of the driven head results in less attenuation of the net maximum bending stress.
- (3) The combined effect of (1) and (2) promotes crack initiation in the inner sheet, if the thickness is less than 0.064 inches [21]. Otherwise, initiation tends to occur in the outer sheet.

The countersunk rivets used in longitudinal splices provide weak clamping, and so cracking generally initiates at the edges of holes in the outer sheet.

Initiation in the outer sheet of a splice is beneficial from an inspection standpoint, and can be further assured by appropriate detail design of the splice. For example, in the DC-10, the pad-up doublers are given a finger-shaped profile to ensure that any fatigue cracking will start in the outer skin.

When dome-head rivets are used, inspection and damage tolerance requirements should be considered carefully. For the reasons given above, cracking can start in the hidden inner sheet. While the dome-head rivets might enhance the fatigue endurance of the splice, extra internal inspection of the fuselage might be necessary to ensure safety. Such inspection is much more expensive than external inspection. Dome-head rivets are used in the splices in some short range transport aircraft, and are commonly used in splice repairs.

4.9.3 Initiation Life

Data presented earlier indicate that crack initiation in splices may be occurring early in the life of the airframe of Boeing aircraft. Detectable cracks have developed at 50% of airframe life in some cases, and striation counts on the Aloha 737 show a long hidden growth period.

In common with many fatigue situations, the initiation life of splice cracks is extended by any measures that reduce the maximum stress, reduce the stress amplitude, and avoid or reduce stress concentrations. These include:

a. Reducing the local (hoop) stress by:

- reducing cabin pressure differential;
- using pad-up of the skin at the splice;
- using more or stiffer frames and straps.

b. Reducing the load transfer at the rivet hole by:

- increasing the number of rows;
- increasing the rivet diameter and reducing pitch;
- increasing friction load transfer (surface treatment, clamping force and area);
- using flexible rivets to equalize the load transfer at the inner and outer rows (eg, 2117-T4);
- bonding the faying surfaces.

c. Reducing hole-edge maximum stress and/or tensile stress amplitude by:

- avoiding a knife-edge countersink;
- avoiding marks, eg, drill marks along the length of the hole;
- increasing the interference between rivet shank and hole after installation (close tolerance hole, extra driving force);
- cold-working the holes prior to rivet installation, using, eg, FTI split sleeve techniques, and Douglas stress-coining techniques.

d. Reducing the secondary bending stress at the critical rivet row by:

- increasing the separation between the outer rows of the splice;
- reducing the applied (hoop) stress;

- using a rivet installation that will attenuate the peak bending moment (refer to earlier discussion);
 - attaching the splice to a stringer to inhibit rotation.
 - minimizing any extra eccentricity introduced by pad-up of skin thickness, doublers, etc.
- e. Reducing the pillowing bending stress at the rivet row by:
- reducing cabin pressure differential;
 - using pad-up of the skin at the splice;
 - locating the stringer further away from the critical row.
- f. Minimizing fretting by:
- using appropriate interfaying surface treatments;
 - minimizing clamping pressure (use less driving force).
- g. Minimizing stress concentrations due to corrosion by:
- using appropriate materials and surface treatments (alclad, anti-corrosion paints);
 - sealing the splice to prevent moisture ingress (polysulphide sealant);
 - providing good drainage and ventilation in the surrounding structure;
 - maintenance.

In view of the number of parameters that affect crack initiation life in a splice, the difficulty of estimating peak stresses around the rivets, and the uncertainties associated with fretting, the fatigue life of splices must invariably be verified by test. Ideally, a combination of coupon tests and full scale tests is used. Full scale fatigue tests in Western Europe are now continued to at least twice the planned life of the airframe. Corrosion tests must be done on a small scale, and it is difficult to simulate service conditions; therefore some uncertainty as to durability will always remain.

For preliminary design purposes, it is possible to estimate fatigue life from S-N data for splices of approximately similar design, if the specific data is available.

The early crack initiation in Boeing splices is probably the result of very high local stresses caused by stress concentration and out-of-plane bending. This potential scenario was discussed earlier in connection with the loading environment of a splice. Such characteristics may not be unique to Boeing aircraft.

4.9.4 Distribution in Initiation Life

If the distribution of initiation life has a high scatter in relation to the period of growth to first link-up, there will be a relatively long period of growth from first link-up to fuselage failure - 10,000 cycles or more. Only one or 2 cracks may develop in a frame-bay before the first link-up. The subsequent rate of growth of the lead crack will be comparatively slow as it negotiates successive substantial ligaments.

If there is low scatter in initiation life in relation to the period of growth to first link-up, there will be a short period of growth from first link-up to fuselage failure - about 1000 cycles. A large cluster of MSD with uniform crack lengths will develop in the frame-bay in question. After the first link-up, the lead crack will progress quickly through successive ligaments until it occupies the full frame-bay.

Low scatter in initiation life in relation to the crack growth period to first link-up is clearly undesirable, but many fuselage splices may fall into this category. As discussed earlier, despite relatively low applied hoop stress in the range 10 to 17 ksi, the peak stresses in a single shear countersunk splice are likely to be close to or above the yield strength of the material. In such situations, fatigue crack initiation life will be short, and initiation scatter, as defined, will be relatively low.

An example of low initiation scatter in a laboratory specimen with a high level of secondary bending is in Figure 4-18. The specimen comprised only 2 rows of 1/8 inch diameter dome-head rivets in 0.032 inches thick 2024-T3 sheet. It failed soon after the first crack appeared, because of lateral distortion allowed by the pin-joined mounting arrangement. Nevertheless, crack initiation is evident at most of the rivet holes.

Some splice locations on Boeing aircraft appear to be susceptible to a near worst case scenario. As mentioned earlier, the initiation life of Boeing splices is relatively short, and fairly uniform MSD has developed in some cases.

Douglas splices appear to have a longer initiation life, but it is not known whether the initiation scatter is high or low. A DC-9 which had accumulated 66,500 flights in 15 years of service was tested by Douglas to 208,000 flights. A tear down inspection showed no cracks at the rivet holes.

The situation can be improved by any of the measures discussed earlier for extending the crack initiation life of splices, since these will also tend to promote greater scatter in the initiation life. Ideally, measures that substantially increase structural weight should be avoided.

4.10 Crack Growth Characteristics

Many of the characteristics of MSD crack growth have already been described during the discussion of MSD scenarios, failure modes and initiation characteristics, and can be summarized as follows:

- a. There is a period of relatively slow growth under the rivet heads followed by faster growth beyond the rivet heads. In 737 splices, the hidden growth period is roughly similar in magnitude to the visible growth period to first link-up. A similar relationship seems likely on other Boeing aircraft.
- b. Clusters of cracks develop in one or more frame-bays, away from frames and straps. The cracks may be uniform or non-uniform to varying degrees. The clusters may be broad, ie, stretch across most of the width of a frame-bay, or they may be narrow, ie, consist of only 2 or 3 cracked rivet holes in close proximity.
- c. Eventually 2 adjacent cracks link-up to form a lead crack. This lead crack grows much faster than other, unlinked cracks, and tends to dominate subsequent crack growth within a frame-bay. The link-up of remaining cracks and uncracked holes

proceeds one ligament at a time until there is a linked crack spanning most of the frame-bay.

- d. As would be expected, this link-up process is rapid in the case of uniform crack lengths, and relatively slow in the case of non-uniform crack lengths.
- e. The combination of a large MSD cluster and uniform crack lengths results in a short period between the first link-up and the development of a lead crack across the frame-bay.
- f. On the other hand, either small clusters or non-uniform cracks will result in a relatively long growth period after first link-up. On the 747 the growth period from first link-up to failure is between 1,000 and 10,000 cycles depending on cluster size and uniformity.
- g. The frame-bay crack is slowed down temporarily by circumferential frames and straps, but otherwise grows rapidly until it reaches critical length, which is typically one to 2 frame bays for a single lead crack.
- h. If there is MSD in the next frame-bay, the lead crack will grow faster and its critical length will be shorter. These effects may be negligible for MSD at an early stage. They may be pronounced if the MSD in the adjacent bay has formed a significant lead crack, or if it exists as a large cluster of uniform, advanced cracks.
- i. In narrow-body aircraft, the lead crack may turn in a circumferential direction and form a skin flap. This may occur before or after the crack has become unstable, and may occur at either the first or second frame encountered. The change in direction is likely to occur abruptly and close to the frame.

While the development of potentially dangerous uniform crack lengths may depend primarily on initiation characteristics, there is a possibility that uniformity is promoted by

a catch-up mechanism during the growth phase of MSD. There are 2 potential catch-up scenarios:

- a. The equalization of the lengths of cracks on opposite sides of the same rivet hole;
- b. The equalization of the tip-to-tip lengths of cracks at 2 adjacent rivet holes.

Park and Atluri [26] have found theoretical evidence of a catch-up mechanism for cracks at the same rivet hole. They found that a slow, steady catch-up occurred over the whole growth period. When the effects of rivet expansion were included in the model, an additional catch-up mechanism was active for short cracks - for the period where both cracks are within the range $0.2 < c/r < 1.46$, where r is rivet hole radius, and c is the crack length measured from the edge of the hole.

Any catch-up at different rivet holes (catch-up scenario b) would have to result from a reduction in load transfer at the cracked rivet hole. It cannot result solely from crack interaction, since the known stress intensity factor solution for unequal collinear cracks [27] indicates that the longer crack will always grow faster than the shorter crack. While interaction between a crack and a nearby rivet hole can be expected to promote earlier initiation and faster growth of a crack at the hole, these events would not represent a catch-up mechanism.

More experimental and theoretical analysis is needed to determine whether a significant catch-up mechanism exists during the growth phase of MSD.

The shape of the leading edge of a splice crack is significant, particularly in the early stages of growth. The shape of the crack front will be influenced by the initiation site, the local geometry and the local stress state, in particular by the amount of out-of-plane bending.

In a single shear splice the bending stress is high, and so the crack front at the hidden, faying surface of the sheet may be well in advance of the visible crack tip. This phenomenon will be referred to as tunnelling. In preliminary testing for this project, tunnelling of up to 0.1 inches beyond the edge of the rivet was detected in specimens

with high bending stress. The tunnelling was visually evident from distortion in the sheet, and was confirmed using low frequency eddy current.

After the crack has grown through the thickness of the sheet, the leading edge adopts an approximate quarter-elliptical shape, and the amount of tunnelling reduces. In a 2 or 3 row splice in 0.040 inches thick 2024-T3 sheet the tunnelling is approximately equal to the material thickness, ie, the hidden crack front is about 0.040 inches ahead of the visible crack front. Fractographic examination indicates that the amount of tunnelling in through cracks remains fairly constant.

The fact that tunnelling reduces away from the rivet is evidence that out-of-plane bending is higher close to a rivet than mid-way between rivets.

The effect of out of plane bending on overall crack growth rates is not known, and research is needed in this area. While bending has a major effect in reducing crack initiation life, there is no obvious reason to expect that it will increase the overall crack growth rate substantially - the average stress intensity factor through the thickness of the sheet due to bending is approximately zero. Nevertheless, it remains possible that the stress intensity factor gradient through the thickness might in some way influence the crack growth rate, and so tests should be done.

4.11 Load Redistribution During Crack Growth

The stress in the structure immediately surrounding a large crack or multiple smaller cracks will increase with aggregate crack length. This load transfer will not be diminished by loss of pressure due to leakage. Even when considerable crack bulging takes place, loose insulation material is likely to act as a bladder to maintain local pressure loading on the structure.

For analysis purposes, the increased average stress in frames and straps is usually expressed as a stress concentration factor, analogous to the stress concentration factor associated with notches.

The stress concentration in the frames and tear straps ahead of a crack in a 737 panel is shown in Figure 4-19. The stress concentration increases with crack length, and may depend on crack location relative to the stringers. For a crack across a full frame-bay, close to a stringer, the stress concentration factors are 2.3 in the strap and 1.6 in the frame.

These figures can be taken as roughly indicative of the effective stiffness of adjacent structure in the presence of a splice crack across a full frame-bay.

4.12 Modelling of Initiation and Growth of MSD

4.12.1 Scope of Models

One challenging aspect of the analysis of MSD is that all stages of crack development, from initiation to critical length, must be considered:

- a. Crack initiation is important because initiation patterns affect cluster size and uniformity, and hence the growth period to failure.
- b. Early growth under the rivet head is important because the growth rate influences the inspection threshold.
- c. The visible period of growth is important because growth rates, crack interaction, and link-up characteristics affect the period of detectable growth to failure, and so influence inspection frequency and technique.
- d. Finally, the late phase of growth, involving large lead cracks and interaction with other, less advanced MSD in adjacent frame-bays, is important because it defines the fail-safe characteristics of the fuselage structure.

A large range of potential crack configurations must be modelled, and to do so efficiently requires several models of limited scope.

For the purposes of this thesis, the scope of MSD models will be described with reference to 3 phases of growth: initial, intermediate and final. These classifications are for descriptive purposes only, and are not defined strictly:

a. Initial Phase

The initial phase of crack growth includes initiation and growth of a through crack just beyond the rivet head. The crack is strongly influenced by local geometry and rivet/sheet interaction.

b. Intermediate Phase

The intermediate phase covers the visible phase of multiple crack growth until a lead crack of several inches has developed. While cracks are small, local effects due to the rivet hole and the countersink are likely still to be present, but might be neglected. However, the wider stress distribution within the splice must be taken into account. After several link-ups, significant changes in stress distribution probably occur in neighbouring skin panels. These might also be neglected, or the boundary stresses of the model might be updated periodically using a separate, larger scale model.

c. Final Phase

The final phase covers crack growth from the link-up of several rivets in one frame-bay until fuselage failure. Local effects due to rivet geometry and rivet/sheet interaction are negligible, but the changes in stress distribution in neighbouring skin panels must be taken into account. A full stiffened shell model of the fuselage may be needed. An elastic-plastic analysis may also be warranted. The effects of crack bulging must be included in the analysis of crack trajectory.

A review of the literature indicates that only NASA is doing substantial work on initial phase modelling.

Several researchers are working on intermediate phase models. Some of these use iterative finite element techniques that involve remeshing at each step in crack growth.

Others use hybrid techniques involving combinations of new stress intensity factor solutions, finite element techniques and boundary element techniques. Only one full attempt at modelling intermediate MSD with Rooke's superposition and compounding method has been made.

NASA, Boeing, and BAe Airbus have reported on efforts to model the growth of large longitudinal cracks. Only BAe has described work specifically on final phase MSD. Considerable resources are required to model final phase MSD, and so it could be several years before comprehensive results are published. Previously discussed work by Tong on residual strength and by Kosai and Kobayashi on crack curving, will contribute to accurate modelling of final phase growth.

A brief review of analytical models of MSD crack growth is given in the following subsections.

4.12.2 Modelling of Initiation and Initial Phase Growth

The loading on a sheet in the vicinity of a rivet hole is complex and depends on a number of factors already discussed in connection with initiation characteristics.

Both the loading and the crack growth during the initial phase are 3-dimensional, and so a 3D finite element model is needed to represent them fully. However, it may be feasible to use a 2D approach in some cases.

Stress concentration factors and stress intensity factors are available in the literature for 2D situations and some 3D situations.

Newman [28] reports that NASA will shortly publish 3D stress concentration data for straight-shank and countersunk holes subjected to tension, bending and pin loads. This data has been generated using a finite element stress analysis code, FRAC3D, which uses 20-node isoparametric elements. Typical results for a countersunk hole under tension are given in Figure 4-20. This data indicates that for a typical countersunk hole the stress concentration factor, K_t , ranges from about 2.5 at the sheet surfaces to 4.1 at the shoulder of the countersink.

The FAA and NASA [29] are also working on 3D stress intensity factor solutions for corner cracks at holes. Three dimensional finite element analysis has been used to generate solutions for a wide range of crack sizes, crack shapes and hole radii. A compendium of this type of data will eventually be published.

Despite the 3D nature of initial phase crack growth at a countersink, Dawicke et al [30] have succeeded in modelling it using a 2D finite element code, FRANC. The key features of the model from the point of view of initial phase growth are as follows:

- the countersink is modelled by a plain hole with the average diameter of the countersink;
- in order to simulate the expansion of the rivet in the hole, the rivet is modelled as an annular mesh subjected to internal pressure;
- the crack trajectory is set to follow the direction $K_{II}=0$;
- bending is neglected.

The initiation site, crack trajectory and growth rate compared favourably with tests on single rivet coupon specimens. Rollers were used to inhibit secondary bending in these tests.

It is possible to avoid dealing directly with the initial growth phase by regarding initiation as occurring when cracks appear beyond the rivet heads. Sampath and Broek [31] took this approach, by using fatigue test data on comparable splices to predict initiation statistically.

4.12.3 Modelling of Intermediate Phase Growth

Dawicke et al's 2D model, introduced in the last subsection, covers the intermediate phase as well as the initial phase. It uses a two dimensional layered finite element approach. Each layer (sheet) is represented by a separate 2D finite element mesh, with a separate database for each layer. The layers are connected by linear spring elements representing

net 2D rivet and hole behaviour. Rivets are modelled as a separate layer with linear springs between the layers representing the rivet and hole. The springs can connect the rivet and hole at one node to represent pin loading, or at several nodes to represent bearing stress. The rivet itself is modelled as an annular mesh subjected to internal pressure, in order to simulate rivet expansion. Joint eccentricity and secondary bending are not represented.

Triangular and quadrilateral elements with quadratic shape functions are used. Dawicke notes that these elements result in relatively simple and robust remeshing algorithms.

LEFM assumptions are used to calculate the stress intensity factors at the crack tips. The singularity at the crack tip is modelled by shifting one of the mid-side nodes of a triangular element to the quarter position; this is now a widely used technique that reduces the number of elements needed to obtain an accurate stress distribution around the crack tip.

Automatic remeshing is performed by deleting elements near the crack tip, and then remeshing the area, taking into account the predicted new location of the crack tip.

Dawicke's procedure for performing a crack propagation analysis on a graphics work station is as follows:

- a. Interactively input initial cracks in the mesh.
- b. Calculate K_I and K_{II} for all crack tips.
- c. Using the calculated K values, predict the crack growth directions and increments. For this purpose Dawicke programmed da/dN vs ΔK relationships obtained experimentally using centre-cracked tension (CCT) specimens of the same material and thickness. The crack trajectory was assumed to be in the direction of $K_{II} = 0$.
- d. Remesh around the new location of the crack tip.
- e. Repeat steps b to d.

The predicted crack growth rates were compared with data from tests on narrow lap splice coupons held by one rivet. Secondary bending was restrained by rollers. Significant scatter is evident in Dawicke's coupon test data, but average values appear to agree with predictions. However, the range of ΔK covered in the tests was very limited (10 to 11 ksi $\sqrt{\text{in}}$), and not fully representative of a true MSD environment. The experimental and theoretical crack trajectories appear to agree closely.

Park and Atluri [26] have recently developed an analytical solution for the stress field due to multiple collinear cracks with loaded crack surfaces, and have incorporated it into an efficient computer model of intermediate crack growth, which can be run on a PC. The solution must be applied iteratively, and a finite element stress analysis of the splice is also required. However, because no remeshing is involved, the technique is relatively fast. The analysis technique is known as the Finite Element Alternating Method (FEAM).

The model does not account for bending stress, but accounts for the following:

- biaxial stress;
- rivet flexibility;
- adhesive bonding of the joint;
- variation in hoop stress across the joint.;
- contact stress distribution between rivets and holes;
- prestress of the hole due to rivet expansion.

The process of computation is summarized as follows:

- a. A coarse finite element analysis of the complete splice is performed to determine rivet loads in the cracked row for a particular crack configuration.
- b. The cracked row is modelled as a strip of loaded holes subjected to hoop stress and bypass stress on opposing edges. Rivet loads from step a are applied as distributed loads.
- c. This strip model of the cracked row is analyzed 2 ways:
 - (1) The stress field in the absence of cracks is determined analytically.

- (2) The stress field due to multiple collinear cracks with loaded crack surfaces is determined analytically. The crack loads are chosen as the reverse of those obtained in c(1) at the crack lines.
- d. The stress fields of c(1) and c(2) are summed. This gives a stress field with zero stress at the crack surfaces (as is the case in reality) but with incorrect boundary conditions. The Schwarz alternating technique is then used to correct the stress intensity factors until they converge to accurate values. This process involves reversing the residual boundary tractions, and using the finite element method and the new collinear crack solution to calculate the associated adjustments to the stress intensity factors. It is not excessively costly because the same mesh is used for all finite element calculations.
- e. The Forman crack growth equation is used to grow the cracks from the given starting configuration. Growth rates are assumed to be constant for a specified interval.

Park and Atluri have not yet compared their model with experimental results, but have used it to find evidence of the existence of a catch-up mechanism between cracks of different lengths at the same hole. This was discussed earlier in connection with MSD crack growth characteristics.

Dawicke and Newman [32] (NASA) are developing an alternating indirect boundary element (AIBE) technique for modelling MSD crack growth. This technique involves iterative use of the boundary element method. It is still at the development stage, but comparisons of theoretical and experimental results for a row of unloaded holes with unequal cracks have been close.

Sampath and Broek [31] have used the superposition of known stress intensity factor solutions to predict MSD crack growth in fuselage lap splices. The model is 2-D but does take into account out of plane bending. It also predicts crack initiation using statistical data from tests on similar splices. The model is not compared directly with experimental results, but has predicted crack distributions and growth periods that are reported to compare well with those observed on actual aircraft.

Relevant features of interest in Sampath and Broek's model are as follows:

- a. A run is terminated when the cracks have linked across 10 rivet holes out of 20 in a frame bay. The model includes 5 frame bays, but there is no attempt to model linked cracks extending beyond the bounds of one frame bay.
- b. Initiation at a particular hole is made dependent on the local stress level and on the scatter in available fatigue life test data for the splice in question.
- c. The local stress is determined by a finite element analysis of a frame bay, taking into account the effect of frames, tear straps and stringers.
- d. A Paris law is used for crack growth, and scatter in growth rate is accounted for by using a Weibull distribution for the value of the constant C .
- e. The values of ΔK for each crack configuration are computed by superposition of known solutions for tension, wedge loading and bending of a single crack in a finite plate. The method of accounting for the interaction of cracks is not fully described. It is believed to involve subdivision of the cracked row into finite strips, each containing one crack; the associated edge correction factor is assumed to have an effect equivalent to that of crack interaction.

4.12.4 Modelling of Final Phase Growth

The modelling of final phase MSD is an extension of intermediate phase modelling. The behaviour of large splice cracks formed by MSD are strongly influenced by the non-linear response of the surrounding stiffened shell structure. The consequences of stiffener failure must also be considered. Since a finite element model of a large fuselage section is costly to run, a global/local strategy must be used for modelling; this involves using a global (fuselage) analysis to define and update the boundary conditions for a local analysis of one or more skin panels.

Several modelling levels could be used to represent MSD growth fully. Starnes [33] reports that NASA is evaluating 5 levels, ranging from a relatively coarse global shell

model for determining the internal load distribution and global response of the shell, to a highly refined local detailed stress analysis model for determining crack growth characteristics from fracture mechanics principles.

