ARL-STRUC-TM-506 AR-005-604

AUSTRALIA

DEPARTMENT OF DEFENCE

DEFENCE SCIENCE AND TECHNOLOGY ORGANISATION

AERONAUTICAL RESEARCH LABORATORY

MELBOURNE, VICTORIA

Aircraft Structures Technical Memorandum 506

A REVIEW OF AUSTRALIAN AND NEW ZEALAND INVESTIGATIONS ON AERONAUTICAL FATIGUE DURING THE PERIOD APRIL 1987 TO MARCH 1989 (U)

Edited by

G.S. JOST

Approved for Public Release

APRIL 1989

AR-005-604

DEPARTMENT OF DEFENCE
DEFENCE SCIENCE AND TECHNOLOGY ORGANISATION
AERONAUTICAL RESEARCH LABORATORY

Aircraft Structures Technical Memorandum 506

A REVIEW OF AUSTRALIAN AND NEW ZEALAND INVESTIGATIONS ON AERONAUTICAL FATIGUE DURING THE PERIOD APRIL 1987 TO MARCH 1989

Edited by

G.S. JOST

SUMMARY

This document was prepared for presentation to the 21st Conference of the International Committee on Aeronautical Fatigue scheduled to be held in Jerusalem, Israel, on June 19 and 20, 1989.

A review is given of the aircraft fatigue research and associated activities which form part of the programmes of the Aeronautical Research Laboratory, Universities, the Civil Aviation Authority, the Australian aircraft industry and the Defence Scientific Establishment, New Zealand. The major topics discussed include the fatigue of both civil and military aircraft structures, fatigue damage detection, analysis and repair and fatigue life monitoring and assessment. Australia, (JET)

DSTO ARL MELBOURNE

POSTAL ADDRESS: Director, Aeronautical Research Laboratory,
P.O. Box 4331, Melbourne, Victoria, 3001, Australia

CONTENTS

9.1 INTRODUCTION

This review of Australian investigations on aeronautical fatigue in the 1987 to 1989 biennium comprises the collective inputs from the Australian and New Zealand organisations listed below. The author acknowledges with gratitude the contributions of those shown against each of the items in the Review.

The abbreviations and addresses of contributing organisations are as follows:

ARL	Aeronautical Research Laboratory, PO Box 4331, Melbourne, Vic. 3001.
ASTA	Aerospace Technologies of Australia, PO Box 4, Port Melbourne, Vic. 3207.
CAA	Civil Aviation Authority, P.O. Box 367, Canberra, ACT. 2601.
HDH	Hawker de Havilland Ltd, PO Box 30, Bankstown, NSW. 2200.
HDHV	Hawker de Havilland (Vic) Pty Ltd GPO Box 779H, Melbourne, Vic. 3001.
MU	Monash University, Clayton, Vic. 3168.
RMIT	Royal Melbourne Institute of Technology, GPO Box 2476V, Melbourne, Vic. 3001.
UOM	University of Melbourne, Parkville, Vic. 3052.
UOS	University of Sydney, NSW, 2006.
DSE	Defence Scientific Establishment, Auckland, New Zealand.

9.2 FATIGUE PROGRAMMES ON MILITARY AIRCRAFT

9.2.1 Mirage IIIO Fatigue Substantiation (J.M. Grandage - ARL)

Previous Australian ICAF reviews have described the Mirage damage tolerance assessment programme which was in its closing stages at the time of the last review [1]. The RAAF Mirage fleet has now been fully retired. However it is worth recording that the crack growth modelling methods used in the damage tolerance assessment could not be validated against experimental data from full-scale testing. For this reason the results of the damage tolerance assessment were not used as a basis for continuing life substantiation. For the short remaining life required, the wing was substantiated using a safe-life based on the Swiss full-scale test [2], and fuselage frame 26 was substantiated using a simplified safety-by-inspection approach in which an easily inspectable area was assumed to be an indicator of the condition of frame 26 in general.

9.2.2 DADTA of RAAF Macchi MB326H Aircraft (P.J. Foden - HDHV)

Earlier work on this programme was reported in the previous ICAF National Review [1]. Since then an extensive coupon test programme has been completed. This covered both constant amplitude and flight-by-flight loading on steel and aluminium alloy coupons having a representative range of thicknesses.

A main spar and centre-section which were removed from a fleet aircraft after 3000 hours service have been run for a further 8000 hours of RAAF fighter conversion squadron spectrum flight-by-flight loading. This spar was also modified to incorporate several over-drilled holes which have been found in service aircraft. Failure finally occurred at a different location altogether, in the inboard section of the lower spar cap.

The project is delayed at present by the inability of our computer crack-growth model to predict the test results with sufficient accuracy to give the confidence to apply it to other areas. When this problem is resolved the project will be completed, and the expertise gained then may be applied to predict lives for other aircraft in the RAAF fleet which do not have such a good data base of service experience and test results.

9.2.3 GAF Nomad (L.T. Tuller - ASTA)

Progress on the Nomad full-scale fatigue test has been reported regularly in past ICAF Reviews and the technical results of the overall programme are published regularly in Project Notes and Certification Reports.

One of the more significant aspects highlighted by the fatigue testing programme has been the discovery of the fatigue-sensitive nature of the stub-wing [1]. Following the failure of the second stub-wing front spar upper cap at the wing strut pick-up fitting position in 1985 after only 37,000 flights (the first stub-wing also failed at this location after 138,000 flights), the stub-wing was rebuilt with a new spar. The opportunity was taken to rework all holes in fatigue-prone regions of the new spar using the cold-expansion process.

To the end of February 1989 the stub-wing had survived over 100,000 flights. By 50,000 flights a crack had propagated from the primary fatigue-critical hole to the nearby free edge and another had initiated at the opposite position in the hole. There has been no measureable growth of this second crack since that time. This is an encouraging result, the full implications of which will become clear only after the cracking has been allowed to run its full course and fractography has been carried out.

It was reported in the previous Review [1] that the test loading was to be increased when the stub-wing had reached 50,000 flights with the aim of forcing failure of the main wing. This plan has been deferred so that cracking in the fatigue-critical cold-expanded hole may be quantified under normal loadings.

The total Nomad fatigue programme comprises some 22 subprogrammes or work packages which are currently in progress. Briefly, these include the following:

(a) full-scale fatigue test
(b) flight testing programme and ground calibration for a dedicated flight test aircraft
(c) component fatigue tests on
  (1) flight control system components
  (2) wing strut interfaces
  (3) life-improvement designs for (1)
(d) fatigue life improvement of the critical stub-wing region based on coupon testing. The effect of cold-working on holes with edge-margins from 1.5 to 0.8 in stub-wing spar sections is being quantified.
(e) miscellaneous investigations such as fracture toughness testing and residual stress estimation
(f) fatigue certification for the various Nomad models operating in their respective roles.

The total number of crack sites discovered in the fatigue test airframe now exceeds 330, although the great majority of these are, as previously reported, in the "nuisance" category. Details of all cracking are provided in [3].

9.2.4 F-111 Wing Pivot Fitting Boron/Epoxy Doublers (B.C. Hoskin, A.A. Baker, R. Jones - ARL)

In order to improve the fatigue performance of the Australian F-111 fleet, ARL, at the request of the RAAF, is developing a system of boron/epoxy doublers for application to the upper plate of the wing pivot fitting (WPF). As shown in Fig. 1, two doublers are used for each WPF; they are adhesively bonded to the D6ac ultra-high strength steel plate of the WPF and also to the aluminium alloy wing skin. These doublers are much thicker than any other composite doublers used by ARL to date, having a maximum thickness in excess of 100 plies or about 15 mm.

Finite element analyses predicted that the doublers would lead to a stress reduction of around 30% in certain fatigue-critical areas of the WPF (namely at the run-outs of integral stiffeners on the inside of the upper plate) and strain surveys on a full-scale wing verified this. In the full-scale tests the wing was loaded by the same jack arrangement as is used in the standard cold proof load test (CPLT) to which all F-111's are regularly subjected in a rig designed by Hawker de Havilland, Victoria (Fig. 2). Because the test wing was one salvaged from a crashed aircraft, and to which "boiler-plate" repairs had been made, the strain survey was conducted at 40% of the proof loads; however, the wing with doublers was subsequently satisfactorily tested to 100% of the proof loads under ambient conditions. In an attempt to simulate CPLT conditions as far as the doublers were concerned, the WPF was then enclosed in a chamber and cooled to -40°C. The wing was satisfactorily tested to 100% proof down-load and then to 98% proof up-load; at that stage the test wing failed in the area of a boiler-plate repair.

Considerable effort has been devoted to materials engineering aspects of this development especially as regards the choice of adhesive and the surface treatment of both the steel and the aluminium alloy prior to bonding. Briefly, a procedure based on grit-blasting followed by silane application was adopted, in conjunction with the use of FM 73 adhesive. Likewise, the establishment of the cure cycle required detailed investigation. The durability of the doubler system, and especially of the adhesive bonds is currently being examined by fatigue tests on large dog-bone specimens; the load spectrum being applied here includes simulated CPLT's.

A procedure, and the associated equipment, permitting fitment of the doublers to the aircraft wing in-situ was developed. In order to minimise thermal mis-match problems arising because of the substantially smaller value of the expansion coefficient of boron/epoxy compared with steel and (especially) aluminium alloy, the doublers are applied with the wing under load. The application procedure was validated first on the structural test wing. A prototype doubler system was successfully applied to one wing of a fleet aircraft in 1987 and has performed satisfactorily to date. A full doubler system was applied to another aircraft in late 1988; this aircraft is scheduled to undergo CPLT, which is the most severe condition for the doublers, during 1989.

9.2.5 **Pilatus PC-9/A Fatigue Substantiation** (I.D. McArthur - HDH)

The PC-9/A aircraft is the Australian-built version of the Swiss PC-9 turbo-prop trainer now entering service with the RAAF. Some 67 PC-9/As are to be built over the next four years.

Because no full-scale fatigue test has been carried out on the PC-9, HDH has been contracted to provide input to a fatigue substantiation programme on this aircraft. Pilatus design and test data have been analysed and control points and associated strain gauge locations are being decided. In brief, the areas considered fatigue sensitive are (1) for normal g loading: the cockpit longerons, main spar and wing bend region, (2) for normal g loading and landing loads: wing inner skins and splices and (3) for lateral gusts and spinning: the fin spar and attachments.

9.2.6 **Pilatus PC-9/A Fatigue Test** (J.M. Grandage - ARL)

The RAAF is acquiring a fleet of 67 Pilatus PC-9 trainer aircraft manufactured by Hawker de Havilland. Since no full-scale fatigue test on either the PC-9 or its predecessor, the PC-7, has been performed, the RAAF has commissioned a full-scale test to be carried out at ARL. The test is scheduled to commence in 1991.

Flight strain data will be recorded as an input to the test, first as an aid in developing the test loading sequence and secondly as a data set for comparing with strains to be measured on the fatigue test airframe. It is expected that loads will be applied to the wings, landing gear, fuselage and horizontal and vertical tailplanes. The severity of the loading sequence will be based on a combination of RAAF operational predictions, flight trials data and the fatigue loading data monitored throughout the fleet at the time. Loads will be monitored by fatigue meters fitted to the whole fleet, and it is also planned to fit AFDAS strain recorders to 13 aircraft. Current work is aimed at defining fatigue critical areas and recommending strain gauge locations for the flight test aircraft, the fatigue test aircraft and the AFDAS installation.

9.2.7 F/A-18 Fatigue Test Programme (A.D. Graham - ARL)

The aim of the F/A-18 full-scale fatigue test programme is to establish the economic life of the F/A-18 aircraft as operated by the RAAF and the Canadian Air Force. The magnitude of the total task has led to an agreement with Canada to share the full-scale testing. Canada will test the centre fuselage/wing structure and Australia (ARL) will test the aft fuselage and empennage structure.

The Australian test will require the simultaneous application of dynamic and manoeuvre loading to the test article in order to achieve confidence that the test results are a valid representation of service usage. In flight, the dynamic loading on the empennage is induced by vortices which are generated by the leading edge extensions of the wings at high angles of attack (AOA), Fig. 3. For angles of attack between 10° and 26° the vortices sweep over the wings and impinge upon the horizontal stabilators, exciting the stabilator natural frequencies, predominantly those at 45 Hz and 89 Hz, peak response occuring at an AOA of about 18°. For angles of attack between 16° and 42°, the vortices impinge upon the vertical tails, and excite two dominant modes, a torsion mode at 45 Hz with a peak response at an AOA of about 20°, and a primary bending mode at 15 Hz with a peak response at around 23°. These situations are indicated diagrammatically in Fig. 4.

The Australian test will endeavour to apply manoeuvre loads to the structure using air springs distributed over the tail and stabilator surfaces whilst applying simultaneously, in the correct time sequence, the dynamic buffet loads using high frequency electro-hydraulic shakers. The test is scheduled to begin in 1992 and to reach two lifetimes (12,000 hrs) by 1994.

9.2.8 F/A-18 Bulkhead Test (G.W. Revill - ARL)

ARL is fatigue testing one of the three bulkheads which distribute the wing loads into the fuselage of the F/A-18 aircraft. The FS 488 bulkhead, Fig. 5, is 2150mm wide by 1350mm high and was machined from a 137mm thick plate of 7050 T7451 aluminium alloy. The loads are transmitted to the bulkhead by two pins for each wing with each pin carrying a load of about one meganewton at the maximum vertical acceleration of 9.25g. The ARL test rig is shown schematically in Fig. 6.

These bulkheads on the first 30 aircraft of the Australian fleet differ from those in later aircraft. The flange fillet profiles, Fig. 7, of the earlier bulkheads are such that their fatigue lives might be inadequate. It is intended that these early bulkheads be modified to conform to the later design at about 2000 flying hours (one third of required life). A fatigue test was required to prove that these aircraft would be safe until then. While similar tests have been carried out by Northrop, a bulkhead with this fillet profile was not included. Further, the Northrop tests excluded the longeron recess fillet so the ARL test is to validate this and other critical areas to the full required life.

The loading sequence used by Northrop was derived for the US Navy and was intended to represent an extremely severe loading spectrum. The life under this spectrum was to be divided by a scatter factor of two to assess the safe life. However, calculations based on the early records of Australian usage indicated that it was at least as severe as the Navy test spectrum: the life under that spectrum should be considered an estimate of life under service loading and a scatter factor of three would then need to be applied to obtain a safe life. On this basis the fatigue test was run to 6000 simulated flying hours with the unmodified flange fillet profile to demonstrate a safe life of 2000 hours. The bulkhead is now awaiting delivery of the jigs and tooling that will be used to modify the fillet. When the modification is complete the test will be continued to failure but the flange fillets will be excluded from test consideration and any cracks subsequently found there will be removed: attention will focus then on the suspect nearby longeron recess fillet region.

Having completed 6000 simulated flying hours the test article has been the subject of extensive NDI, with and without applied load, using dye penetrant and eddy-current examination and no cracks have been identified. Further examinations will be made before the test continues. The ARL SPATE equipment, Section 9.4.8, has also been used to carry out stress surveys of the areas of interest and one of these is shown in Fig. 8.

Before the fatigue test began, silicone rubber moulds were made of critical areas to provide a record of surface finish and defects. It is now believed that the IVD (Ion Vapour Deposition) coating on the bulkhead surface retained some of the silicone compounds which later caused difficulties in properly bonding the strain gauges and may have reduced the sensitivity of the dye penetrant testing.

A series of coupon tests is underway to check the ratio of lives calculated under the Australian and US sequences. Because bulkhead material has been difficult to obtain, the initial series of tests are using specimens of 2L65 material (similar to 2014): the results so far do not contradict the calculations of the severity of the early Australian usage. This result will be confirmed with 7050-T7451 specimens as soon as that material is available.

9.2.9 Fatigue Monitoring of RAAF Aircraft (P.J. Foden - HDHV)

9.2.9.1 F/A-18

The final implementation phase for the F/A-18 Maintenance Data and Service Life Monitoring System (MD&SLMS) was implemented during 1988 when the acceptance testing was completed on the central computing facility software at HdHV and formal project completion was acknowledged by the RAAF. The backlog of flight data for analysis by the central facility has been significantly reduced during 1988 and fleet management reports on airframe fatigue life have been forwarded to the RAAF. The damage rates on structural fatigue are being more closely monitored. Software enhancements to the unit computing facilities are allowing airframe and engine fatigue data to be processed and reviewed on a weekly basis.

During 1989, the strain data and aircraft useage information recorded by the central computer will play an important role in determining the Australian useage spectrum. This information will contribute to the loading spectrum for the fatigue test on the F/A-18 aft fuselage structure (Section 9.2.7).

9.2.9.2 F-111

The successful completion of AFDAS into the prototype F-111 aircraft in 1985 has led to adoption of the system on a wider basis throughout the fleet. Ten aircraft are being modified and fitted with the MK111 AFDAS system. This will lead to better overall fleet monitoring for all aircraft.

A DADTA study on the aircraft structure has commenced. The activities to date have centred on the derivation of a representative spectrum for the Australian fleet aircraft. The intention is to apply this spectrum to the General Dynamics suite of software and hence ascertain a benchmark for the performance of Australian aircraft compared to American aircraft.

9.2.9.3 Nomad

The incorporation of AFDAS fatigue monitoring equipment to the Nomad aircraft has been suspended by the RAAF until further flight test data have been evaluated. The present concern is the fatigue performance of the stub wing in rugged military operations.

9.2.9.4 C-130

A gradual change in the spectrum usage profile has been detected since the DADTA analysis in 1983. The spectrum is presently being examined and the impact of profile changes to the crack growth performance in some structural areas is under evaluation.

9.2.9.5 Macchi MB 326 H

Activities have centred on the selection of aircraft for the formation of the lead fighter training and fleet support aircraft for the RAAF. The selection criteria for these aircraft will depend on the level of fatigue damage on the wings and the modification status of the aircraft to undertake the new role. The fitting of AFDAS to aircraft under this new role is not likely because of financial and reduced life-of-type considerations.

9.2.10 Helicopter Gear Fatigue Life (K.F. Fraser - ARL)

Under programmes sponsored by the Royal Australian Navy (RAN), ARL has estimated the safe fatigue lives of main rotor gear box (MRGB) gears for the Wessex Mk 31B and the Sea King Mk 50 helicopters operated by the RAN. For both of these aircraft the manufacturer specified fatigue-discard lives for some gears. The aim of both programmes was to substantiate, or revise, safe lives for Australian operating conditions. A UK study which suggested that promulgated lives for Wessex gears could be very optimistic provided by strong impetus for the work on Wessex. The work on Wessex was completed in 1978 and that for Sea King in 1986.

In both programmes the helicopter manufacturer provided gear fatigue data in the form of cycles to failure as a function of transmitted power (directly related to gear tooth stress). Loads data in the form of in-flight gear torque measurements were acquired in Australia over an extended flying period (about 250 hours in each of two aircraft for each programme) for normal aircraft operations.

The fatigue stress of major interest for gears is the tooth root bending stress and in the case of two gears in simple mesh each tooth will experience one load cycle per revolution. The tooth root bending stress is proportional to applied torque. For helicopters, the rotational speed of the mechanical system is essentially constant so the rate of application of load cycles is a known constant. Hence the fatigue life expenditure can be estimated if the torque spectrum (total time in various torquebands) is known.

In the Wessex programme total times in 10 torquebands were accumulated and a ground station analysis was used to estimate safe fatigue lives for Australian operating conditions. Analysis of the acquired data revealed that the MRGB gear fatigue lives promulgated by the manufacturer were valid for RAN operating conditions [4].

In the Sea King programme the gear fatigue data were incorporated in an airborne microprocessor program and the fatigue life usage of critical gears was estimated and indicated in real-time during flight [5].

The Sea King programme revealed that, for RAN operating conditions, the fatigue lives of MRGB gears should exceed the LOT of the aircraft. The programme also demonstrated the practicality of estimating gear fatigue life usage during flight [6]. The techniques developed could be implemented in comprehensive helicopter health and life usage monitoring systems which are likely to come into service across helicopter fleets in the foreseeable future [7].

