REVIEW OF AERONAUTICAL FATIGUE INVESTIGATIONS IN THE UNITED KINGDOM DURING THE PERIOD MAY 2001 TO APRIL 2003

compiled by J E Moon and A B Mew

2.8 FRACTURE MECHANICS AND DAMAGE TOLERANCE

2.9 OTHER ASPECTS OF FATIGUE

2.1 INTRODUCTION

This review summarises aeronautical fatigue investigations carried out in the United Kingdom during the period May 2001 to April 2003. The format of the paper is similar to that of recent UK ICAF reviews; the topics covered include developments in fatigue design tools, fatigue loads measurement, fatigue of metallic structural features including repair, full scale fatigue testing, developments in fatigue monitoring, fatigue in composite materials and structures, fracture mechanics and damage tolerance. A list of references related to the various items is given at the end of the paper.

The authors gratefully acknowledge the contributions generously provided by colleagues in the aircraft and associated industries, universities and at QinetiQ. The names of the principal contributors, and their affiliations, are shown in brackets after the title of each item. In addition, contact details are provided in the list at the end of the report.

2.2 DEVELOPMENTS IN FATIGUE DESIGN TOOLS

2.2.1 Fatigue Crack Growth Predictions using the Boundary Elements analysis code BEASY (Sharon Mellings, John Baynham, Robert A Adey, Computational Mechanics BEASY, Tom Curtin, Computational Mechanics Inc.)

Continued development and improvement is progressing in the area of fatigue crack growth predictions made with the BEASY analysis code. The automatic crack growth feature that was introduced at the time of ICAF 2001 has been extended and improved. Additional features include stiffened panels analysis, crack growth in residual stress fields, model extraction from finite element models and improved post-processing features. Ongoing work is proceeding on increasing the accuracy and robustness of the calculations whilst reducing the computational effort required.

Crack Analysis in Stiffened Panels Structures with Combined Boundary and Finite Element Analysis: A project has been running involving the development of crack growth simulation in stiffened panel structures. This development has produced an analysis tool specially designed for the computation of fatigue life in airframe structures. In this project stiffening ribs and beams are modelled with beam elements and the panel itself is modelled using a boundary element analysis zone. The couplings between the constituent parts of the analysis model are made via elements representing mechanically fastened connectors, such as rivets. In the project these connections have been developed to allow full non-linear connections to be represented. This development also allows the analysis of mechanically fastened panel structures, such as overlapping panel structures and doubler panels to be modelled using similar techniques.

Automatic Fatigue Crack Growth in Residual Stress Fields using Boundary Elements Analysis: Residual stresses are often built into materials during the manufacturing process and may also be introduced due to surface treatment to reduce the risk of fatigue failure. This project has been run to allow these residual stresses to be applied to a crack growth model along with the external mechanical loading to predict the fatigue life of cracks introduced into the part. This allows the effect of different residual stresses to be investigated easily. The residual stresses field can be taken from existing finite element analysis results and applied directly to the crack itself thus simulating the impact of the residual stresses on the stress intensity factors and hence the fatigue life computed. As the crack grows during the analysis, the residual stress field is continually reapplied to introduce the residual stress effect on the new crack surfaces.

Automatic Crack Growth using Finite Element Models: Analysis of structural parts is often performed with finite element analysis methods. However these are not usually suitable for crack growth analysis calculations. A project has been running to allow the results from a finite element analysis model to be used to generate a smaller BEASY sub-model. This is a boundary element analysis model generated from the finite element model localised in the region where the crack is to be analysed and the results from the finite element analysis are used to generate boundary conditions on the model. Using this sub-model, a crack can be introduced and automatic crack growth analysis can be performed. This allows for full stress analysis to be performed and the detailed stress intensity factors to be computed on the part without the need for the user to generate a specialist BEASY model.

2.2.2 Development of a New Plug and Play Fatigue Solver for Fracture Mechanics (A Chilton, nCode)

There are many possible methods to account for crack growth - as well as modelling the relationship of crack growth with loading sequence and localised yielding of the material at the crack tip. Organisations typically combine a bespoke collection of methods for their standard analysis - and no one combination is the same. It is important that engineers can reproduce their historical analyses using modern analysis codes.

nCode Crack Growth (nCG) allows companies to do just that. It was designed as a ‘crack growth architecture’, and allows full and easy integration of legacy, current and future technologies (including Finite and Boundary element integrated calculations). The nCG user interface is common to all approaches, and so allows easy transfer of personnel from project to project.

Any bespoke or commercial materials database can be seamlessly integrated - while the nCode materials database allows for any format material data to be stored and referenced when required. Advanced investigation tools such as back calculation, multiple value analysis for inputs such as stress scale factors or initial crack size provide the engineer with more confidence in the design they are analysing.

nCG works with the advanced signal processing capability provided by nCode’s toolsets - and when linked with the SN and EN products provides a signal processing, fatigue and damage tolerance capability.