To deal with final phase MSD growth, the approach must account for:

- non-linear load/displacement response;
- large deformation behaviour;
- effects such as pillowing (important in determining crack trajectory related to flapping failure modes);
- changing crack trajectory (requires an adaptive mesh).

Starnes, Miller of Boeing [34] and Cummins of BAe Airbus [35] have reported on recent efforts to model final phase crack growth. Of these, only BAe has so far reported on the modelling of final phase cracks in splices.

Cummins describes the simulated growth of MSD in 3 frame bays, starting from fairly uniform cracks at all rivet holes, and ending with 'net section yield' in one frame-bay followed by fast fracture of the 3-bay splice. Details of the model are not provided.

Miller gives a description of Boeing's fuselage model. There is a narrow body version typical of the 727, 737 and 757; and there is a wide body version typical of the 747, 767 and 777. It is implemented in NASTRAN, and obtains direct output of modes I, II, and III stress intensity factors by means of NASTRAN's solid crack tip element CRAC3D.

Boeing's model is summarized as follows:

- the skin is idealized as quadrilateral thin plate elements;
- the solid crack tip element interfaces with the skin elements by means of rigid bar elements around its boundary - this approach gives accurate values for modes I and II, but it has not been possible to verify the accuracy for mode III;
- stringers are idealized as beam elements having 6 degrees of freedom;

- floating frames are idealized as a single beam element between stringers, while shear-tied frames are idealized as a series of beams and plates;
- rivetted tear straps are idealized as separate plate elements, while bonded tear straps are idealized as thicker skin elements;
- stringer clips are idealized as tapered beam elements;
- fasteners are modelled using rigid body elements for rotational and axial displacements, and spring elements for transverse displacements;
- differential pressure is simulated by point loads at the nodes of the skin elements;
- longitudinal pressurization stress is applied via the boundary conditions;
- a typical model has 26,000 degrees of freedom.

Boeing have compared predicted stress distributions, crack growth rates and crack trajectories with those from full scale tests of a narrow body fuselage and a wide body fuselage. The predictions are for skin cracks, and agreement is generally good. Predictions for splice cracks have not yet been made.

Figure 4-21 illustrates a typical Boeing narrow body structure, and a comparison of measured and predicted hoop and longitudinal stresses is in Figure 4-22. Hoop stresses match closely, but measured longitudinal stresses are lower than predicted, possibly for reasons related to fixture design rather than errors in the analysis.

The features of a typical Boeing wide body structure are summarized in Figure 4-23. There are some significant differences with the narrow body structure:

- the frames are attached to skin via shear ties between stringers;
- there are no tear straps, but there are .025 inch pad-ups under frames and stringers;

- the skin is .075 inch thick instead of .036 inch.

A comparison of measured and predicted hoop and longitudinal stress distributions for the wide body panel is in Figure 4-24. There is fairly close agreement on hoop stress distribution. Measured longitudinal stresses are lower (by 20%) than the predicted values (the reverse of what is shown in Boeing's illustration).

5 PROJECT DEFINITION

5.1 Overall Research Needs

This project definition follows from the definition and analysis of the MSD problem in previous sections. It describes the overall research work needed to resolve the issues associated with MSD, and identifies the work chosen for this thesis.

Research work in several areas related to MSD in fuselage splices is needed to improve the safety and economy of existing and future commercial transport aircraft. These include:

- a. Experimental and theoretical research to quantify the effects of various parameters on the initiation patterns and growth characteristics of MSD.
- b. The development of semi-empirical computer models of all phases of MSD growth, to help in analyzing the fail-safe characteristics and maintenance requirements of existing and future fuselage designs.
- c. The development of an inexpensive coupon-type testing system for the initial and intermediate phases of MSD; this is needed both for general research purposes, and as a cost-effective supplement to full-scale testing for the verification of crack initiation and growth characteristics.
- d. The development of efficient techniques for inspecting fuselages for cracks at rivet holes.
- e. The development of specific regulatory guidelines based on the foregoing research.

5.2 Research Topics Chosen for This Thesis

Current international research efforts appear not to include the development of an inexpensive but realistic coupon-type test system for MSD. In view of the potential economic benefits in the fatigue testing of splices, it was decided to attempt such a development.

To complement this work, it was also decided to attempt the development of a computer program to predict intermediate phase MSD crack growth. The aim was to base the program on a technique pioneered by Rooke [36], which involves the compounding as well as the superposition of known stress intensity factor solutions. A program based on this technique might not be as accurate as a finite element program, but is potentially much simpler, quicker and less expensive to use. As mentioned in Section 4, Sampath and Bræk have produced a program based on superposition, but without using Rooke's compounding technique. The approach followed was substantially different from the one they used.

These particular 2 tasks were chosen for several other reasons:

- they would have practical relevance to aircraft manufacturers and airlines;
- they would not duplicate work by other researchers;
- they would be achievable within the limited resources available;
- if successful, they would open up further important avenues of research;

5.3 Concept Specification for the Test Specimen

From the information gained during the MSD problem analysis, including the testing of prototype specimens, it was concluded that a realistic simulation of MSD within a frame-bay could probably be performed in a standard uniaxial load frame using a specially designed specimen. The important parameters that would have to be simulated, and the test system features needed to perform the simulation inexpensively have been included in a concept specification for the coupon-type test system.

The concept specification is as follows:

- a. The hoop stress and out-of-plane bending stress distribution experienced by the specimen at the critical row of rivets in the uncracked condition must match distributions obtained from strain gauge measurements of the aircraft or from an accurate finite element model of the aircraft.
- b. The specimen must simulate the redistribution of these stresses (load transfer to surrounding structure) that occurs as cracks progress.
- c. The geometry and local conditions at the critical rivet row in the specimen must match those at the critical row on the aircraft, but full replication of the aircraft splice is not necessary. Important parameters to match for similitude include bypass stress (rivet load), rivet forming load (clamping force), interfaying treatments (bonding, primers, sealants), and hole preparation.
- d. The specimen must provide enough usable splice length to permit MSD to be initiated naturally and to grow under realistic conditions until several rivet holes have linked up. It is not intended to represent a splice length greater than one frame-bay, typically 20 inches.
- e. The timing of the appearance of cracks and the crack growth characteristics of the specimen must resemble full scale test data for a comparable splice. In particular, there must be similarity in the extent and uniformity of the MSD cluster(s) just prior to the first link-up.
- f. It must be possible to batch-produce the specimen at low cost, so that repetitive testing for statistical purposes is feasible (when needed).
- g. The mounting scheme must be such that specimens between 10 and 20 inches in width can be tested using standard load frames.
- h. The load-frame should not have to meet rigorous standards of stiffness and alignment, so that it can be locally manufactured, using beams, hydraulic

components and instrumentation that are likely to be available already in the test department of an aircraft company. (Desirable requirement)

- i. The specimen concept must be such that it is possible to design a specimen quickly and inexpensively to match a specific aircraft situation.
- j. An standard methodology (eg, finite element techniques) should be defined, which can be used inexpensively and with confidence in the design and analysis of future specimens. (Desirable requirement)

The foregoing test system concept includes all the significant loading conditions associated with MSD within a frame-bay, except longitudinal stress and in-plane shear stress. The complexity of the test system would be considerably increased by attempting to apply these loading conditions realistically.

As discussed earlier in connection with splice loading, the effects of longitudinal stress can only be estimated, since there are no comparative studies in the literature of splice crack growth under uniaxial and biaxial stress. In the specimen, crack initiation life is expected to be shorter than on the aircraft, and early crack growth is expected to be faster. The growth rates of cracks that have appeared beyond the rivet head are expected to be represented fairly accurately.

The effect of static or alternating in-plane shear stress on MSD crack initiation and growth is potentially important, but actual MSD scenarios suggest that it may only be a significant loading condition in a minority of splice locations. Therefore, the omission of shear stress in the test system does not diminish its usefulness.

Some consideration was given to the use of the Poisson effect to generate biaxial stress with a uniaxial load. Conceptually, this could be done by bonding uniaxial graphite-epoxy composite to the sides of the specimen, and then anchoring the composite to a rigid compression frame. Graphite-epoxy is very stiff, and so, with good detailed design, a significant Poisson effect could probably be achieved. The full Poisson stress, ie, 33% of applied hoop stress, would not be realized. However, this might not be a concern, since, according to strain gauge measurements made by Phillips and Britt on a 737 [37], the longitudinal skin stress near a stringer may be less than 25% of hoop stress.

6 DESIGN OF TEST SPECIMEN

6.1 The Key Design Problem

The concept specification for the full coupon-type test system was given in the section on project definition. The specimen is the only component in the system which is intended to be the subject of special research and development. The other components are commonly available equipment in an aircraft structural test department.

The key problem is finding a specimen concept that will simulate the aircraft stress environment in the cracked condition as well as the uncracked condition. Such a specimen has not been designed before. Schmidt [1] highlighted the problem recently in commenting on the activities of the FAA-sponsored international committee on Widespread Fatigue Damage; he noted that coupon specimens are not representative of crack growth after the occurrence of initial damage, because the increase in net stress is unrealistic.

6.2 Alternative Concepts

Several designs of specimen were considered in arriving at the optimum approach reflected in the concept specification.

The simplest form of lap splice specimen consists of 2 rivetted sheets clamped between plates at each end and pin-joined to the load-frame. This type of specimen is commonly used by aircraft manufacturers for fatigue life tests on longitudinal splices. Over 30 examples from a test program by one company were examined, and only a few had developed multiple cracks. In most cases, cracks had started at one of the first 3 rivets on one side of the specimen. These had formed a lead crack which grew through other rivets holes until the specimen failed.

Tests on preliminary specimens, conducted as part of this project, confirmed this tendency to premature failure from one side of the specimen.

The main reasons for this shortcoming in existing types of splice specimen are as follows:

- a. An asymmetric crack distribution causes one side of the specimen to be more compliant, and to extend more than the other. This difference in displacement results in untypically large stresses at the crack tips, which promotes further asymmetry in the crack configuration.
- b. The Poisson effect creates a biaxial stress state in the central area of the specimen, which tends to reduce hole-edge stress concentrations slightly.
- c. The end rivets are adjacent to the free edges of the specimen, and so have less clamping effect than the other rivets; consequently, crack initiation is likely to occur earlier at the end rivet holes.

Clearly, while this type of lap splice specimen might be adequate for 'safe-life' testing, or for establishing an inspection threshold, it is not useable for investigating either the initiation or the growth characteristics of MSD.

Several relatively straightforward modifications were considered, in order to make it more representative of a real splice:

- a. Crack initiation at the outer rivets can be delayed by, eg, cold working the holes, avoiding countersinking, using rivets or bolts with strong clamping force, and reinforcing the ends of the splice with rivetted doublers.
- b. Cracks that have developed on one side of the specimen can be discouraged from growing too quickly to the edges by placing crack-stoppers at the edges. Rivetted doublers and partial side straps can be used for this purpose as well as for delaying crack initiation at the end rivets (see a above). Side straps also serve to off-load the end rivets of the splice, and thereby delay crack initiation.
- c. The ends can be rigidly clamped instead of pin-joined, so that there is less difference in displacement between the cracked side and the uncracked side of the specimen in the event of an asymmetric crack distribution. Wide hydraulic clamps in a substantial load-frame would be needed to achieve the necessary rigidity.

The feasibility of modification a was confirmed by testing a preliminary specimen with dome-head rivets, in which the 2 outer pairs of rivets in the critical row were reversed. As mentioned in the problem analysis, with dome-head rivets, cracks tend to initiate in the sheet containing the driven head; hence, initiation was delayed at the reversed rivets. MSD initiated in the central area of this specimen, and a 4 inch linked crack developed before the specimen failed.

Unfortunately, modification a does not avoid the development of untypically high net section stresses.

Modification b was found to be more difficult than might appear at first sight. Preliminary specimens were manufactured to test 2 different approaches to the modification, but these were inconclusive, since the modifications became the source of nuisance cracks. Photographs of the specimens are in Figure 6-1 and Figure 6-2. It was concluded that careful detailed design, including the use of cold worked holes, would be needed to achieve success.

It was discovered that a generally similar approach had been used in a test program sponsored by the FAA, and that cold-worked holes and dome-head rivets had been used to avoid nuisance cracks. A full test report has not yet been published.

Modification b, like a, was abandoned because it does not avoid the development of untypically high net section stresses, and because a better approach was discovered.

The third modification was rejected because it would require the use of expensive equipment that is not widely available.

6.3 Photoelastic Study

A birefringence photoelastic study was done on one preliminary specimen:

- to determine if such a study could yield useful information on splices;

- to investigate the stress distribution within and around a single shear lap splice;
- to determine the stress transferred to rivetted side doublers of the kind mentioned in the last subsection, and the stress concentrations associated with such a design.

The specimen was the same type as that shown in Figure 6-1. It was 7 inches wide, and was fabricated with 0.040 inch thick 2024-T3 sheet and 2 rows of 5/32 inch diameter 2117-T4 dome-head rivets. Four doublers 1 inch x 5 inches were rivetted along the edges of each sheet; these did not continue across to the other sheet. A 1/8 inch diameter rivet was located at the most highly stressed end of each doubler, to reduce the peak stress at this location.

Sheets 0.010 inch thick of birefringent plastic were bonded to each side of the splice, leaving clearance holes for the rivets. The specimen was then statically loaded to 6,000 lb and viewed through a polariscope.

The polariscope images were digitally processed into contour plots of maximum shear strain, using a system being developed by Komorowski of the NRC [38].

The contour plots of maximum shear strain are shown in Figure 4-4. Since the applied stress was uniaxial, these contours also give a rough indication of the tensile stress distribution in the splice.

Secondary bending stress is high along the edge of the clamped area. It can be seen from the contours that this edge runs approximately along a tangent to the rivet holes. In the critical sheet, the stress along this line reduces to a low value, indicating that the bending stress mid-way between rivets is approximately 80% of the far field stress.

6.4 The Optimum Concept

These early studies provided insight into the problem of specimen design and into the fatigue behaviour of rivetted joints, but they offered only partial solutions to the key problem.

The optimum solution was developed from these initial ideas, and consists of a pin-joined specimen with continuous straps bonded along each edge, and with profiled doublers bonded across its width above and below the splice. This concept has been implemented as the 10 inch wide specimen shown in the assembly drawing GE301-1 in Appendix A and the photographs in Figure 6-3. The width was dictated by available equipment and facilities, and should be considered as the minimum for MSD tests. The main features of this design are as follows:

- a. The bonded side straps fulfil several functions:
 - (1) They avoid the development of nuisance cracks at the edges of the specimen.
 - (2) They prevent cracks growing through to the edge of the specimen.
 - (3) They moderate the increase in net section stress as cracks grow, and thereby simulate the load transfer characteristics of a fuselage. The stiffness of the straps can be adjusted to achieve the desired results.
 - (4) They make it unnecessary to load the specimen through rigid clamps in a rigid load-frame. A pin-joined mounting scheme (as shown in the drawing) and a locally manufactured load-frame can be used, to minimize costs.
- b. The doublers have 2 functions:
 - (1) They control the nominal stress distribution across the width of the splice. Their position, profile and stiffness can be adjusted to produce the required distribution.

- (2) They provide extra thickness of material under the clamping plates, and thereby minimize the possibility of nuisance fatigue cracks in this area.
- c. The bending stress experienced by the sheet at the critical rivet row can be varied by changing the spacing between the 2 outer rivet rows of the splice.
- d. The rivet load and bypass stress at the critical rivet row can be varied by changing the diameter and stiffness of the rivets at the other rows, and by changing the number of rivet rows in the splice.
- e. The mounting scheme comprises 2 pairs of reusable clamping plates and close tolerance loading pins. The pins are held by fork-ends (not shown in the assembly drawing), and a universal joint is incorporated in the load-frame, to ensure that the specimen automatically aligns itself with the applied load.

The clamping plates and pins, though reusable, are an integral part of the specimen, and their detailed design includes several important features:

- a. The loading holes in the specimen and plates are a close tolerance fit with the pin. Their position and the squareness and parallelism of the edges of the plates and specimen are dimensioned to allow the specimen and plates to be aligned quickly and accurately.
- b. The clamping bolts are close-pitched and positioned close to the loading edge of the plates, in order to minimize fretting of the specimen. The corners of the loading edges of the plates have a small radius for the same reason.
- c. The bolt pattern ensures good clamping and maintains the alignment of the close-tolerance loading holes with the minimum number of bolts.
- d. The material used for the plates is AISI 4340 steel quenched and heat treated to an ultimate strength of 125 ksi. This choice was made to obtain infinite fatigue endurance and reasonable toughness within an acceptable size and handling weight. The close tolerance fit between the plates and loading pins also serves

this purpose by minimizing stress concentration at the holes (which can exceed a factor of 10 with a loose fit).

6.5 Finite Element Model of Specimen

Finite element analysis was used to develop the optimum specimen concept, and this experience was used to define a simple but effective model that can be used for tailoring the specimen to specific aircraft requirements. Using this model, it will be possible to design a specimen of any width to match closely the uncracked and cracked hoop stress distributions in a particular frame-bay in an aircraft. The model was implemented using DISPLAY 3 and NISA 2 commercial software, but could be implemented in any other comparable software.

In the finite element model, sheet metal (including the clamping plates) is represented by 8-node quadrilateral thin shell elements of varying thicknesses, and rivets are represented by omni-directional spring elements attached at each end to a node of a shell element. The stiffness of the springs is determined from a semi-empirical formula: alternatives are contained in a textbook by Niu [7] and in a paper by Swift on the design of the DC-10 [39]. The 2 sheets of the lap splice are constructed separately in the same plane, and are then connected by the spring elements. Finally, the 2 sheets are brought together until they overlap, and the nodes at the side straps are merged to complete the connection. The process is illustrated in Figure 6-4. Cracks are represented by leaving selected nodes of shell elements unmerged.

The model is not required to reproduce the stress distribution exactly in the immediate vicinity of rivets and cracks, and so the assumptions implicit in the model can be stated as follows:

- a. All bond lines are assumed to be rigid.
- b. Out-of-plane bending is omitted. The splice is treated as entirely 2-dimensional¹, ie, as having no eccentricity. This treatment is only possible mathematically, and is not a physical representation.

- c. The effect of rivets on the membrane stress distribution are assumed to be accurately represented, except in the immediate vicinity of the rivet, by springs acting in the plane of the model, as described above.
- d. The effect of cracks on the membrane stress distribution is assumed to be accurately represented, except in the immediate vicinity of the crack tips, by disconnecting (leaving unmerged) the nodes of adjacent elements. There is also an unavoidable approximation in this approach, in that the lengths of cracks cannot be reproduced exactly - the accuracy depends on the distance between nodes.

The stress distribution obtained with this model for the uncracked specimen loaded to 10,000 lb is shown in the contour plot in Figure 6-5. This contour plot shows only one sheet in half the splice, because of symmetry. The precise area covered can be determined from the diagram of the finite element model in Figure 6-4.

The general appearance of the stress contours around the rivets is similar to that of the photoelastic images in Figure 4-4. For a more direct comparison, maximum shear stress contours from the finite element model were also plotted, and are in Figure 6-6. The general similarity remains. These comparisons indicate that the simplified representation of rivets is acceptable for modelling the general stress distribution in the specimen.

The stress contours ahead of the splice (1 to 2 inches away from the rivets) display a negative stress gradient in the direction of the splice. In an aircraft, such a gradient would be expected, since the skin panel pillows out under the action of cabin pressurization. In this context, the specimen contours are generally similar to contours obtained by NASA [33] from a fuselage finite element model.

The mesh used in the model is visible as white lines in Figure 6-6. These are slightly distorted to give an exaggerated indication of nodal displacements. The size and grading of the mesh were chosen to provide a short run time and acceptable accuracy. Accuracy is discussed in the section on test results. The run time on a Silicon Graphics Iris workstation is 0.5 to 1.5 hours, depending on whether a 1/2 model or full model is used. This run time and the relative simplicity of the model would allow the design of a specimen to be done iteratively in a few days at reasonable expense in CPU time.

An example of the stress distribution predicted for a cracked specimen is in the contour plot in Figure 6-7. The crack configuration approximates one experienced during testing. There is one large, linked crack on one side of the specimen, running through rivets #2 to #4, and one smaller crack through rivet #5. There is a low stress area on each side of both cracks, and a high stress area at every crack tip.

Comparing this stress distribution with the uncracked distribution in Figure 6-5, it can be seen that, while the net section has reduced by over 35%, the nominal stress (ie, the net stress) in the uncracked area has risen by only about 10%. For this comparison, the nominal stress was read from the contour plot along a line 1 inch from the critical rivet row. This line is fairly close to the rivets, but outside the area of major stress perturbations caused by the (simulated) rivets.

The uncracked and cracked stress distributions predicted by the finite element model were eventually checked against strain gauge measurements from an actual specimen. The agreement was good. The comparison is discussed in more detail in the section dealing with test results.

6.6 Detailed Design of the Specimen

The materials, splice geometry and stress distribution of the first specimen built to the full specification were chosen to be roughly representative of current generation commercial transport aircraft. This approach was considered to be sufficient for proving the concept, design methodology and manufacturing procedures, and was also expected to yield useful test data on the characteristics of MSD.

The design of the splice in the specimen was based on typical splice construction, as reviewed earlier in the MSD problem analysis. It consists of 0.040 inch thick 2024-T3 clad sheet and 5/32 inch diameter 2117-T4 countersunk rivets. The countersink formed a knife-edge in the outer sheet of the splice. The rivet pitch and row spacing are both 1 inch.

In order to avoid extra complexity at the proof-of-concept stage, it was decided to omit pad-up doublers and interfaying sealant. The 2 skin sheets are simply rivetted together

in the splice area without any surface treatment. In this regard, the Design resembled a disbanded splice in an early 737.

The hoop stress distribution across the specimen was chosen to represent distributions measured by Boeing in narrow and wide body aircraft [34]; these are reproduced in Figure 4-22 and Figure 4-24. The Boeing measurements were taken between stringers, but the stress gradients are assumed to be similar to the gradients near a splice located on a stringer. The Boeing data shows some sharp discontinuities; it was assumed that these occurred because there were insufficient strain gauges to form a smooth plot.

The wide body (probably 747) stress distribution is curved across the whole frame-bay, with a peak value at the mid-bay location. The narrow body (probably 737) stress distribution is relatively flat in the central area between the frames, probably because the mid-bay tear strap is preventing full pillowing of the skin. In both cases, the edge stresses are about 90% of the mid-bay stresses.

It was not possible to represent either of the Boeing stress distributions exactly, since the splice in the specimen is only half the width of a frame-bay. The aim was to produce a central section with a flat distribution, and a gradual curve down to about 90% of maximum value.

The stress distribution across the specimen, as predicted by the finite element model is shown as a graph in Figure 8-5. It is relatively flat over the central 4 inches of the specimen, and then curves gradually downwards to a value at the edge which is 80% of the maximum value. At the outer rivets the value is 90% of maximum. The graph applies to a line 1 inch ahead of the critical rivet row. Stresses along this line are considered as the nominal stress applied to the splice. It is not possible to measure a meaningful nominal stress any closer to the splice, because the perturbations caused by the splice are too strong.