9.2.11 Black Hawk Fatigue Substantiation (K.F. Fraser - ARL)

Under a programme sponsored by the RAAF, ARL has been tasked to undertake a fatigue substantiation programme for the S-70A-9 Black Hawk helicopter operated by the Australian Army. The programme will involve the measurement of flight parameters and selected load parameters relating to the powertrain, rotor system, flight control system and airframe. A major aim of the programme is to compare mission severity for Australian operations with the theoretical mission spectrum which forms the basis for the promulgated fatigue lives of components. It is proposed that data will be collected on sample aircraft for a limited period. Initial work is being undertaken on the integration of suitable data system elements for airborne data logging and ground station data recovery.

9.3 FATIGUE IN CIVIL AIRCRAFT

9.3.1 Douglas DC-9 Rear Pressure Bulkhead Attachment (R.B. Douglas - CAA)

A massive crack has recently been discovered in the rear pressure bulkhead in an Australian registered DC9-30 aeroplane. The defect consists of a 24-inch circumferential crack in the stem of the T to which the bulkhead web attaches. The general arrangement is shown in Fig. 9, and part of the fracture surface in Fig. 10. The aircraft had achieved 36079 cycles, 40336 hours. Laboratory examination [8] has confirmed the crack as multiple origin fatigue. The initiation sites are on the forward face, and extend over most of the crack length. The crack had entered an unstable growth phase and complete failure - probably catastrophic - was imminent.

The crack was detected visually from the aft face during a scheduled D check. However, it is alarming that cracking of such an extent was present at the initial inspection threshold. It is also noted that, since the cracks initiate on the forward face under the web, they are not visually detectable until they penetrate through the thickness, even though extensive multiple site damage may be present.

The aircraft operator has noted that through-cracks of smaller circumferential length would also be difficult to detect, since even this large crack was not obvious. Access is difficult and requires removal of APU and air conditioning equipment, and thorough cleaning to remove insulation, sealant etc.

The most effective inspection would appear to be an Eddy-current technique, and in fact such a technique is given by the manufacturer's inspection instructions as an option. The Australian operator has now developed an enhanced Eddy-current procedure.

Although the investigation is not yet complete, the indications are that the inspection threshold and repetitive intervals need to be reduced, that the Eddy-current inspection method is the preferred option, and that aircraft which have previously been inspected visually need re-inspection.

9.3.2 Cessna Series 400 Spar Strap Modification (S. Dutton - HDH)

CAA Airworthiness Directive Cessna 400/40, Amendment 8, has reduced the safe life of all series 400 (402B, 421B, 402C, 421C) wing spars. To comply with the AD, aircraft life extension must be achieved by spar replacement.

As an alternative to spar replacement, HDH has designed an external fail-safe strap for critical regions of the lower spar cap such that in the event of spar failure, the strap will carry subsequent wing loads: the design has CAA approval. There are some detail differences between straps for various models within the series to accommodate various engine installations, spar cross sections and service loadings.

The straps (up to four laminates) are riveted through the lower spar cap at existing rivet positions. Prior to installation, and with these rivets removed, the spar cap is inspected for cracks with an Eddy-current detector using a technique which scans each (vacant) hole. If cracks are discovered, the modification is not proceeded with.

The strap is designed using Damage Tolerance philosophy. An initial (manufacturing) flaw is assumed to exist and the rate of growth is estimated using in-flight measured stresses. To assure safety, these rates are then assumed to accelerate by a factor of 3.5 to account for variability. Inspection intervals are then set using this accelerated growth rate curve and from established statistical data such that there is a 95% probability of detection. By this approach no safe-life is guaranteed but the wing is serviceable subject to a clean inspection result.

Fig. 11 shows typical details of the extent and location of the straps.

9.3.3 Janus Glider Wing Fatigue Test (R.B. Douglas - CAA, L.A. Wood - RMIT)

Progress on this test has been reported in previous ICAF Reviews: the present situation is detailed in [9]. The test specimen comprises a complete tip-to-tip wing assembly. The starboard wing is one which was badly damaged in a major accident, and which has been fully repaired using a variety of techniques. The port wing was purchased new from the factory. A total of 300 electrical resistance strain gauges were installed during repair/manufacture.

The test load spectrum was derived from flight loads measurements on a number of gliders in service in Australia. It is a composite spectrum representing 15 separate flight profiles covering a mix of ab-initio and post-solo training, cross country flying, competition flying, aerobatics, and both aerotow and winch launching in the ratio of about 2.2:1 respectively and at an overall average of about 0.6 flights per hour. The spectrum is divided into 6 load ranges (12 levels) and the loads are applied in a random sequence in blocks of 29434 turning points representing 294 flight hours. Each block takes about 32 hours of rig running time. The highest loads of +6g and -2.6g occur once every 6000 simulated hours and are applied manually. The load levels have been factored up from those measured, in order to achieve the now commonly used design stress level of 300 MPa at 9g ultimate load. These were based on the results of flight and test rig strain surveys and a detailed finite element analysis.

Detailed inspections are conducted at intervals of 1000 flight hours and include measurement of vibration modes and frequencies every 3000 hours. As at December 1988 the test had achieved 12000 simulated hours, at which time a major inspection was carried out.

No major structural problems have been found so far. The most recent incident was the failure of a whiffle tree link (steel cable) during the 6000 hour 6g (5.9g achieved) limit load application. The safety link held and no other damage was incurred; all similar links have since been replaced as a precaution.

There have been numerous instances of skin cracking in the starboard (repaired) wing which all appear to be associated either with repairs or with damage incurred in the aircraft crash, some of which was deliberately left unrepaired and some of which was undetected. These cracks, most of which would be repaired immediately if found in service, do not measurably affect the stiffness, mode shapes or frequencies and are thus not considered to be significant structurally. Some have been repaired, others are being monitored.

One defect has been found in the port (factory supplied) wing, involving skin cracking at the lower surface skin/root rib/spar junction. This crack was initially detected and repaired early in the test. Recent failure of the associated strain gauges, and visual indications, now indicate failure of the repair. This is being monitored.

Static deflection checks have shown that the wing stiffness is unchanged. Vibration checks have shown that with one exception the modes and frequencies are also unchanged. The exception is a 3% drop in the frequency of the first anti-symmetric mode. This mode involves rotation in the centre section joint and is most likely the result of wear and/or lubrication changes in the wing attachment pin and socket joints. This is being monitored at each major inspection. The test rig is shown in Fig.12.

An investigation into mechanisms of failure in sailplane gel coats has been completed. Premature gel coat cracking can result in potential cracking of the composite matrix. The cracking was found to result primarily from incorrect formulation of the gel coat [9(e)].

9.3.4 Bell 214ST Main Rotor Drag Brace (R.B. Douglas - CAA)

In March 1988 an Australian registered Bell 214ST helicopter experienced severe in-flight vibrations and control difficulties and was forced to ditch in the Timor sea. A heavy sea was running and, on touchdown, one main rotor blade struck a wave and the helicopter overturned. After evacuation of the passengers and crew the aircraft sank; all 15 on board were subsequently rescued.

When the wreck was recovered after 6 weeks in 30 fathoms, it was found that one main rotor blade drag brace had failed in fatigue [10]. The failure had occurred in the 1.25-inch diameter threaded portion of the rod, under the innermost of two nuts which clamp the blade attachment yoke at the outer end of the brace. The assembly is shown in Fig. 13, the broken drag brace and the fracture surface after cleaning in Fig. 14.

The plane of the fracture was located three threads in from the face of the nut and was normal to the axis of the brace. A secondary fatigue crack had propagated from the thread adjacent to the face of the nut and had intersected the primary crack, releasing a small wedge shaped section which was not recovered. Cracks were also evident in two threads inboard from the primary crack.

Despite an extensive laboratory fractographic and metallurgical examination, no specific cause for the failure has been found. There was no corrosion found in the threads; there was, however, some damage to the cadmium plating which was not found on the other drag brace. It has been suggested that some loss of nut torque may have caused the failure; this has not been confirmed.

The manufacturer has conducted a flight strain and loads survey and a fatigue test program in endeavour to explain the failure. Whilst test failures were produced, they occurred under the outer nut and thus did not reproduce the service failure which remains unexplained.

The manufacturer has subsequently introduced a redesigned drag brace of substantially increased sectional area and with some refinements, including a material change to 15/5 PH stainless steel.

9.3.5 Aerospatiale SA330J Puma Main Rotor Spindle (R.B. Douglas - CAA)

In May 1987 an Australian registered Aerospatiale SA330J Puma helicopter experienced a failure of one of the two flapping hinge lugs on one main rotor blade spindle during an off-shore flight with 17 people on board. The resulting out-of-balance forces caused severe vibrations and control difficulties; however, the crew were able to descend and hover-taxi to a barge where the machine was landed safely.

Laboratory examination [11] of the fracture surfaces revealed the presence of fatigue. The fatigue cracking had initiated in an area damaged by fretting on the inner surface of the lug. This fretting resulted from the inner bearing race of the flapping hinge pin assembly contacting the lug face. The offending spindle is shown in Figs. 15 and 16; note the bend in the remaining lug.

Measurement of the crack propagation rate showed that one lug fracture propagated, from a crack depth of 3.4mm to complete failure, in approximately 180 start/stop cycles. Taking the average start up/shut down time of 1 hour 15 minutes for this operation, it was estimated that this occurred over a period of 225 flight hours and the crack in the lower section of the lug propagated from 4.3mm to failure in 100 cycles i.e. 125 hours. The time since last overhaul was reported to be 989 hours. The head had also been removed for minor repairs 398 hours prior to failure, but had not been disassembled at that time.

The investigation also brought to light the fact that the maintenance manual required disassembly and inspection of the lugs for fretting and cracking at 800 hours. This inspection had been overlooked for this aircraft, partly because the requirement was not highlighted as a safety-of-flight critical inspection (there had been previous similar failures overseas); and also because the operator believed that all heads in his fleet had been factory modified to a later configuration which incorporated an anti-fretting modification.

9.4 FATIGUE DAMAGED STRUCTURE : ANALYSIS, DETECTION AND REPAIR

9.4.1 Development of Non-Linear Finite Element Software (G.P. Steven - UOS)

The fracture zones associated with fatigue failure or the cold worked regions near holes involve material being stressed beyond yield. For such situations a Finite Element code needs to be able to analyse such physical states.

The (elastic) STRAND 5 FE code, which has been totally developed and sourced in Australia, is being extended to cover both non-linear material and non-linear geometric behaviour. At present some benchmark problems have been analysed in both these categories [12].

The final outcome is to be a fully developed easy-to-use non-linear code with full graphics pre- and post-processing.

9.4.2 Stress Analysis of Cold-Expanded Holes (G.S. Jost, R.P. Carey - ARL)

The work on two-dimensional plane-stress and plane-strain finite element analyses of cold-expanded holes in large plates [13] reported in the previous Review has been extended to a three-dimensional case in which hole diameter is equal to plate thickness [14]. A comparison of this case with the two-dimensional solutions showed that, as expected, interior stresses and strains for the 3-D case were well approximated by the 2-D plane-strain solutions. However, surface stresses and strains for the 3-D case were poorly approximated by 2-D plane-stress solutions, the greatest discrepancy occurring in the region of greatest interest, i.e. near the hole.

A comparison has also been made of the plane-strain 2-D finite element solution referred to above [13] with analytically-derived solutions using deformation theory [15]; the agreement between the two is remarkably close, Fig. 17. Reference [15] gives closed-form expressions, which are readily evaluated, for all stresses and strains of interest associated with the cold-working of holes in an elastic-perfectly plastic material. The effect of reaming is also examined there and is found to have a negligible effect in relieving residual stresses near the hole.

9.4.3 Fatigue Life Enhancement (J.G. Sparrow - ARL)

Earlier work on the Boeing split-sleeve cold-expansion process and its benefits in fatigue life enhancement was reported in the previous Review [1]. Whilst most overseas studies have concentrated on the application of the process to relatively thin sections, ARL effort has been directed more towards the assessment of the fatigue life enhancement of thick (8-35mm) sections. Fatigue tests have been conducted on thick aluminium alloy specimens and simple joints with holes cold-expanded, either as a single thickness of material, or as part of a stack-up between aluminium or steel side plates. Increases in fatigue life by a factor of at least 6 for open hole specimens and somewhat less than 3 for bolted joint specimens were achieved. However, as the fatigue failures in the bolted-joint specimens with cold-expanded holes initiated by fretting several millimeters away from the holes, the full potential benefits of cold expansion may not have been attained in this case [16].

A comparison of fatigue lives of thick aluminium sections with and without the benefits of cold-expansion has demonstrated that fatigue improvement by a factor of 7 can be achieved even if small (0.75mm) residual fatigue cracks were present before cold-expansion. The ratio of the lives of holes cold-expanded with such small residual cracks compared to the remaining lives of similarly cracked specimens without cold-expansion was approximately 45. Cold-expanded hole fractures displayed a marked disparity in crack depths adjacent to the two faces of the specimens. Considerable differences were evident in crack depths and fatigue crack areas at failure between cold-expanded and non-cold-expanded hole specimens. These findings have ramifications in the damage tolerance assessment of aircraft structures, in particular with regard to crack inspection techniques adopted, assignment of inspection intervals and nomination of critical crack sizes [17].

9.4.4 Use of Adhesive Inserts to Extend the Fatigue Life of Cracked Fastener Holes (J.F. Williams - UOM)

In aircraft wings the aerodynamic loads are transferred from the wing skin to the spars by means of bolted and riveted joints. These joints are one of the main causes of cracking and often determine the fatigue life of the structure. Although recent studies have shown that the use of solid bonded inserts can significantly increase fatigue life, this has been done on uncracked holes and the numerical analyses have concentrated on holes containing cracks unrepresentative of realistic damage.

This project is investigating the on-site repairability of holes containing existing, real cracks by bonding into the hole a (stiffer) steel sleeve using epoxy resin and avoiding the (current) necessity of having to return the structure to a major depot for reworking. The use of a sleeve, of course, minimises the opening and closing of the crack under repetitive fatigue loading and also allows the rivet or bolt to be inserted or removed as desired. This repair may then be augmented by the use of an externally bonded patch.

A series of experiments was performed on pre-cracked specimens in both the repaired and unrepaired conditions at constant amplitude loading to determine their respective fatigue lives. The reason for doing this was to establish that the method will work for holes containing existing cracks and to establish the maximum crack size that can be successfully repaired by this method. The repaired component was then tested under the same load spectrum. When using a steel sleeve and an external patch no crack growth occured and tests were stopped after a life increase of 100-fold had been achieved [18].

9.4.5 Collaborative Bonded Bush Investigation (G.S. Jost, J.M. Finney - ARL)

The use of adhesively bonded bushes in cracked holes to extend subsequent fatigue life has been discussed previously [1]. Preliminary experimental studies, Section 9.4.4 and [18,19], have indicated that the process does achieve very useful increases in life, at least under constant amplitude loading. A collaborative programme is now being set up between the RAE and ARL aimed at further quantifying the effectiveness of this process under variable amplitude loading.

Simple aluminium alloy specimens, each containing an open cracked hole into which a steel bush has been adhesively bonded, will be tested under FALSTAFF loading at two stress levels. Other variables being considered for inclusion in the testing programme are the type of adhesive, adhesive thickness and crack shape. The number and extent of all the possible variables are being deliberately limited to achieve test data for all combinations of those variables being varied: subsequent statistical analysis should then be free of problems associated with missing data.

9.4.6 Strain-Energy-Density in Fatigue (M. Heller - ARL)

By considering the remote loading of a cold-expanded hole it has been shown that the local strain energy density field and the global work do not remain in phase. The observed increase in fatigue life [20] may be understood in terms of this phase change: in general it may be either beneficial or detrimental depending upon the sign of the applied load, i.e. it may result in energy adding to or subtracting from the mean energy level. It has been used to explain the failure of stiffener runouts in the wing pivot fittings of F-111 aircraft and to assist in designing the repair and local changes in stiffener geometry [21].

9.4.7 MultiPaRe Interacting Fatigue Cracks (D.G. Ford - ARL)

Since the last ICAF Conference this general purpose fatigue reliability program has been restructured in terms of (Pascal) linked lists of records for loading actions, damage and cracking. There will now be no theoretical limit on the number of critical regions included, and the program will allow up to the same number of separate loading actions. (It may become convenient to remove this restriction; the reprogramming is trivial). The multiplicity of loading actions has forced consideration of turning point times as the simplest means of allowing for relative sequencing.

The calculation of crack growth or of cumulative damage in a structure follows by the solving of differential equations. In current practice damage is integrated numerically (though crudely, see Section 9.4.10) whilst the only way to predict crack growth with interaction is via direct simulation, often with very long sequences. These heavy computational loads have motivated improvements, included in MultiFaRe, which are described below.

In MultiFaRe, damage or crack rates may be found via direct load simulation but there is also provision for special short sequences which (albeit in different order) supply terms in quadrature formulae. Special quadrature formulae have been developed which are accurate to the fifth order in distribution function but have equal weights at particular evaluation points as in Gauss-Legendre formulae. These points correspond (through exceedance levels etc) to special points on a Goodman diagram of range-pair densities. If their turning-points are sequenced to preserve these range-pairs and if range-pairs uniquely define crack growth, short "typical" sequences may be designed which match single crack growth now predicted by long simulations, including any form of preload or overload interaction which is differentiable.

For several cracks such sequences may be repeated and interleaved with other loading actions to shorten combined-load sequences. Such interleaving, which includes phase relations, requires turning point times to allow for appropriate stressing under combined loading.

The development of MultiFaRe now awaits first order reliability, general validation and documentation.

9.4.8 Experimental Stress Measurement (J.G. Sparrow - ARL)

The SPATE 8000 thermoelastic stress analyser continues to find application in stress mapping of specimens under loading in electro-hydraulic fatigue testing machines, and in the study of aircraft components under test in the laboratory. Examples of the latter include an evaluation of stress in a boron/epoxy reinforcement attached to the F-111C wing pivot fitting; and an assessment of a critical area of an F/A-18 fuselage bulkhead. Thermoelastic measurements have also been employed in NDE of composite specimens, in the examination of impact damage growth [22] and in the study of crack extension of a metallic substructure overlaid with a composite reinforcement panel. A test methodology based on the SPATE 8000 has been developed for the study of heating and cooling phenomena in grossly rapidly deformed tensile metallic bars [23]. When used in conjunction with a large deformation finite element analysis this yields an excellent representation of the true stress-strain curve and the plastic state variable Z used in the Stouffer-Bodner theory of incremental plasticity. A theoretical study has demonstrated a procedure for decomposing bulk stress data (measured by SPATE) into individual stress components [24].

In the previous Review mention was made of the experimental and theoretical demonstration of the need for a modification of Kelvin's treatment of the thermoelastic effect. Kelvin's law states that, under adiabatic conditions, the rate of change of temperature of a dynamically loaded body is directly related to the rate of change of the principal stress sum. It has now been shown that the thermal response of a cyclically loaded body is a function not only of the dynamic component of stress, but also of the static component. This finding has led to the suggestion that residual stresses within a material might be detected using this phenomenon. As a practical demonstration of the use of the SPATE 8000 thermoelastic stress analyser for the measurement of residual stress, a simple test specimen was loaded in a four-point bending rig to produce a known residual stress distribution across the width of the specimen. SPATE scans of a 2024-T351 aluminium alloy specimen containing residual stresses and a similar specimen without such stresses are shown in Figs. 18. The SPATE derived residual stresses are shown in Fig. 19 compared to the predicted values based on strain gauge results [25-29].

Instrumentation for holographic interferometry has been extended by the acquisition of a powerful pulsed laser. This laser will initially be deployed in developing techniques for holographic stress measurements under laboratory conditions and in the NDE of composite panels following earlier NDE work using a continuous laser [30]. Holographic interferometry, using a low power continuous He/Ne laser is also being applied to the detection of ill-fitting compressor blades using a technique demonstrated in a preliminary study. The possible extension to an in-situ procedure using the pulsed laser is to be investigated.