Based on the finite element results, it was concluded that the specimen would represent the hoop stress gradients in Boeing aircraft closely enough for a valid study of MSD.

6.7 Manufacture of the Specimen

A full set of engineering drawings of the specimen and its components are in Appendix A. Tolerances have been designed to ensure similarity between specimens for statistical purposes, while minimizing the cost of manufacture.

A fully detailed manufacturing procedure for the specimen excluding clamping plates is in Appendix B. It contains the results of considerable process development effort. The metal to metal bonding process was proven by special lap shear tests. Also, the bonds have so far survived over 400,000 cycles in full specimen testing without showing any signs of degradation.

The heat treatment and preliminary machining of the clamping plates were performed by a subcontractor. Hardness tests and final machining were done at the NRC.

6.8 Strain Gauge Installation

An engineering drawing of a typical strain gauge installation is in Drawing GE310-1 in Appendix A. This installation requires 20 strain gauges, and will provide adequate coverage for most purposes. Stresses in areas that cannot be instrumented, may be obtainable from the finite element model.

Strain gauges are placed on opposite sides of the sheet at each location, so that bending stress and membrane stress can be inferred from the readings.

Seven locations provide the nominal stress across the splice. These lie along a line 1 inch from the critical rivet row. Finite element modelling has shown that this is an appropriate location for reading nominal stress. A photograph of these strain gauges on one side of the specimen is in Figure 6-8.

Three locations provide the maximum secondary bending stress in the splice. These lie along a line which is tangential to the inner hole of the countersinks in the critical row, and are positioned mid-way between 2 rivets. Photoelastic analysis, described earlier, showed that this would be an appropriate location for measuring secondary bending stress.

It is also the standard location chosen by the AGARD Panel on Fatigue Rated Fastener Systems [40]. Their choice was based on work by Schutz and Lowak [9] on secondary bending in splices.

One of each pair of these secondary bending gauges must be placed inside the splice. To provide clearance for the hidden gauge, a shallow groove 0.020 x 0.25 inches is milled into the other sheet, as defined on Drawing GE310-1. The hidden strain gauges must be installed and wired before the specimen is rivetted together. The installation design provides adequate internal and external clearances for the gauge and associated solder tabs and wiring. The installation is designed to withstand fatigue testing, but it is likely that cracks in the specimen will eventually damage all 3 pairs of secondary bending gauges.

The clearance groove for the hidden gauge is expected to have only a small effect on the stress distribution in the critical sheet of the splice, since it is located at the free, low-stressed end of the opposite sheet.

The form and fit of the hidden strain gauge installation has been tested in a specimen. Unfortunately, glue migrated into the splice and clearance groove during curing, and so the strain gauge could not be functionally tested. The MSD test described in this thesis was conducted on another specimen which was only equipped with external strain gauges.

7 EXPERIMENTAL EQUIPMENT AND PROCEDURE

7.1 Overview of Test System

The specimen, including clamping plates, was mounted in a locally manufactured frame, and was loaded by a single, servo-controlled hydraulic jack. The arrangement could be subjected to accurate constant amplitude loading of at least 8,000 lb with a stress ratio of 0.02 at frequency of 5 Hz without risk of instability. The maximum capacity of the system used was 13,000 lb in tension.

Stresses in the specimen were measured via a 20-channel strain gauge multiplexing and display system.

Crack growth was measured using a travelling microscope. The initial appearance of cracks was detected visually. Eddy current inspection equipment was also investigated for initial detection, and for measuring cracks that were not through the thickness of the sheet.

Crack configurations were recorded photographically with the help of a Diffracto Sight (D Sight) aided-visual inspection system. Crack growth measurements were made from the negatives using a travelling microscope, in order to evaluate D Sight as an inspection method for fuselage skin cracks.

Still photographs of crack link-up were taken just before and during failure of the ligament, with the help of a video-microscope.

Data reduction was performed on Silicon Graphics and SUN workstations, using specially prepared Mathematica programs.

7.2 Load-Frame

The specimen, hydraulic jack, and load cell were mounted in a simple, sturdy framework of I-beams, as shown in the photograph in Figure 7-1. This framework was assembled

locally, and can be adjusted in length and width to suit different sizes of specimen. There were ball joints at top and bottom of the frame, to ensure that the load was applied in line with the centre-line of the specimen. The lower one was clamped to hold the jack firmly after the assembly had been aligned under load. The upper one was left free permanently.

The specimen was held in fork-ends, which also helped in ensuring correct alignment of the load.

The load cell was installed above the specimen.

High strength studs were used to connect the various components together. These were pre-tensioned using spiral washers, to prevent fatigue failure.

The maximum tensile capacity of the jack was 13,000 lb, with a supply pressure of 3,000 psi. The flow rates of the servo-valve and the flexibility of the specimen limited the frequency of load application to 7 Hz. To ensure stability, a frequency of 5 Hz was used for all testing.

The jack was supplied with oil from large-capacity hydraulic pumps, via a hydraulic service module (HSM). The HSM contained high and low pressure supply valves, as well as an accumulator to minimize supply pressure fluctuations.

The servo-valve on the jack, the supply valves on the HSM, and the load-cell output signal were electrically connected to a control system, which automatically regulated the applied load. The control system also shut down the hydraulic pressure or provided warnings in the event of pre-defined malfunctions.

All principal hydraulic and control system components were supplied by MTS Corporation, and so there were no difficulties associated with system integration.

The control system was the MTS FlexTest DSC System, which is fully described in the User's Manual [41]. The system was calibrated in accordance with this manual. The shunt calibration approach was used for the load-feedback circuits.

The load programs were created using the program generation facilities provided by MTS T/RAC software. These are fully described in the User's Manual [42].

The FlexTest Digital Servo Controller (DSC) is a multi-channel system. It provides complete system servo control for up to four test stations, including:

- load control and stroke control;
- feedback selection;
- program source selection;
- limit detection;
- set point and span adjustments;
- servo loop adjustments, in either a manual or auto tuning mode;
- transducer calibration adjustments.

The controller consists of 3 chassis mounted in a single cabinet. A separate PC loaded with special FlexTest software is required to operate the controller. The arrangement is shown as a block diagram in Figure 7-2, and a photograph is in Figure 7-3. The 3 chassis are as follows:

- a. The Real Time Active Controller is a multiple processor unit that provides all memory, control logic, and A/D and D/A conversion functions.
- b. The Analog Chassis provides the transducer signal conditioning and valve drive functions that link the Controller to the mechanical system.
- c. The Hydraulic Control Panel contains the power outlets to the HSM supply valves, and manual controls for activating and deactivating the loading system. It also contains monitoring circuits, independent of the Controller, which can be programmed to flag warnings or shut down the mechanical system.

Only one test station was activated for the test. The system was tuned manually using a duplicate specimen. While the basic load programs were created separately, the upper and lower peak loads were readjusted in real time using the set point and span controls. This was necessary for fine tuning, but also because the peak values of load feedback tended to drift slightly as the oil warmed up.

7.3 Strain Gauge System

A strain gauge system was designed and assembled that controlled and recorded the output of 20 independent strain gauges: 14 x 350 Ω for the nominal stress locations, and 6 x 120 Ω for the bending locations. The lower resistance for the bending locations was dictated by size constraints.

The bridge circuits were assembled locally and connected to a Hewlett Packard 3497A Data Acquisition/Control Unit. The Hewlett Packard system and the cabling to the gauges can be seen in the photograph of the load frame in Figure 7-1. A photograph of the strain gauge installation on one side of the specimen is in Figure 6-8.

The system was programmed to scan gauge output on demand, and record the data in hard copy. It was also programmed to supply graphical output of processed stress data, so that events could be monitored during the test.

Data reduction to report quality graphs was performed after the test, using a Mathematica program.

7.4 Measurement of Crack Growth

Cracks were measured directly on the specimen using a 60X travelling microscope, which could be read accurately to 0.0005 inch. The photograph in Figure 7-4 shows the microscope in position for measurements.

An unsuccessful attempt was made to use marker bands as a method of measuring crack growth. This method involves interrupting the normal fatigue loading and applying an equal or lower maximum load at a different stress ratio for a limited number of cycles. The crack surfaces are examined under a microscope after the test, and the distance between marker bands is measured. For practical purposes, the bands must be made easily visible under an optical microscope without compromising the validity of the crack growth data. This goal has been achieved by other researchers on 0.080 inch thick 2024-T3 sheet. It was not achieved on the 0.040 inch sheet used on this project, and the

attempt was abandoned. Thinner sheet allows plane stress conditions to exist through more of the thickness at the crack tip, and thereby allows more material near the crack tip to deform plastically during crack extension. This might have helped to make the marker bands less visible in the 0.040 inch sheet.

Travelling microscope measurements were made from the centre of the rivet to the crack tip. The centre of the rivet was judged as the centre of a dimple placed in the head of the rivet during manufacture. The crack tip was taken as the end of a visible fissure in the metal. A fibre optic light source was used to illuminate the crack from different angles, to ensure that the crack tip had been identified correctly. During the early part of the test, the cracks were tight, and so there was some uncertainty in the position of the crack tip. Later in the test, crack opening angles and displacements were relatively large, and the crack tip could usually be located with confidence to within ± 0.001 inch.

The position of each rivet was also measured relative to a datum line marked on the specimen. This later allowed all crack growth curves to be plotted on common axes. The datum line was originally the reference for the position of each rivet hole during manufacture of the specimen, and was permanently marked at the intended centre of the first rivet.

The net measurement accuracy of crack tip position relative to the datum line is estimated to be within ± 0.002 inch, based on repeatability tests performed during normal test measurements.

This measurement accuracy and the uncertainty in crack tip position lead to a net estimated accuracy in crack tip position of within ± 0.003 inch for most readings. The error is potentially higher for short cracks.

The low scatter in the experimental crack growth curves appears to support this estimate of measurement accuracy. The curves are irregular in places, but, in most cases, the irregularity appears to be either isolated or part of a trend unrelated to measurement accuracy.

7.5 Initial Detection of Cracks

Experience during the testing of preliminary specimens with thicknesses in the range 0.032 to 0.040 inch indicated that cracks could be detected visually before or just after they had emerged from beneath the rivet heads. This was fortunate, since inspecting frequently for the first cracks in a specimen was found to be tedious and time consuming with other methods. The best results were obtained when viewing at an angle in diffused daylight. In this way, the crack is revealed by the surface disturbance ahead and around it.

Eddy current was found to be less capable, and less reliable than visual and D Sight inspection techniques:

- a. For cracks that were not through the thickness, relatively wide low frequency probes were needed, and these could not be positioned close enough to the rivet head to gain an advantage over visual detection. Custom-manufactured as well as standard probes were tried.
- b. For through the thickness cracks, a narrow high frequency custom-made probe was found to be a useful supplement to visual inspection in some cases, but it was too awkward to use regularly.

Another disadvantage with some eddy current probes was their tendency to scratch the surface of the metal.

The foregoing comments regarding eddy current inspection apply only to its usefulness on this project, and not to its usefulness in aircraft inspection.

All the comments in this subsection apply to crack detection in unpainted 0.040 inch sheet. They may not apply to painted or thicker sheet.

7.6 Monitoring of Link-up

A sample of 2 link-ups was monitored using a video-microscope. Photographs from one of these are in Figure 8-4, and are discussed later in connection with test results.

7.7 Diffracto Sight Inspection and Photography

Diffracto Sight or D Sight is a new technique for inspecting surfaces. Komorowski of the National Research Council [43] is currently developing procedures and equipment for applying it to the inspection of aircraft, in particular to the inspection of fuselage splices.

In its simplest form, the technique involves reflecting light off the target surface at an angle of about 30 deg, and reflecting it back along almost the same path using a retro-reflective screen. The target is viewed from slightly above the light source. Areas of the surface sloping away from the viewer appear light, while areas sloping towards the viewer appear dark. This difference in intensity is in the secondary image of the imperfection returned by the retro-reflective screen [44].

Ambient light must be reduced, and so the screen and light source are usually enclosed in a portable hood.

A prototype aluminum hood incorporating mirrors for compactness was used for the test. The same hood had been used for inspecting aircraft splices for cracks and corrosion. A photograph of the hood in position on the load-frame is in Figure 7-5.

The hood was modified to incorporate an adjustable fibre optic light source, and the specimen was viewed through a 35mm camera equipped with a telephoto lens. The aperture of the light source was modified to simulate a point source of light as far as possible. The light source was placed directly in front of, and slightly below the centre-line of the telephoto lens, in a position that did not interfere with the image of the rivet rows.

For maximum definition, the D Sight photographs were taken with Kodak Tech Pan film at a film speed setting of 25 ASA, and the negatives were developed in Technidol. The exposure was 1 second at f16. The intensity of the light source was adjusted to maintain this exposure setting.

D Sight photographs were printed for this thesis by electronically scanning the original photographs, enhancing the digital images, and printing them on a high definition laser printer. Despite these efforts, the reproductions do not have the definition and contrast of the original negatives.

Measurements of crack tip position were made from the negatives using a light table and a 60X travelling microscope.

To facilitate these measurements, and to provide an accurate means of scaling from the photographs, the specimen was permanently marked with 8 lines at 1 inch spacing. These lines were cut lightly in the surface cladding with a sharp blade. This approach was found to produce accurate and durable datum lines which were clearly visible in D Sight negatives.

All D Sight crack measurements were made with reference to the datum lines. It was not generally possible to distinguish between the position of each datum line and the centre of the associated rivet.

A scaling factor was defined as the distance between adjacent datum lines on the specimen divided by the same distance measured on the negative. It was computed for every ligament on every negative.

It was found that the scaling factor varied by up to 20% across the specimen. Consequently, all D Sight measurements were scaled by the local scaling factor for a particular ligament, and not by an average value for the negative. The variation in scaling factor across the specimen was probably due to a combination of perspective, slight inclination of the specimen with respect to the line of sight, and telephoto lens distortion.

A discussion of the effectiveness of D Sight for inspecting cracks is in the section dealing with test results. It includes photographs and a comparison between the crack growth curves obtained from direct measurement and D Sight measurement.

The choice of photography to record D Sight images was made because it was found to give better results than available standard video and digital equipment. The results represent the best achievable with 35mm camera equipment.

It is likely that better results could be achieved with a high definition television and state-of-the-art image enhancement software. A D Sight system incorporating these improvements was not available.

7.8 Test Sequence

An outline of the test sequence is given below.

The specimen and clamps were mounted under the load-cell, and the load feedback reading was offset to the weight of one pair of clamping plates.

The alignment of jack and specimen under a nominal load of 1,000 lb were checked using a theodolite.

The load required to produce the planned maximum fatigue stress of 13.5 ksi in the centre of the specimen was estimated using the finite element model as 7,219 lb.

Before the fatigue test was started, static loads were applied in steps of 1,000 lb up to the planned maximum fatigue load, and strain gauge readings were taken at each step. Readings were also taken as the load was reduced.

A constant amplitude, sinusoidal fatigue load was programmed for a specific number of cycles, interlocks were set, and the fatigue loading was started. The minimum load was set to produce a stress ratio of 0.02. The set point and span settings of the control system were adjusted to fine tune the maximum and minimum fatigue loads to the desired values. The command and feedback load waveforms were checked on an oscilloscope.

The test was interrupted when:

- the fastest crack tip had grown approximately another 0.020 inch;
- a new crack appeared;
- a link-up was imminent.

After each growth interval, some or all of the following readings were taken:

- a. Travelling microscope readings of crack tip positions were almost always taken.
- b. A D Sight photograph was taken in 60% of cases; sometimes several photographs were taken to compare images at zero load and full load, or to ensure correct focus in different locations.
- c. A scan of the strain gauges was recorded when significant changes in the stress distribution had occurred.

If a new crack appeared, a set of readings was taken immediately.

If a link-up was imminent, the load was applied in small batches of cycles. Eventually the load was applied one cycle at a time under manual control, until the ligament failed. When possible, crack measurements were taken up to the last cycle before failure. Photographs were taken with the help of a video microscope just before and during the failure process.

The test was terminated when a single crack had developed across the whole specimen between the side straps.

8 TEST RESULTS

8.1 Objectives of Test

One specimen manufactured to the final specification was tested using the equipment and methods described in the section on experimental procedure. The objectives of the test were as follows:

- a. To confirm that the specimen can be used to create and monitor realistic MSD.
- b. To observe the crack initiation and growth characteristics of the splice configuration tested.
- c. To define a criterion for crack link-up which can be applied in computer models.
- d. To validate the finite element model used in the design of the specimen: firstly, as a reliable means of designing other lap splice specimens with different configurations; and secondly, as a means of estimating the full stress distribution in an existing specimen.
- e. To evaluate Diffracto Sight (D Sight) as a technique for inspecting and measuring cracks in fuselage skin.

All objectives were achieved, and useful information was gathered on the characteristics of MSD crack growth in a typical lap splice.

The following sub-sections give a detailed description of crack initiation and development, the mechanism of link-up, the behaviour of overlapping cracks, the redistribution of stress as cracks progressed, and the accuracy of the finite element model in predicting stresses in cracked as well as uncracked specimens.

8.2 Overview of Crack Initiation and Growth

An overview of all crack growth curves for the test is in Figure 8-1. The main events during the development of the cracks are described chronologically below.

D Sight photographs of selected crack configurations throughout the test are in Figure 8-2, and are used to illustrate the description of crack growth.

All crack growth graphs in this thesis, regardless of scale, use the same co-ordinate system for crack tip position in the longitudinal axis. For all practical purposes, this places the centre of rivet #1 at co-ordinate 1 inch, the centre of rivet #2 at 2 inches, and so on. In reality, manufacturing tolerances result in slight variations in rivet position relative to these round-number co-ordinates. A full discussion of the datum system for crack tip position was given in the section on experimental procedure.

Crack growth proceeded as follows:

- a. Cracks were first observed on each side of rivet #3 (3L and 3R) at 336,563 cycles, with visible lengths of 0.01 and 0.06 inches respectively. These were not visible on the D Sight negatives.
- b. Cracks appeared at 2L, 5L and 5R before the cracks at rivet #3 had grown half-way across the rivet pitch. D Sight photographs a and c show the cracks at rivets #2 and #5 soon after they had appeared. Photographs a and b are the same, except that the latter was taken with the load removed. Cracks 3L and 3R remain visible, but crack 2L is no longer visible - this was also the case on the negative.
- c. Cracks subsequently appeared at 2R and 4L. Photograph d shows the crack configuration shortly afterwards. It shows a total of 7 cracks at 4 rivets.
- d. The cracks at rivet #3 dominated at this stage. 3L soon overlapped with 2R, while 3R linked with 4L. The acceleration in the growth of 2L after this overlap and link-up can be seen clearly in the curves in Figure 8-1. This acceleration indicates that the overlap acted in the same way as a link-up in extending the

length of the lead crack. Photographs e and f were taken before and after link-up respectively.

- e. Crack 4R appeared shortly after link-up 3R/4L. Photograph g, taken later, shows the lead crack running from tip 2L to 4R, and cracks on each side of rivet #5. Distortion signifies a hidden crack at 6L.
- f. Crack 4R subsequently linked with 5L. The effective lead crack then stretched from tip 2L to tip 5R, a total length of 3.76 inches, as seen in photograph h. The acceleration in the growth of tips 2L and 5R associated with this last link-up can be seen clearly in the crack growth curves.
- g. The lead crack was growing rapidly at this stage. Nevertheless, opposing cracks had time to grow ahead of it at 6L and 7L, which indicates that these cracks had initiated when the lead crack was much shorter, possibly well before the first link-up. Photographs i and k show the lead crack approaching the cracks at 6L and 7L respectively. The last photograph, l, shows the lead crack running from 1L to 7R.
- h. The lead crack grew through both the end rivets without there being any signs of earlier crack initiation at these rivets. Lack of prior initiation could be explained by the fact that the nominal stress was significantly lower at the end rivets than at the other rivets.
- i. The deceleration of both crack tips due to the bonded side straps can be seen in the crack growth curves.
- j. The test was terminated after both tips had penetrated the glue fillet at the straps. The fully cracked specimen withstood fatigue loading without any apparent distress.

8.3 Individual Crack Growth Curves

Larger scale crack growth curves are shown in Figure 8-3. The graphs include theoretical curves, which are discussed later in connection with computer model results. The curves

are formed as a succession of straight lines drawn between adjacent points; both the lines and the points are shown. This approach allows any aberrations in the data, such as widely spaced points and points that are not consistent with the trend, to be seen easily.

While there are occasional aberrations in the data, the smoothness of the curves, even at high magnification indicates that measurements of crack tip position were consistent and accurate, and that material behaviour was consistent.

8.4 Crack Growth Rates

Spot values of crack growth rates given in the later subsection on link-up were calculated from the experimental data by fitting a polynomial curve to the data points, and then differentiating the polynomial analytically at the point of interest.

A cubic curve fit over 4 to 7 points - up to 3 on each side of the point of interest - was found to give reasonably consistent results over the full range of data.

However, this approach cannot be relied upon to give accurate spot values of da/dN in all situations: the curve fit might not be the best one, particularly when less than 7 points are available; moreover, small differences in curve fit tend to be amplified when curves are differentiated.

It was decided that comparisons of theoretical and experimental crack growth data would be made by comparing full crack growth curves.

8.5 Evidence of Catch-up Mechanism

Crack growth rates were studied closely to see if there was any evidence of a mechanism in the early (visible) stages of MSD whereby shorter cracks might catch up with longer cracks. Some researchers have hypothesized that such a mechanism might contribute to

the development of the relatively uniform type of MSD seen on some aircraft. Two potential catch-up situations were considered:

- a. Cracks emanating from opposite sides of the same rivet - in this case crack length is measured from the centre of the rivet.
- b. Cracks at adjacent rivets - in this case, crack length is measured from tip to tip.

The cracks at rivet #3 provided evidence that a catch-up mechanism might exist for the former situation. Cracks 3L and 3R were first observed with a difference in length (as measured from the centre of the rivet) of 0.065 inches. The shorter crack, 3L, consistently grew faster. When the next crack (2L) appeared 48,000 cycles later, the difference in length had reduced to 0.032 inches. The difference continued to decrease, until it became 0.003 inches shortly before link-up 3R/4L and overlap 2R/3L, but interaction with other cracks may have played a role during this later period. Strain gauge readings indicated that the stress 1 inch ahead of each ligament was approximately equal during the catch-up period.

The test provided insufficient evidence to form any conclusions in case b (cracks at adjacent rivets).

8.6 Characteristics of Link-up

The characteristics of crack link-up were studied closely during the test:

- to determine whether link-up occurs as a result of plastic collapse of the remaining ligament, or unstable crack growth of one of the opposing cracks, or some other mechanism.
- to attempt to define a criterion for crack link-up that can be applied to computer models of MSL crack growth.

2

PM-1 3½"x4" PHOTOGRAPHIC MICROCOPY TARGET
NBS 1010a ANSI/ISO #2 EQUIVALENT

PRECISION^{FM} RESOLUTION TARGETS

A detailed study of this nature has not been published previously. Other researchers have only observed that final ligaments are small, and appear to fail as in a monotonic static test.

For ligament 3R-4L, the last ligament size recorded was 0.080, and link-up occurred within the next 50 cycles. The last recorded crack growth rates were 0.00036 in/cycle for 3R and 0.00025 in/cycle for 4L. These growth rates may not be accurate, for reasons given earlier in this section, but are recorded to give some idea of their order of magnitude. The ligament size immediately prior to link up was smaller than 0.080. Post-linkup measurements indicate that the final ligament size 3R-4L was between 0.01 inches and 0.08 inches measured in the longitudinal direction (ie, along the splice). Crack separation in the hoop direction (in the line of the applied load) was approximately 0.02 inches. Link-up occurred diagonally across the gap between the 2 cracks. Visible surface distortion of the metal in and around the link-up area indicated that considerable local yielding took place during link-up.

In the case of ligament 4R-5L, the ligament size immediately before failure was 0.061 inches, and final failure involved plastic distortion and necking across the ligament. The last measurable growth rates of both cracks were around 0.009 in/cycle. Crack separation in the hoop direction was approximately 0.03 inches.

Failure occurred progressively over 2 cycles, and necking occurred during each cycle. Both cycles were applied slowly, with the load frame under manual control. Video microscope photographs of the ligament at maximum load during each of the 2 cycles were taken, and are similar to those for ligament 5R/6L, shown in Figure 8-4. During the first cycle, the ligament necked severely across its full width along a S-shaped path. The presence of necking was confirmed by illuminating the ligament from different angles. Less pronounced distortion occurred along a straighter, more direct path joining the original crack tips. During the second cycle, additional necking occurred, and the ligament fractured along the S-shaped path.

The small separation of the crack tips in the hoop direction appears to have created a complex stress state in the ligament, and so final failure may have involved shearing. Nevertheless, it was clear that failure was preceded by plastic distortion and necking

across the full width of the ligament, and that it was not the result of unstable growth of one or other of the crack tips prior to full yielding of the ligament.

In the case of ligament 5R-6L, the ligament size immediately prior to failure was 0.096 inches, and the mechanism of failure appeared similar to that of ligament 4R/6L. The crack growth rates prior to failure were around 0.002 and 0.001 in/cycle respectively. The separation of the cracks in the hoop direction was approximately 0.035 inches.

Failure occurred progressively over 3 cycles. During the first cycle, the crack tips grew in the longitudinal direction and then turned towards each other. Slight surface distortion appeared in a line joining the tips which indicated that yielding had occurred across the ligament. These features are visible in the video microscope photograph in Figure 8-4a. During the second cycle (Figure 8-4b), pronounced necking occurred along a S-shaped path. Weaker distortion occurred along a more direct path between the original crack tips. As the load was increasing, a sound was heard that indicated that fracture might have occurred; however, neither a fracture line nor any significant crack growth could be seen. During the 3rd cycle (Figure 8-4c), full separation occurred along the S-shaped path without any obvious sounds of fracture.

In the case of ligament 6R-7L, the last ligament size recorded was 0.181 inches, 6 cycles before failure, and the appearance of the failed ligament was generally similar to the previous ones. The last measurable crack growth rates were around 0.004 and 0.002 in/cycle respectively. The separation of the cracks in the hoop direction was approximately 0.01 inches.

The foregoing observations of 4 link-ups led to the following conclusions, applicable to 0.040 inch thick 2024-T3 sheet:

- a. The link-up of approaching cracks with a normal separation of up to 0.030 inch involves plastic distortion and necking prior to final fracture. The precise mode of failure is complex and might include shearing in some cases.
- b. The process of necking and final fracture of the ligament may take several cycles when the lead crack is long.

- c. The ligament size at which this failure process starts is small, and increases either with the size of the larger crack, or with the aggregate crack size, or with both; it was not possible to distinguish between these possibilities.

8.7 Overlapping Cracks

The test also provided the opportunity to observe the behaviour of 2 overlapping cracks. Cracks 2R and 3L initially followed a pattern of growth generally similar to that of 3R and 4L. However, their separation in the hoop direction was much greater - approximately 0.07 inches at the time of overlapping. This was sufficient to delay link-up until an overlap of 0.17 inches had developed. The crack growth rates peaked at around 0.0002 and 0.0003 in/cycle respectively at the time of overlap, and reduced sharply thereafter (Figure 8-3b and c).

8.8 Crack Approaching a Rivet Hole

The large scale graph of crack 7R shows a gradual increase in growth rate after the crack tip had passed the mid-way point between the rivets, until the crack tip had disappeared beneath rivet head #8. During the period covered by this graph, the effective length of the crack was over 6 inches, and the growth rate was very high, rising from 0.001 to 0.004 inches per cycle.

8.9 Stress Distribution and Validation of Finite Element Model

The finite element model has 2 purposes. Its primary purpose is to determine in advance whether a particular specimen design will meet the specification. This is a generic purpose, and implies that the model will be relied upon to provide accurate stress data for specimens which have different geometries to the present one. Its secondary purpose is to supply stress data on an existing specimen in areas which have not been instrumented. For neither purpose is it necessary for the model to represent closely the stress distribution in the immediate vicinity of the rivets, but it must be possible to obtain a reliable estimate of the bypass stress.

The finite element model was validated for both purposes by comparing the measured and predicted stresses along a line 1 inch ahead of the critical rivet row. This comparison was made for a cracked as well as an uncracked specimen. For the former, an advanced MSL crack configuration was chosen, in order to test the model in a situation where considerable stress redistribution had occurred. The configuration comprised a 2.765 inches crack from 2L to 4R, with a 0.423 inch tip to tip crack at rivet #5. In the model, shown in Figure Figure 6-7, the crack lengths were 2.75 inches and 0.5 inches respectively. Graphs overlaying the measured and predicted stress distributions for the uncracked and cracked cases are in Figure 8-5 and Figure 8-6.

The graphs indicate that the finite element model is adequate for its purposes in its present form. In the uncracked case, the predicted stresses are lower by an amount varying between 0.2 and 0.6 ksi (4% of maximum stress). This is within the accuracy normally expected of general purpose strain gauges. In the cracked case, the predicted stress varies between 0.7 ksi above and 2.1 ksi (12%) below the measured stress. The region of maximum difference is at rivet #5. The crack at rivet #5 was longer in the model than in the specimen; the associated reduction in local stiffness could explain the 12% underestimate in the nominal stress.

A further comparison was made between the model and a photoelastic test specimen, in order to judge qualitatively whether the model reasonably represented the stress distribution in the vicinity of the rivets. This comparison has already been discussed in connection with specimen design. The specimen involved was not identical to the main test specimen; nevertheless, the comparison was sufficient to give confidence that the model could be used for estimates of bypass stress.

The design of the finite element model is described in the section dealing with design of the specimen. The aim was to produce a simple model that was easy and quick to use. To this end several simplifications were used, including a relatively coarse mesh, a simplified representation of rivets, and the assumption of an infinitely stiff bond line. Greater accuracy in the model could have been achieved, for example, by using a finer mesh, without changing the basic modelling approach; the penalty would have been a much longer running time (2 hours or longer instead of 1/2 hour).

8.10 Redistribution of Stress as Cracks Progress

The hoop stress measured along a line 1 inch ahead of the critical rivet row reduced almost to zero at strain gauges in line with cracked areas, and rose progressively at gauges in line with uncracked areas. Graphs of the hoop stress distributions corresponding to 4 of the crack configurations in Figure 8-2 are in Figure 8-7. The general trend in hoop stress during the last 15% of the crack growth period is illustrated in the carpet plot in Figure 8-8.

Since the test specimen had been designed to be representative of an aircraft, the nominal stress remained moderate even when the MSD was at an advanced stage. The maximum stress at 1 inch from the rivet row was measured at 14 ksi at the start of the test, and never rose above 26 ksi at any of the 7 strain gauge positions across the specimen.

The bending stress distribution 1 inch ahead of the rivets is plotted together with the membrane stress in Figure 8-7. The bending stress at the edge of the clamped area of the joint, ie, along approximately a tangent to the rivet holes, was probably much higher than these values.

8.11 Detectable Length of Cracks

The first cracks in the specimen, at rivet #3, were detected together when their visible lengths were 0.01 and 0.06 inches respectively. These were tight cracks, which were only clearly visible under favourable lighting conditions. Nevertheless, they could probably have been detected earlier, if the load frame had been programmed for shorter blocks of loading cycles.

As the test progressed, there was less difficulty in detecting new cracks. Cracks appearing after the first ones at rivets #2, #3 and #5 could be detected easily before they grew beyond the rivet heads by the surface distortion that preceded them.

Eddy current inspection was not used during this test. It was used during preliminary tests performed as part of the problem analysis phase of this project, and did not offer any significant advantages over visual inspection for initial crack detection. An assessment

of eddy current as a means of initial detection of cracks was given earlier in connection with experimental procedure.

D Sight in the form used was no better than unaided visual inspection for the detection of new cracks. Its potential usefulness in a wider context is discussed in the next section.

8.12 Effectiveness of D Sight for Inspecting Cracks

D Sight photographs were taken of 53 out of 94 crack configurations measured during the test. Sometimes, several photographs were taken of a particular crack configuration, to investigate focus, camera settings and relative image quality at zero load. A total of 73 D Sight photographs were taken. The equipment used for this purpose has already been described in connection with experimental procedure.

Measurements taken from the negatives were used to plot crack growth curves for comparison with the curves obtained by direct measurement of the cracks. A sample of D Sight curves are shown in Figure 8-9. All 73 photographs were measured and are included in the applicable graphs; therefore, there are sometimes several points at one cycles value.

The curves for all the cracks show scatter of within ± 0.015 inch in the crack tip position. At the scale used in the figures, the curves still appear relatively smooth. This degree of scatter was expected from measurement inaccuracy, and from the effect of scaling measurements from a negative image which was less than 1/6 full size.

Taking the scatter into account, the crack growth curves obtained by direct and D Sight measurement show good agreement. The D Sight measurements of crack length were generally on the high side. The difference was relatively small, up to 0.020 inch, for the earlier cracks at rivet holes #2, #3 and #5. The difference was largest, up to 0.050 inch, for the small cracks in the path of the lead crack. The degree of surface distortion near the tips of these short cracks seemed to be quite high, and may have led to the larger overestimate. It is reassuring that most errors in the D Sight crack measurements were conservative.

Under maximum load, the early, tight cracks could be seen easily on photographic prints after they had extended 0.070 inch beyond the edge of the rivet head. Even before the first link-up, some new cracks could be detected before they appeared from beneath the rivet heads.

Detectability, as judged by relative sharpness and contrast in photographs, reduced at zero load compared to full load. Short cracks in the early stages of MSD may not be visible. Figure 8-2 a and b show a full load and zero load situation respectively. The cracks at rivet #3 remain visible at zero load, while the crack at 2L disappears.

The foregoing observations apply to the inspection of an unpainted splice under laboratory conditions, using a prototype aircraft inspection hood, a fibre optic light source and 35 mm photographic equipment. The presence of paint would probably reduce detectability and measurement accuracy. The use of a high definition video camera and digital image enhancement could improve the results.

Detectability and measurement accuracy might decrease when using D Sight on an aircraft unless a suitably manoeuvrable platform system is provided. The ambient lighting conditions would not matter, since D Sight is self-contained in this respect.

The test has shown that D Sight used with commonly available photographic equipment can locate MSD crack tips in an unpainted splice reasonably accurately. While the test conditions were ideal, it is estimated that results approaching those described above could be obtained under hangar conditions.

8.13 Summary Review and Discussion of Test Results

Cracks were initiated naturally at 4 out of 8 rivets before any link-up occurred, and corresponded to the non-uniform type of MSD evident in actual MSD scenarios. Since 2 of the 8 rivets were in a lower stress environment close to the side straps, it could be considered that the record was cracks at 4 rivets out of 6 prior to link-up.

Strain gauge measurements at the start of the test confirmed that the nominal stress distribution in the specimen was as intended, ie, representative of the hoop stress gradients measured on aircraft.

The stress distribution measured after several link-ups corresponded well with that predicted using the finite element model, and demonstrated that the increase in net section stress with crack growth was being moderated in the same manner that it would be on an aircraft.

Thus, it can be said that the test reproduced MSD under realistic conditions, and that information gained from the test has general relevance to the design and maintenance of commercial transport aircraft.

By extrapolation, specimens which might in future be designed to simulate the relevant parameters of a splice on a particular aircraft could be expected to give a useful forecast of crack initiation and growth characteristics for that aircraft.

For the particular splice configuration tested, several observations can be made from the test, which might have wider significance:

- a. Cracks first appeared beyond the rivet heads at approximately 337,000 cycles. about four times later than cracks have appeared in some splices on Boeing 737 aircraft. Initiation life and early growth rates are sensitive to a number of parameters, discussed earlier, including local hoop stress, out-of-plane bending stress, rivet clamping force, rivet expansion, depth of countersink and faying surface treatment. It was not feasible, nor was it attempted in this project to match all these details for any particular Boeing aircraft. Nevertheless, the test results are useful in demonstrating the need for close matching of several parameters, if reliable initiation life data is required. Crack growth rates of intermediate MSD are less sensitive than initiation life to rivet details, and so it should be easier to achieve similitude.
- b. At least 3 rivet holes developed cracks independently, ie, before crack growth had caused any significant redistribution of stress.

- c. At other rivet holes, crack initiation probably occurred early, but may have been accelerated by load transfer. In most cases, the cracks tended to appear when opposing cracks came close enough to interact strongly; this is unlikely to have occurred unless the new cracks had initiated much earlier.
- d. There was evidence that a weak catch-up mechanism might exist for unequal cracks at the same rivet hole.
- e. The crack initiation and growth periods in the specimen are summarized and compared with available data on the earliest cracks on several types of Boeing aircraft as follows:

(1) Appearance of first cracks beyond rivet heads:

Specimen	337,000 cycles
Boeing	21,500 to 79,000 cycles

(2) Period from first appearance to first link-up:

Specimen	70,000 cycles
Boeing	9,000 to 42,000 cycles

(3) Period from first link-up to link-up of several rivets:

Specimen	6,000 cycles to 8 rivets (8 inches)
Boeing	1,000 to 10,000 cycles to 18 rivets (18 inches)

The visible growth periods in the specimen are quite close to those in Boeing aircraft. This agreement provides further confirmation that the crack growth characteristics observed in the specimen can be considered as generally representative of those in aircraft.

- f. The growth rate became almost asymptotic after three link-ups had produced a lead crack 4 inches long. At a length of 6 inches the growth rate of the lead crack was over 1 inch per 1000 cycles without any close interaction with holes or other

cracks. On narrow body aircraft, this rate would be equivalent to 1 inch in 4 months.

- g. The mechanism of crack link-up involved a complex pattern of plastic distortion and necking across the full width of the ligament prior to fracture. Fast fracture prior to yielding did not occur.
- h. Link-up occurred at fairly small crack separations of 0.06 to 0.18 inches, and the size of the final ligament prior to link-up increased with crack size.
- i. The fact that plastic failure of the ligaments of longer cracks occurred over 2 or 3 cycles in the specimen, indicates that the surrounding structure was providing a measure of displacement control, allowing yielding in the ligament to provide local stress relief. A similar phenomenon could occur in an aircraft splice.
- j. Two cracks overlapped with a normal separation of 0.07 inches. The measured stress distribution and crack behaviour suggest that overlapped cracks could be modelled as linked cracks.

As a result of observations g and h, link-up was defined in the computer model of MSD crack growth as occurring when the cracks, with Irwin plastic zone correction added, touch each other. This approach has 3 attractions:

- a. It results in reasonably accurate predictions using linear elastic fracture mechanics (LEFM), as discussed later in connection with the results of the computer model.
- b. It predicts larger final ligaments for longer cracks, as was observed on the test specimen.
- c. A plastic zone correction to crack length is logical. The crack tip plastic zone size in thin sheet is relatively large, because the stress state is approaching plane stress over a large proportion of the thickness. Moreover, crack interaction creates large plastic zones even with modest lead crack lengths.

For the purpose of comparing theory and experiment, the final ligament size is regarded as the one that precedes obvious necking of the ligament. The stress relief in the ligament referred to in i above can prolong the period of necking by a few cycles, but any such delay in final fracture is unlikely to be significant in predictions of overall crack growth periods.

The finite element model was found to be an accurate and cost-effective means of predicting the general stress distribution in cracked as well as uncracked specimens. The same approach is recommended for future specimens.

9 DESCRIPTION OF COMPUTER MODEL OF MSD CRACK GROWTH

9.1 Choice of Modelling Approach

Like the test specimen, the computer model of MSD crack growth is intended primarily to represent the situation at the critical row of a longitudinal skin splice on a real aircraft.

It was decided to limit the model initially to representing crack growth from emergence from under a rivet head to the link-up of several rivets within a single frame-bay, ie, intermediate phase growth. There were several reasons for this:

- a. The intermediate phase of crack growth constitutes most of the detectable period of crack growth, and, from a maintenance point of view, is the period of primary interest.
- b. It can be modelled 2-dimensionally, whereas the initial phase (growth under the rivet head) should be modelled 3-dimensionally.
- c. The modelling of a large lead crack interacting with other MSD cracks over 2 or more frame bays, ie, final phase MSD, would have involved substantial additional complexity.

There are 2 broad approaches to modelling intermediate phase crack growth. A potentially accurate but costly approach is to use finite element modelling to compute the stress intensity factor (SIF) for each crack tip, and then to remesh and repeat the calculation after each growth increment. An approximate, but quicker and less expensive approach is to use the superposition and compounding of known exact SIF solutions to arrive at the SIF for each crack tip.

It was decided to evaluate the latter approach by developing a program and testing its predictions against actual crack growth measured on the test specimen.

9.2 Overview of Theory

The process of predicting crack growth curves for any case involving constant amplitude loading of sheet metal is a semi-empirical process, as follows:

- a. Estimate the mode I stress intensity factor range (ΔK) for the known material, crack geometry, component geometry and loading.
- b. Estimate the instantaneous crack growth rate (da/dN) from available curves of da/dN vs ΔK for wide centre-cracked tension specimens; for similitude, the material properties, thickness, and stress ratio must be the same.
- c. Select a growth interval (dN) and compute the growth increment (da).
- d. Repeat a to c until the crack growth curve has been assembled.

On an aircraft, the stress intensity factor range, ΔK , is simply the value of K at maximum operating pressure. With the test specimen, it is necessary for practical reasons to use a small positive value of stress ratio, R . In this case:

$$\Delta K = (1 - R) K_{\max} \quad (8)$$

In a splice joint, an accurate estimation of K is made difficult by the complex stress distributions around and within the joint. Since relatively low stress levels are employed in fuselage splices, the stress/strain response is linear throughout most of the splice. Consequently, the stress distribution can be represented as the superposition (sum) of several simpler stress distributions. Some or all of these may correspond to stress distributions for which exact K solutions are already available in the literature. Those that do not may be amenable to solution using Rooke's compounding method. This method can produce accurate solutions, but in practice the accuracy will depend on the simplifying assumptions that might have to be made.

The compounding method involves calculating the resultant stress intensity factor (K_r) using the following equation:

$$K_r = K_0 + \sum_1^n (K_n - K_0) + K_e \quad (9)$$

where:

K_0 = SIF with no boundaries present;

K_n = SIF with only the nth boundary present; boundaries are cracks, holes, free edges, etc;

K_e = SIF contribution from boundary to boundary interaction.

The derivation of this equation can be found in Rooke [36]. The estimation of K_e is complex and not practicable for the present purpose. Its value has been assumed to be negligible. The validity of this and other assumptions will be assessed in the light of the program's overall accuracy in comparison with test results.

Some qualitative comments can be made regarding the likely significance of K_e . Rooke points out that the value of K_e can be expected to be small, provided that the boundaries are not too close to each other. In the use of compounding for collinear cracks, described below, the boundary giving rise to the largest K_n is usually the adjacent crack; other cracks are of much less significance. Therefore, it can be reasoned that K_e might only be significant when the adjacent crack is itself adjacent to a long crack. In such situations, the overall growth pattern would in any case probably be dominated by the long crack. Thus, it is possible that K_e has little significance in the context of intermediate MSD crack growth.

9.3 Choice of Superimposed Stress States

The general case of a lap splice located on a stringer between 2 frames is as shown in Figure 9-1. The critical location for cracking is the outer skin along the line of the upper rivet row.

The critical area can be isolated and approximated as the free body diagram also shown in Figure 9-1. For clarity, only 3 rivets are shown; these have been given an arbitrary crack configuration.

For this analysis, the section of splice under consideration is considered to behave as part of an infinitely long splice joint. The loading environment of such a splice has already been described in the section on MSD problem analysis. It is modelled as follows:

- a. The applied (nominal) stress, σ_0 , is non-uniform.
- b. The effect of the rivets is approximated as point loads acting on the skin. This appears to be a reasonable approach for intermediate MSD, since the effects of the rivet, including clamping, are localized, and might have only a small effect on crack growth beyond the rivet head.
- c. These point loads and the bypass stress, σ_{bp} , are in equilibrium with the applied stress.
- d. Out of plane bending due to the combined effects of joint eccentricity and skin pillowing is represented in the free body diagram by a superimposed state of pure bending.
- e. The boundary conditions at each end of the section are those of an infinitely long sheet, and are not shown in the figure.

This loading condition can be represented by the superposition of 3 simpler loading conditions, A, B and C, shown in Figure 9-2:

- a. Loading condition A comprises a row of cracks and holes in a single sheet subjected to non-uniform tension σ_{bp} .
- b. Loading condition B comprises a row of loaded holes and cracks, where the total load is balanced by a non-uniform tensile stress $\sigma_t = \sigma_0 - \sigma_{bp}$ (which will be referred to as the transfer stress) acting in the opposite direction to the loads. The

loads are those exerted on the sheet by each rivet, and act at the locations of the rivet centres.

- c. Loading case C consists of a row of cracks and holes subjected to pure out-of-plane bending.

In these loading cases, cracked rivet holes are assumed to be equivalent to simple cracks running from tip to tip, or from tip to edge of hole as the case may be; the holes are ignored in further analysis. Uncracked rivet holes are treated as such.

The stresses in case B can be shown to be equivalent to one half the sum of stresses of loading cases D and E, also shown in Figure 9-2. Loading case D comprises a row of cracks and holes subjected to a non-uniform tensile stress σ_t . Loading case E consists of a row of cracks and holes subjected to self-equilibrating loads (wedge loads) acting at the locations of the rivet centres, and equal in magnitude to the unidirectional rivet loads.

The SIFs associated with loading cases A, C, D and E can be summed to arrive at the total K, as follows:

$$K_{total} = K_A + 0.5 (K_D + K_E) + K_C \quad (10)$$

Using more descriptive suffixes, this equation can be rewritten as:

$$K_{total} = K_{bp} + K_{transfer} + K_{bend} \quad (11)$$

where

$$K_{transfer} = 0.5 (K_{ltension} + K_{wedge}) \quad (12)$$

These variable names are as used in the computer program. The suffix 'ltension' stands for 'load transfer tension'.

9.4 Calculation of K for Each Stress State

In cases A and D (cracks under simple tension), the compounding equation can be applied by considering the crack of interest to be paired successively and independently with each other crack or uncracked hole. There are known solutions for:

- 2 collinear cracks in an infinite sheet;
- A crack approaching a hole in an infinite sheet.