Both in-plane and shadow moire techniques are regularly applied in the laboratory particularly in the monitoring of damage growth in composite specimens under fatigue loading [31].

9.4.9 Sensitivity of Fatigue Crack Growth Prediction to Data Representation (J.M. Finney - ARL)

Currently, there is no fatigue crack growth model which accurately predicts growth under all circumstances without restorting to experimental calibrations. In the case of the Wheeler model, the value of the retardation exponent "m" is obtained by comparing predicted crack growth curves using several m-values with an experimental curve obtained under spectrum loading, and selecting the closest match. Experience has shown that the exponent is stress-scale and crack-shape dependent.

An examination has been conducted to determine any influence also of the method chosen to represent the base growth rate data on the exponent (and hence on subsequent predicted life) [32]. Four representations of the same growth rate data were used :

1. $da/dN - \Delta K$ data tabulations with no restriction of R on ΔK, i.e. $\Delta K = (1-R)K_{max}$.
2. As 1. except that $\Delta K = K_{max}$ for $R < 0$ (i.e. truncation of the stress range at zero).
3. Curve fitting the $da/dN - \Delta K$ data using the Forman equation, $da/dN = C(\Delta K)^n/[(1-R)K_c - \Delta K]$
4. Correlating the R-effect on $da/dN - \Delta K$ by the Schijve equation,
   $\Delta K_{eff} = \Delta K\,(0.55 + 0.35R + 0.1R^2)$, and
   using a single best-fit curve.

For a flight-by-flight load sequence used for testing Mirage aircraft the following m-calibrations were obtained for centre-cracked tension specimens.

Wheeler m-exponent calibrations for Mirage aircraft test sequence

$da/dN - \Delta K$ data representation method	stress	
	13.33 MPa/g	17.98 MPa/g
Data tabulations:		
(i) full-range definition of ΔK	0.70	1.70
(ii) truncated definition of ΔK	0.55	1.00
Curve fits:		
Forman equation (with (ii) above)	0.85	2.30
ΔK_{eff} (with (i) above)	0.30	0.85

Not only is the m-calibration value dependent on stress scale, but it is also quite sensitive to the method of representing the base growth rate data: at both stress levels there is a variation of approximately 3:1.

It is thus apparent that, for any practical use of the Wheeler model, the growth rate data representation method should be stated alongside any reported calibration values. Such information is rarely given and two questions arise:

1. What "errors" in predicted crack growth life can occur if calibration values are taken from the literature and applied without any knowledge of the method used to represent the basic da/dN - ΔK data? Considering only the representation schemes above, it was found that predictions may be in error by factors ranging from 1.5 to 3.1.

2. Is prediction independent of data representation method if the method used for prediction is the same as that used for the calibrations? Two cases were examined, one requiring interpolation for m with a change in stress scale but all other conditions remaining as for the calibration. In this case predicted crack growth was virtually unaffected by the data representation method - with a maximum difference of only 11%. The second case involved a stress scale requiring an extrapolation for m, and a different geometry and crack shape. Variations in crack growth life by a factor of up to 2.8 were obtained, thus highlighting the danger in extrapolating calibration values and indicating that calibrations should cover the full range of stresses for which predictions are required.

These conclusions will apply to all crack growth prediction methods requiring experimental calibration.

9.4.10 Summation of Continuous Damage (D.G. Ford - ARL)

Because service loads and those often applied in fatigue tests come from continuous distributions, the summation of fatigue damage leads to integration which, from the nature of the problem, must generally be numerical.

There are two general approaches. The first is to fit analytical expressions to the load and fatigue data and then integrate the damage analytically, if this is still possible. The second is the direct use of an integration formula. In principle both are good methods but for inexplicable reasons the standard integration procedure for damage is based on stepwise calculation of the damage rate, a method so crude as to escape consideration in texts on calculus or numerical analysis. The inherent errors are also heightened by the non-linear scale in conventional presentations of exceedance data.

In the method of [33], the variable of integration is logarithmic exceedance rather than load or stress level. The new procedure is less ambiguous and more rapid in execution: it is also properly convergent. It avoids differencing the exceedance spectum and the arbitrary choice of midpoints on non-linear scales. The accuracy thus gained may be retained by using any of the standard numerical integration rules.

9.4.11 Mixed-mode Fatigue Crack Growth (Y.C. Lam - MU)

Although the strain-energy-density factor range ΔS has been proposed as a parameter useful for mixed mode fatigue crack growth predictions, in its existing form it is found to be incompatible with the concept of crack closure. ΔS is consequently being modified into ΔSp by taking into account crack closure. As such, mode I fatigue crack growth data could also be used for mixed mode fatigue crack growth prediction.

A preliminary experimental investigation indicates that ΔSp could predict satisfactorily the crack growth rates and trajectories at various R ratios. The work is continuing.

9.4.12 Effect of Residual Stress on Fatigue Crack Growth (Y.C. Lam, J.R. Griffiths - MU)

A technique of introducing and accurately measuring residual stress has been developed. The effect of residual stress re-distribution as a crack propagates and its effect on fatigue crack growth is being investigated.

In addition, a method of introducing beneficial compressive stress by intermittent heating is being investigated. Preliminary investigations indicate that significant improvements in fatigue life could result.

9.4.13 Shot Peening and Fatigue Resistance (J.Q. Clayton - ARL)

The use of components subjected to life-enhancing materials processing in military aircraft is increasing dramatically. Surface treatments include those designed to enhance fatigue resistance; the use of these processes routinely during manufacture has focussed attention on both the performance of the treated component in service, and the benefit consistently attainable from the processing. Reference [34] describes research into the effects of peening with steel shot or glass beads on fatigue in a high strength aluminium alloy used in the F/A-18 aircraft, and shows that the introduction of surface imperfections during peening can degrade the fatigue resistance.

9.4.14 Combined Fatigue/Environmental Box Beam Test (K. Watters - ARL)

A box beam comprising a metal substructure and a graphite/epoxy skin containing impact damage has been tested to failure under an environmental fatigue load sequence. Prior to the fatigue test, an extensive strain survey of the critical area of the graphite/epoxy skin was performed. The box beam is shown in Fig. 20 mounted in its test rig.

The box beam was 2.5m long, of 0.6m chord and 0.15m thick. Its metal substructure was comprised of four aluminium alloy spars and, on the tension side, an aluminium alloy skin reinforced by four steel booms. The graphite/epoxy skin on the compression side was a 7mm thick continuous plank made from 56 plies of XAS-914 material according to the layup $[\pm 45, O_2]_{7S}$. Seven impacts were made on the graphite/epoxy skin in the highly stressed middle area of the beam. Two impacts were midway between spars, three impacts were adjacent to a spar and two impacts were over a spar cap. Impactor mass ranged from 1.0 to 2.0 kg and drop height ranged from 1.2m to 1.8m giving impact energies in the range 11.8J to 35.3J. All impact damage fell into the barely visible category.

The loading configuration was three-point bending combined with a small torque. A truncated version of the FALSTAFF load sequence was used with all load cycles in the range level 8 to level 17 removed. The environmental aspect of the test consisted first of conditioning the graphite/epoxy skin to 1% moisture content and maintaining that level. Secondly, in conjunction with load cycling, a three level temperature spectrum of -20°C (for 10% of cycles), 20°C (75%) and 95°C (15%) was applied. The thermal cycle was based on a block of 200 flights so that 30 thermal cycles were applied during a test lifetime. A dwell of 5 sec was applied for loads above level 25 in the hot segment.

One lifetime of loading was applied to the box beam prior to moisture conditioning and without thermal cycling or load dwell. For this lifetime the maximum load level (32) in the FALSTAFF sequence was set at 70% of the ultimate design load (UDL) of the box beam. No damage growth was observed. Then the graphite/epoxy skin was moisture conditioned and loading resumed, but this time with thermal cycling and load dwell. Maximum load level was again 70% UDL and a further two lifetimes of loading were applied with no observed damage growth. Maximum load level was then raised to 80% UDL and one further lifetime of testing applied, again with no observed damage growth. Maximum load level was then raised to 90% UDL and the graphite/epoxy skin failed on the first occasion the load reached the maximum level, which happened to be in the hot segment. The graphite/epoxy skin failed in compression in a reasonably straight chord line adjacent to the middle of the box beam and encompassing several of the damage sites. The nature of the failure is currently being investigated.

9.4.15 Thermomechanical Analysis of Damage in Composites (M. Heller - UOM)

The coupling between mechanical deformation and thermal energy in an elastic material has enabled the development of a unique non-contact method for investigating the stress distribution over the surface of a body. However, until recently the formulation [35] did not fully account for the interaction between the available mechanical energy and the thermal energy in the case of a composite material. In particular, the effects of moisture and mean load had previously been neglected. This revised theory [36] predicted that when a body is subjected to a single frequency cyclic loading, the temperature response should not only be at the loading frequency, as inferred by Kelvin's Law, but there should also be a component at twice that frequency. Subsequent experiments were performed to detect this predicted second harmonic response [26] and the results conclusively confirmed such a non-linear response, thereby strongly supporting the validity of the revised theory.

In the present work [37] attention is focused on the use of this theory for problems associated with repair technology and impact damage in composites. In one case a numerical analysis was conducted and used to enable comparison with an experimentally determined temperature profile. Attention was then focused on the use of experimentally observed changes in the local temperature field as a measure of the severity of impact damage in a graphite epoxy laminate.

The work to date indicates that temperature measurements may be used to determine the stress distribution on the surface of a unidirectional composite repair and have the potential for detecting and monitoring crack growth below the patch. In another case large changes in the temperature profiles were observed on both sides of an impact damaged composite specimen, allowing the presence of damage to be detected. In contrast to most traditional methods of non-destructive inspection, the present approach reflects the interaction of load, geometry, material and damage in a non-destructive fashion.

A parameter, D_t, related to the variance of the surface temperature field, has been proposed as an indicator of the severity of damage for an impacted composite [38]. Subject to further experimental validation, the authors believe that this parameter may prove to be useful in the analysis of problems associated with damaged composite laminates.

The methodology considered to date involves the use of direct temperature measurements. However, an alternative approach would be to measure surface displacements and compute the temperature field.

9.4.16 Repair of Surface Flaws (R. Jones - ARL)

Surface flaws of up to 40mm long and 6mm deep in an 11mm thick 2024 aluminium alloy have been repaired using an externally bonded patch. When tested under constant amplitude loading the repaired specimens were found to have a more than 20-fold increase in fatigue life [39]. This work has subsequently been continued in conjunction with Melbourne University.

9.4.17 Fundamentals of Composites Failure Analysis (R. Jones - ARL)

A review of the current status of design and damage assessment in composites has been undertaken [40]. This contains a detailed description and summary of test methodology and life prediction laws. It was concluded that current fatigue life prediction methods are inadequate and that alternative methods for damage assessment are required.

It has been shown that the more advanced forms of Tsai-Hill theory predict a failure envelope virtually identical to that predicted by energy density theory [41]. This raises the level of confidence in the Tsai-Hill theory which had previously been seen as essentially a curve fitting exercise. However, it must be stressed that since Tasi-Hill is based on the von Mises theory for metals it cannot be used for problems associated with failure by delamination. This is due to the fact that, for metals, the von Mises equivalent stress is associated with the location of maximum plasticity rather than the direction of crack growth.

For impact damage it has been shown [42] that the path-dependent T^* integral is a good tool for assessing severity provided that the growth of the damage is self-similar. On the other hand energy density theory does not suffer from this restriction. An advantage of energy density (i.e. W) is that the critical values can be determined from standard tests and do not require a series of special fracture related tests in order to find the toughness in modes I, II and III. Nevertheless T* is a very useful fracture parameter. Both T* and W are used at ARL and are particularly simple to calculate.

Closed form solutions have recently been obtained for a circular delamination under hydrostatic pressure. This solution involves large displacement and post-buckling theory, and has been confirmed by a series of finite element analyses [43].

The generic shape of the residual strength curve is as previously predicted numerically viz: "as the size of the damage increase a stage is reached after which a further significant increase in size does not result in a further reduction in static strength". This generic behaviour was also predicted for damage to adhesively bonded composite to metal joints. Tests at ARL on specimens representative of an F/A-18 step-lap joint have confirmed this prediction [44].

9.4.18 Composite Repairs (J. Paul - ARL)

The use of energy density theory applied to repaired, impact damaged composites has predicted that, provided global bending is restrained, the increase in the residual strength (RS) after repair can be estimated by the following simple formula:

$$RS\ (repaired)/RS\ (unrepaired) = W\ (unrepaired)/W\ (repaired)$$

where W is the energy density in the laminate in the region of the patch. Whilst this formula applies to multiaxial loading it reduces, in the case of uniaxial load, (and when the patch has the same effective modulus as the parent laminate) to

$$RS\ (repaired)/RS\ (unrepaired) = S\ (unrepaired)/S\ (repaired)$$

where S is the stress under the patch in the load direction.

This approximation was subsequently evaluated by a series of tests on a 50-ply AS-4/3501-6 $[(+45_2,-45_2,0_4)90]_S$ with and without a $[0_2,\pm 45,\mp 45,0_2]_S$ patch. Predicted failure strains were in good agreement with those measured experimentally [45].

9.4.19 Variable Amplitude Loading and Patching Efficiency (A.A. Baker - ARL)

The use of adhesively bonded composite patches to repair cracks in metallic components has been reported in several previous ICAF reviews. Recently, a collaborative program was undertaken between the National Aeronautical Establishment - Canada, and the Aeronautical Research Laboratory - Australia to assess the efficiency of patching under FALSTAFF loading. The 2024-T3 specimens, as depicted in Fig.21 were: a) prepared at ARL, b) pre-cracked at NAE, c) returned to ARL for patching, and d) tested under modified FALSTAFF loading at NAE.

The test results and the types of specimen tested are indicated in Fig. 22. In all cases the patches were boron/epoxy and adhesives were AF 163, FM 73 or acrylic Flexon 241. The pre-bonding surface treatments for panels repaired with adhesives AF 163 and FM 73 was grit blast (with alumina grit) followed by wash with a 1% aqueous solution of epoxy silane GPS (1) and dry at 70°C whereas for the acrylic adhesive the silane treatment was omitted. The initial crack lengths were approximately 5mm and 25mm, giving crack-to-width ratios of 0.03 and 0.16 respectively.

The patched specimens were tested as honeycomb sandwich panels; this approach [46] was taken to: a) avoid curvature of the panels due to residual stress which arise on patching due to thermal expansion mismatch between the composite patch material and aluminium specimen, and b) minimise bending under load due to the neutral-axis offset caused by the bonded patch. Bending stiffness produced by the honeycomb configuration is considered to simulate the level of restraint to secondary bending experienced in most aircraft structure.

In the FALSTAFF program employed in these tests the maximum stress was 248 MPa (36 ksi) tension; compression loads as normally applied in FALSTAFF were eliminated to avoid buckling of single unpatched panels. The peak stress was almost twice that employed in the standard program (138 MPa) and this stress was applied twice in each FALSTAFF block. One block of FALSTAFF loading contains 200 flights and each flight is nominally equivalent in length to one hour of flying.

The results to date, Fig. 22, [47] indicate that a very high level of patching efficiency is obtained for all of the adhesives investigated. It is of particular interest to note the excellent performance demonstrated by the acrylic adhesive, in view of its ambient temperature cure. In general, a life extension of over 200 FALSTAFF blocks was obtained, nominally 4 x 10^4 hours of (1 hour) flights. Catastrophic failure occurred in some specimens during application of the high loads when crack growth was well advanced, by disbonding or by fracture of the patch.

Future work in the collaborative programme will evaluate the effect of realistic temperature and humidity cycling. In these tests the compressive loads will be included.

9.4.20 Fatigue of Fibre Composite Materials and Components (D.S. Saunders - ARL)

(1) Impact-Damage Growth in 56-ply Coupons

Fatigue testing of impact-damaged CFC material (XAS-914C) has continued over the past two years. The experimental programme has concentrated mainly on the following areas of work:

(a) the effects of load spectrum modification (including load holds) on damage (delamination) growth,

(b) the effects of hot/wet environments on fatigue-life behaviour,

(c) the combined effects of (a) and (b), above, on the fatigue-life behaviour.

The results of work reported in [48] and [49] demonstrated that, under ambient conditions, the fatigue-life behaviour of impact damaged XAS-914C was unaffected by load spectrum modifications, Figs. 23 and 24. More recent preliminary results for severe hot/wet conditions demonstrated that the degradation in fatigue life was of the same magnitude as the measured degradation in static compressive strength [50], Fig. 25. Severe degradation in fatigue life was demonstrated when load holds were introduced into the load spectrum for testing impact-damaged coupons under hot/wet conditions [50], Fig. 26. Thus, as the glass transition temperature is approached (T_g being lowered by moisture ingress) under hot/wet environments, then it appears that time-dependent properties play a more significant role in the damage growth and/or ultimate failure process. Static testing of impact-damaged and impact-damaged-and-fatigued coupons is to be undertaken as an extension of this programme.

(2) Impact-damage growth monitoring

Time-of-flight ultrasonic C-scanning has been used to map delamination growth in thick coupons, [49,51], and development of the in-situ C-scanning frame (Section 9.4.22) has continued to permit more frequent mapping of delamination growth during fatigue testing. Delamination growth (under compression-dominated FALSTAFF) has been mapped for 56-ply XAS-914C material, $[\pm 45, 0_2]_{7S}$ and will be compared with delamination growth in 50-ply AS-4/3501 material. In the case of this material an appropriate loading spectrum, derived from flight loads data for Australian F/A-18 aircraft, will be used. (Preliminary C-scanning of the impact damage imparted to the 50-ply AS-4/3501 laminates has shown that the configuration of the damage is quite different to that imparted to the XAS-914C laminates tested in previous work.)

Delamination growth has also been studied using thermal emission measurements (SPATE) [22] and the change in level of emission related to a damage parameter which was found to increase monotonically with number of programs of fatigue loading. The level of thermal emission effectively represents a measure of the compliance of the region over which the scan is performed and so can be used to map damage growth.

(3) Fatigue Behaviour of Composite-to-Metal Joints using Bolted Fasteners

A study of the behaviour of composite-to-metal joints using bolted fasteners is in the preliminary stages and so far a number of specimen designs has been tested. One severe limitation of the specimens studied was the large degree of antibuckling restraint necessary for testing, since only small levels of secondary bending are believed to be appropriate to composite wing skins. (The testing to date has also used a spectrum which has a large number of high compressive loads.) This testing programme led to the adaption of an RAE design which has been found to be appropriate to the work in hand.

To date, a number of 50-ply AS4-3501 coupons has been tested under a MCAIR F/A-18 fatigue test spectrum. The most significant "failures" in the present series of tests have been appreciable damage in the fastener hole regions and fastener failures. The first aspect of the failure involves wear-out of the composite such that it changes the dimensions of the fastener holes in the composite, and cracking (delamination) occurs between fibre layers. Wear-out appears to involve erosion of resin and fibres from the ply layers but at different rates depending on the orientation of the ply layers with respect to the principal loading direction, Fig. 27. Exposure of the fibres and uneven wear ensues both in the "parallel-sided" region and in the region of the 100-degree countersink. Change in hole size in the composite material with number of programs of the MCAIR spectrum is being measured. Hole wear in composite coupons will be assessed for other spectra to determine if load spectrum influences fatigue life of holes for bolted fastener systems.

Ultimate failure of the specimen could be considered to occur when the fastener fails. Using the modified RAE test geometry, initial fastener failure occurred in 100 to 200 programs. It was found, however, that by replacing fasteners on failure the fatigue test could be continued but fastener life appeared to be reduced with each successive change. This is believed to be due to wearout of the fastener hole in the composite coupon producing larger bending stresses on the fastener which then fails sooner under fatigue loading. Fastener failure, however, has caused some concern in the experimental programme because it was not intended that this aspect would be studied in detail and in some cases the failure of fasteners imparts additional damage to the coupons, which complicates the experiments.