In the current version of the program, the graphical solutions (β geometry factors) catalogued by Rooke and Cartwright [27] are used. They have been programmed as polynomial curve fits with linear interpolation. The use of analytical expressions would be more accurate, and could be investigated as a program improvement.

A nominal (far field) stress is needed to convert the β factors into K values. In case A this is σ_{bp} . In case D this is $\sigma_t (= \sigma_0 - \sigma_{bp})$. Distributions along the splice of σ_0 and σ_{bp} in the uncracked condition are obtained by finite element analysis or direct measurement using strain gauges. The value in the uncracked condition, at the longitudinal (x) coordinate of the crack tip of interest is the value used in computing K.

Consistent with the assumption that the splice joint behaves as though it were infinitely long, no edge correction is used in computing K.

The presence of circumferential stiffeners, ie, frames and straps, interferes with the assumption of an infinite splice in 2 ways:

- a. The stiffeners give rise to a non-uniform hoop stress in the uncracked condition. This is accounted for in the model by imposing the required non-uniform nominal stress distribution.
- b. The stiffeners moderate the increase in stresses caused by load transfer from cracked sections of the splice even more than would the assumed infinitely wide unstiffened sheet. This effect is currently not accounted for in the model, and might lead to an overestimation of the growth rates when MSD is fairly advanced.

If later found necessary, a fairly straightforward empirical correction based on aggregate crack length could probably be devised.

In case E (cracks subjected to wedge loads), there is a known simple analytical solution, catalogued by Rooke and Cartwright, for an isolated crack:

$$K_{wedge} = \frac{P}{t \sqrt{\pi a \left(\frac{a+s}{a-s} \right)}} \quad (13)$$

for the left and right crack tips respectively, where s is the distance in the longitudinal direction from the centre of the crack to the line of action of the wedge load. The value of P for a particular crack is calculated from:

$$P = \frac{\sigma_t}{p \cdot t} \quad (14)$$

where p = rivet pitch, and t = sheet thickness. The value of transfer stress σ_t at the longitudinal co-ordinate of the centre of the crack is used in the above equation. If a crack traverses several rivet holes, the associated wedge forces are resolved into a single resultant for use in the known solution.

There is no known solution for the wedge load case that takes into account the interaction of other cracks also subjected to wedge loads. If no adjustment is made for interaction, it is likely that the net effect on overall K would be small in many cases, because:

- K_{wedge} is itself typically relatively small, and
- K_{wedge} tends to decrease as crack length increases.

Therefore, in the interests of simplicity, interaction in the wedge load case is assumed to be negligible. Since interaction is being neglected, there is no need for compounding.

In case C (out of plane bending at the rivet line), the following 2 approaches to estimating K have been considered:

- a. Assume that out of plane bending has a negligible effect on crack growth rate, ie, neglect K_{bend} .
- b. Assume that out of plane bending increases the effective SIF range (ΔK) by an amount proportional to the surface bending stress in the sheet:
 - (1) Estimate or measure the bending stress distribution along the splice.
 - (2) Apply the known solutions in Rooke and Cartwright for cracks at a hole subjected to pure bending.
 - (3) Apply an empirical factor to the values of K so obtained, and add them to the overall K .

An option to use either approach is included in the computer model, but a full study is needed to determine how best to model the effect of bending on crack growth characteristics. Initial indications are that neglecting bending gives satisfactory results.

9.5 Estimation of Crack Growth Rate from ΔK

As discussed in the section on damage tolerance analysis, the usual method of estimating the crack growth rate, da/dN , is to obtain the value from experimental data of da/dN vs ΔK . No solely theoretical method has yet been devised. For similitude with aircraft fuselage skin, the experimental data must be obtained from tests on wide, centre-cracked tension specimens using the same material, thickness and stress ratio. In these tests, ΔK is inferred from the boundary conditions using known analytical solutions.

Only a limited amount of such experimental data is available in the literature. The program currently only incorporates data for 0.040 inches thick 2024-T3 at a stress ratio of 0.02. This data was published by Dawicke et al [30] in graphical form only, and is reproduced in Figure 9-3. It was produced under NASA's Aircraft Structural Integrity

Program. For the purposes of the computer program, it has been reduced to 4 visually fit straight lines in the form of the Paris equation:

$$\begin{aligned}
 \Delta K < 7 & \quad \frac{da}{dN} = 10^{-9.47} \Delta K^{4.12} \\
 7 \leq \Delta K < 17 & \quad \frac{da}{dN} = 10^{-8.56} \Delta K^{3.08} \\
 17 \leq \Delta K \leq 32 & \quad \frac{da}{dN} = 10^{-10.87} \Delta K^{4.97} \\
 \Delta K > 32 & \quad \frac{da}{dN} = 10^{-14.29} \Delta K^{7.29}
 \end{aligned} \tag{15}$$

where ΔK is in $\text{ksi}\sqrt{\text{in}}$, and da/dN is in in/cycle .

The scatter in da/dN in the NASA data varies between $\pm 19\%$ and $\pm 28\%$ for a given value of ΔK in the range 4 to 30 $\text{ksi}\sqrt{\text{in}}$. A review of other published data for 2024-T3, indicates that scatter of this magnitude is typical.

The NASA data only covers a ΔK range of 4 to 32 $\text{ksi}\sqrt{\text{in}}$. As can be seen from the above equations, the program extrapolates beyond these values in a way that is likely to overestimate the crack growth rate of very short cracks (low ΔK), and to underestimate the growth rate of long cracks or cracks separated by a small ligament (high ΔK). In the case of intermediate phase MSD, this range of ΔK covers most of the crack growth period, and so any error in predicting overall growth period is likely to be small.

9.6 Calculation of Crack Growth Curves

The crack growth curve for each crack is built up iteratively from a given initial crack length, by using the instantaneous crack growth rate, da/dN , of each crack tip to project further growth, da , over a selected number of cycles, dN . The interval is selected so as to produce a growth increment of approximately 0.010 inches in the fastest growing crack tip. This criterion has been found to produce satisfactory results without excessive computation time, but can easily be changed if necessary.

After each growth interval, the cumulative length of each crack is plotted against the cumulative cycles. When the separation of approaching crack tips is small enough to meet the link-up criterion, the growth curves of the approaching cracks are terminated.

9.7 Link-up Criterion

Observations of link-up characteristics on the test specimen (0.040 inch thick 2024-T3) suggest that a semi-empirical formula based on the combined plastic zone size of the approaching cracks could be used to describe the final ligament size. It was noted that regardless of the crack configuration, final failure occurred by plastic collapse, and that the ligament size just prior to failure tended to be larger for longer cracks. Such a formula could be applicable to any intermediate phase MSD conditions in which the approaching cracks were approximately collinear.

It may be possible to use a similar approach to the modelling of overlapping cracks. Observations on the test specimen indicate that the growth rates of approaching cracks, even with a fairly large normal separation (0.07 inches), are as high as those of 2 collinear cracks. Test measurements also indicate that the stress distribution around an overlapped crack with a normal separation of 0.07 inches is similar to that of a linked crack. Therefore, it may be feasible to model overlapped cracks as linked cracks. Moreover, the same link-up criterion could be used in all cases - the associated error in the predicted overall MSD growth period would probably be small.

The criterion chosen to define link-up in the computer model was that the approaching cracks, with Irwin plastic zone correction included, must touch. The choice of plastic zone correction - Irwin, Dugdale or other - can be reviewed later in the light of any further experimental data. It is unlikely that the choice would make a major difference to the predicted overall growth period, since the growth rate is always high just prior to link-up.

To maintain consistency in the model, this correction to crack length is included in all calculations of K . The general effect of this correction on growth rates was small. The effect on the growth rate of approaching (highly interactive) cracks was significant, and tended to improve the correlation with experimental growth rates.

Use of a plastic zone correction is an artificial means of compensating for the existence of a relatively large plastic zone, while continuing to use the relatively simple approach of linear elastic fracture mechanics (LEFM) to define and compare stress fields in the vicinity of crack tips.

In the broader context of fracture mechanics, it is interesting that the use of such a correction in this case seems to result in a reasonably accurate prediction of when link-up will occur, while also improving the prediction of crack growth rate in situations of high crack tip plasticity.

9.8 Addition of New Cracks

Since the program does not predict crack initiation, the initiation sequence must be included in the program. The starting crack configuration is input as a data file. Subsequent crack initiation is made subject to conditions which are written into a sub-program. This sub-program is run at the beginning of every crack growth iteration, to see if the conditions for adding the next crack have been met.

An example of crack initiation conditions and the associated sub-program are given in the next section.

9.9 General Features of Computer Program

The key theory, assumptions and approximations embodied in the computer program have been described above.

The application of this theory to a multiple crack situation involves programming complexities. The use of many multi-dimensional array variables, nested Do loops, etc, has been unavoidable. For this reason, the structure of the program has been kept as simple as possible, and copious notes have been included to aid comprehension. The program has been implemented in Mathematica code. This is a high level language which allows the code to be written in a concise and readable form. It also incorporates all required mathematical and graphical functions. The Mathematica system is such that

programs can be modified, debugged, run, interrupted, interrogated and restarted interactively. Minor changes and alternative graphical output can be programmed quickly and easily. Data files in almost any format can be read and manipulated with only a few commands. The program can be run on a PC or work station equipped with Mathematica software.

A full listing of the source code is in Appendix C. The structure of the program has been made clear by giving headings to each module and major iteration loop. Headings and notes are distinguished from program code by enclosing round brackets and stars - eg, (* TEXT *). Mathematica commands always start with an upper case letter, and their operands are enclosed in square brackets. Variables have been given descriptive, fully lower case names, and have also been described in the notes. Minor iteration loops and If sequences have been indented to make them readable.

The main algorithm is summarized as follows:

- 1 Input joint geometry, starting crack configuration and cycle count, material yield strength, nominal hoop stress distribution, and bypass ratio distribution.
- 2 Input whether bending effects are to be included in computations.
- 3 Establish functions to describe bypass and transfer stress distributions.
- 4 Define graphs of geometry factors (β factors) from Rooke and Cartwright.
- 5 Define array variables whose sizes depend on joint geometry.
- 6 Start main iteration loop.
- 7 Increment the cycle count except on the first iteration.
- 8 Compute the crack increments and the new crack lengths except on the first iteration.
- 9 Determine if it is time to add any new cracks. If so, add the new cracks.

- 10 Define/redefine array variables whose dimensions depend on the crack configuration.
- 11 Define/redefine the interpolation functions for the graphs of β factors (these functions depend on the crack configuration).
- 12 Compute the crack tip coordinates of the new crack configuration excluding plastic zone corrections and add them, together with the current cycle count, to the plot matrices.
- 13 Look up or compute β factors, as applicable.
- 14 Compute K_{bp} and $K_{transfer}$ for each crack tip.
- 15 Compute the Irwin plastic zone correction for each crack tip using $K_{interim}$ ($= K_{bp} + K_{transfer}$).
- 16 Compute corrected crack lengths.
- 17 ***Repeat steps 13 to 14 once*** using the corrected crack lengths, and then proceed to step 18.
- 18 Determine if any link-ups have occurred. If so, adjust the crack lengths accordingly, and ***repeat steps 9 to 17***.
- 19 Look up β factors for bending (if applicable)
- 20 Compute K_{bend} (if applicable).
- 21 Compute $K_{total} = K_{bp} + K_{transfer} + K_{bend}$
- 22 Compute da/dN for each crack tip.
- 23 Compute dN for next iteration.

- 24 **Repeat steps 7 to 22** until all rivets have linked up, or until a specified number of iterations have been completed.
- 25 Create individual crack growth plots for each tip, and store them in graphics format.
- 26 Create, store in graphics format, and display a combined plot of all cracks.

Note: One of the above steps might correspond to several modules in the program.

9.10 Summary of Assumptions

Several simplifying assumptions were made in the theory that underpins the program, and in writing the code. Most of these have already been mentioned. They are summarized as follows:

- a. The problem was reduced from that of a splice structure with complex, and not fully defined interactions between components to one of a single, infinite sheet with holes and cracks, subjected to a relatively simple system of loading.
- b. In the application of all known stress intensity factor solutions to this simplified model, it was assumed that:
 - (1) all cracks are collinear, and lie on the centre-line of the rivet row;
 - (2) a hole with 2 cracks can be replaced with an equivalent crack with the same tip locations;
 - (3) a hole with 1 crack can be replaced by an equivalent crack running from the actual tip to the furthest edge of the hole;
- c. The effect of circumferential stiffeners is approximated by imposing a non-uniform distribution of nominal hoop stress, which can be adjusted to match the

actual distribution in the aircraft skin panel in question. This approach is valid for the uncracked condition, but may not be as accurate with advanced cracking.

- d. Boundary interaction was ignored in the application of Rooke's compounding technique.
- e. Crack interaction was neglected for the wedge loading and bending cases.
- f. For long cracks, the wedge load was computed as the resultant of all the wedge loads at each rivet location within the crack.
- g. Out-of-plane bending was assumed to have no effect on crack growth rates in the data output for this thesis, but the program code includes an option to factor it in if desired.
- h. In applying all known solutions, regardless of the crack configuration, the nominal stress is taken as the hoop stress in the uncracked condition at a location in line with the crack tip 1 inch ahead of the rivet row. The hoop stress distribution 1 inch ahead of the rivet row must be estimated or measured for input to the program.
- i. In applying all known solutions, the crack length is adjusted to include the Irwin plastic zone correction.
- j. Link-up of 2 opposing crack tips is assumed to occur when the adjusted crack tips touch.
- k. The relationship between crack da/dN and ΔK in an aircraft splice is assumed to be correctly represented by the graphical data published by Dawicke et al (Figure 9-3).
- l. Dawicke's data covers a ΔK range of 4 to 32 $\text{ksi}\sqrt{\text{in}}$, and has a scatter of around $\pm 20\%$. It was approximated by 4 Paris law equations covering an unlimited range of positive ΔK . Crack growth rates might be underestimated at very high ΔK , and overestimated at very low ΔK .

- m. The curves of geometry factors in Rooke and Cartwright for the collinear cracks and crack approaching a hole were programmed as approximate polynomial curve fits. The program does not extrapolate beyond the envelope of the graphs given, and so crack growth rates might be underestimated in some extreme crack configurations.

The size of the error associated with each assumption listed above is not expected to be large, but it can be greatly magnified in the final growth rate by the power law relationship between da/dN and ΔK . Overall error factors of 2 or more are not uncommon in estimating crack growth rates, even for constant amplitude loading, particularly if some growth takes place at high values of ΔK .

In view of the complexity of the real loading conditions, and the number of assumptions that have to be made to achieve an estimate of crack growth rate, it is impractical to attempt a detailed error analysis.

A realistic approach to assessing and improving the accuracy of the program is to compare it with test results, and this is the approach taken in this thesis.

10 RESULTS OF COMPUTER MODEL

10.1 Comparison of Theoretical and Experimental Crack Growth Curves

To evaluate the MSD crack growth model, it was programmed to initiate cracks in the same sequence as the test specimen. The 2 sets of crack growth curves were then compared.

This sequencing was achieved by gearing the initiation of a new crack to the length of an existing crack. For example, in the test, crack 2L was first observed at a length of 0.128 inches (from the rivet centre) when crack 3L had reached a length of 0.324 inches. In the model, crack 2L was initiated when 3L had reached the same length of 0.324 inches, and was given the same starting length of 0.128 inches. Cracks 5L and 5R were geared to crack 3R; crack 2R was geared to crack 3L; crack 4L was geared to crack 3R; and so on. The Mathematica sub-program 'addcrackprog' written to obtain this sequence is in Appendix D.

An overview graph showing the predicted growth curves for all cracks is in Figure 10-1. The corresponding experimental graph in Figure 8-1 has been overlayed, so that the 2 can be compared directly. The option to exclude bending effects was used for the predicted curves, and cracks were not 'grown' beyond the end rivets. The stress data for the program was obtained from the finite element model of the specimen rather than from strain gauge measurements. Consequently, the crack growth predictions do not rely on any experimental data from the specimen.

There is close agreement between the predicted and experimental MSD crack growth. Particular points of interest are as follows:

- a. The total growth period from initiation of the first crack to the link-up of all 8 rivets was about 75,000 cycles for the specimen and about 60,000 cycles for the computer model - 20% less. The difference is of the same magnitude as the scatter typically found in experimental graphs of da/dN vs ΔK .

- b. The growth period from initiation to first link-up (a double link-up) was 70,000 cycles for the specimen; this period was 93% of the total growth period to link-up of all 8 rivets. The same ratio is evident in the predicted crack growth.
- c. Allowing for the generally faster growth rate of the predicted curves, there is an approximate congruency between the experimental and predicted curves: they have the same overall size and shape; and significant events such as link-ups and step changes in growth rate (due to link-up elsewhere) occur at approximately the same points.
- d. There is general similarity in the crack configurations at any particular time. This can be seen particularly clearly in the development and position of link-ups on each graph.

The predicted ligament sizes just prior to link-up compare with the measured values as follows (all measurements are in inches):

<u>Ligament</u>	<u>Measured</u>	<u>Predicted</u>
3R/4L	0.01 - 0.08	0.06
4R/5L	0.06	0.11
5R/6L	0.10	0.11
6R/7L	0.18	0.14

This accuracy is adequate for the purpose of predicting crack growth curves, since the crack growth rate just prior to link-up is very high.

The larger scale graphs of individual crack growth curves in Figure 8-3 allow a more detailed assessment of the computer model to be made. In these graphs the predicted curve has been shifted such that its initiation point coincides with that of the experimental curve. Both sets of curves are formed by joining points with straight lines. The points are shown only on the experimental curves, where they serve to highlight any aberrations in the data.

The following observations from the larger scale curves point to some potential improvements in the computer model:

- a. The visible growth periods to link-up of the early, independent cracks, ie, 2L, 3L, 3R, 5L and 5R, were underestimated by 0%-18%.
- b. The visible growth periods for the right tip of the lead crack, ie, 4R, 6R, and 7R, were underestimated by 25%-35% - a larger underestimate than for the early cracks.
- c. These same curves also show less acceleration as they approach link-up than those of the early cracks.
- d. The visible growth periods of small cracks which were under the dominant influence of the approaching lead crack for most of their visible growth, ie, 2R, 4L, 6L and 7L, were overestimated by 24%-40% - the reverse of the situation with the early, independent cracks and the lead crack.

Observations c and d indicate that the computer model is underestimating ΔK in 2 situations:

- for a large crack close to an approaching small crack or uncracked rivet hole;
- for most of the growth of a small crack approaching a large crack.

The underestimate in both cases can be explained by the fact that the computer program does not extrapolate outside the envelope of the Rooke and Cartwright curves of β factors. Therefore, it should be possible to correct the problem by programming the solutions in analytical rather than graphical form. As indicated earlier, this would in any case be a logical program improvement.

Observations a and b show that, while the error in predicted growth rate of independent or dominant cracks is small, it has a tendency to increase with crack length. There are

several possible explanations for this conservative trend in the model. Since an increase in crack length is synonymous with an increase in ΔK , these possibilities include:

- a progressive error in the estimate of ΔK for programming reasons of the general type just discussed;
- a progressive error in the estimate of ΔK because assumptions in the theory are not completely valid in the context of the specimen;
- a progressive error in the da/dN vs ΔK data.

In order to narrow down the range of possibilities, more comparisons between the model and test data will have to be done.

While some potential improvements in the program have been identified, the program has so far demonstrated remarkable accuracy. The discrepancies between the individual theoretical and experimental crack growth curves are relatively small when judged against the scatter observed in typical da/dN vs ΔK test data, and against the uncertainties involved in modelling structure as complex as a fuselage splice joint.

Notwithstanding initial success, more test data is needed to validate the program - even for lap splices using 0.040 inches thick 2024-T3 aluminum.

In addition, da/dN vs ΔK data is needed for other thicknesses of 2024-T3 sheet. Data covering a wide range of ΔK at low values of stress ratio are not available in the literature.

The running time of the program is very short. It took less than 10 minutes running time on a SUN work-station to produce the crack growth curves for the test specimen. About 300 growth steps (iterations of the full program) were used to grow cracks from first appearance at rivet #3 to link-up of all 8 rivet holes.

That such good agreement between theory and experiment has been achieved in such a fast program justifies further development of this method of analysis of MSD.

10.2 Comparison of Uniform and Non-uniform MSD

The program was used to investigate the growth characteristics of exactly uniform cracks at all rivet holes.

The results in Figure 10-2 indicate that the total visible growth period to the link-up of all rivets for this worst case MSD scenario would be 30,000 cycles - 50% of that predicted for the scenario in the test specimen.

This information illustrates clearly the potential seriousness of large clusters of uniform cracks.

It also demonstrates how the computer program can be used to guide maintenance policy, or to perform parametric analyses to improve splice design.

11 CONCLUSIONS

The objectives of this thesis were to define the MSD problem in practical terms, to analyze its causes and effects as far as possible from available information, and to define and initiate a research project that would be of value to the aircraft industry.

Sections 2 and 4 of this thesis contain a comprehensive definition and analysis of the MSD problem. Based on this analysis, 2 complementary research projects were defined - one experimental, and one theoretical. The remainder of the thesis describes the results of these 2 projects.

The problem analysis revealed that no concept existed for a coupon-type specimen which would approximate on-aircraft conditions, and allow realistic MSD to be created and studied in a laboratory environment. Therefore, a specimen concept has been developed that is suitable for studying MSD in splices of up to 20 inches in width - equivalent to one frame-bay.

A companion finite element model has been developed. This will facilitate the design of future specimens to simulate the conditions on a particular aircraft in the cracked condition as well as in the uncracked condition.

To prove the specimen concept, it has been implemented in a 10 inch wide version. The materials, splice geometry and stress distribution of this first specimen were chosen to be roughly representative of current generation commercial transport aircraft.

The test was successful in reproducing MSD under realistic conditions. Specifically:

- a. Cracks were initiated naturally at 50% of rivets before any link-up occurred, and corresponded to the non-uniform type of MSD evident in actual MSD scenarios.
- b. Strain gauge measurements at the start of the test confirmed that the nominal stress distribution in the specimen was as intended, ie, representative of the hoop stress gradients measured on aircraft.

- c. Strain gauge measurements after several link-ups indicated that net section stresses were being moderated in the same way that load transfer moderates them in real fuselage structure.
- d. The overall growth rates of visible cracks were roughly comparable to those observed in full scale testing of Boeing aircraft.

By extrapolation, future specimens designed to correspond exactly to the splice configuration on a particular aircraft can be expected to give a useful forecast of crack initiation and growth characteristics for that aircraft. However, this extrapolation should be verified by detailed comparisons with full scale tests.

The mechanism of crack link-up involved a complex pattern of plastic distortion and necking across the full width of the ligament prior to fracture. Fast fracture prior to yielding did not occur. Link-up occurred at fairly small crack separations of 0.06 to 0.18 inches, and the size of the final ligament prior to link-up increased with crack size. A link-up criterion based on plastic zone size is proposed for modelling purposes.

The measured stress distribution and crack behaviour associated with 2 cracks that overlapped during the test indicate that overlapped cracks with a separation in the hoop direction of up to 0.070 inches can be modelled as linked cracks.