It is intended that the testing of bolted fastener systems under hot/wet environments will be undertaken during the next phase of the experimental work.

(4) Moisture Absorption by Carbon Fibre Composite Laminates.

Another aspect of the environmental studies which has been under investigation for the past two years has been in the moisture uptake of carbon fibre composite laminates. Most importantly, this work has resulted in an accumulation of moisture absorption data on thermally spiked XAS-914C and AS4-3501, [52]. The effects of thermal spiking have been incorporated in computer simulations of moisture absorption by graphite/epoxy laminates.

9.4.21 NDE Research (D.R. Arnott - ARL)

Non-Destructive Evaluation research conducted at ARL now focusses on the development of quantative methods in ultrasonics and Eddy-current non-destructive evaluation. One highlight from each of the research programs is presented. A theoretical treatment [53] of ultrasonic surface waves (Rayleigh waves) propagated along the surface of a metallic component with a corner of any angle containing a crack at any angle has shown that the depth of a uniform crack may be calculated from measurements of the separation of modulation peaks in the frequency spectrum for both forward scattered and backscattered waves, Fig. 28. Experiments conducted with a laser pulse source and a "pinducer" receiver successfully evaluated a fatigue crack of 3 millimetre depth. Work toward the evaluation of sub-millimetre cracks and work to address the closure problem is planned.

Eddy-current research [54, 55] has focussed on the development of theoretical models of coil response to cracks in thin metal plates starting with Maxwell's equations and using a current vortex formalism. The research has now successfully modelled coil responses to through-cracks in thin plates with excellent agreement between the theoretical model and experimental data. Success with the second layer problem, where a through-crack in a thin plate is located above or below a second plate of arbitrary thickness, has practical significance to overlapping aircraft skins and to cracks beneath fasteners. Research to investigate the inverse problem (i.e. to predict crack behaviour from inspection of coil response) is in progress.

9.4.22 Developments in Practical NDI (G. Clark - ARL)

The application of non-destructive inspection methods to composite materials and components is an area of major interest, particularly with the acquisition by the RAAF of the F/A-18 aircraft. The potential long-term effects of impact damage occurring during service are not yet fully understood, and in order to assess the significance of such damage it is first necessary to develop NDI methods which can locate the damaged area and reveal its nature and extent. At ARL an ultrasonic technique, depth C-scanning, has been developed [56, 57] for use with a large immersion ultrasonic system; this technique uses time-of-flight ultrasonic data, combined with a more conventional C-scanning approach, to determine both the extent and depth location of each delamination in the damaged region. A sophisticated data analysis and display system permits real-time display of defect maps which are colour-coded to assist with interpretation; resolution is sufficient to allow layer-by-layer display of the damage envelope. A portable version of the depth C-scanning system has been developed, Fig. 29, to permit the acquisition of ultrasonic signal data from impact-damaged coupons under test or from damaged aircraft components in the field. These data can then be transferred to the larger system for more sophisticated processing, analysis and display. The system is currently the subject of field trials, Fig. 30.

Work is currently under way to investigate the feasibility of in situ NDI of critical components. Various approaches are under consideration, including Eddy-current, magnetic field leakage, and methods based on optical fibre fracture. The objective is to develop a sensor system which can be permanently mounted in known critical regions of an airframe and, by means of either real-time monitoring or signal storage, to detect the development of cracking. Areas in which critical defect sizes are small are of particular interest, and any useful approach must feature high levels of both sensitivity and stability. To date, promising results have been obtained using solid-state magnetic field sensors. These sensors have been tested on fatigue cracks in specimens simulating high-strength steel aircraft components; results are being evaluated.

9.4.23 Patch Durability field Trial (T.G. Hill, A.A. Baker - ARL)

A series of demonstration metal patches was applied to an Ansett Boeing 727-200 commercial airliner in July 1984 to study the effect of real (non-simulated) exposure on the durability of a number of surface treatment/adhesive combinations. All of the patches were formed from aluminium alloy 2024-T3 clad, although two different thickness of patch were used. The adhesives were standard aerospace film and paste epoxy adhesives and the surface treatments included state-of-the-art and experimental preparations. Eight patches were applied to the lower fuselage aft of the wheel well and four to a horizontal stabilizer leading edge. All patches were placed to avoid the possibility of engine ingestion.

Since the patches were applied, the aircraft has been in regular domestic service within Australia for three years and it also spent nine months in Jamaica on lease. During this time the bonded patches have received no privileged treatment. For instance, the eight fuselage patches are painted with the standard polyurethane paint scheme and that area is washed every two days with a proprietary aircraft washing compound to remove dirt and hydraulic oil stains.

The four horizontal stabilizer patches are in an unpainted area subject to severe erosion. The only detectable deterioration has been direct erosion of the adhesive under the upstream edge of these patches. This situation is monitored at normal maintenance checks but no special treatment is given.

The patches have now been flown in excess of 10000 hours as at December 1988.

9.4.24 Graphite/Epoxy Environmental Exposure Programme (R.J. Chester - ARL)

Graphite/Epoxy (gr/ep) composite materials are widely employed in the structure of the F/A-18 aircraft; indeed, 34% of the external surface area and 9% of the dry structural weight is gr/ep. These materials are currently replacing conventional aluminium alloys for both major and minor components, bcause they offer weight reductions due to their high specific strength and stiffness and because they are generally resistant to degradation by fatigue and corosion.

In order to support the RAAF in maintenance of the gr/ep components on the F/A-18, ARL has initiated a major program in association with the Tropical Exposure Site at MRL-Queensland. The experiments are designed to evaluate the influence of cyclic loading and tropical exposure on a range of gr/ep specimens which represent various structural details of the aircraft; honeycomb sandwich specimens, monolithic moisture absorption coupons, adhesively bonded joints and mechanically fastened joints will all form part of the trial. The sandwich beam specimens consist of two ten-ply skins bonded to aluminium honeycomb core and as such are typical of various components on the F/A-18. These beam specimens contain either representative forms of damage (such as impact damage or teflon inclusions) or representative repairs. Constant amplitude cyclic loading at a strain level of 2500 microstrain is applied to all specimens in blocks of 100,000 cycles at a frequency of 0.013 Hz. Between these loading blocks the beams are removed from the loading rigs and subjected to thermal cycling (-50°C + 105°C) and 40 overload cycles at 3500 microstrain. These conditions are designed to simulate as closely as possible the conditions of actual aircraft usage.

Damage assessment is carried out by ultrasonic C-scanning, compliance measurements and residual strength measurements at the end of the trial. To date, the specimens have experienced from 300,000 to 500,000 cycles and no damage growth has been observed by C-scan.

9.5 REVIEW OF STRUCTURAL FATIGUE RESEARCH AND DEVELOPMENT IN NEW ZEALAND (H. Levinsohn - DSE)

The small New Zealand R&D effort in aeronautical fatigue is centred at the Defence Scientific Establishment (DSE) in Auckland. DSE shares some facilities with the Auckland Industrial Development Division (AIDD) of the Department of Scientific and Industrial Research (DSIR) and the University of Auckland Engineering School laboratories.

The DSE project on aircraft fatigue is dominated by the need to maintain the airworthiness of existing aircraft types. Past fatigue problems in the RNZAF fleet of jet attack and trainer aircraft drove home the need for increased self-sufficiency in aircraft structures engineering. The problem of maintaining older aircraft structures is becoming more severe because the jet attack and maritime surveillance aircraft are being updated with modern electronic equipment, with the intention of extending their operational lifetime.

Airborne Fatigue Recorder

New Zealand does not have the resources to conduct full scale fatigue testing. Alternatively DSE has allocated a significant effort, by New Zealand standards, to developing a fatigue load measurement system. This has been named the aircraft structural strain sequence recorder (A3SR), with the major initiative directed towards the design and construction of an airborne microprocessor component known as ADARS (airborne data acquisition and recording system). The retrieval of data from ADARS is achieved with the use of a desk-top computer hardware and software package known as the ground support unit (GSU). ADARS and the GSU are the major A3SR components, but the whole system also includes a variety of lesser items necessary to produce suitable data displays and strain gauge checking facilities. ADARS has 8 primary channels of which 7 will be connected to strain gauges and one to an accelerometer. The sampling rate for these channels will be 150 Hz with a frequency response cut-off at a maximum of 30Hz. Three additional channels will be used to record air speed, altitude and temperature at intervals of no less than one sample per minute. The system will record strain turning points at a maximum resolution of 15 microstrain, step-width. ADARS is made up of 2 plug-in elements, namely the data acquisition module (DAM) and the data memory module (DMM), Fig. 31 (a). The DAM unit is permanently mounted in the aircraft, but the DMM unit is unplugged and transferred to the GSU, Fig. 31 (b), for the purpose of data recovery. The DMM stores the strain turning points using a RAM of 2 Mbytes with a lithium battery backup, capable of retaining the data for up to 48 days. The records are headed by information of which the main items are the identification of the aircraft, the pilot code, the time and date, and the sortie type. The turning points are also time annotated at one second intervals. It is intended to develop an aircraft fatigue recorder (AFR) from the A3SR, by ultimately incorporating into the DAM a suitable crack growth model applicable to RNZAF operational conditions. In the meantime A3SR data will be used for laboratory experimental work to derive or modify existing crack growth models. Validation trials will be carried out by subjecting aluminium aircraft alloy test coupons to compressed fatigue load spectra derived from the A3SR data.

The object of this work is to create a position whereby the RNZAF can predict structures problems rather than having to react to them. The plan is to fly a prototype system (A3SR/ADARS) by April 1989 and produce a prototype AFR by the end of 1990. The initial flight trials will be done with an RNZAF jet trainer aircraft. One of the main features of the AFR will be the interface between the fatigue monitoring system and the pilot. Research will also be needed into ways of presenting fatigue data to assist the pilot to conserve structural life.

Damage Tolerance Assessment

Although cumulative damage tolerance analysis methods have many attributes, these cannot easily be applied to older RNZAF aircraft containing fatigue and corrosion repairs which were not evaluated in full scale tests. Unfortunately, deterministic fracture mechanics methods to define the durability aspects of repairs need to be refined to improve their reliability.

Part Substitution

The availability and cost of spare parts for RNZAF aircraft has become a problem as the aircraft fleet becomes older. DSE has embarked on a project to assess the suitability of high quality castings which could be produced economically by local industry. The newly developed high strength aluminium/copper/silver casting alloy is being investigated. Improvements to mechanical properties by hot isostatic pressing (HIPing) of castings made from this alloy are being evaluated. The effect of HIPing on internal porosity defects is also being studied. The question of whether gas and/or shrinkage cavities are converted to linear defects or eliminated by solid phase diffusion welding needs to be addressed. Consequently, the DSE investigation is becoming involved with the surface chemistry of casting defects.

The project will advance to the stage of certifying such components for use in aircraft and the validation schedule will include spectrum loading fatigue testing.

Strain Gauging

Reliable strain gauge installations are a key to the development of the high performance fatigue recorders which are needed for the accurate monitoring of loads in existing and future aircraft types. Concern is raised about the inconsistent levels of reliability experienced with strain gauges in service. Improvements in this area are clearly needed. In particular, users require more information about the long term reliability of given types of strain gauge installations under severe operating environments. DSE is addressing some of these questions, but is concerned that there are no published quality assurance standards for the design of installations and testing of strain gauge bridge circuits. This issue is of prime importance if the fatigue lives of individual aircraft are to be determined from strain measurements on service aircraft.

9.6 BIBLIOGRAPHY ON THE FATIGUE OF MATERIALS, COMPONENTS AND STRUCTURES

The fourth volume of this work by J.Y. Mann covering the period 1966 to 1969 has been submitted to the publisher (Pergamon Press) for publication later in 1989. It contains 5709 references and will join the first three volumes of the Bibliography covering the periods 1838-1950, 1951-1960 and 1961-1965 respectively.

9.7 REFERENCES

1. Jost, G.S. A review of Australian investigations on aeronautical fatigue during the period April 1985 to March 1987. Minutes of the 20 ICAF Conference, Ottawa, June 1987.

2. Grandage, J.M. A chronological record of the Swiss Mirage fatigue test. ARL Structures Tech. Memo. 479, December 1987.

3. Tuller, L.T. GAF Nomad Model N22 Certification Report N22-7202 - Fatigue test crack and repair summary.

4. King, C.N. RAN Wessex Mk 31B Helicopter - Fatigue life of main rotor gear box. ARL Aeropropulsion Report 170, March 1986.

5. King, C.N. and Fraser, K.F. Fatigue life estimation for the main rotor gear box in Sea King helicopters operated by the Royal Australian Navy. ARL Aeropropulsion Report 171, July 1986.

6. Fraser, K.F. In-flight computation of helicopter transmission fatigue life expenditure. AIAA Journal of Aircraft, Vol. 20, No. 7, July 1983.

7. Fraser, K.F. and King, C.N. Helicopter transmission fatigue life estimation. AIAA Journal of Aircraft, Vol. 25, No 5, May 1988.

8. Hole, B. Examination of a cracked rear pressure bulkhead attach tee section from a DC-9 aircraft. Specialist Report X-2/89, Materials Evaluation Facility, Civil Aviation Authority, Canberra, Australia, February 1989.

9. RMIT Composite Sailplane Program:
   (a) Overview - L.A. Wood and N. Mileshkin
   (b) Full scale fatigue test - D.R. Collyer and C.A. Patching
   (c) Specimen testing - G.W. Randall
   (d) Finite element modelling - M.L. Scott
   (e) An investigation of premature failures in sailplane gel coats - P.D. Coates and M.L. Scott
   Royal Aeronautical Society in Australia, Aerospace 88 Australia, Sydney, 10-11 October 1988, pp. 103-134.

10. Romeyn, A. Drag brace assembly fracture - Bell 214ST. Specialist Report X-9/88, Materials Evaluation Facility, Civil Aviation Authority, Canberra, Australia, July 1988.

11. Romeyn, A. Main rotor spindle fracture - Puma. Specialist Report 7/87, Materials Evaluation Facility, Civil Aviation Authority, Canberra, Australia, June 1987.

12. Care, G. and Steven, G. Benchmark testing for plane elasticity non-linear analysis. Finite Element Methods in Engineering. Proc. 5th Int. Conf. in Australia on Finite Element Methods, Melbourne, 19-21 August 1987, pp. 75-82.

13. Carey, R.P. Computed stress and strain distributions under interference-fit and after cold-working. ARL Structures Tech. Memo. 466, August 1987.

14. Carey, R.P. Three dimensional computation of stress, strain and strain energy density under interference-fit and after cold-working of holes. ARL Structures Tech. Memo 478, January 1988.

15. Jost, G.S. Stress and strains in a cold-worked annulus. ARL Aircraft Structures Report 434, September 1988.

16. Mann, J.Y., Sparrow, J.G. and Beaver, P.W. Fatigue characteristics of joints with holes cold-expanded in a multi-layer stack. To be published in Int. J. Fatigue.

17. Pell, R.A., Beaver, P.W., Mann, J.Y. and Sparrow, J.G. Fatigue of thick-section cold-expanded holes with and without cracks. Submitted to Fatigue and Fracture of Engineering Materials and Structures.

18. Heller, M., Hill, T., Williams, J.F. and Jones, R. Increasing the fatigue life of cracked fastener holes using bonded repairs. Theoretical and Applied Fracture Mechanics Vol. 11, No. 1, 1989.

19. Heller, M. Some problems concerning the fatigue life enhancement of aircraft structures. Ph.D. Thesis, University of Melbourne, Submitted April 1989.

20. Jones, R., Wong, A.K., Heller, M. and Williams, J.F. The role of energy related parameters in life enhancement. Proc. 5th Int. conf. in Australia on Finite Element Methods, Melbourne, Eds. J.F. Williams and L.K. Stevens, August 1987, pp. 244-246.

21. Paul, J. and Molent, L. Applications of energy density theory in cyclic plasticity. Theoretical and Applied Fracture Mechanics, Vol. 10, No. 1, 1988, pp. 43-48.

22. Jones, R., Heller, M., Lombardo, D., Dunn, S. and Saunders, D. Thermoelastic assessment of damage growth in composites. J. of Composites (in press).

23. Molent, L. and Dunn, S. A test methodology for an investigation into heating and cooling phenomena in grossly deformed tensile metallic bars. ARL Structures Tech. Memo. 482, December 1987.

24. Ryall, T.G. and Wong, A.K. Determining stress components from thermoelastic data - a theoretical study. ARL Structures Tech. Memo. 491, June 1988.

25. Wong, A.K., Sparrow, J.G. and Dunn, S.A. Recent work on the stress dependence of the thermoelastic parameter. SPIE, Vol. 817, San Diego, USA, August 1987.

26. Wong, A.K., Sparrow, J.G. and Dunn, S.A. On the revised theory of the thermoelastic effect. Journal of Physics and Chemistry of Solids, 49(4), 1988, pp. 395-400.

27. Wong, A.K., Dunn, S.A. and Sparrow, J.G. Residual stress measurement by means of the thermoelastic effect. Nature, 332 (1988) pp. 613-615.

28. Dunn, S.A., Wong, A.K. and Sparrow, J.G. On the mean stress dependence of the thermoelastic parameter for three high strength alloys. ARL Structures Report 433, May 1988.

29. Dunn, S.A., Lombardo, D. and Sparrow, J.G. The mean stress effect in metallic alloys and composites. Paper presented to Int. Conf. on Stress and Vibration, London, 14-16 March 1989.

30. Rumble, S.J. Experimental aspects of using time-averaged holographic interferometry to detect barely-visible impact damage in a graphite/epoxy composite plate. ARL Structures Tech. Memo. 467, June 1987.

31. Watters, K.C. In-plane Moire optical processing at ARL. ARL Structures Report 428, July 1987.

32. Finney, J.M. Sensitivity of fatigue crack growth prediction (using Wheeler retardation) to data representation. JTEVA, Vol. 17, No. 2, March 1989.

33. Ford, D.G. The summation of continuous damage. ARL Structures Report (in preparation) 1989.

34. Clayton, J.Q. and Clark, G. The effect of steel shot and glass bead peening treatments on the fatigue resistance of 7050-T76351 aluminium alloy. Proc. Conf. Aust. Fracture Group Symposium, Melbourne, November 1988.

35. Thomson, W. On the thermoelastic and thermomagnetic properties of matter. Q.J. Maths., Vol., 1855, pp. 55-77.

36. Wong. A.K., Jones, R. and Sparrow, J.G. *Thermoelastic constant or thermoelastic parameter.* J. Physics and Chemistry of Solids, Vol. 48, 1988. pp. 749-753.

37. Jones, R., Tay, T.E. and Williams, J.F. Thermomechanical behaviour of composites. Composite Material Response : constitutive relations and damage mechanisms, Applied Science Publishers, London, 1988, pp. 49-60.

38. Heller, M., Dunn, S.A., Williams, J.F., and Jones, R. Thermomechanical analysis of composite specimens, J. Composite Structures (in press).

39. Jones, R., Molent, L., Baker, A.A. and Davis, M.J., Bonded repaired metallic components: thick sections. Theoretical and Applied Fracture Mechanics, 9, 1, 1988, pp. 61-70.

40. Jones, R., Paul, J., Tay, T.E. and Williams, J.F. Assessment of the effect of impact damage: some problems and answers. Theoretical and Applied Fracture Mechanics., 9, 3, 1988, pp. 83-95.

41. Molent, L., Paul, J. and Jones, R. *Criteria for matrix dominated failure.* ARL Structures Report 432, 1988.

42. Tay, T.E., Williams, J.F. and Jones, R. Application of the T* integral and S criteria in finite element analysis of impact damage at fastener holes in graphite epoxy laminates under compression. J. Composite Structures, 7, 4, 1987, pp. 233-253.

43. Partridge, C., Waechter, R.T., Williams, J.F. and Jones, R. Understanding delamination behaviour in a circular laminate. Theoretical and Applied Fracture Mechanics (in press).

44. van Blaricum, T.J., Bates, P. and Jones, R. An experimental investigation into the effects of impact damage on the compressive strength of step-lap joints. Eng. Frac. Mech. (in press).

45. Jones, R. and Paul, J. Fibre composite repairs to damaged structural components. Plenary Lecture, 6th Int. Conf. on Composite Structures, Glasgow, July 1989.

46. Baker, A.A. Fibre composite repair of cracked metallic aircraft components - practical and basic aspects. Composites, 18, September 1987.

47. Raizenne, M.D., Heath, J.B.R. and Benak, T. TTCP PTP-4 collaborative test program - Variable amplitude loading of thin metallic materials repaired with composite patches. National Aeronautical Establishment Laboratory Technical Report LTR-ST-1662, 1988.

48. Clark, G. and van Blaricum, T.J. Load spectrum modification effects on fatigue of impact-damaged carbon fibre composite coupons. Composites, 18, July 1987, pp. 243-251.

49. Saunders, D.S. and van Blaricum, T.J. Effect of load duration on the fatigue behaviour of graphite/epoxy laminates containing delaminations. Composites, 19, May 1988, pp. 217-228.

50. Saunders, D.S., Clark, G., van Blaricum, T.J. and Preuss, T.E. Graphite/epoxy laminate testing programme at DSTO Aeronautical Research Laboratory. Presented at a workshop sponsored by the Commonwealth Advisory Aeronautical Research Council at ARL, 18-19 May 1988; to be published in Journal of Theoretical and Applied Fracture Mechanics.

51. Clark, G. and Preuss, T.E. Use of time of flight C-scanning for assessment of impact damage in composites. Composites, 19 (2), 1988, pp. 145-148.

52. Clark, G., Saunders, D.S., van Blaricum, T.J. and Richmond, M.E. Moisture absorption in graphite/epoxy laminates. To be published.

53. Doyle, P.A. and Scala, C.M. Depth measurement for corner cracks of arbitrary angle using Rayleigh waves. Ultrasonics (submitted 1989).

54. Burke, S.K. and Rose, L.F.R. Eddy-current detection of short cracks in plates. Review of Progress in Quantitative NDE (accepted 1989).

55. Burke, S.K. A benchmark problem for calculation of Z in Eddy-current NDE, Journal of NDE (accepted 1989).

56. Preuss, T.E. and Clark, G. Use of time-of-flight C-scanning for assessment of impact damage in composites. Composites, 19, 2, March 1988, pp. 145-148.

57. Preuss, T.E. and Clark, G. Some developments in ultrasonic C-scanning of composites. Proc. AINDT, Brisbane, 1988.

58. Broughton, W.R. and Chester, R.J. The development of a portable ultrasonic facility for NDI of graphite/epoxy composites. J. Australian Institute for NDT, 23, 1986, pp. 38-40.

59. Chester, R.J. A comparison of sealants for protecting strain gauges during tropical exposure. Exp. Tecnniques, 11, 1987, pp. 22-23.

60. Chalkley, P.D. and Chester, R.J. Interim report on environmental durability program. ARL Tech. Memo 397, 1988.

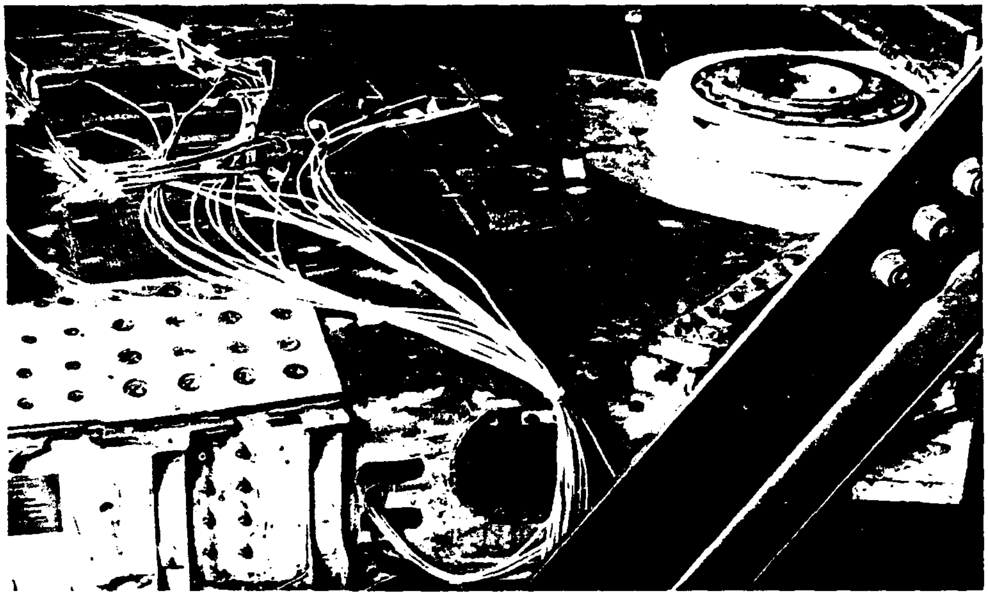


FIG. 1 BORON/EPOXY DOUBLERS FOR F-111 WING PIVOT FITTING

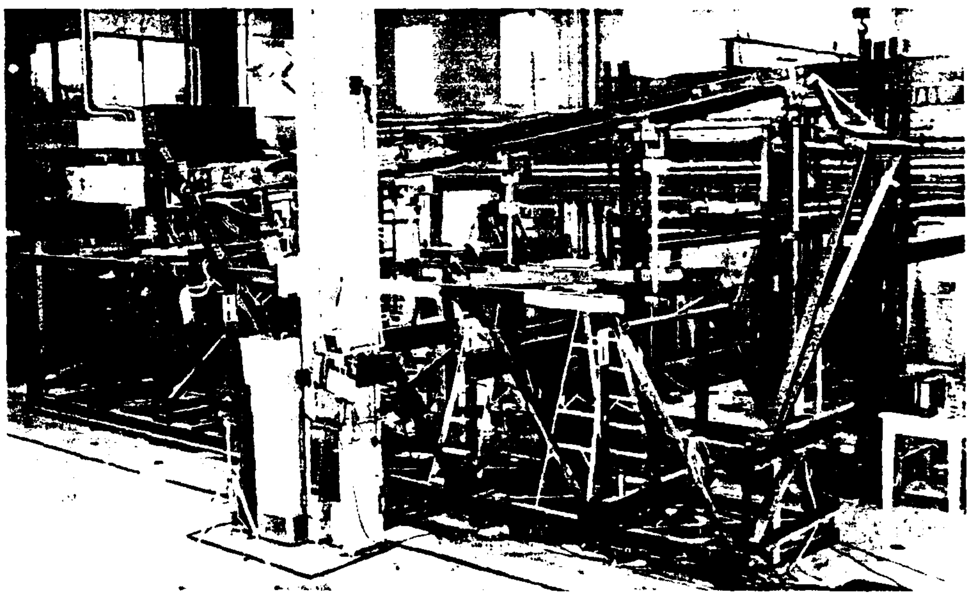


FIG. 2 F-111 WING WITH BORON/EPOXY DOUBLERS UNDER STATIC TEST (80% POSITIVE PROOF LOAD)

FIG. 3 VORTICES FROM LEADING EDGE OF F/A-18

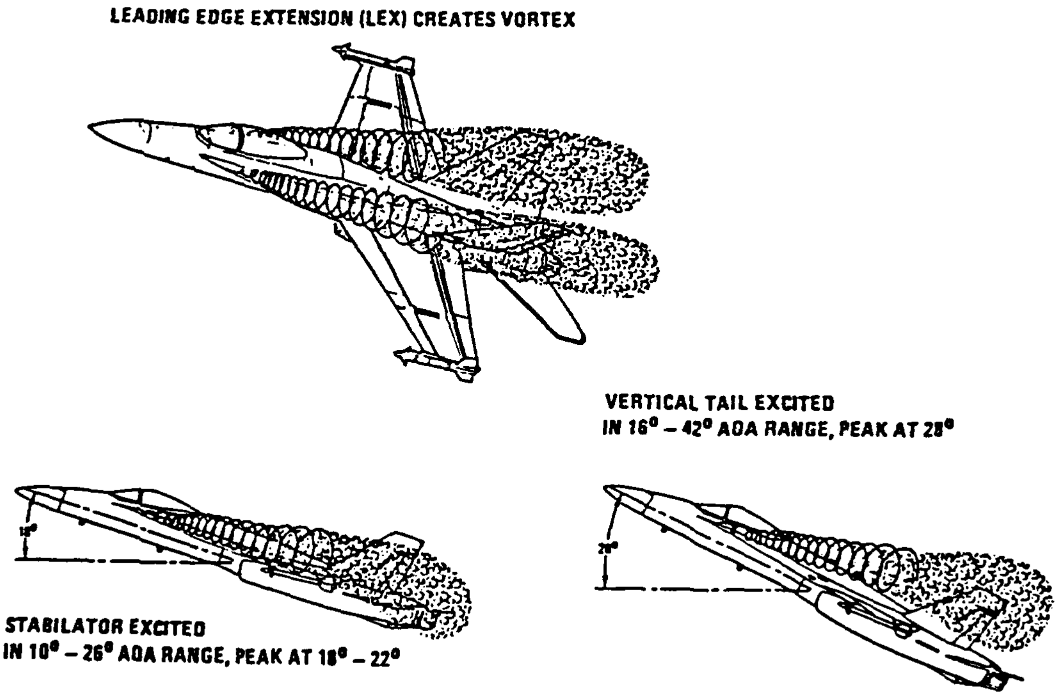


FIG. 4 VORTEX LOADING OF F/A-18 EMPENNAGE

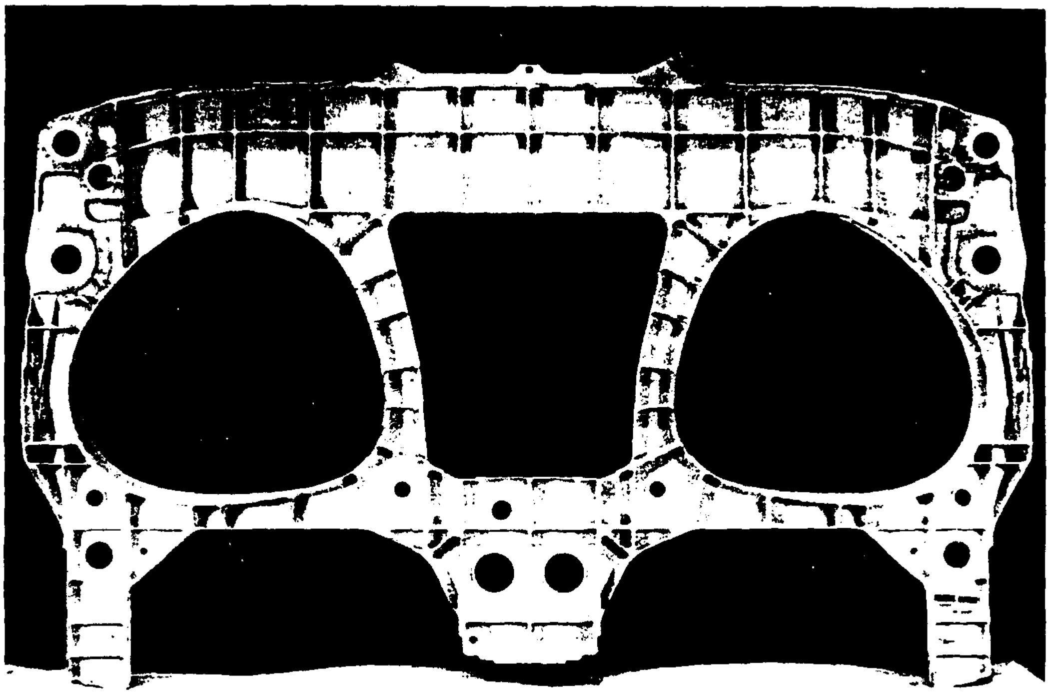


FIG. 5 F/A-18 BULKHEAD 488

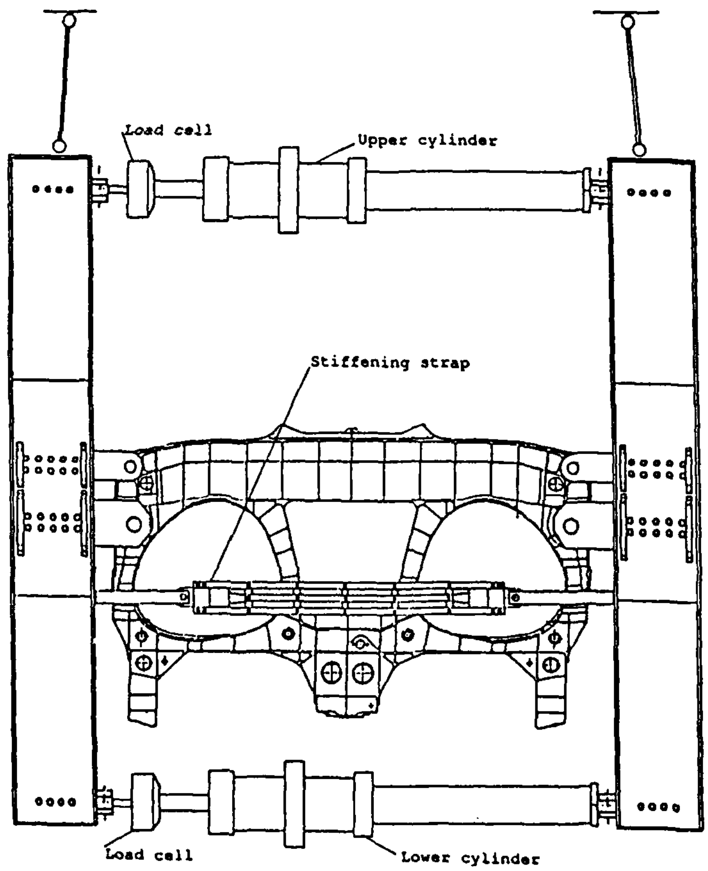


FIG. 6 F/A-18 BULKHEAD FATIGUE TESTING RIG

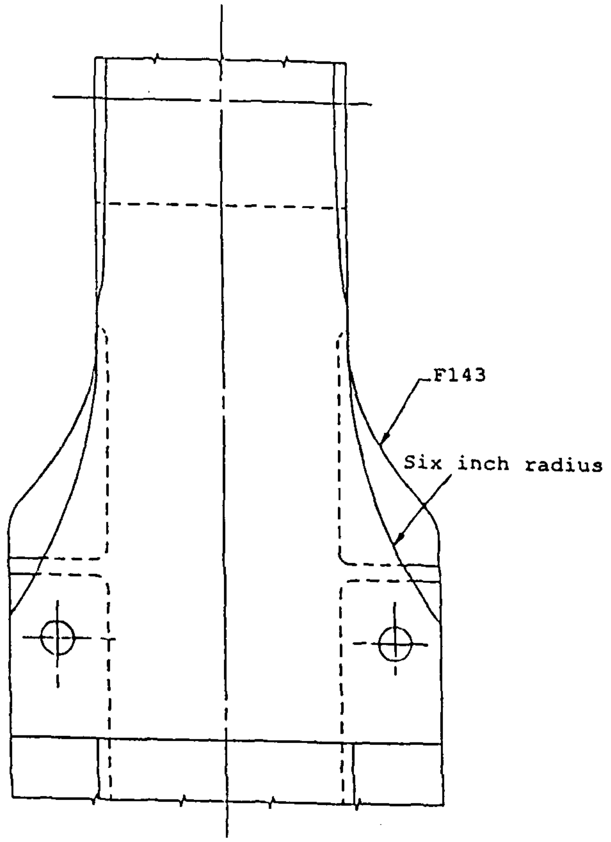


FIG. 7 F/A-18 BULKHEAD FILLET PROFILES

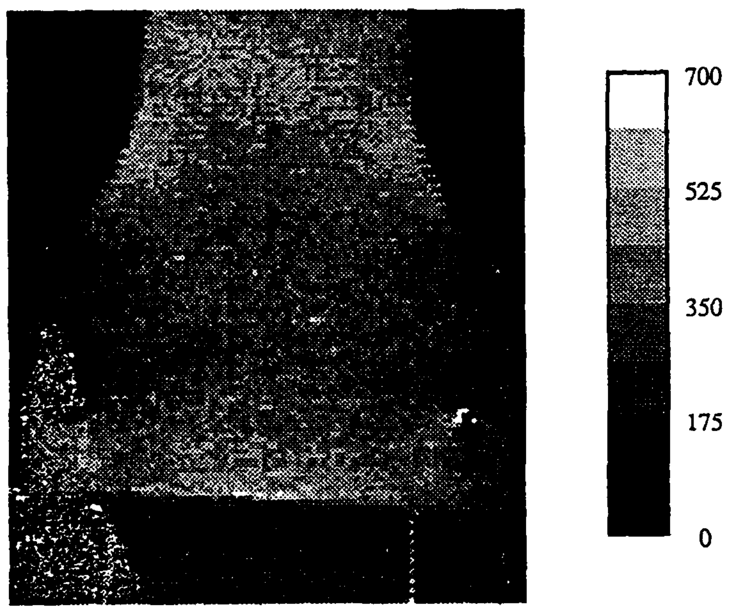


FIG. 8 SPATE SCAN OF F/A-18 F143 FILLET STRESSES
(UNCALIBRATED UNITS)