The test provided evidence that a weak catch-up mechanism might exist for 2 unequal cracks at the same rivet hole.

The finite element model was found to be an accurate and cost-effective means of predicting the general stress distribution in cracked as well as uncracked specimens. The same approach is recommended for future specimens.

In a parallel study, the MSD test was used to investigate the effectiveness of D Sight, a new and potentially more efficient method of inspecting fuselage splices and other structure. Using D Sight with commonly available photographic equipment, early MSD cracks in the specimen were measured from negatives with a consistency (scatter) of within ± 0.015 inch. The length of cracks tended to be overestimated by amounts generally less than 0.020 inch. Cracks could be seen easily in photographic prints when

their visible length exceeded 0.070 inch. The results confirm the potential usefulness of D Sight as an inspection tool.

The MSD problem analysis identified the need for and the feasibility of developing a cost effective computer model of MSD crack growth covering the period of most interest from a maintenance standpoint - the period from first appearance beyond a rivet head to the link-up of several cracks within a frame bay. Therefore, a computer program was developed which estimates crack growth curves for MSD in fuselage splices using Rooke's superposition and compounding method. The program accepts inputs of splice geometry, material properties, uncracked hoop and bypass stress data, and crack initiation criteria. It produces multiple crack growth curves in graphical form.

To evaluate the computer model, it was programmed to initiate cracks in the same sequence as the test specimen. The 2 sets of crack growth curves were then compared. There is close agreement between the predicted and experimental MSD crack growth. The model was slightly conservative, by 20%, in its prediction of overall growth period from the appearance of the first crack to the link-up of all 8 rivets.

The model routinely applies Irwin's plastic zone correction to all crack lengths, and link-up is defined as occurring when approaching, corrected cracks touch. This link-up criterion appears to work well.

On a SUN workstation, crack growth from initiation to linked cracks totalling 7 inches was performed in less than 10 minutes.

That such good agreement between theory and experiment has been achieved in such a fast program justifies further development of this method of analysis of MSD.

While some potential improvements in the program have already been identified, it might be usable without modification for lap splices using 0.040 inch thick 2024-T3 aluminum, subject to further experimental verification.

The program was used to predict the growth characteristics of exactly uniform cracks at all rivet holes. The results indicate that the total visible growth period to the link-up of all rivets for this worst case MSD scenario would be 50% less than for less uniform

MSD, such as occurred in the test specimen. They highlight that accurate knowledge of MSD crack initiation and growth characteristics is needed to formulate a safe and efficient maintenance policy for a particular aircraft.

Research into MSD and associated damage tolerance analysis, test and inspection methods is likely to continue internationally for many years. The work is needed to support existing fleets, and to improve new designs. The results achieved in this thesis have potential safety and economic benefits for the airline industry, and the research should be extended. The following are some specific recommendations:

- a. Since the concepts underlying the test specimen and the computer model have been validated, it would now be appropriate to attempt to apply them to a specific aircraft. Detailed design and test data would be needed to do this, and so the active participation of an aircraft manufacturer would be necessary.
- b. In addition, considerable basic research can be done. The test specimen should be used to do a study of all parameters likely to affect initiation, initial phase growth and intermediate phase growth. These parameters have already been discussed in this thesis.
- c. The computer model should be used in the same study to help decide on an efficient test matrix. The model cannot be used to study the same number of parameters as the specimen, but can be used evaluate the effects of stress level, stress gradients, splice geometry, rivet stiffness, skin thickness, and skin material on crack growth curves.
- d. Comparisons of predicted and measured growth curves should be made during this parametric study, to improve the accuracy and scope of the computer model.
- e. The feasibility of extending the computer model to cover final phase growth, including the estimation of residual strength, should be considered.

FIGURES

Figure 2-1 Photograph of Aloha Airlines 737 after explosive decompression (Tong October 1989).

Figure 3-1 Residual strength curves for a hypothetical panel illustrating the effect of variations in fracture toughness and panel width (Broek).

Figure 3-2 Residual strength diagram for simple stiffened panel (Vlieger and Broek AGARD-AG 176).

Figure 4-1 Typical fuselage monocoque structure (Niu).

Figure 4-2 Types of skin splice: a. Single shear lap splice; b. Single shear butt splice; c. Double shear splice.

Figure 4-3 Visualization of loads on a single shear lap splice.

Figure 4-4 Photoelastic contours of maximum shear strain for a preliminary specimen shown in Figure 6-1: a. Manufactured head side; b. Driven head side.

Figure 4-5 Rivet details from Aloha Airlines 737 showing MSD (Tong Oct 89).

Figure 4-6 MSD in an unidentified 737 (Swift March 1991).

Figure 4-7 Distribution of MSD crack sizes in an unidentified 727 - total 43,400 cycles (Mayville and Warren).

Figure 4-8 Sketch of MSD on 727 Line #342 - total 39,500 cycles (Bobo).

Figure 4-9 Sketch of early MSD crack distribution in retired 737 subjected to additional pressure cycling by Boeing - total 79,000 cycles (Goranson).

Figure 4-10 Crack growth curves prior to link-up in Boeing test of retired 737 (Goranson).

Figure 4-11 Location of skin splices in the 747-100SR (Maclin).

Figure 4-12 Final crack distribution and cumulative crack growth curve in Boeing test of 747-100SR (Maclin).

Figure 4-13 Crack distribution and cumulative crack growth curve for splice at stringer S44R in Boeing test of 747-400 (Maclin).

Figure 4-14 Crack distribution and cumulative crack growth curve for splice at stringer S44L in Boeing test of 747-400 (Maclin).

Figure 4-15 Crack distribution and cumulative crack growth curve for splice at stringer S6R in Boeing test of 747-400 (Maclin).

Figure 4-16 Flapping failure across 2 frame-bays in a Boeing test of a 737 (Goranson).

Figure 4-17 Calculated residual strength for a lead crack with and without MSD in stiffened sheet; tensile strength=83 ksi, thickness=0.063 inch, stiffener spacing=12 inches (Tong et al April 92).

Figure 4-18 Multiple crack initiation in a preliminary specimen, consisting of 0.032 inch 2024-T3 sheet, 7 inches wide, held with 2 rows of 1/8 inch 2117-T4 dome-head rivets.

Figure 4-19 Stress concentration in tear straps and frames of a narrow-body fuselage caused by longitudinal skin cracks (Miller et al).

Figure 4-20 Calculated stress concentration around a countersunk hole (Newman and Harris).

Material: Skin and tear straps, 2024-T3 clad;
stringer and frames, 7075-T6 clad

Radius: 74 in

Skin: 0.036 in thick

Tear straps: 0.036 in thick

Stringers: 0.028-in-thick hat sections
9.25-in spacing

Frames: 0.040-in-thick Z-sections,
20-in spacing

Figure 4-21 Typical Boeing narrow-body structure (Miller et al).

Figure 4-22 Mid-plane skin stresses (ie, excluding any out-of-plane bending stresses) in a Boeing narrow-body skin panel (Miller et al).

Figure 4-23 Typical Boeing wide-body structure (Miller et al).

Figure 4-24 Mid-plane skin stresses (ie, excluding any out-of-plane bending stresses) in a Boeing wide-body skin panel (Miller et al).

Figure 6-1 Preliminary specimen with rivetted side doublers; nuisance fatigue crack started under doubler.

Figure 6-2 Preliminary specimen with rivetted partial side straps; failed from nuisance fatigue crack under strap.

Figure 6-3 Optimum specimen concept, incorporating full length bonded side straps and full width bonded doublers, implemented as 10 inches wide specimen: a. Countersunk side; b. Driven side.

Figure 6-4 Process of constructing finite element model of MSD test specimen.

Figure 6-5 Hoop stress distribution in uncracked MSD test specimen under 10,000 lb load, predicted using finite element model: a. Range 2 to 23 ksi in steps of 1.5 ksi; b. Range 15.5 to 19 ksi in steps of 0.25 ksi.

Figure 6-6 Maximum shear stress distribution in uncracked MSD specimen under 10,000 lb load, predicted using finite element model.

Figure 6-7 Hoop stress distribution in cracked MSD test specimen under 10,000 lb load, predicted using finite element model.

Figure 6-8 Strain gauge installation on one side of the MSD test specimen; specimen is fully cracked and under full load.

Figure 7-1 Load-frame with specimen installed.

Figure 7-2 Block diagram of FlexTest Digital Servo Controller.

Figure 7-3 Experimental arrangement of FlexTest DSC Controller.

Figure 7-4 Travelling microscope in position for measurements of crack tip position.

Figure 7-5 Diffracto Sight hood installed on load frame to photograph specimen.

Figure 8-1 Overview of all crack growth curves measured on MSD test specimen; co-ordinates of crack tips are in inches from origin 1 inch to left of first rivet; rivet co-ords are 1, 2, 3, 4, etc; same in all other graphs.

Figure 8-2 D Sight photographs of a sample of crack configurations in MSD test specimen (3 pages).

e. Less than 50 cycles before link-up of 3R and 4L; 405,852 cycles.

f. Overlap of 2R and 3L, link-up of 3R and 4L, and cracks at 2L, 4R (new), 5L and 5R; lead crack effectively runs from 2L to 4R; 405,900 cycles.

g. Lead crack approaching 5L; distortion signifying hidden crack at 6L; 409,250 cycles.

h. One cycle before link-up of 4R and 5L; plasticity of ligament makes it appear that link-up has occurred; distortion at 6L is more pronounced, but no crack is visible yet; 409,418 cycles.

Figure 8-2 (continued)

Figure 8-2 (continued)

Figure 8-3 Comparison of theoretical and experimental crack growth curves for the MSD test specimen; data points retained in the experimental curves; 1=L and 2=R, eg, crack[3,2] = crack on R. (3 pages)

Figure 8-3 (continued)

Figure 8-3 (continued)

Figure 8-4 Microscope photographs of progressive failure over 3 cycles of ligament 5L/6L during link-up: a. First cycle - cracks turn, slight necking; b. Second cycle - severe necking; c. Third cycle - fracture.

Figure 8-5 Comparison of measured and predicted hoop stress distributions across the specimen at 1 inch from the critical row in the uncracked condition; measured is the higher curve.

Figure 8-6 Comparison of measured and predicted hoop stress distributions across the specimen at 1 inch from the critical row with advanced MSD; measured is the higher curve.

Figure 8-7 Hoop stress (upper curves) and bending stress distributions at 1 inch ahead of cracked rivet row in MSD test specimen; associated crack configurations are d, f, h and j in Figure 8-2.

Figure 8-8 Variation of measured hoop stress across the specimen as cracks advanced; last 15% of the test shown.

Figure 8-9 Comparison of crack growth curves obtained by measurement of D Sight photographs (shown with data points) with those obtained by direct measurement of the specimen.

Figure 9-1 Free body diagram of critical row in a single shear lap splice.

Figure 9-2 Superposition of several simpler loading conditions to approximate the net loading of a single shear lap slice.

Figure 9-3 Crack growth rate vs stress intensity factor range for centre cracked tension specimens of 2024-T3 0.040 inch thick at a stress ratio of $R=0.02$ (Dawicke et al).

Figure 10-1 Overview of all crack growth curves (the lower set) predicted for the test specimen using computer program 'growprog'; experimental curves shown to demonstrate close agreement.

Figure 10-2 Overview of predicted crack growth curves assuming simultaneous crack initiation at all rivet holes.

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APPENDIX A. Drawing set for MSD test specimen.

REQUIRED TO BE CLOSE TOLERANCE FIT IN PLATES MATCHED PAIR OF 302-1

TOLERANCES X ± .05 XX ± .01 XXX ± .005 (UNLESS NOTED)		DRAWN BY G. F. STAGN		DESIGN PAR PROJECT JGH 00		NATIONAL RESEARCH COUNCIL OF CANADA INSTITUT DE RECHERCHES EN AEROSPACE STRUCTURES ET MATERIAUX		MATERIAL MILD STEEL		QUANTITY 2 P/N A221	
UNITS INCHES		SCALE FULL SIZE		DATE 28 MAY 93		APPROVED GFE		ASSEMBLY DRG. 1st 301 1		QUANTITY 1	
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STRAIGHT WITHIN .005

MATCHED PAIR TO HAVE SAME SHORT DIMENSION WITHIN .01
EDGES B & C SQUARE TO EDGE A WITHIN .005

REMOVE ALL SHARP EDGES AND CORNERS

MATERIAL ALUMINUM ALLOY SHEET 2024-T3 ALCLAD 90-A-250/5 .040 THICK

TOLERANCES		DRAWN BY		DESIGNER		MATERIAL	
XX ± .01	UF EASTAUGH	PROJECT		PROJECT		AS ABOVE	
XXX ± .005	PROJECT	JCH 00		JCH 00		QUANTITY	
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APPROVED	APPROVE	DATE		DATE		DATE	
GFE						A --- (JE) SHIP 1	
National Research Council Canada INSTITUTION FOR AEROSPACE RESEARCH STRUCTURES AND MATERIALS LABORATORY				Consail national de recherches Canada			
SKIN PIECES MATCHED PAIR

STRAIGHT WITHIN .005

REMOVE ALL SHARP EDGES AND CORNERS

MATERIAL ALUMINUM ALLOY SHEET 2024-T3 ALIAD 100 A-25025 040 10011

TOLERANCES XX ± .01 XXX ± .005 UNLESS NOTED UNITS INCHES		DRAWN BY G. CASTAGNA PROJECT JGH 00 SCALE 1/4		DESSINE PAR PROJECT ECHELLE		National Research Council Canada Institut pour l'AEROSPACE RESEARCH STRUCTURES AND MATERIALS LABORATORY		MATERIAL AS ABOVE QUANTITY 4 PER ASSY ASSEMBLY DOW 100 10011		MATERIAL GRANITE QUANTITY 100 10011	
APPROVED GJE		DATE 7 APR 93		SHEET 1		TRAP		A		100 10011

STRAIGHT WITHIN .005

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MATERIAL ALUMINUM ALLOY SHEET 2024-T3 ALCLAD 00-A-250/5 .040 THICK

TOLERANCES XX ±.01 XXX ±.005 (UNLESS NOTED)		DRAWN BY GFC/STAGH		DESIGNE PAR PROJECT JGH 00		PROJECT PROJECT		MATERIAL AS ABOVE		QUANTITY 4 PER ASSY ASSEMBLY Dwg 106 514-1	
UNITS INCHES		SCALE 1/4		FORMELLE 1/4		DATE 7 APR 93		National Research Council Canada INSTITUT FOR AEROSPACE RESEARCH STRUCTURES AND MATERIALS LABORATORY		REVISION A	

TOLERANCES XX ± .01 XXX ± .005 SURFACES FINISHED (BY THE)		DRAWN BY DESIGNER PROJECT LIGHT OIL SCALE 1:4 DATE 18 MAY 53		NATIONAL RESEARCH COUNCIL OF CANADA RESEARCH FOR CANADA RESEARCH FOR AERO SPACE RESEARCH AND MATERIALS		MATERIAL ALUMINUM QUANTITY 4 Pcs 1/2 x 1/2 x 1/4 1/2 x 1/2 x 1/4		APPROVED (BY)	
(1)

Piston

REMOVE ALL SURFACE FILLS AND CORNERS

MATERIAL ALUMINUM 2024 T3 1/2 x 1/2 x 1/4

SCALE 1/4

SPECIMEN SHOWN WITH MANUFACTURED HEAD OF RIVETS UPPERMOST

MM STRAIN GAUGE CEG-500C 120 075 PER 100
MOUNTED ON OPPOSITE SIDES OF THE SHEET
SHEET GROOVES AS SHOWN IN DETAILED VIEW TO
CLEAR HIDDEN GAUGE

MM STRAIN GAUGE CEG-500C 120 075 PER 100
MOUNTED ON OPPOSITE SIDES OF THE SHEET

DETAIL OF GAUGE INSTALLATION

SCALE FULL SIZE

GRID CENTRE LINES

CLEARANCE GROOVE
FOR HIDDEN GAUGE
DEPTH 0.016 MIN
0.020 MAX

SCALE 4X

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APPENDIX B. Manufacturing procedure for MSD test specimen.

MANUFACTURING PROCEDURE FOR SPECIMEN GE304-1

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WARNING

The preparation and application of isopropyl alcohol, 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:

- a drilling and countersinking fixture (locally manufactured - available)*
- an adjustable riveting machine*
- an accurate sheet metal guillotine*
- an oil-free grit-blasting machine and fine aluminum oxide grit*
- compressed nitrogen*
- vacuum pump and hoses*
- vacuum-bagging supplies as used for bonding and curing of metal and composites*
- an autoclave*

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 the following components from this sheet using a guillotine:

Item 1	GE306-1 Skin Pieces Matched Pair	Qty 2 per
Item 3	GE307-1 Short Strap	Qty 4 per
Item 4	GE308-1 Long Strap	Qty 4 per
Item 5	GE309-1 Doubler	Qty 4 per

An accurate, powered guillotine is needed to achieve the required tolerances. The 1.00 x 0.20 chamfers in Item 5 should be hand-ground (small quantities) or milled (several sheets clamped together).

2. 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. For the marks in the y direction (perpendicular to the splice), an optional method of marking is to use a razor blade with *light* pressure, such that only the clad layer is disturbed. If this method is used, continue the cut marks 1/2 inch beyond the rivet rows. They will serve as reference marks for measurements and scaling from D-Sight photographs taken during fatigue testing.
3. Clamp together the 2 skin pieces (Item 1) with an overlap of 3.00 (per drawing GE304-1). 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.
4. 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.
5. Clamp the skin pieces again as described above, and countersink (100 deg) 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.

6. 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.
7. Prepare Items 1,3,4 and 5 for bonding with FM73 as follows:
 - a. Clean with isopropyl alcohol, until a KimWipe tissue lightly dampened with alcohol remains clean after vigorous rubbing.

Note: Hereafter, components must be handled so as to avoid any possibility of contamination of the surfaces to be bonded.

- b. Repeat step a using Methyl Ethyl Ketone (MEK).
 - c. Apply masking tape to each side of Item 1 over the area defined in Figure 1 (this area will not be bonded).
 - d. Grit-blast both sides of all components using fresh #80 (fine) aluminum oxide grit. Use a machine that will not deposit oil on the treated surfaces, and purge the machine of used grit before starting. Grit-blast the surfaces just enough to leave a completely matt finish. Refresh the grit when its effect diminishes. Grit-blasting may result in distortion of the thin sheet. If necessary, restore the flatness of the components by further grit-blasting of the concave surface. After grit-blasting, remove the masking tape, and clean loose material from the components by blasting them with compressed nitrogen.

Note: Do not subsequently solvent-clean or wipe 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.

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.

- f. Brush (using a high quality soft bristle brush) or spray the *grit-blasted surfaces only* with corrosion-inhibiting primer Cyanamid BR127 to a thickness of .0001 to .0003. Brushes or spray equipment must be thoroughly cleaned with MEK prior to use. Application techniques should be practised so as to achieve the correct thickness of primer. 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. Air dry the primer for 30 minutes, and then oven cure it for 30 minutes at 120 ± 6 degC (250 ± 10 degF). After curing, the thickness of the (semi-transparent) primer should be checked against standard colour coupons. 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 the splice area is left exposed; this approach will allow the pieces to be rivetted together (later) without exposing other treated areas to contamination.
- g. 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.

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.

8. If hidden strain gauges are required, install the gauges and terminal strips per drawing GE310-1 on the hidden surface of the countersunk sheet. Use standard procedures, as recommended by the manufacturer, but do not prepare an excessive surface area for each strain gauge. Check that the finished gauge installations do not interfere with the opposite sheet, when the 2 sheets are brought together.
9. Cut 2 pieces $2\frac{7}{8} \times \frac{7}{8}$ of FM73M (or FM73M-OST) supported film adhesive. Place these at each end of the mating face of one of the skin pieces, leaving $\frac{1}{16}$ clearance from the free edges of the skin piece. If necessary, warm the FM73 pieces slightly (43 degC max) using a heat gun to help tack them in place.
10. Place a thin strip (between $\frac{1}{64}$ and $\frac{1}{32}$ 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 rivetted area during curing. Extend both ends of each strip $\frac{1}{4}$ inch beyond the mating surfaces, as shown in Figure 2. One end of each strip will remain unattached until the 2 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; use a piece of non-stick Tacky Tape backing paper to apply the pressure.
11. If required, 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.
12. Bring the mating surfaces of the splice joint together, and clamp the specimen loosely using close fitting bolts through several rivet holes. Tape rivets in place in some of the remaining holes, in order to ensure that the specimen remains correctly aligned during riveting. Then tighten the bolts to clamp the specimen firmly.

13. Rivet the skin pieces together in random sequence. Use a powered, adjustable rivetting machine. Set the machine to produce the required driven head dimensions. Unless otherwise specified, the driven head diameter should be 0.24 ± 0.01 . This dimension determines the clamping force applied by a given rivet.
14. 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.
15. The uncured FM73 in the rivetted specimen will have a shop life of up to 30 days at 24 degC, and so the remainder of this MP should be completed within this shop life.
16. Cut pieces of FM73M (or FM73M-OST) und. size by between 1/16 and 1/8 on all dimensions corresponding to each of the following components:

Item 3	GE307-1 Short Strap	GE307-1	Qty 4
Item 4	GE308-1 Long Strap	GE308-1	Qty 4
Item 5	GE309-1 Doubler	GE309-1	Qty 4

If these pieces of FM73 will not be used within 8 hours, seal them in a plastic bag, and store them in a freezer.

17. Stick the FM73 pieces to Items 3, 4 and 5, leaving between 1/32 and 1/16 clearance with all edges. Working with one side of the rivetted specimen at a time, stick the straps and then the doublers to the specimen. Ensure that the straps are flush (± 0.005 per drawing) with the edges of the specimen. Where the short straps and skin pieces butt together, the gap must be 0.04 ± 0.01 (per drawing). The doublers must be 4.00 ± 0.01 (per drawing) from the centre of the specimen. Note that the edges of straps, doublers and skin do not necessarily have to be flush at the 2 extreme ends of the specimen. If the FM73 is not tacky enough to hold the straps and doublers in place, use a heat gun to warm the required area slightly (43 degC max). Only do this when the component is correctly positioned,

since any subsequent attempt to adjust it will probably damage the FM73. Strong manual pressure for a few seconds is usually enough to hold components in place until all components on one side of the specimen have been positioned accurately; the heat gun can then be used to tack the components firmly.

18. Vacuum bag the specimen using standard techniques. Care is needed:

- to avoid loss of glue from the bonded areas,
- to prevent glue from migrating and contaminating the lap joint,
- and to obtain good glue fillets.

To these ends, special techniques and precautions have been developed, as follows:

- a. Place the specimen between 2 layers of impermeable 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. Use a fresh outer bag for each specimen. The vacuum hose mushroom attachment must be placed at least 6 inches from one end of the specimen, and the bag must be made long enough to accommodate it.
- c. Do not use any fibreglass breather material or other padding except as follows:
 - Lay a thin rectangular layer over the driven heads of the rivets; this layer must only slightly overlap the rivet heads, and must be positioned such that there are no creases in the plastic which could allow glue to migrate from the side straps to any rivets.
 - Place several stepped layers in the area of the hose attachment, in order to provide enough airway to properly evacuate the bag, while avoiding excessive wrinkling of the bag near the specimen.