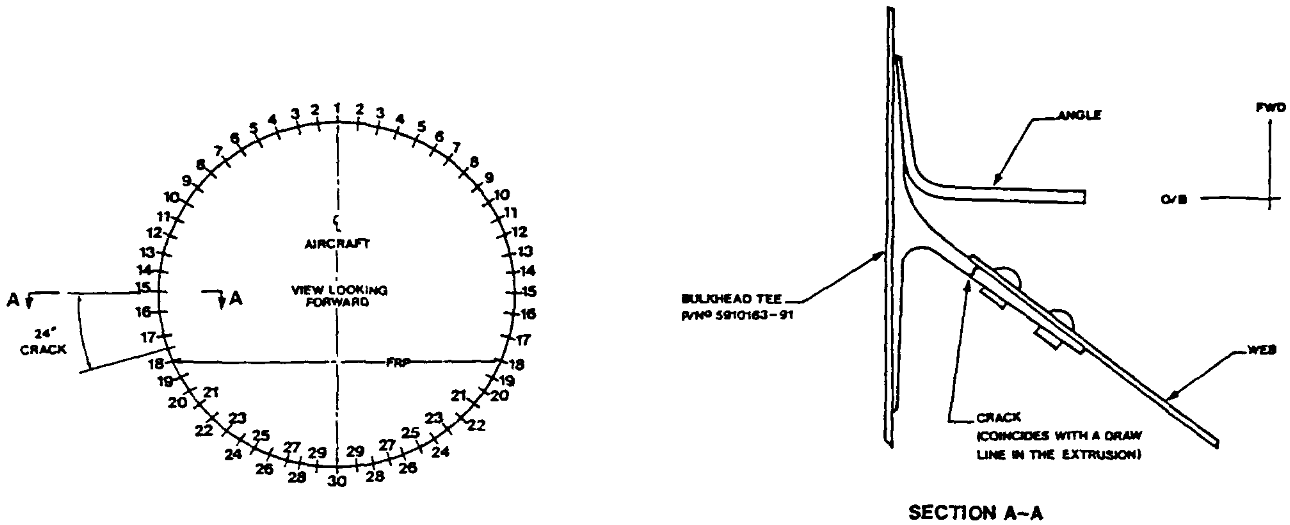


FIG. 9 DOUGLAS DC-9 REAR PRESSURE BULKHEAD ATTACHMENT

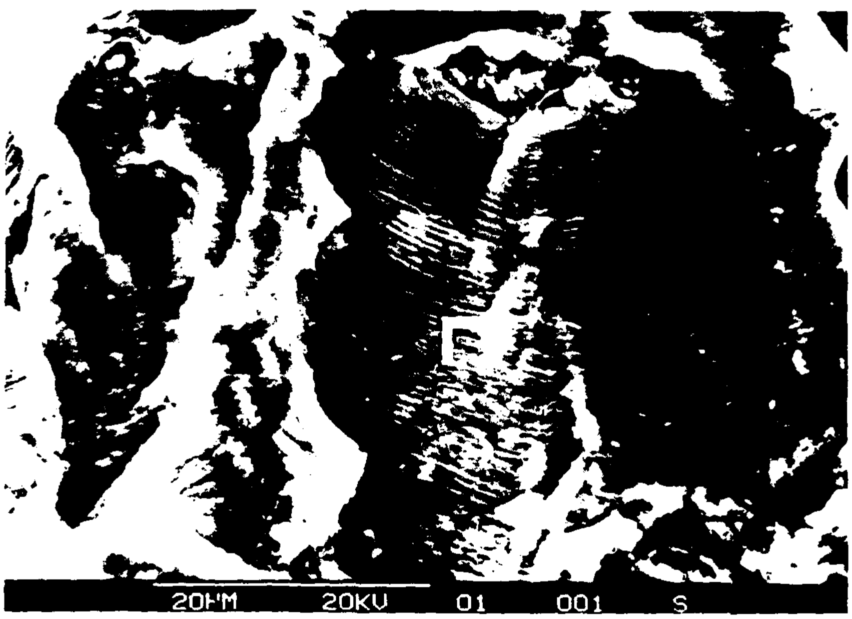


FIG. 10 DOUGLAS DC-9 BULKHEAD ATTACHMENT FRACTURE SURFACE

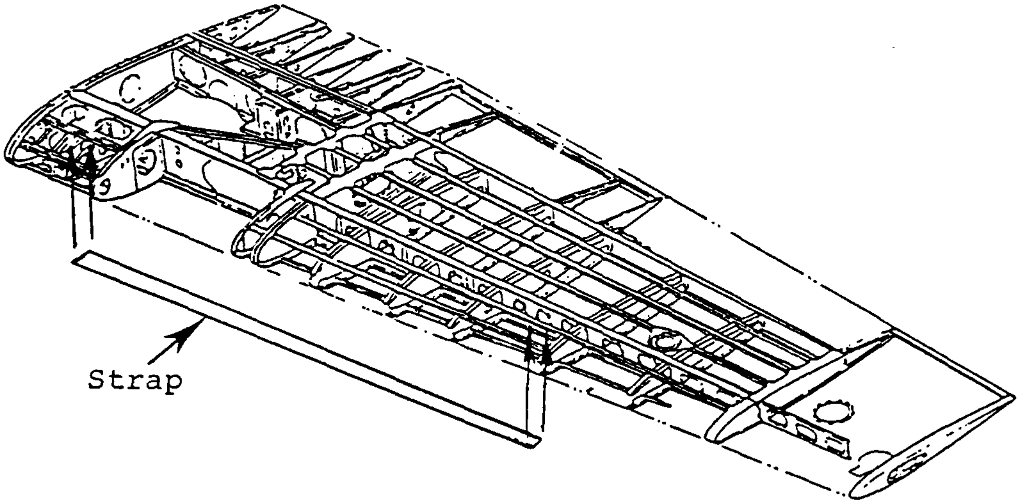


FIG. 11 CESSNA SERIES 400 SPAR REINFORCEMENT STRAP

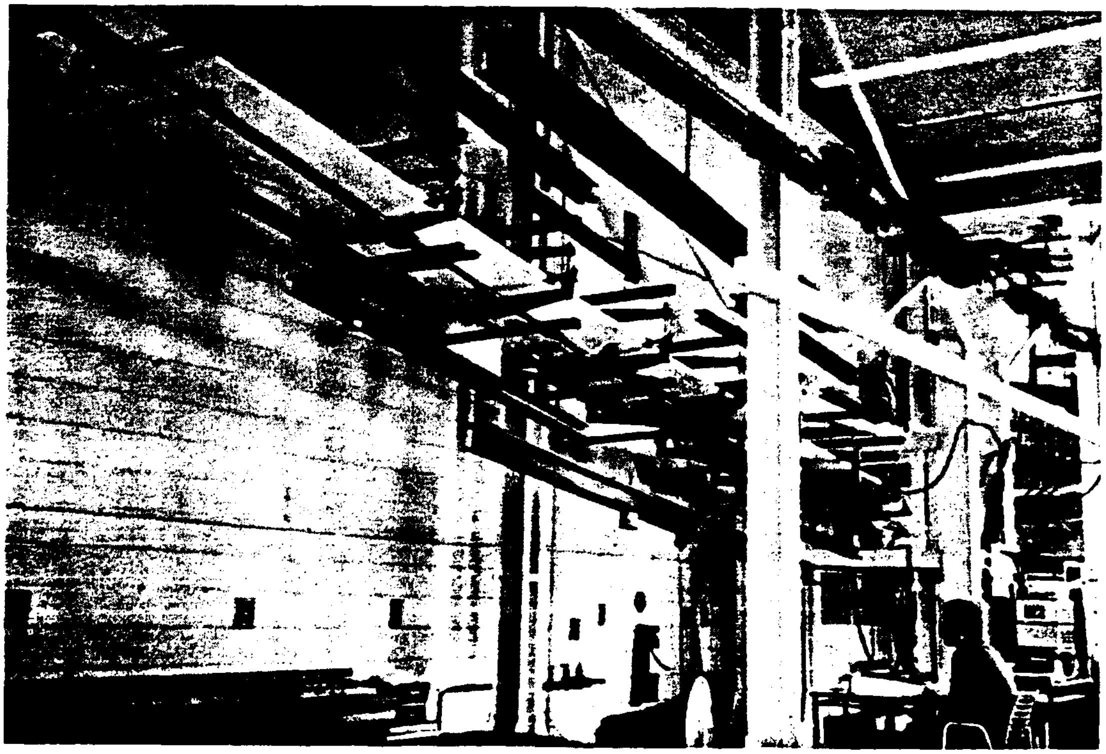


FIG. 12 JANUS GLIDER WING FATIGUE TEST

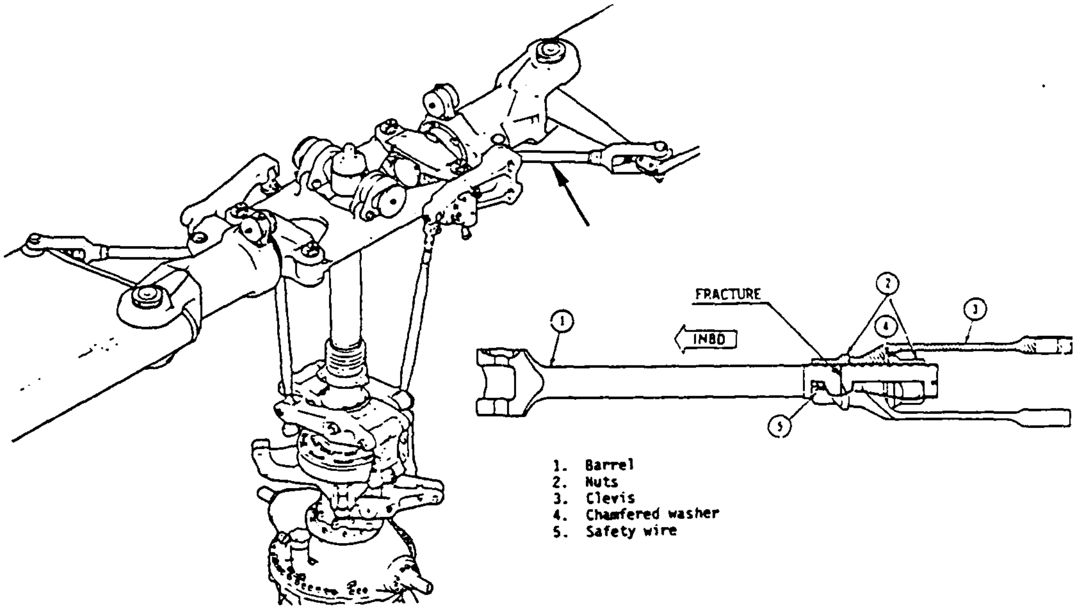


FIG. 13 BELL 214ST MAIN ROTOR HEAD AND DRAG BRACE

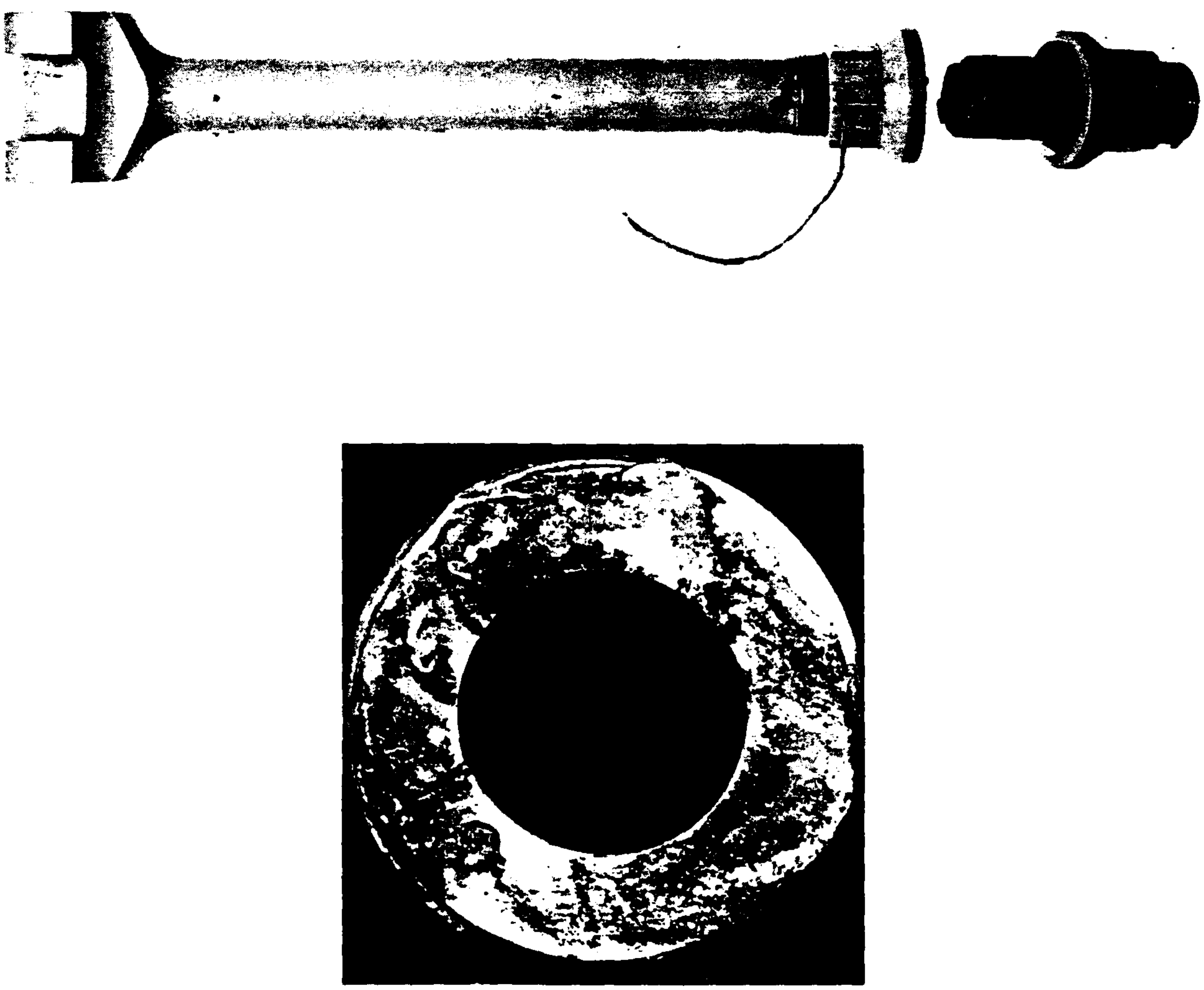


FIG. 14 BELL 214ST DRAG BRACE FAILURE AND FRACTURE SURFACE

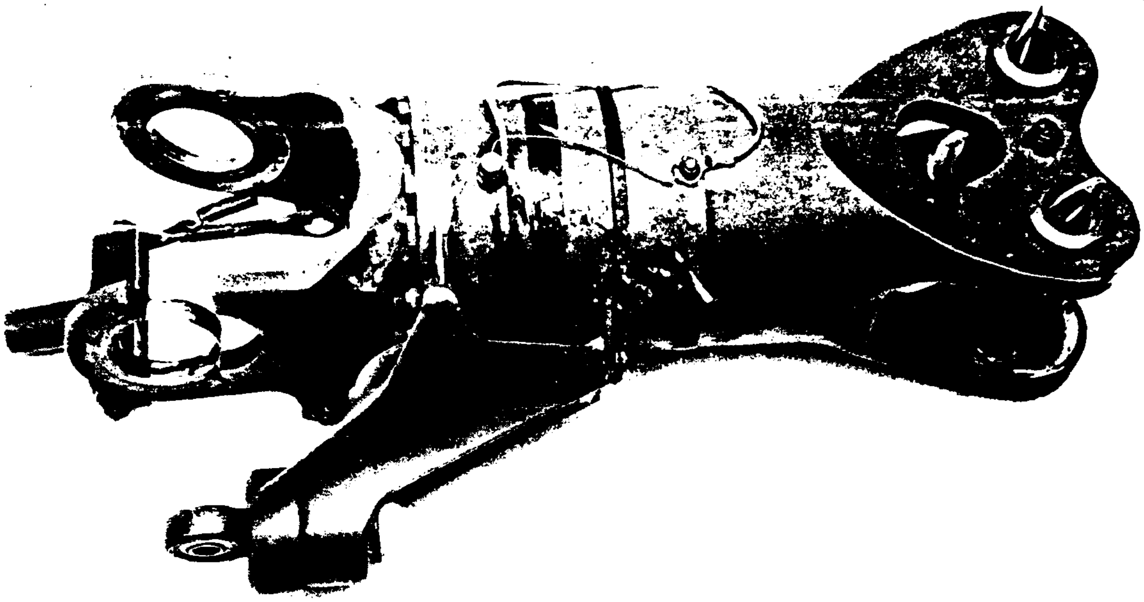


FIG. 15 AEROSPATIALE SA330J PUMA MAIN ROTOR SPINDLE FAILURE

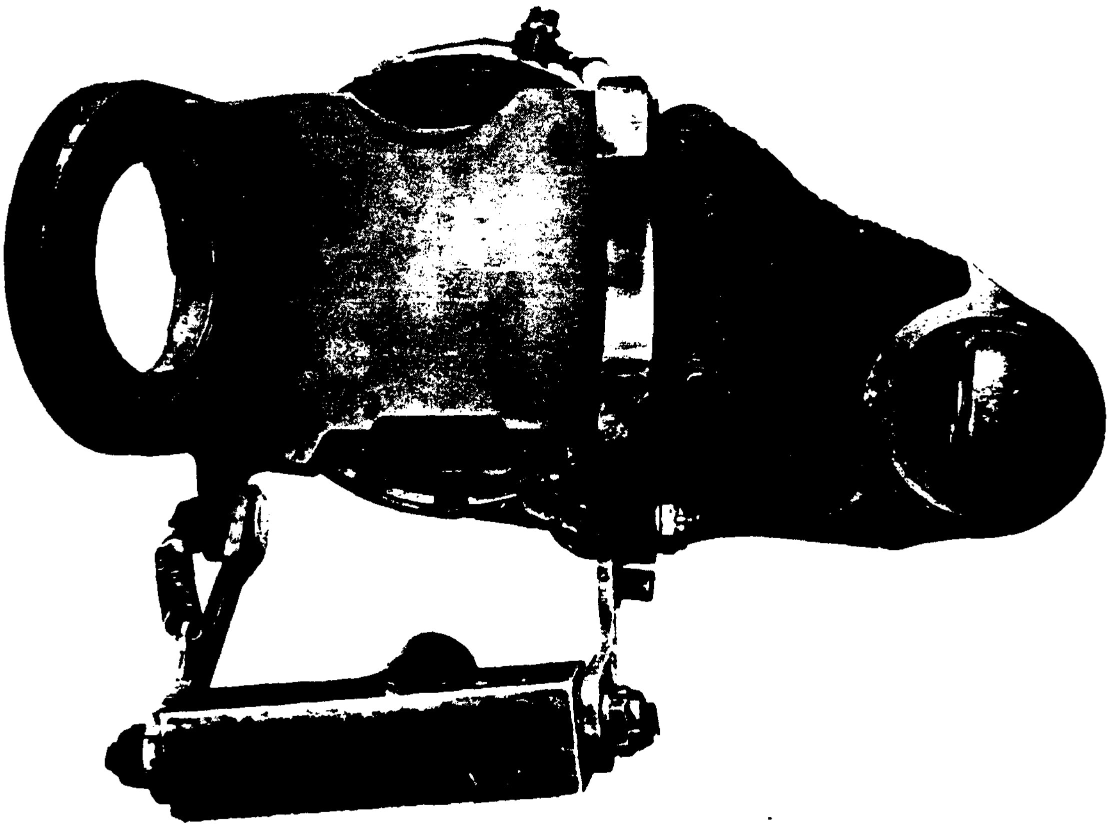


FIG. 16 SA330J PUMA MAIN ROTOR SPINDLE FRACTURE SURFACES

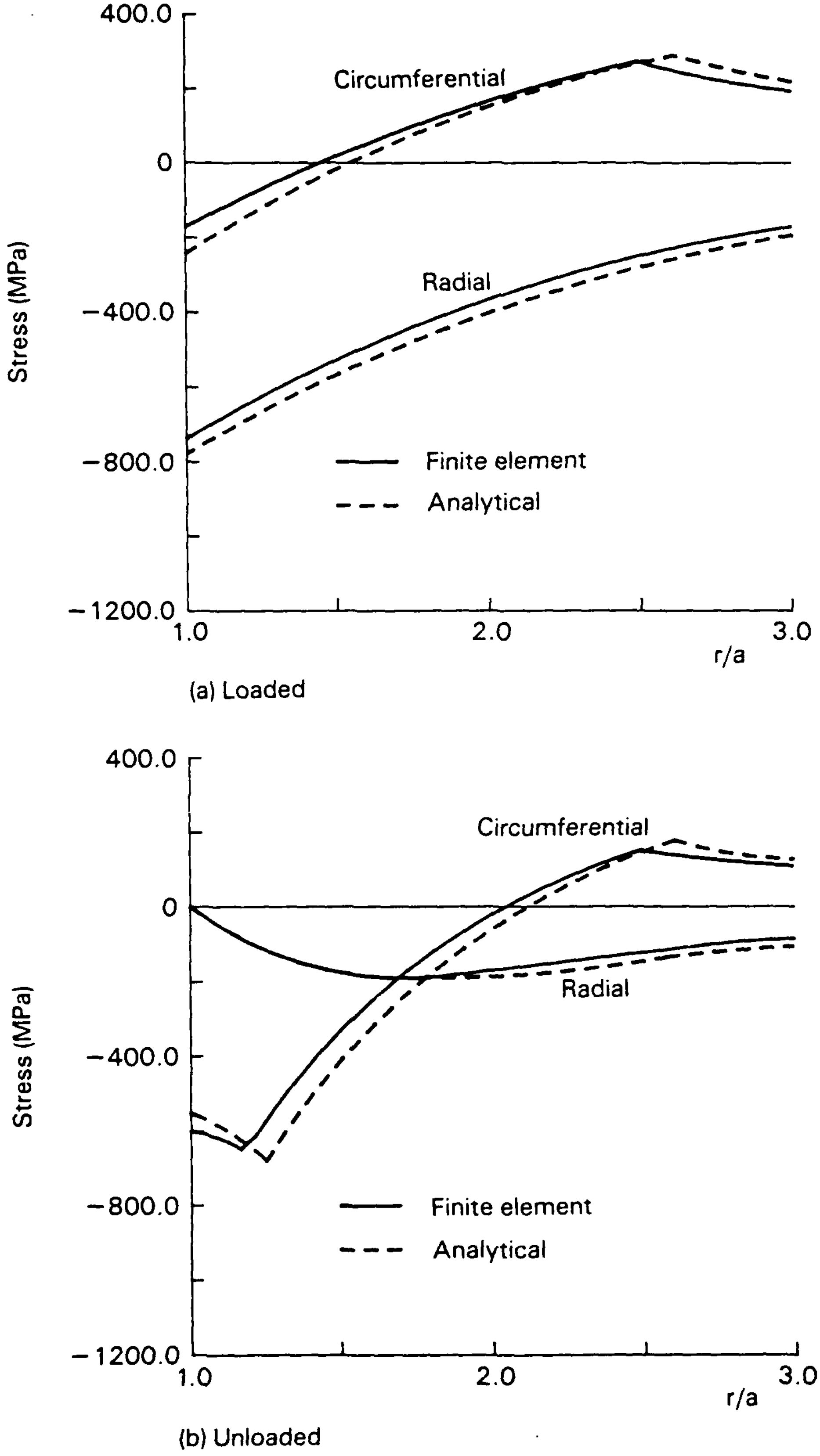
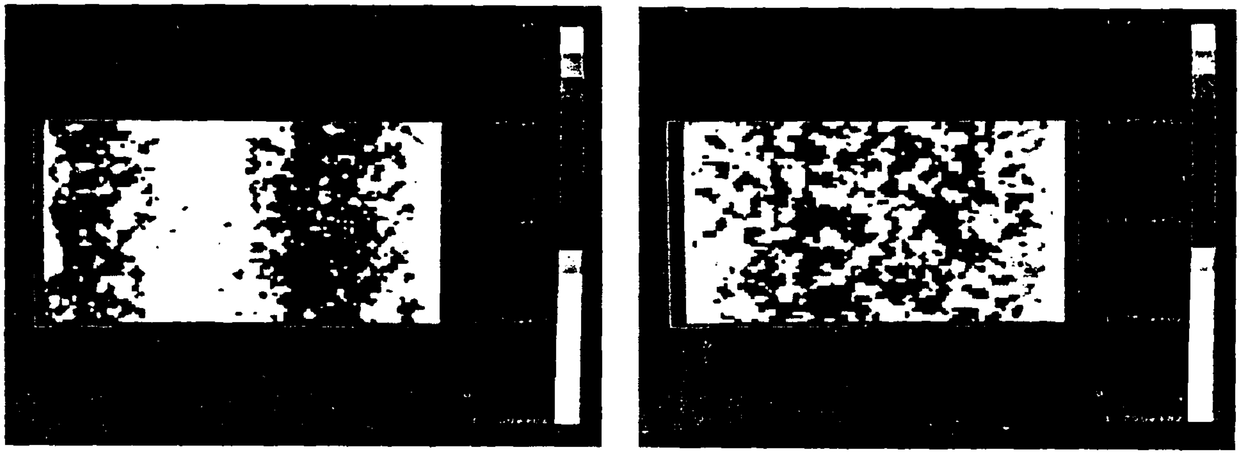


FIG. 17 COMPARISON OF IN-PLANE FINITE ELEMENT AND ANALYTICAL STRESSES



(b) (a)

FIG. 18 SPATE SCANS ACROSS BEAM (a) WITHOUT AND (b) WITH RESIDUAL STRESS

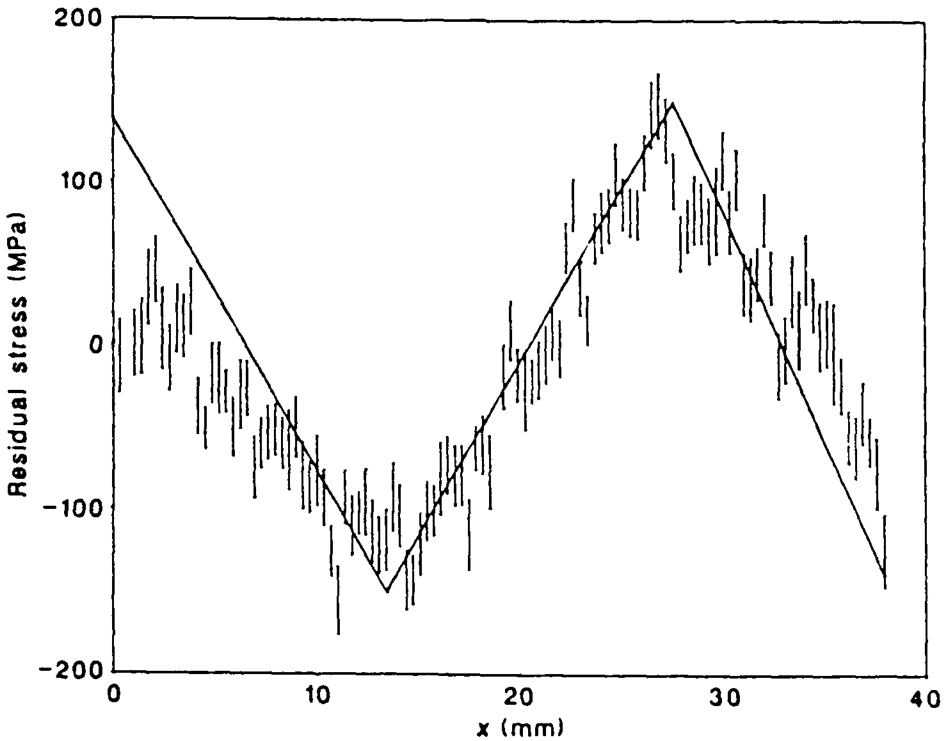


FIG. 19 AVERAGED SPATE AND STRAIN GAUGE STRESSES ACROSS BEAM CONTAINING RESIDUAL STRESSES

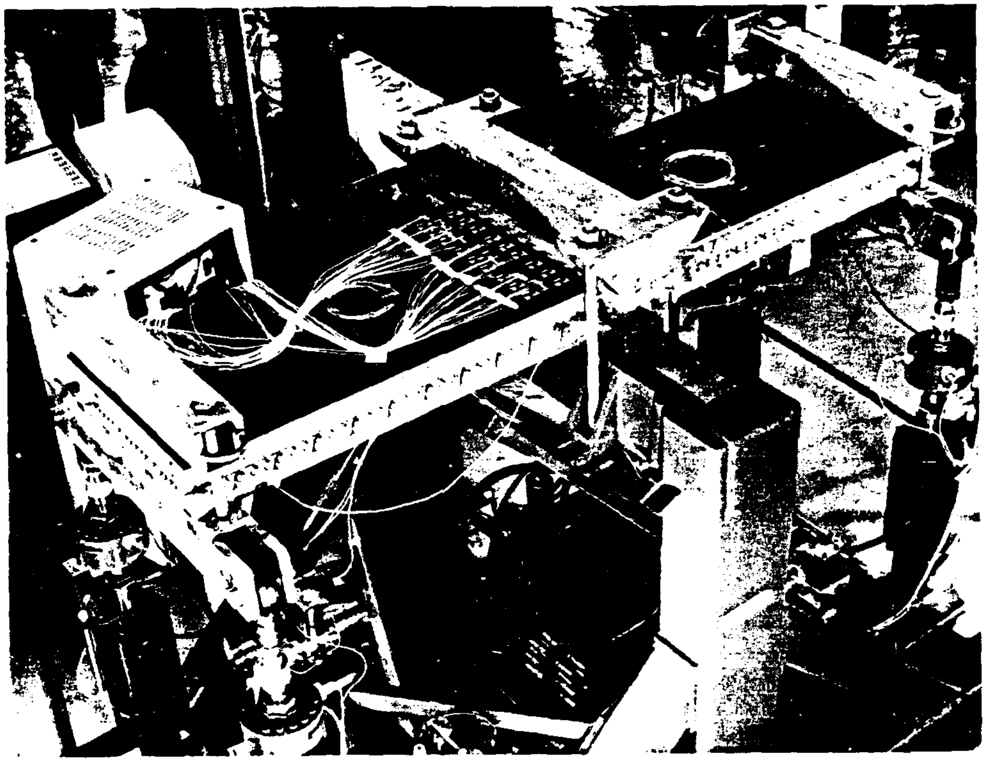
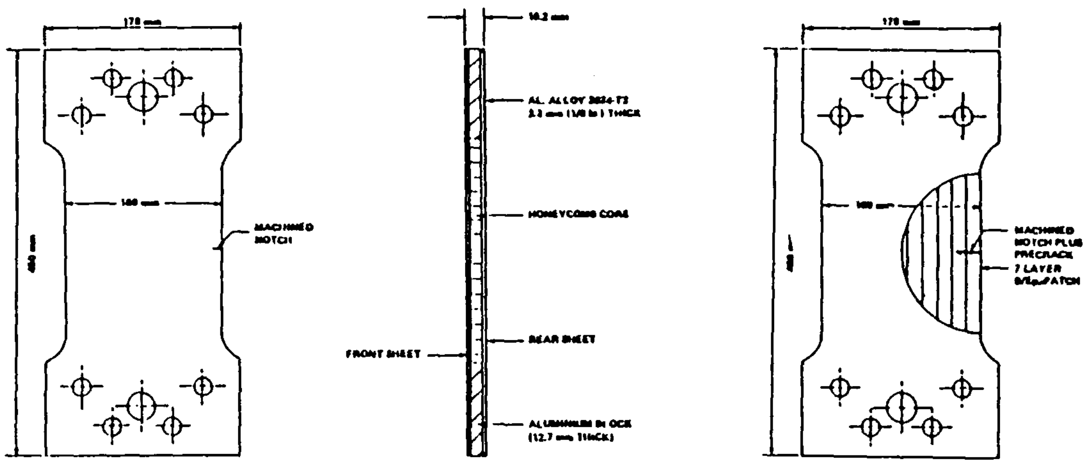