- d. Place an even fillet of Tacky Tape along the edges of the lap joint. Join this fillet with the exposed ends of Tacky Tape emerging from each end of the joint, to form a closed loop. The objective is to prevent glue migrating into the joint from any source.
 - e. When everything is positioned satisfactorily, apply suction. Smooth out wrinkles as far as possible. 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. Ensure that there are no wrinkles, creases or fillets anywhere that might allow glue to migrate in an unwanted manner. Maintain suction until the specimen has been cured (see below).
19. Cure the FM73 in an autoclave at 40 ± 5 psi to the following schedule:
- Heat up in 30 minutes to 120 degC ($250 \pm \text{degF}$)
 - Hold 60 minutes at $120 \pm 3 \text{ degC}$ ($250 \pm 5 \text{ degF}$).
- During the cure, the specimen should be restrained (eg, with shims and weights or clamps) so as to minimize warping. Maintain the restraint until the specimen has cooled. The photographs in Figure 3 show a cured specimen still in the vacuum bag.
20. Remove the glue flashing around the outer edges of the specimen. Use a strong, sharp knife or other suitable tool. If any aluminum must be removed, ensure that the dimensions on drawing GE304-1 are maintained.
21. Machine the 2 x 1.251 holes and the 44 x 9/16 holes (a NC program already exists at NRC Building M4).
22. Inspect the finished specimen to drawing GE304-1 and record any discrepancies.
23. If required, install external strain gauges per drawing GE310-1, following the manufacturers recommended procedures. Do not prepare a larger surface area for each gauge than necessary.

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PM-1 3½"x4" PHOTOGRAPHIC MICROCOPY TARGET
NBS 1010a ANSI/ISO #2 EQUIVALENT

PRECISIONSM RESOLUTION TARGETS

FIGURES

- 1 Areas of sheets to be masked during grit-blasting.
- 2 Application of S-M Tacky Tape as barriers to prevent glue migration into the rivetted area.
- 3 Photographs of both sides of specimen in vacuum bag after curing in an autoclave; additional 'Tacky Tape' barrier shown around 3 sides of rivetted area is optional.

Figure 1 Areas of sheets to be masked during grit-blasting.

Figure 2 Application of S-M Tacky Tape as barriers to prevent glue migration into the rivetted area.

Figure 3 Photographs of both sides of specimen in vacuum bag after curing in an autoclave; additional 'Tacky Tape' barrier shown around 3 sides of rivetted area is optional.

APPENDIX C. Computer program 'growprog' for predicting crack growth curves for MSD in fuselage lap splices.

(* PROGRAM TO COMPUTE STRESS INTENSITY FACTORS AND
CRACK GROWTH RATES OF MULTIPLE SITE FATIGUE CRACKS
IN A FUSELAGE LONGITUDINAL SKIN SPLICE *)

(*

NOTES:

- (1) The program computes crack growth curves for intermediate phase MSD crack growth, ie, from small but visible cracks to link-up of several rivet holes. A graph giving an overview of all crack growth curves is automatically displayed on the screen and filed as 'alltips' when the program has been run. Graphics files of individual curves are created but not displayed. They must be converted to printable Post Script code using the UNIX 'psfix' command, eg, psfix tip31>tip31.ps.
- (2) The program allows an option for including out-of-plane bending effects, if required. Only secondary bending is computed. Pillowing bending is not computed.
- (3) All fuselage and joint loading conditions are included in the program except for longitudinal stress and in plane shear stress. The influence of stiffeners (frames and straps) is accounted for in the data input of stress distribution and bending factor distribution - as contained in data files "stress.dat" and "bendpos.dat".
- (4) Position along the rivet row is defined by coordinate x, with the origin located 1 inch to the left of the centreline of the first rivet.
- (5) Data files must comprise a series of values separated by spaces (any number) or commas. Files are required as follows:
 - a. Create a data file "crack.dat" containing a list of distances of crack tips from rivet CL as measured using a travelling microscope. If there is no crack on one side of a hole, enter 0 for that tip. If there is a linked crack, enter 99. The data file must not start or end with 99 or any other value>1.

These distances will be used to construct crack tip and hole edge coordinates, tips[i,j], based on an origin 1 inch to the left of the first rivet, where i = rivet no and j = 1 for left and

2 for right. In tips(i,j), the values 99 retain the meaning in Note (6)a. In x(i,j), these are translated into an array of crack tip coordinates only; for this purpose, a hole edge is treated as a crack tip if the other edge is cracked.

- b. Create a data file "stress.dat" containing the values of the nominal hoop stress applicable to each rivet location in sequence from left to right. In the MSD test specimen, and in actual aircraft skin panels, there is a longitudinal stress gradient. Input the values for an uncracked joint at a line close to the joint, but away from any local stress concentrations at the rivet holes, eg, at 1 inch from the critical row. The values should be the average over one rivet pitch.
- c. Create a data file "bpratio.dat" containing the values of the bypass ratio applicable to each rivet location. Bypass ratio is the ratio of the local bypass stress to the local nominal stress. It can be obtained from measurements or finite element analysis. The value of bpratio increases with proximity to frames and straps.
- d. Create a data file "bendpos.dat". Enter factors for each rivet location that represent the reduction in secondary bending due to any bending constraint provided by frames and straps.

*)

(* OPEN FILE FOR KEY OUTPUT WITH STUDY # AS EXTENSION msdoutput.# *)

```
idno=InputString("Enter study #: ");
file=ToString(StringForm["\"msdoutput.\"\"\",idno]);
file=OpenWrite[file];
Write[file,OutputForm["\n"]];
Write[file, OutputForm[StringForm[
    "OUTPUT OF \"growprog\" STUDY #\"\",idno] ]];
```

(* INTERACTIVE INPUT OF JOINT AND CRACK GEOMETRY *)

```
sameasbefore= InputString("Do you wish to use the
same joint\geometry and material as already entered\nduring
the current Mathematica session? (y)/n: ");
```

```
If[
    sameasbefore=="n",
```

```
t= Input["Enter sheet thickness in inches: t = "]; (* Thickness
is constant *)
```

```

yieldstrength=Input("Enter yield strength of the material in ksi:
  yieldstrength= ");
p= Input("Enter rivet pitch in inches: p = "); (* Pitch is constant *)
n= Input("Enter total no of rivets: n = ");
d= Input("Enter rivet hole effective dia: d = ");
r= 0.5*d;

stressratio= Input("Enter stress ratio of constant amplitude
fatigue loading: stressratio = ");

l1= Input("Enter distance between outer\nrivet row of lap joint
and equivalent 'clamped' edge: l1 = ");
l3= Input("Enter distance between 2 outer\nrows of lap joint: l3 = ");
l2= 0.5*l3;

  ];

pi= 3.14159;

kbendquery=InputString("Do you wish to exclude secondary\n
bending in the computation of crack growth rate? (y)/n: ");

(* INPUT DISTRIBUTION OF NOMINAL HOOP STRESS sigma(x) AND
ESTABLISH FUNCTIONS FOR BYPASS STRESS AND TRANSFER STRESS *)

stressdata= ReadList("stress.dat", Number);

stressdistrib= {};

bpratiodata= ReadList("bpratio.dat", Number);

bpratiodistrib= {};

Do[
  stressdistrib= Append[stressdistrib, 1+(i-1)*p];
  stressdistrib= Append[stressdistrib, stressdata[[i]] ];
  bpratiodistrib= Append[bpratiodistrib, 1+(i-1)*p];
  bpratiodistrib= Append[bpratiodistrib, bpratiodata[[i]] ],
  {i,1,n}
];

stressdistrib= Partition[stressdistrib, 2];
stressz= Fit[stressdistrib, {1,z,z^2}, z];

bpratiodistrib= Partition[bpratiodistrib, 2];
bpratioz= Fit[bpratiodistrib, {1,z,z^2}, z];

bpratio[x_]:= bpratioz /. {z->x};
sigma[x_]:= stressz /. {z->x}; (* Nominal stress *)
sigmabp[x_]:= sigma[x]*bpratio[x]; (* Bypass stress *)
sigmalt[x_]:= sigma[x]*(1-bpratio[x]); (* Stress transferred at the
critical rivet row *)

```

(* DEFINE GRAPHS FOR beta[i,j,q] THE BETA FACTOR FOR A CRACK "i"
DUE TO INTERACTION WITH A REMOTE CRACK "q" IN A TENSION FIELD
IN AN INFINITE PLATE *)

(* Graph 1 - 2 collinear cracks, furthest tip - Rooke
& Cartwright Fig 74 *)

```
curve1a= InterpolatingPolynomial[{ {0,1.04}, {1,1.03},
    {4,1.02}, {6,1.03}, {8,1.04}, {9,1.05} }, z];
curve1b= InterpolatingPolynomial[{ {0,1.15}, {2,1.14},
    {4,1.13}, {6,1.13}, {8,1.15}, {9,1.17} }, z];
curve1c= InterpolatingPolynomial[{ {0,1.31}, {2,1.31},
    {4,1.31}, {6,1.3}, {8,1.3}, {9,1.31} }, z];
curve1d= InterpolatingPolynomial[{ {0,1.5}, {2,1.48},
    {4,1.45}, {6,1.44}, {8,1.46}, {9,1.51} }, z];
curve1e= InterpolatingPolynomial[{ {0,1.67}, {2,1.62},
    {4,1.59}, {6,1.58}, {8,1.6}, {9,1.64} }, z];
curve1f= InterpolatingPolynomial[{ {0,1.81}, {2,1.75},
    {4,1.72}, {6,1.71}, {8,1.74}, {9,1.78} }, z];
curve1g= InterpolatingPolynomial[{ {0,2.4}, {2,2.3},
    {4,2.26}, {6,2.24}, {8,2.28}, {9,2.34} }, z];
```

(* Graph 2 - 2 collinear cracks, nearest tip - Rooke
& Cartwright Fig 75 *)

```
curve2a=InterpolatingPolynomial[{ {0,1.0}, {2,1.01}, {4,1.03},
    {6,1.04}, {8,1.1}, {9,1.18} }, z];
curve2b=InterpolatingPolynomial[{ {0,1.04}, {2,1.06}, {4,1.08},
    {6,1.11}, {8,1.21}, {9,1.37} }, z];
curve2c=InterpolatingPolynomial[{ {0,1.08}, {2,1.09}, {4,1.12},
    {6,1.17}, {8,1.32}, {9,1.53} }, z];
curve2d=InterpolatingPolynomial[{ {0,1.15}, {2,1.17}, {4,1.22},
    {6,1.3}, {8,1.52}, {9,1.8} }, z];
curve2e=InterpolatingPolynomial[{ {0,1.35}, {2,1.38}, {4,1.46},
    {6,1.59}, {8,1.9}, {9,2.34} }, z];
curve2f=InterpolatingPolynomial[{ {0,1.5}, {2,1.58}, {4,1.68},
    {6,1.85}, {8,2.25}, {9,2.8} }, z];
curve2g=InterpolatingPolynomial[{ {0,1.66}, {2,1.74}, {4,1.85},
    {6,2.05}, {8,2.53}, {9,3.15} }, z];
curve2h=InterpolatingPolynomial[{ {0,1.8}, {2,1.9}, {4,2.03},
    {6,2.26}, {8,2.78}, {9,3.48} }, z];
curve2i=InterpolatingPolynomial[{ {0,2.4}, {2,2.52}, {4,2.73},
    {6,3.08}, {8,3.83}, {9,4.5} }, z];
```

(* Graph 4 - collinear hole and crack, nearest tip - Rooke & Cartwright Fig 110 *)

(* Note: For most current lap joints, $r/c < 0.2$. Therefore, graph 3 - Rooke & Cartwright Fig 109 is not needed (betahole[i,j,k]=1 (approx) for furthest tip). Also, graph 4 can be limited to curves for $r/c = 0.1$ and 0.2 . *)

```
curve4a= InterpolatingPolynomial[{ {0,1.0}, {0.2,1.005}, {0.4,1.01},
  {0.6,1.015}, {0.8,1.065}, {0.9,1.13} }, z];
curve4b= InterpolatingPolynomial[{ {0,1.02}, {0.2,1.03}, {0.4,1.04},
  {0.6,1.07}, {0.8,1.19} }, z];
```

(* DEFINE GRAPHS 5 AND 6 FOR betabend *)

(* Graph 5 - single crack at a hole: pure out of plane bending - Rooke & Cartwright Fig 196 *)

```
curve5= Fit[{ {1,0}, {1.2,.68}, {1.3,.76},
  {1.4,.81}, {1.6,.88}, {2.0,.92},
  {3.0,.90}, {4,.87}, {5.0,.84}, {6.0,.84},
  {50,1.0} }, {1,1/z,1/z^2,1/z^3}, z];
```

(* Graph 6 - 2 cracks at a hole, pure out of plane bending - Rooke & Cartwright Fig 196 *)

```
curve6= Fit[{ {1,0}, {1.2,.68}, {1.3,.77},
  {1.4,.83}, {1.6,.90}, {2.0,.96},
  {3.0,.99}, {5.0,1.0}, {6.0,1.0}, {100,1} },
  {1,1/z,1/z^3,1/z^5,1/z^7}, z];
```

(* DEFINE ARRAYS SIZED BY RIVET COUNT *)

Array[tips,{n,2}]; (* Crack tip and hole edge coords in pairs *)

Array[spacing,{n,2}]; (* Spacing of adjacent crack tips or hole edges *)

Array[growpoints,{n,2}]; (* 4D matrix containing coords of crack tip positions vs cycles for plotting crack growth curves. *)

```
Array[growplot, {n, 2}]; (* Graphics object of each crack
                           growth curve *)
```

```
(* INITIALIZE SOME VARIABLES FOR THE MAIN ITERATION LOOP *)
```

```
Do[
  growpoints[i, j] = {},
  {i, 1, n}, {j, 1, 2}
];
```

```
(* INPUT INITIAL CRACK TIP COORDINATES INTO MATRIX datapairs AND
   INITIAL CYCLES COUNT INTO VARIABLE cycles *)
```

```
data = ReadList["crack.dat", Number]; (* See notes above *)
datapairs = Partition[data, 2];
```

```
cycles = Input["Enter the cycles applicable to the\n
starting crack configuration: cycles = "];
```

```
(* START OF MAIN ITERATION LOOP *)
```

```
limit = Input["Enter a limit on the number of iterations\n
to be performed: limit = "];
```

```
Do[ (* See near end of program for closure of this Do loop *)
```

```
Print[""]; Print[""];
Print[StringForm[
```

```
  "ITERATION # '", iteration]]];
```

```
Write[file, OutputForm[StringForm[
  "ITERATION # '" \n", iteration]]];
```

```
If[
  Sum[datapairs[[i, j]], {i, 1, n}, {j, 1, 2}] >= 99 * 2 * (n - 1),
  Break[]
]; (* This condition closes the iteration when
    all ligaments are linked *)
```

```
(* INCREMENT THE CYCLE COUNT cycles ON THE
SECOND AND SUBSEQUENT ITERATIONS BY AN AMOUNT
BASED ON THE MAX da/dN VALUE COMPUTED
FOR THE LAST ITERATION *)
```

```
If[
  iteration!=1,
  dn=Floor[0.01/maxdadn]; (* Sets a constant growth increment
                           of 0.01 in fcr all values of dadn *)
  cycles=cycles+dn
];
```

```
(* COMPUTE CRACK INCREMENTS da AND ADD THEM TO datapairs *)
```

```
If[
  iteration!=1,
  da={};

  Do[
    da=Append[da,dadndawicke[i,j]*dn],
    {i,1,m},{j,1,2}
  ]; (* Creates a vector of crack increments *)

  next=1; (* Initializes the value of next *)

  Do[
    Which[
      datapairs[[i,1]]==datapairs[[i,2]]==0,
      Goto[hop],
      datapairs[[i,j]]==99,
      Goto[hop],
      datapairs[[i,j]]<99,
      (* This means that we are now at the next crack tip
         and can add the next increment *)
      datapairs[[i,j]]=datapairs[[i,j]]+da[[next]];
      next=next+1
    ];
    Label[hop],
    {i,1,n},{j,1,2}
  ]
];
```

```
(* REPEAT PART OF ITERATION LOOP IF LINKUP HAS OCCURRED *)
```

```
( (* Open procedure *)
```

```
Label[linkuprepeat];
```



```
(* DETERMINE WHETHER ANY NEW CRACKS ARE TO BE ADDED
   TO THE CURRENT CONFIGURATION *)
```

```
<<addcrackprog; (* This is a separate Mathematica mini-program.
                  If no crack are to be injected, the file
                  must exist with no text. *)
```

```
(* REDEFINE THE VARIABLES THAT CONTROL THE SIZES OF ARRAYS
   SO THAT THEY CORRESPOND WITH THE NEW CRACK CONFIGURATION *)
```

```
m=0;
y=0;
```

```
Do[
  If[
    datapairs[[i,1]]==datapairs[[i,2]]==0,
    y=y+1,
    If[
      datapairs[[i,1]]!=99,
      m=m+1
    ]
  ],
  {i,1,n}
]; (* Redefines m and y for the changed crack configuration *)
```

```
(* SET UP ARRAY VARIABLES WHOSE SIZE
   DEPENDS ON CRACK CONFIGURATION *)
```

```
Array[x,{m,2}]; (* Crack tip coords in pairs - excludes
                  uncracked holes *)
```

```
Array[a, m]; (* Crack half-lengths *)
```

```
Array[o,m]; (* Crack crack centre coords *)
```

```
Array[h,y]; (* Coords of centres of uncracked holes *)
```

```
Array[bcrack,{m,m}]; (* Crack to crack separation for
                      interaction - 1st digit represents primary crack *)
```

```
Array[bhole,{m,y}]; (* Crack to hole separation for interaction *)
```

```
Array[c,{m,y}]; (* Crack centre to hole centre separation for
                  interaction *)
```

```
Array[beta,{m,2,m}]; (* Beta[i,j,q] for crack to crack interaction,
```

where $K_0 = \sigma[i] \cdot \sqrt{\pi \cdot a[i]}$ - σ is
bypass stress or transfer stress, as applicable *)

Array[betahole, (m,2,y)]: (* Beta[i,j,k] for hole to crack interaction
where $K_0 = \sigma[i] \cdot \sqrt{\pi \cdot a[i]}$ - σ is
bypass stress or transfer stress, as applicable *)

Array[s,m]: (* Distance between cg of rivet load(s) and centre
of crack *)

Array[betabend, (m,2)]: (* Beta factor due to bending, where
 $K_0 = \sigma_{\text{bend}}[i] \cdot \sqrt{\pi \cdot a[i]}$ *)

Array[cgholes,m]: (* Coords of point of act on of wedge loads *)

Array[bendfactor,m]: (* Ratio of bending stress to nominal stress *)

Array[wedgeload,m]: (* Hypothetical wedge load equal in magnitude
to the unidirectional rivet load; for cracks containing several
rivet holes, wedgeload is computed as a single resultant *)

Array[kbp, (m,2)]: (* SIF due to bypass stress *)

Array[kwedge, (m,2)]: (* SIF associated with wedge loads at rivet
locations *)

Array[nooffirstrivet,m]: (* No of first rivet in a crack *)

Array[noofflastrivet,m]: (* No of last rivet in a crack *)

Array[klntension, (m,2)]: (* SIF associated with a nominal stress
equal to the stress transferred at the critical rivet row *)

Array[ktransfer, (m,2)]: (* Net SIF due to load transfer *)

Array[kinterim, (m,2)]: (* Interim SIF for computation of Irwin
plastic zone correction to use in next
iteration for K; only one iteration programmed *)

Array[kbend, (m,2)]: (* SIF due to out of plane bending caused
by joint eccentricity *)

Array[ktotal, (m,2)]: (* Total SIF *)

Array[dadndawicke, (m,2)]: (* Crack growth rate using centre-cracked
tension specimen data for 0.040 2024-T3 R=0.02
from Dawicke Apr 92 *)

Array[dadnpart1, (m,2)]: (* Crack growth rate using centre-cracked
tension specimen data for 0.080 2024-T3 from Part1 &
Schijve Jan 92 *)

Array[plasticzone, (m,2)]: (* Plane stress monotonic plastic zone
size based on Irwin/Broek *)

Array[ligament, (m,2)]: (* Size of ligament associated with each
crack tip *)

(* DEFINE INTERPOLATION FUNCTIONS FOR THE GRAPHS
FROM Rooke and Cartwright *)

```
graph1[i_,j_,q_]:=
(
  u= Min[a[i]/bcrack[i,q], 0.9];
  v= Min[a[q]/bcrack[i,q], 10];
  Which[
    0<=v<0.4,
    1+(curve1a-1)*(v/.4) /.{z->u},
    0.4<=v<1,
    curve1a+(curve1b-curve1a)*(v-.4)/(1-.4) /.{z->u},
    1<=v<2,
    curve1b+(curve1c-curve1b)*(v-1)/(2-1) /.{z->u},
    2<=v<3,
    curve1c+(curve1d-curve1c)*(v-2)/(3-2) /.{z->u},
    3<=v<4,
    curve1d+(curve1e-curve1d)*(v-3)/(4-3) /.{z->u},
    4<=v<5,
    curve1e+(curve1f-curve1e)*(v-4)/(5-4) /.{z->u},
    5<=v<=10,
    curve1f+(curve1g-curve1f)*(v-5)/(10-5) /.{z->u}
  ]
); (* Linear interpolation. No extrapolation
   from programmed curves; therefore, output may
   be low for very close crack tips and/or
   a small crack with large interacting crack *)
```

```
graph2[i_,j_,q_]:=
(
  u= Min[a[i]/bcrack[i,q], 0.9];
  v= Min[a[q]/bcrack[i,q],10];
  Which[
    0<=v<0.2,
    1+(curve2a-1)*v/0.2/.{z->u},
    0.2<=v<0.4,
    curve2a+(curve2b-curve2a)*(v-0.2)/(0.4-0.2) /.{z->u},
    0.4<=v<0.6,
    curve2b+(curve2c-curve2b)*(v-0.4)/(0.6-0.4) /.{z->u},
    0.6<=v<1,
    curve2c+(curve2d-curve2c)*(v-0.6)/(1-0.6) /.{z->u},
    1<=v<2,
    curve2d+(curve2e-curve2d)*(v-1)/(2-1) /.{z->u},
    2<=v<3,
    curve2e+(curve2f-curve2e)*(v-2)/(3-2) /.{z->u},
    3<=v<4,
    curve2f+(curve2g-curve2f)*(v-3)/(4-3) /.{z->u},
    4<=v<5,
    curve2g+(curve2h-curve2g)*(v-4)/(5-4) /.{z->u},
    5<=v<=10,
    curve2h+(curve2i-curve2h)*(v-5)/(10-5) /.{z->u}
  ]
); (* Linear interpolation. No extrapolation
   from programmed curves; therefore, output may
   be low for very close crack tips and/or for a large
```

interacting crack. *)

```
graph4[i_,k_]:=
{
  u= Min[a[i]/bhole[i,k], 0.9];
  w= Min[r/c[i,k], 0.2];
  Which[
    0<w<0.1,
    1+(curve4a-1)*(w-0)/(0.1-0) /. {z->u},
    0.1<w<0.2,
    curve4a+(curve4b-curve4a)*(w-0.1)/(0.2-0.1) /. {z->u}
  ]
}; (* Linear interpolation. No extrapolation
   from programmed curves; therefore, output may
   be low for large cracks and crack tips close to hole *)
```

```
graph5[i_,j_]:=
{
  aoverr= a[i]/r;
  curve5/.{z->aoverr}
};
```

```
graph6[i_,j_]:=
{
  aoverr= a[i]/r;
  curve6/.{z->aoverr}
};
```