FIG. 20 COMBINED FATIGUE/ENVIRONMENTAL BOX BEAM TEST



(a) UNPATCHED

PATCHED

FIG. 21 UNPATCHED AND PATCHED HONEYCOMB TEST PANELS

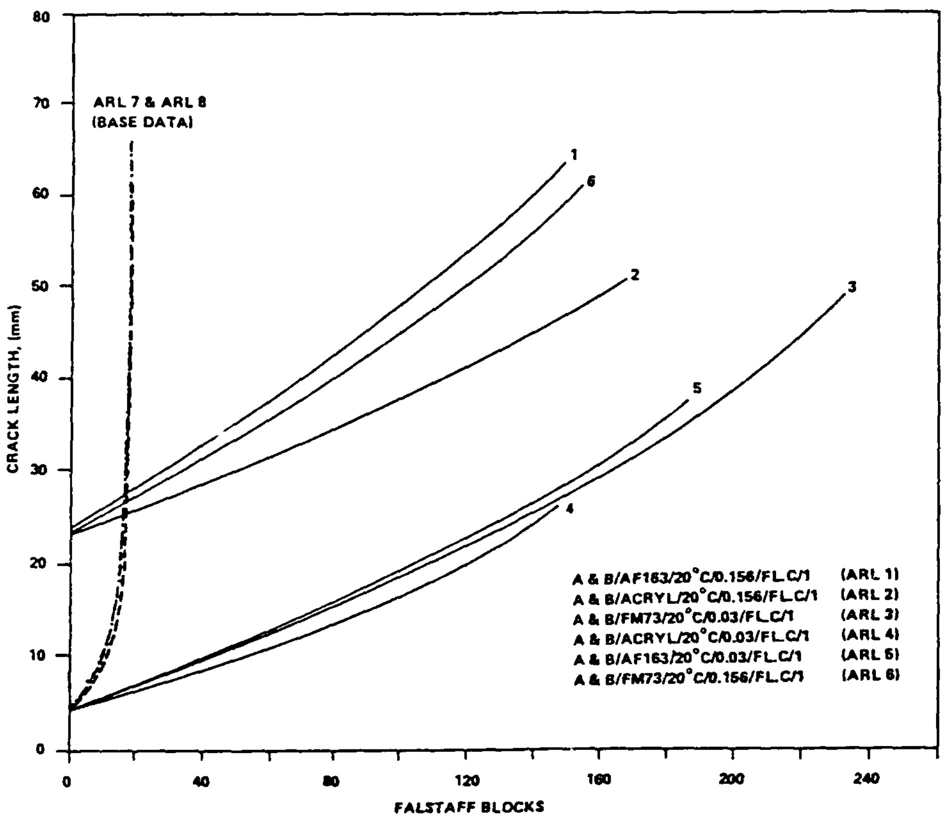


FIG. 22 CRACK GROWTH OF UNPATCHED AND PATCHED TEST PANELS

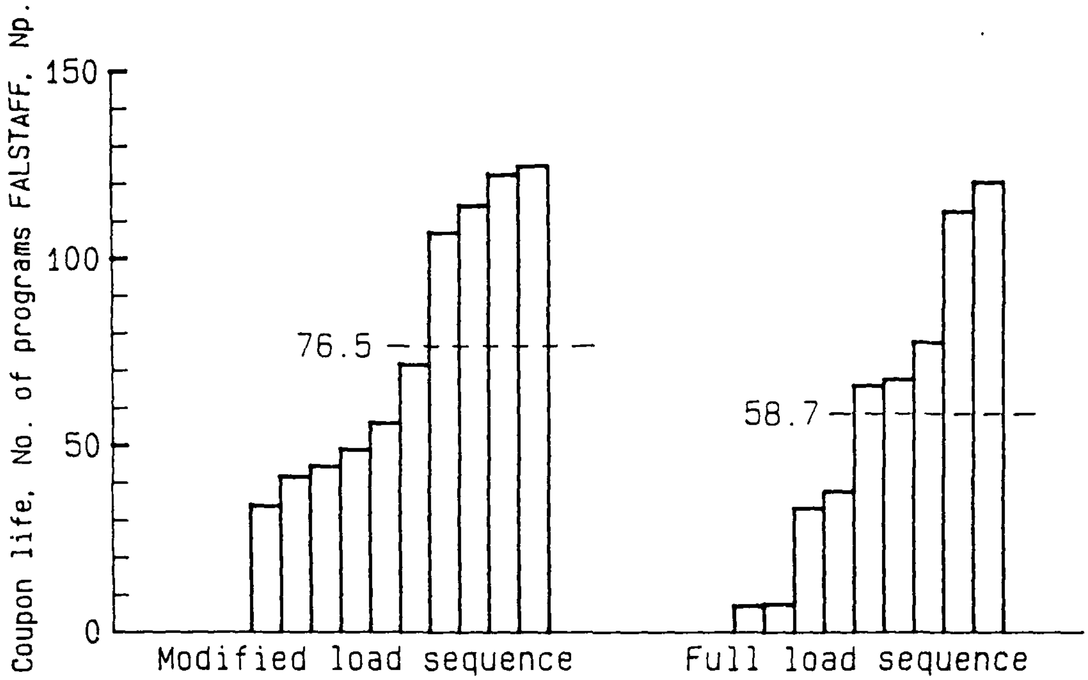


FIG. 23 COMPOSITE SPECIMEN FATIGUE RESULTS SHOWING INDIVIDUAL AND MEAN LIVES

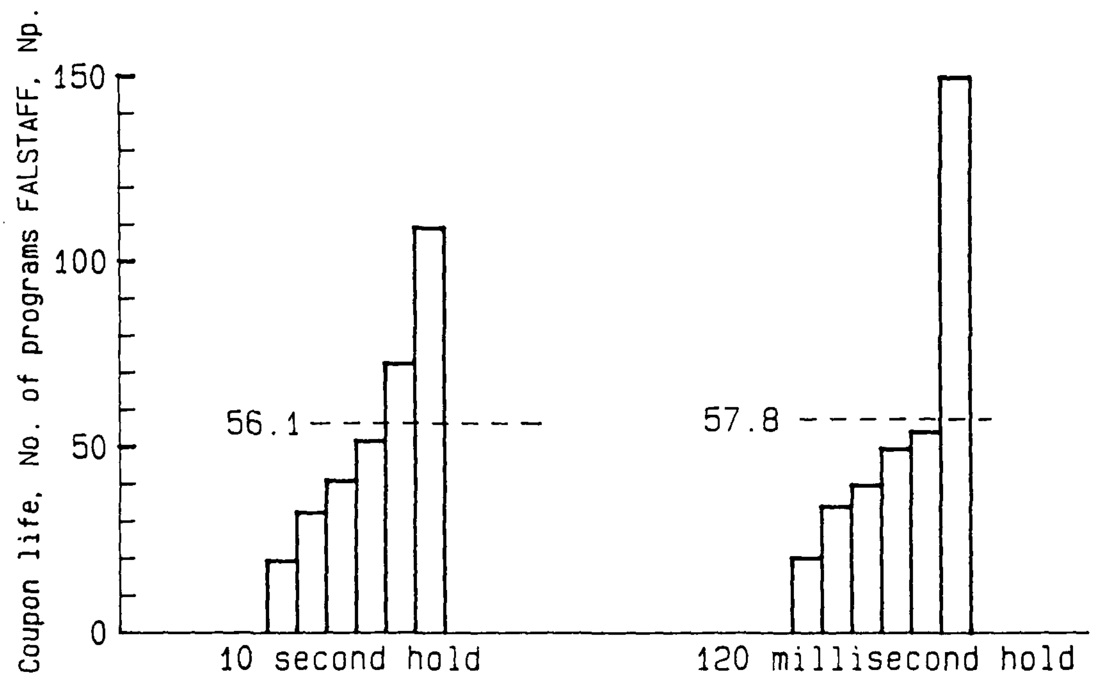


FIG. 24 EFFECT OF HOLD TIME ON LIFE OF IMPACT-DAMAGED COMPOSITE COUPONS UNDER FALSTAFF LOADING

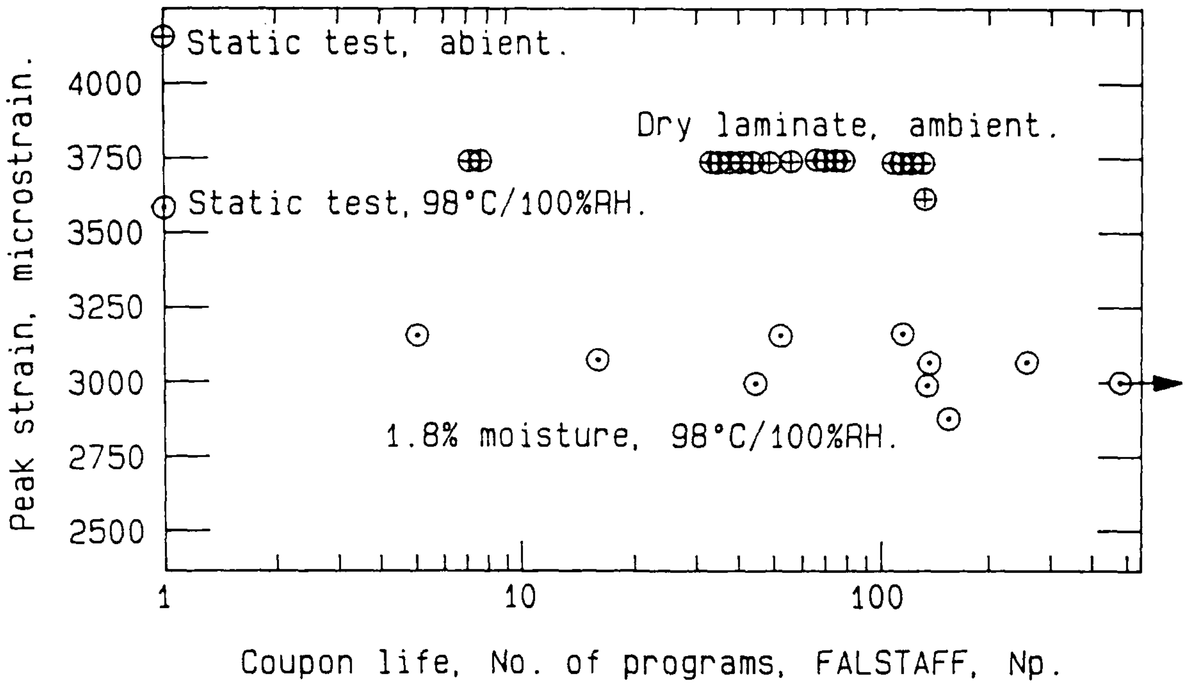


FIG. 25 EFFECT OF ENVIRONMENT ON FATIGUE LIFE OF 56-PLY XAS-914C LAMINATES

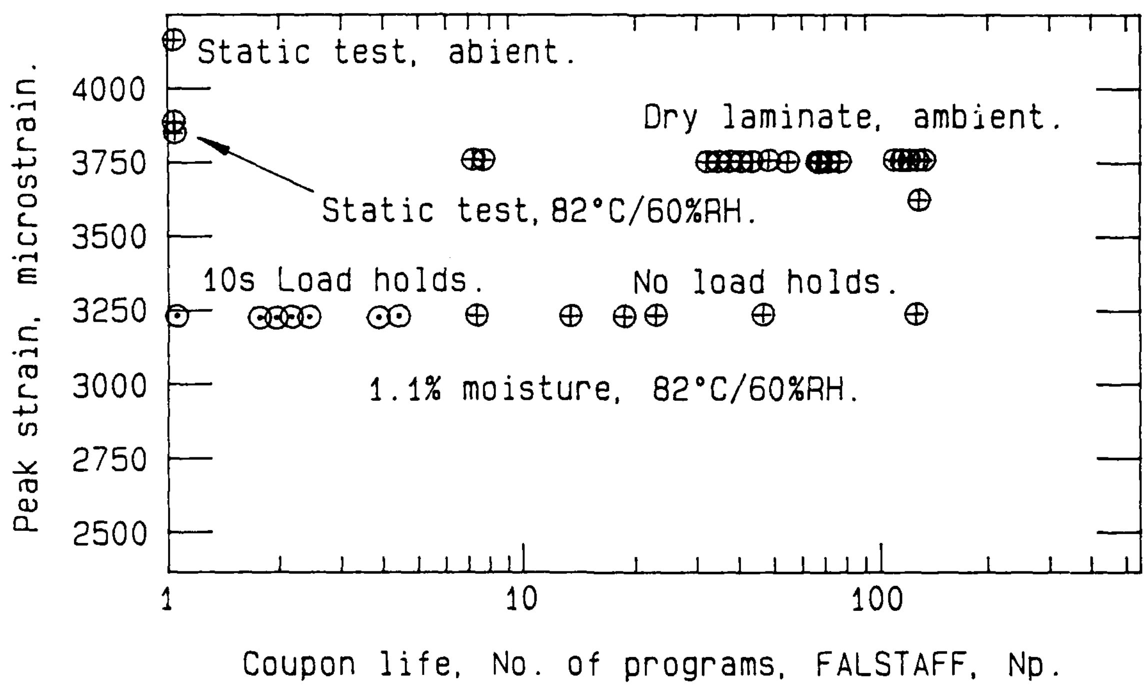


FIG. 26 EFFECT OF HOLD TIME ON FATIGUE LIFE OF 56-PLY XAS-914C LAMINATES

FIG. 27 COUNTERSUNK BOLT FASTENING OF COMPOSITE PANELS - BEFORE AND AFTER FATIGUE LOADING

THEORY

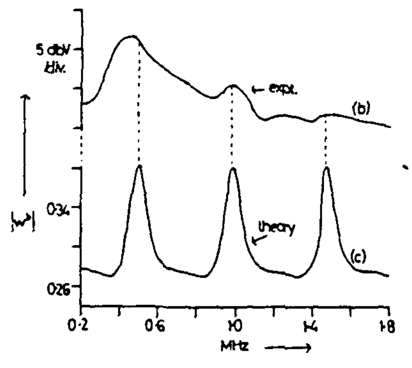
$$W \propto \exp\left(\frac{4\pi i f_R d}{V_R}\right)$$

f_R - Rayleigh wave frequency

V_R - Rayleigh wave velocity

CRACK LENGTH d = 3 mm

ULTRASONICS EXPERIMENT → 3·1 mm



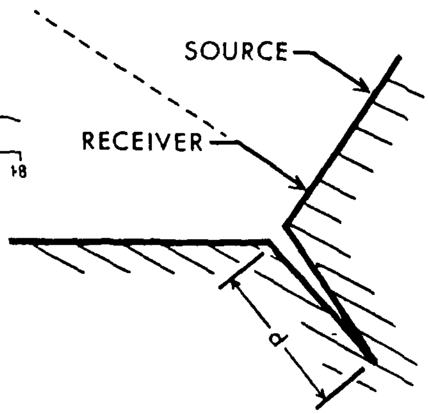


FIG. 28 DEPTH MEASUREMENT OF CORNER CRACKS USING ULTRASONIC RAYLEIGH WAVES

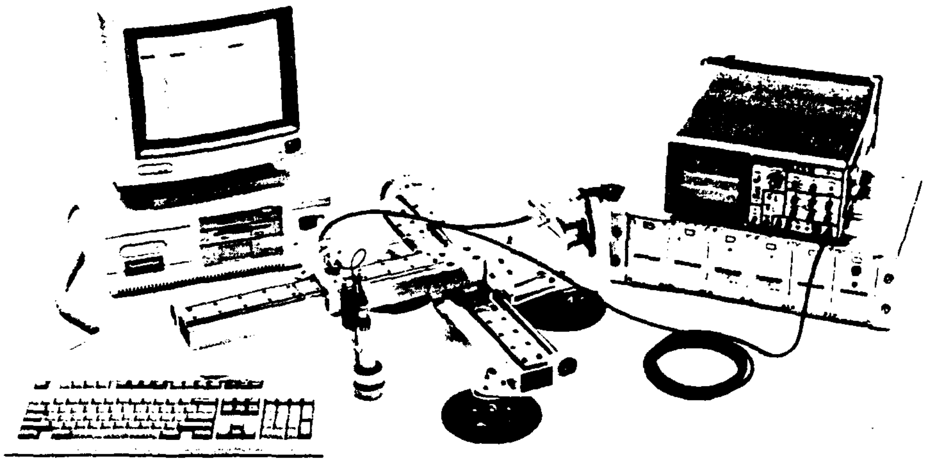


FIG. 29 PORTABLE C-SCAN DEVICE AND PERIPHERALS

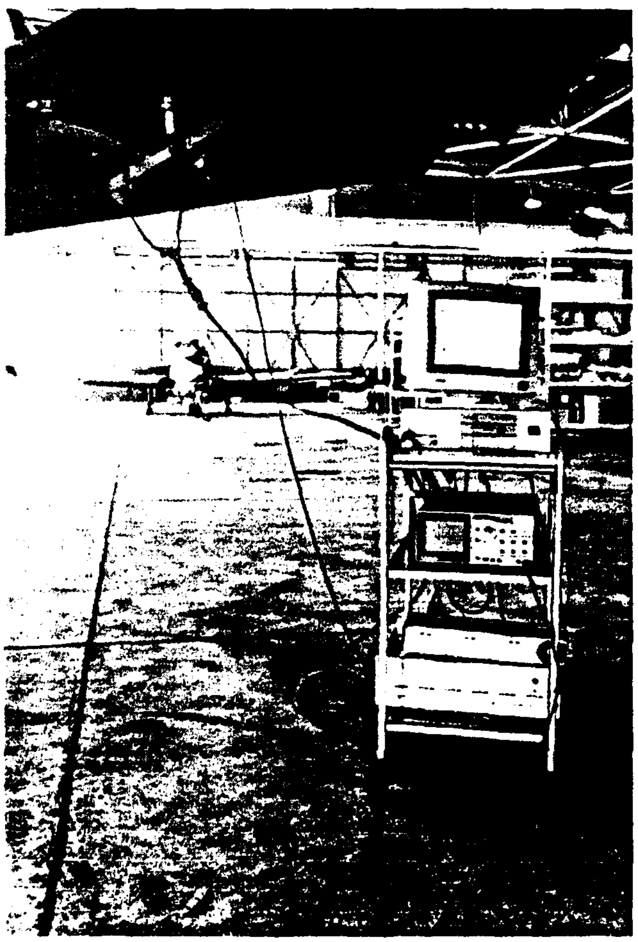
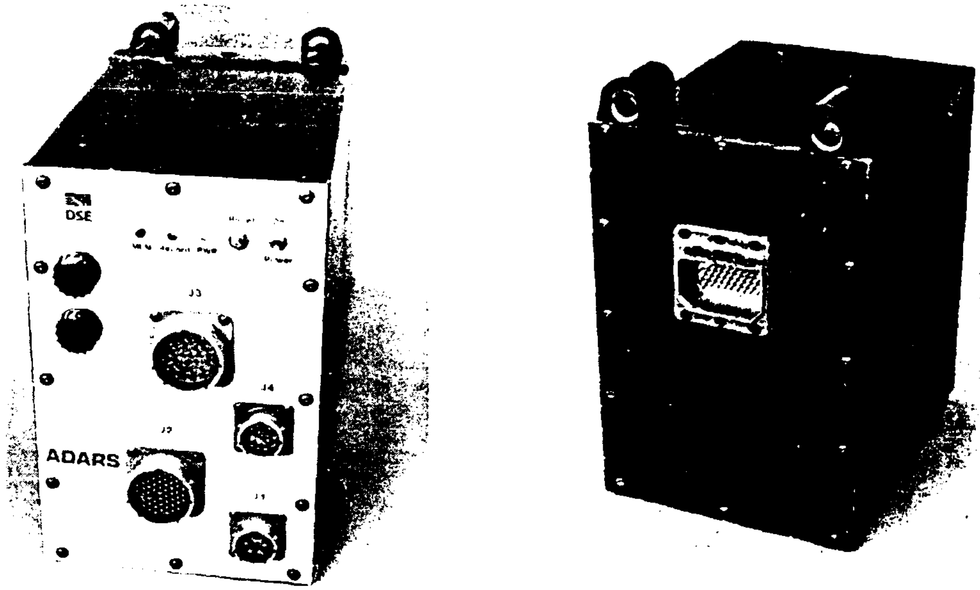
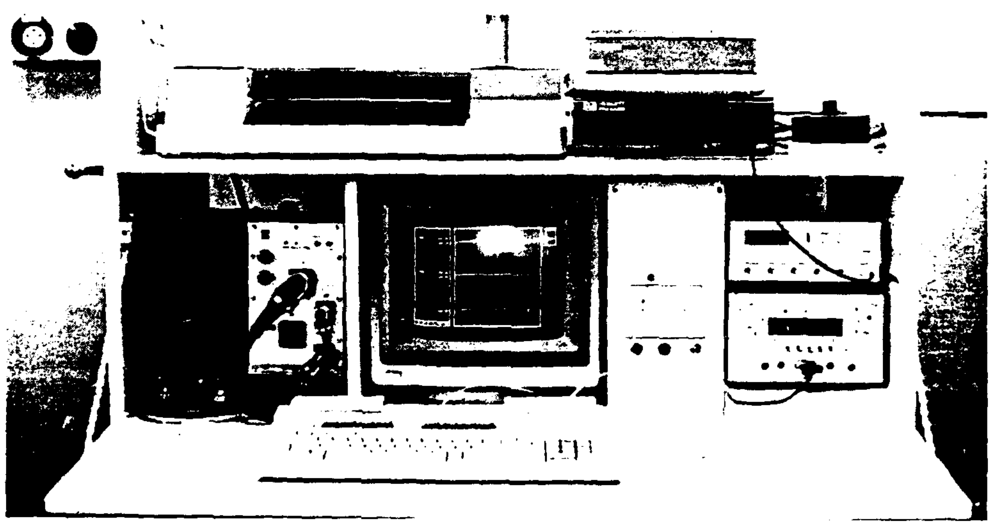


FIG. 30 IN-SITU EXAMINATION OF F/A-18 STABILATOR USING PORTABLE C-SCAN EQUIPMENT



(a) DATA ACQUISITION (LEFT) AND MEMORY MODULES



(b) GROUND SUPPORT UNIT

FIG. 31 AIRBORNE DATA ACQUISITION AND RECORDING SYSTEM (ADARS)

DISTRIBUTION

AUSTRALIA

Department of Defence

Defence Central

Chief Defence Scientist
FAS Science Corporate Management (shared copy)
FAS Science Policy (shared copy)
Director, Departmental Publications
Counsellor, Defence Science, London (Doc Data Sheet Only)
Counsellor, Defence Science, Washington (Doc Data Sheet Only)
S.A. to Thailand MRD (Doc Data Sheet Only)
S.A. to the DRC (Kuala Lumpur) (Doc Data Sheet Only)
OIC TRS, Defence Central Library
Document Exchange Centre, DISB (18 copies)
Joint Intelligence Organisation
Librarian H Block, Victoria Barracks, Melbourne
Director General - Army Development (NSO) (4 copies)

Aeronautical Research Laboratory

Director
Library
Divisional File - Aircraft Structures
Author: G.S. Jost
C.K. Rider
B.C. Hoskin
R. Jones
D.G. Ford
J.L. Kepert
J.M. Finney
J.G. Sparrow
A.K. Patterson
K.C. Watters
J.M. Grandage
A.D. Graham
G.W. Revill
D.J. Sherman
D.S. Saunders
R.G. Parker
P.W. Beaver
L.R. Gratzer
M. Heller
J.J. Paul
G.S. Jost (for ICAF Distribution - 200 copies)
C.K. Rider (for TTCP Distribution - 5 copies)
Chief - Aircraft Materials
A.A. Baker
N.E. Ryan
G. Clark
B.J. Wicks
D.R. Arnott

R.J. Chester
J.Q. Clayton
N.T. Goldsmith
T.G. Hill
Chief - Flight Mechanics and Propulsion Branch
K.F. Fraser
Chief - Aircraft Systems

<u>Materials Research Laboratory</u>
Director/Library

<u>Defence Science & Technology Organisation - Salisbury</u>
Library

<u>Navy Office</u>
Navy Scientific Adviser
Director of Naval Aircraft Engineering

<u>Army Office</u>
Scientific Adviser - Army
Engineering Development Establishment, Library
Royal Military College Library

<u>Air Force Office</u>
Air Force Scientific Adviser
Aircraft Research and Development Unit
Scientific Flight Group
Library
Engineering Division Library
Director General Aircraft Engineering - Air Force
HQ Support Command (SLENGO)
RAAF College, Point Cook

<u>Statutory and State Authorities and Industry</u>
Civil Aviation Authority
Library
C. Torkington
R.B. Douglas
A.J. Emmerson
Aero-Space Technologies Australia
Manager
Library
L.T. Tuller
Ansett Airlines of Australia
Library
J.H. Bibo
Australian Airlines, Library
Qantas Airways Limited
BHP, Melbourne Research Laboratories
Hawker de Havilland Aust Pty Ltd, Victoria
Library
P.J. Foden

University College of the Northern Territory
Library

NEW ZEALAND

Defence Scientific Establishment
Library
Mr H. Levinsohn
RNZAF, Vice Consul (Defence Liaison)
Transport Ministry, Airworthiness Branch, Library

Universities

Canterbury
Library

Auckland
Library
Dr G.D. Mallinson

Wellington
Library

SPARES (20 copies)
TOTAL (349 copies)

Hawker de Havilland Aust Pty Ltd, Bankstown
Library
I.D. McArthur
S. Dutton

Universities and Colleges

Adelaide
Barr Smith Library

Flinders
Library

LaTrobe
Library

Melbourne
Engineering Library
Dr J.F. Williams

Monash
Hargrave Library
Prof I.J. Polmear
Dr J.R. Griffiths
Dr Y.C. Lam

Newcastle
Library

New England
Library

Sydney
Engineering Library
Prof. G.P. Steven

NSW
Physical Sciences Library
Prof. R.A. Bryant, Mechanical Engineering
Library, Australian Defence Force Academy

Queensland
Library

Tasmania
Engineering Library

Western Australia
Library

RMIT
Library
Dr L.A. Wood

AL 149 DEPARTMENT OF DEFENCE

DOCUMENT CONTROL DATA

PAGE CLASSIFICATION	UNCLASSIFIED
PRIVACY MARKING	

1a. AR NUMBER	1b. ESTABLISHMENT NUMBER	2. DOCUMENT DATE	3. TASK NUMBER
AR-005-604	ARL-STRUC-TM-506	APRIL 1989	DEF 89/501

4. TITLE

A REVIEW OF AUSTRALIAN AND NEW ZEALAND INVESTIGATIONS ON AERONAUTICAL FATIGUE DURING THE PERIOD APRIL 1987 TO MARCH 1989

5. SECURITY CLASSIFICATION (PLACE APPROPRIATE CLASSIFICATION IN BOX(S) IE. SECRET (S), CONF.(C) RESTRICTED (R), UNCLASSIFIED (U)).

DOCUMENT	TITLE	ABSTRACT
U	U	U

6. NO. PAGES 45

7. NO. REFS. 60

8. AUTHOR(S)

G.S. JOST (EDITOR)

9. DOWNGRADING/DELIMITING INSTRUCTIONS

10. CORPORATE AUTHOR AND ADDRESS

AERONAUTICAL RESEARCH LABORATORY
P.O. BOX 4331, MELBOURNE VIC 3001

11. OFFICE/POSITION RESPONSIBLE FOR:

SPONSOR -
SECURITY -
DOWNGRADING -
APPROVAL CSTD

12. SECONDARY DISTRIBUTION (OF THIS DOCUMENT) Approved for Public Release.

OVERSEAS ENQUIRIES OUTSIDE STATED LIMITATIONS SHOULD BE REFERRED THROUGH ASDIS, DEFENCE INFORMATION SERVICES BRANCH, DEPARTMENT OF DEFENCE, CAMPBELL PARK, CANBERRA, ACT 2601

13a. THIS DOCUMENT MAY BE ANNOUNCED IN CATALOGUES AND AWARENESS SERVICES AVAILABLE TO....

No limitations.

13b. CITATION FOR OTHER PURPOSES (IE. CASUAL ANNOUNCEMENT) MAY BE [X] UNRESTRICTED OR [ ] AS FOR 13a.

14. DESCRIPTORS

Aircraft
Fatigue (materials)
Aircraft structures

15. DRDA SUBJECT CATEGORIES

0051C
0071L

16. ABSTRACT

This document was prepared for presentation to the 21st Conference of the International Committee on Aeronautical Fatigue scheduled to be held in Jerusalem, Israel, on June 19 and 20, 1989.

A review is given of the aircraft fatigue research and associated activities which form part of the programmes of the Aeronautical Research Laboratory, Universities, the Civil Aviation Authority, the Australian aircraft industry and the Defence Scientific

PAGE CLASSIFICATION
UNCLASSIFIED
PRIVACY MARKING

THIS PAGE IS TO BE USED TO RECORD INFORMATION WHICH IS REQUIRED BY THE ESTABLISHMENT FOR ITS OWN USE BUT WHICH WILL NOT BE ADDED TO THE DISTIS DATA UNLESS SPECIFICALLY REQUESTED.

16. ABSTRACT (CONT.)

Establishment, New Zealand. The major topics discussed include fatigue of both civil and military aircraft structures, fatigue damage detection, analysis and repair and fatigue life monitoring and assessment.

17. IMPRINT

AERONAUTICAL RESEARCH LABORATORY, MELBOURNE

18. DOCUMENT SERIES AND NUMBER	19. COST CODE	20. TYPE OF REPORT AND PERIOD COVERED
Aircraft Structures Technical Memorandum 506	24 1100	

21. COMPUTER PROGRAMS USED

22. ESTABLISHMENT FILE REF.(S)

M3/48

23. ADDITIONAL INFORMATION (AS REQUIRED)