(* ENTER VALUES IN ARRAYS tips[i,j] AND x[i,j]
BASED ON CURRENT VALUES IN datapairs *)

```
Do[
  Which[
    datapairs[[i,j]]==0,
    tips[i,j]=(1+(i-1)*p+((-1.)^j)*r),
    0<datapairs[[i,j]]<99,
    tips[i,j]=(1+(datapairs[[i,j]])*(-1)^j+(i-1)*p),
    datapairs[[i,j]]==99,
    tips[i,j]=99
  ];
  Print[StringForm["tips['','] = '", i, j, tips[i,j] ]],
    {i,1,n}, {j,1,2}
];
```

cracktips= {}; (* Empty vector of crack tip coordinates *)
tiptypes= {}; (* Empty vector of crack tip types: "0" will signify
a hole, and "1" will signify a real crack tip *)

```
Do[
  Which[
    datapairs[[i,1]]==datapairs[[i,2]]==0, Goto[hop],
    datapairs[[i,j]]==99, Goto[hop],
```

```

0<datapairs[[i,j]]<99,
cracktips=Append[cracktips,tips[i,j]];
tiptypes=Append[tiptypes,1],
datapairs[[i,j]]==0,
cracktips=Append[cracktips,tips[i,j]];
tiptypes=Append[tiptypes,0]
];
Label[hop],
{i,1,n},{j,1,2}
];

cracktips= Partition[cracktips,2]; (* Convert vector to pairs
matrix *)
tiptypes= Partition[tiptypes,2]; (* Convert vector to pairs
matrix *)
Write[file,OutputForm[""]];

Do[
  x[i,j]=cracktips[[i,j]];

  Write[file,OutputForm[StringForm[
    "x['',']='',i,j,x[i,j] ]]];

  Print[StringForm["x['',']='', i, j, x[i,j] ],
    {i,1,m},{j,1,2}
];

(* ADD THE CRACK TIP POSITIONS WITHOUT
PLASTIC ZONE CORRECTIONS TO THE PLOT COORDS FILE *)

Do[
  If[
    0<datapairs[[i,j]]<99,
    growpoints[i,j]=Append[growpoints[i,j],{tips[i,j],cycles}]
  ],
  {i,1,n},{j,1,2}
];

(* INITIALIZE PLASTIC ZONE SIZES TO ZERO *)

Do[
  plasticzone[i,j]=0,
  {i,1,m},{j,1,2}
];

```

```
(* INITIATE DO LOOP FOR IRWIN PLASTIC ZONE CORRECTION *)
```

```
Do[
  Print[StringForm["irwin= '",irwin]];
```

```
(* AMEND tips[i,j] and x[i,j] TO INCLUDE PLASTIC
  ZONE CORRECTION *)
```

```
If[
  irwin==2,
  Write[file,OutputForm[""]];
  Do[
    Which[
      j==1,
      x[i,j]=x[i,j]-plasticzone[i,j]/2;
      iconv=2+Floor[(x[i,j]-1)/p];
      tips[iconv,1]=tips[iconv,1]-plasticzone[i,j]/2,
      j==2,
      x[i,j]=x[i,j]+plasticzone[i,j]/2;
      iconv=1+Floor[(x[i,j]-1)/p];
      tips[iconv,2]=tips[iconv,2]+plasticzone[i,j]/2
    ]; (* iconv converts positions in array x to positions
        in array tips *)

    Print[StringForm["plasticx['','] = '",i,j,x[i,j] ]];
    Write[file,OutputForm[StringForm[
      "plasticx['','] = '",i,j,x[i,j] ]]],
      {i,1,m},{j,1,2}
    ];

  Write[file,OutputForm[""]];

  Do[
    Print[StringForm["plastictips['','] = '",i,j,tips[i,j] ]];
    Write[file,OutputForm[StringForm[
      "plastictips['','] = '",i,j,tips[i,j] ]]],
      {i,1,n},{j,1,2}
    ]
  ];
```

```

(* COMPUTE CRACK LENGTHS a[i], CO-ORDS OF CENTRES OF
   CRACKS o[i], CO-ORDS OF UNCRACKED HOLES h[i],
   SEPARATIONS FROM REMOTE CRACKS bcrack[i,q],
   SEPARATIONS FROM UNCRACKED HOLES bhole[i,k], AND
   CENTRE DISTANCES BETWEEN CRACKS AND HOLES c[i,k] *)

(* Note:  "i" is the no of the prime crack;
           "q" is the no of the remote, interacting crack;
           "k" is the no of the uncracked hole (all numbering
           is left to right). *)

Write[file,OutputForm["\n"]];

Do[
  a[i]=0.5*(x[i,2]-x[i,1]);
  o[i]=0.5*(x[i,1]+x[i,2]);

  If[
    irwin==1,
    Print[StringForm[
      "Length of crack # ``: ``", i, 2*a[i] ]];
    Print[StringForm[
      "  Coord of centre:      ``", o[i] ]];
    Print[""];
    Write[file,OutputForm["\n"]];
    Write[file,OutputForm[StringForm[
      "Length of crack # ``: ``", i, 2*a[i] ]]];
    Write[file,OutputForm[StringForm[
      "  Coord of centre:      ``", o[i] ]]];
  ],
  {i,1,m}
];

hh=(); (* Vector of co-ords of uncracked holes *)

Do[
  If[
    datapairs[[i,1]]==datapairs[[i,2]]==0,
    hh=Append[hh,i*p]
  ],
  {i,1,n}
];

Do[
  h[i]=hh[[i]],
  {i,1,y}
];

Do[
  Which[
    i>q, bcrack[i,q]=o[i]-x[q,2],
    i<q, bcrack[i,q]=x[q,1]-o[i],
    i==q, bcrack[i,q]=999
  ];
  Print[""];
  Print[StringForm["Interaction spacing crack `` to crack ``: ``",
    i, q, bcrack[i,q] ]],
  {i,1,m}, {q,1,m}
];

```

```

Do[
  c[i,k]=Abs[o[i]-h[k]];
  Print[""];
  Print[StringForm["Centre spacing crack `` to hole ``: ``",
    i, k, c[i,k] ]];
  bhole[i,k]=c[i,k]-r;
  Print[""];
  Print[StringForm["Interaction spacing crack `` to hole ``: ``",
    i, k, bhole[i,k] ]],
  {i,1,m}, {k,1,y}
];

(* COMPUTE beta FOR CRACK TO CRACK INTERACTION *)

Do[
  If[
    j==1,
    Which[
      q>i, beta[i,j,q]=graph1[i,j,q],
      q<i, beta[i,j,q]=graph2[i,j,q],
      q==i, beta[i,j,q]=1
    ],
    Which[
      q>i, beta[i,j,q]=graph2[i,j,q],
      q<i, beta[i,j,q]=graph1[i,j,q],
      q==i, beta[i,j,q]=1
    ]
  ];

  If[
    beta[i,j,q]==Null,
    beta[i,j,q]=1;
    Write[file, OutputForm[""]];
    Write[file, OutputForm[StringForm[
      "BETA OFF GRAPH a[``]=`` a[``]=`` bcrack[``]=``",
      i,a[i],q,a[q],i,q,bcrack[i,q] ]]]
  ];

  If[
    a[i]/bcrack[i,q]>0.9,
    Write[file, OutputForm[""]];
    Write[file, OutputForm[StringForm[
      "BETA OFF GRAPH a[``]=`` a[``]=`` bcrack[``]=``",
      i,a[i],q,a[q],i,q,bcrack[i,q] ]]]
  ];

  Print[""];
  Print[StringForm["beta[``][``][``] = ``", i, j, q, beta[i,j,q] ]],
  {i,1,m}, {j,1,2}, {q,1,m}
];

```


(* COMPUTE betahole FOR CRACK TO HOLE INTERACTION *)

```

Do[
  If[
    j==1,
    Which[
      h[k]>o[i], betahole[i,j,k]=1,
      h[k]<o[i], betahole[i,j,k]=graph4[i,k]
    ],
    Which[
      h[k]>o[i], betahole[i,j,k]=graph4[i,k],
      h[k]<o[i], betahole[i,j,k]=1
    ]
  ];

  If[
    betahole[i,j]==Null,
    betahole[i,j]=1;
    Write[file, OutputForm[""]];
    Write[file, OutputForm[StringForm[
      "BETAHOLE OFF GRAPH a['']='' bhole['','']='' c['','']=''",
      i,a[i],i,k,bhole[i,k],i,k,c[i,k] ]]]
  ];

  If[
    a[i]/bhole[i,k]>0.9,
    Write[file, OutputForm[""]];
    Write[file, OutputForm[StringForm[
      "BETAHOLE OFF GRAPH a['']='' bhole['','']='' c['','']=''",
      i,a[i],i,k,bhole[i,k],i,k,c[i,k] ]]]
  ];

  Print[""];
  Print[StringForm["betahole['','']= ",
    i, j, k, betahole[i,j,k] ]],
    {i,1,m},{j,1,2},{k,1,y}
  ];
];

```

(* COMPUTE Kbp THE SIF ASSOCIATED WITH THE BYPASS STRESS *)

```

Do[
  If[
    tiptypes[[i,j]]==0,
    kbp[i,j]=0,
    kbp[i,j]= (1 + Sum[beta[i,j,q] - 1, {q,1,m}] +
      Sum[betahole[i,j,k] - 1, {k,1,y}]) *
      sigmabp[x[i,j]]*((Pi*a[i])^0.5)
  ];
  Print[""];
  Print[StringForm["Kbp['','']= ", i, j, kbp[i,j] ]],
    {i,1,m},{j,1,2}
  ];
];

```

```
(* COMPUTE Ktransfer THE SIF ASSOCIATED WITH LOAD TRANSFER *)
```

```
(* Compute Kwedge[i,j] *)
```

```
(* Notes:
```

```
  (1) This is the SIF that would occur due to
      a wedge load wedgeload[i] at each rivet location in
      the crack, using the formulae in Rooke and Cartwright
      p36 and Broek p86.
```

```
  (2) Assume that the total wedge load for a given
      crack acts as a single resultant wedge load.
```

```
*)
```

```
Do[
  nooffirstrivet[i]= Floor[1+(x[i,1]-1)/p]+1;
  noofflastrivet[i]= Floor[1+(x[i,2]-1)/p];
  cgholes[i]= 1+0.5*p*((nooffirstrivet[i]-1)+(noofflastrivet[i]-1));
  s[i]= cgholes[i]-0.5*(x[i,1]+x[i,2]); (* Displacement in x
                                          direction from cg cracks to cg holes *)
  wedgeload[i]= sigma[o[i]]*(1-bpratio[o[i]])*(p*t)*
                (noofflastrivet[i]-noofffirstrivet[i]+1);
  If[
    tiptypes[[i,j]]==0,
    kwedge[i,j]=0,
    kwedge[i,j]= (wedgeload[i]/(t*Sqrt[pi*a[i]]))*
                  Sqrt[ (a[i]+((-1)^j)*s[i])/(a[i]-((-1)^j)*s[i]) ]
  ];
  Print[""];
  Print[StringForm["Kwedge['','] = ", i, j, kwedge[i,j] ]],
        {i,1,m},{j,1,2}
  ];
```

```
(* Compute Kltnsion the SIF that would occur due to a nominal
stress equal to the stress transferred at the critical rivet row *)
```

```
Do[
  If[
    tiptypes[[i,j]]==0,
    kltnsion[i,j]=0,
    kltnsion[i,j]= (1 + Sum[ beta[i,j,q] -1, {q,1,m} ] +
```

```

Sum( betahole[i,j,k] -1, (k,1,y) ))*
sigma1t[x(i,j)]*Sqrt(pi*a[i])
];
Print[""];
Print[StringForm["Kl1tension['','] = '",
    i, j, kl1tension[i,j] ]],
    {i,1,m},{j,1,2}
];

(* Compute Ktransfer *)

Do[
    If[
        tiptypes[{i,j}]==0,
        ktransfer[i,j]=0,
        ktransfer[i,j]= 0.5*(kl1tension[i,j] + kwedge[i,j])
    ];
    Print[""];
    Print[StringForm["Ktransfer['','] = '",
        i, j, ktransfer[i,j] ]],
        {i,1,m},{j,1,2}
    ];

(* COMPUTE Kinterim AND THE IRWIN PLASTIC ZONE SIZE *)

If[
    irwin==1,
    Do[
        kinterim[i,j]=kbp[i,j] + ktransfer[i,j];
        plasticzone[i,j]=(kinterim[i,j]^2)/(pi*yieldstrength^2),
        {i,1,m},{j,1,2}
    ]
],

(* CLOSE DO LOOP FOR PLASTIC ZONE CORRECTION *)

(* Check that last statement is followed by a comma *)
{irwin,1,2}
];      (* Closes irwin Do loop *)

```

```
(* DETERMINE IF LINKUP HAS OCCURRED -
  IF SO ADJUST datapairs AND RESTART ITERATION *)
```

```
restart=0;
```

```
Do[
  Which[
    tips[i,2]==99,
    Continue[],
    tips[i,2]>=tips[i+1,1],
    datapairs[[i,2]]=99;
    datapairs[[i+1,1]]=99;
    restart=1
  ],
  {i,1,n-1}
];
```

```
If[
  restart==1,
  Goto[linkuprepeat]
]
```

```
(* The above Do and If statements add 99s to datapairs
  when linkup has occurred, and then repeat part
  of the current main iteration. The Do statement
  contains the linkup condition, which is that cracks
  or crack and hole - with plastic zone correction
  included - must touch or overlap *)
```

```
); (* Closes procedure containing linkuprepeat *)
```

```
(* IF REQUESTED INTERACTIVELY, COMPUTE Kbend
  THE SIF DUE TO OUT OF PLANE BENDING *)
```

```
(*
Note: Kbend is computed at the surface of the sheet
      where maximum tensile bending stress occurs. In reality,
      Kbend varies through the thickness. Therefore, for the
      purpose of computing crack growth rates, it will
      probably be necessary to factor this value before
      including it in the total SIF. A program modification
      should be made when experimental data is available
      on the effect of out of plane bending on crack
      growth rate under combined constant amplitude
      tension and bending. As yet, there is no such data in
      the literature.
```

```
*)
```

```
If[
  kbendquery=="n",
  (* Open If *)
```

```
bendpositionfactor=Fit[ReadList["bendpos.dat", Number],
  {1,x,x^2}, x];
```

```

empiricalfactor= 0.6; (* This accounts for the difference between
                        Schijve's secondary bending theory and
                        experimental results. Plastic deformation at the
                        rivet/hole interface probably causes a large
                        part of the discrepancy. Therefore, the factor
                        is probably non-linear with applied stress.
                        It probably also varies with joint geometry,
                        and with location relative to the rivet. *)

Do[
  alpha1= Sqrt[12*sigma[o[i]]/((10^4)*t^2)];
  alpha2= Sqrt[12*sigma[o[i]]/((10^4)*8*t^2)];
  tee1=Tanh[alpha1*11];
  tee2= Tanh[alpha2*12];
  bendfactor[i]= empiricalfactor*(bendpositionfactor/(x->o[i]))*
    3/(1+alpha1*tee2/(alpha2*tee1));
  (* Modified version of equation presented in Part1 & Schijve Jan 92 *)

  Print[""];
  Print[StringForm["bendfactor[''] = '", i, bendfactor[i] ]],

  {i,1,m}
];

Do[

  If[
    tiptypes[[i,1]]==0,
    betabend[i,j]=graph5[i,j]*0.39,
    If[tiptypes[[i,2]]==0,
      betabend[i,j]=graph5[i,j]*0.39,
      betabend[i,j]=graph6[i,j]*0.39
    ]
  ];

  Print[""];
  Print[StringForm["betabend['',''] = '", i, j, betabend[i,j] ]];

  If[
    tiptypes[[i,j]]==0,
    kbend[i,j]=0,
    kbend[i,j]=betabend[i,j]*bendfactor[i]*
      sigma[o[i]]*Sqrt[pi*a[i]]
  ];

  Print[""];
  Print[StringForm["Kbend['',''] = '", i, j, kbend[i,j] ]],

  {i,1,m},{j,1,2}
];

(* Close If *)

```

```
(* COMPUTE Ktotal - SIF FOR EACH CRACK TIP OF ARBITRARY LINKED
OR UNLINKED CRACKS IN THE OUTER ROW OF A LAP JOINT UNDER
HOOP STRESS - INCLUDING EFFECTS OF SECONDARY BENDING
DUE TO JOINT ECCENTRICITY *)
```

```
Do[
  Ktotal[i,j]= kbp[i,j] + ktransfer[i,j] + If[kbendquery=="n",
                                             kbend[i,j],
                                             0
                                             ];

  Print[""];
  Print[StringForm["Ktotal['','] = '", i, j, Ktotal[i,j] ]],
        {i,1,m},{j,1,2}
];
```

```
(* COMPUTE CRACK GROWTH RATES BASED ON da/dN VS. deltak
DATA FOR 0.040 INCH 2024-T3 CENTRE CRACKED TENSION SPECIMENS
R=0.02 IN DAWICKE APR 92 FIG 11 *)
```

```
Do[
  deltak= Ktotal[i,j]*(1-stressratio);

  Print[""];Print[""];
  Print[StringForm["deltak['','] = '", i, j, deltak] ];
  Write[file,OutputForm[""]];
  Write[file,OutputForm[StringForm[
    "deltak['','] = '", i, j, deltak] ]];

  dadndawicke1=(10^-9.47)*deltak^4.12;
  dadndawicke2=(10^-8.56)*deltak^3.08;
  dadndawicke3=(10^-10.87)*deltak^4.97;
  dadndawicke4=(10^-14.29)*deltak^7.29;

  Which[
    deltak<4, dadndawicke[i,j]=0,
    4<=deltak<7, dadndawicke[i,j]=dadndawicke1,
    7<=deltak<17, dadndawicke[i,j]=dadndawicke2,
    17<=deltak<=32, dadndawicke[i,j]=dadndawicke3,
    deltak>32, dadndawicke[i,j]=dadndawicke4
  ];

  Print[StringForm["dadndawicke['','] = '", i, j,
    dadndawicke[i,j] ]];
  Write[file,OutputForm[StringForm[
    "dadndawicke['','] = '", i, j,dadndawicke[i,j] ]]],
        {i,1,m},{j,1,2}
];
```

```

maxdadn=Max[Flatten[Table[dadndawicke[i,j],{i,1,m},{j,1,2}]]],
(* maxdadn is needed to compute dn for the next iteration *)

(* CLOSE MAIN ITERATION LOOP *)

(* Check that last statement has been terminated with a comma *)
(iteration,1,limit)
];

Print["ITERATION COMPLETE"];

(* STORE POINTS FOR CRACK GROWTH PLOTS IN FILE growpoints.#
   WHERE # IS THE STUDY NUMBER WHICH WAS INPUT INTERACTIVELY *)

growpointsfile=ToString[StringForm["\"growpoints.\"\"\"", idno]];
growpointsfile=OpenWrite[growpointsfile];

Do[
  Write[growpointsfile, growpoints[i,j]];
  If[
    i==n&&j==2,
    Continue[],
    Write[growpointsfile, OutputForm[StringForm["\", \""]],
    ],
    {i,1,n},{j,1,2}
  ];
Close[growpointsfile];

(* CREATE INDIVIDUAL GRAPHICS OBJECTS FOR EACH GROWTH CURVE
   AND DISPLAY A COMBINED PLOT OF ALL CRACKS ON THE SCREEN *)

(* Note that the plot range is set at 336000 to
   414000 cycles. This may need to be adjusted to suit. *)

alltips={};

Do[
  If[
    growpoints[i,j]=={},
    Continue[],
    growplot[i,j]=ListPlot[growpoints[i,j],

```

```

PlotLabel->FontForm["ALL CRACKS",{"Times-BoldItalic",12}],
PlotJoined->True,
Frame->True,
FrameLabel->(Co-ord,Cycles),
DefaultFont->{"Times-Italic",9},
AxesOrigin->(0,336500),
FrameTicks->({1,2,3,4,5,6,7,8},Automatic),
PlotRange->({0,9},{336000,414000}),
PlotRegion->({0.1,1},{0,1}),
GridLines->({1,2,3,4,5,6,7,8},None),
AspectRatio->1.5,
DisplayFunction->Identity
];
alltips=Append(alltips, growplot[i,j])
],
(i,1,n),(j,1,n),
]; (* All plots on common axes *)

Display["alltips",
Show[alltips,
DisplayFunction->Identity
]
]; (* Creates a graphics file of all cracks on
one plot. This file must be subjected to the
UNIX psfix command before printing *)

Show[alltips,
DisplayFunction->$DisplayFunction
]; (* Gives a screen display of all cracks together *)

(* CLOSE THE OUTPUT FILE msdoutput.# *)

Close[file];

```


APPENDIX D. **Subprogram 'addcrackprog' used with 'growprog' to simulate the crack initiation sequence of the test specimen.**

```
Which[
  datapairs[[2,1]]!=0,
  Continue[],
  datapairs[[3,1]]>=0.324,
  datapairs[[2,1]]=0.128
];
```

```
Which[
  datapairs[[5,1]]!=0,
  Continue[],
  datapairs[[3,2]]>=0.519,
  datapairs[[5,1]]=0.149;
  datapairs[[5,2]]=0.149
];
```

```
Which[
  datapairs[[2,2]]!=0,
  Continue[],
  datapairs[[3,1]]>=0.594,
  datapairs[[2,2]]=0.162
];
```

```
Which[
  datapairs[[4,1]]!=0,
  Continue[],
  datapairs[[3,2]]>=0.621,
  datapairs[[4,1]]=0.154
];
```

```
Which[
  datapairs[[4,2]]!=0,
  Continue[],
  datapairs[[4,1]]>=99,
  datapairs[[4,2]]=0.111
];
```

```
Which[
  datapairs[[6,1]]!=0,
  Continue[],
  datapairs[[5,2]]>=0.429,
  datapairs[[6,1]]=0.122
];
```

```
Which[
  datapairs[[6,2]]!=0,
  Continue[],
  datapairs[[6,1]]>=99,
  datapairs[[6,2]]=0.167
];
```

```
Which[
  datapairs[[7,1]]!=0,
  Continue[],
  datapairs[[6,2]]>=0.471,
  datapairs[[7,1]]=0.157
];
```

```
Which[
  datapairs[[7,2]]!=0,
  Continue[],
  datapairs[[7,1]]==99,
  datapairs[[7,2]]=0.198
];
```

END

0 8-1 1-94

FIN