Developed in partnership with industrial leaders such as BAE SYSTEMS, QinetiQ, Westland Helicopters and the CAA, nCG provides a flexible foundation for researchers and engineering teams, while at the same time providing a secure and auditable environment for damage tolerance analysis within projects.

Reduced technology beta versions are currently under investigation by the developing partners, with a full product release due in the Autumn of 2003.

2.2.3 Design Data (A C Quilter, ESDU International plc, London)

During the period under review, ESDU International has extended the Fatigue - Fracture Mechanics Series with the issue of the following Data Items.

Data Item No. 97024 Derivation of endurance curves from fatigue test data, including run-outs: Work on a Data Item, No. 97024, on the derivation of a mean *S–N* curve from fatigue test data is very nearly complete. The Data Item will be accompanied by a Fortran computer program, which provides a ‘windows-style’ user interface and graphical as well as numerical output. The method used by the program is applicable to data for both metallic and non-metallic materials as well as to data from tests on coupons or specimens representing structural features. The method deals with the low-cycle, high-cycle and endurance limit regimes. One of the main features of the method used by the program is that it takes account of censored data such as unfailed specimens (run-outs) in a mathematically rigorous manner. The program also allows the user to make use of any knowledge of the shape of the curve by allowing certain parameters to be predefined, thus constraining the shape of the curve. The program output includes the equation of the derived curve, the coefficient of variation of the data, a measure of the goodness-of-fit of the curve to the data, and a plot of the curve and the data.

Data Item No. 99002 Computerised crack resistance curves: Work on this Data Item and the accompanying computer program is nearing completion. The Data Item presents a program based on the method described in Data Item No. 85031, “Crack resistance curves”. The program determines either the fatigue crack length that will cause fast fracture of a component under a given applied stress or *vice versa*. The program runs as a module within the Microsoft Excel spreadsheet program and gives results in the form of both numerical and graphical data. All of the resistance curve data presented in Item No. 85031 are available within the program and the facility for users to add their own data is also provided.

ESDU 00932, The Metallic Materials Data Handbook: This Handbook, became available in electronic form in 2001. It is available on CD from which it may be installed locally on a PC or made available over a network. In addition, it is available on the internet via the ESDU web site and it may also be incorporated in a user’s intranet.

The package includes a search facility enabling the user to locate materials meeting certain combinations of user-specified criteria relating to anything from the material form through composition and physical and mechanical properties to suitability to being welded by a particular process. All the graphical data in the Handbook are included as fully interactive figures. The facility to save all or part of the data, both numerical and graphical, relating to a particular material specification is also included.

Improvements effected in 2002 included the highlighting of any revisions made to individual Data Sheets and documentation in the Handbook. Other features currently being considered are the addition of a facility whereby users may select the system of units in which the data are displayed and a feature allowing users to search for materials using US, French, German or commercial material specifications.

Rooke and Cartwright's Compendium of Stress Intensity Factors: Work is currently in progress on the incorporation of Rooke and Cartwright's compendium of stress intensity factors into an ESDU Data Item, in which form it will be available both on CD and via the internet. Completion of the project is scheduled for the end of 2003.

2.2.4 Rapid Assessment of Stress Concentration Factors (J Trevelyan, University of Durham and S Spence BAE SYSTEMS, Warton)

The University of Durham, School of Engineering, is working in collaboration with BAE SYSTEMS to investigate rapid computational methods of assessing stress concentration factors. The work has resulted in a software system *Concept Analyst*, which has been implemented in BAE SYSTEMS. The software is based on the boundary element method (BEM) for ease of re-meshing on a change in geometry. The essential feature of the system is that it contains a high degree of automation of stress analysis, so that a geometry may be sketched interactively, loads applied to line segments, and a rapid analysis proceed. Results compare favourably against Peterson and analytical solutions.

A second essential feature of *Concept Analyst* is its ability to undertake a rapid re-analysis of a component on a design change, e.g. a change in fillet radius, or the movement of a hole. In this event, the BEM model is updated rapidly, taking advantage of the localised meshing in the method, the associated parts of the system matrix updated and a new solution obtained.

Results are typically obtained within a few seconds of *starting* to build the model, and with the re-analysis option several design options can be evaluated in an extremely short space of time, giving accurate K_t values. This technology has the promise to replace handbook estimation of this information. An example of a Concept analysis screen shot is shown in Figure 1

2.2.5 SCONES - Determination of Stress Concentration Factors (S Spence, BAE SYSTEMS Warton)

The SCONES (Stress CONcentration Evaluation System) computer software has been developed jointly by the University of Hull and BAE SYSTEMS. The programme of work has been in progress for a number of years now to produce a system for the efficient and accurate determination of stress concentration factors. Various combinations of both geometry and loading are considered, providing a common tool for Structures engineers across all BAE SYSTEMS aircraft projects. This work is part of a larger activity in the area of rapid stress concentration determination that also includes work at Durham University, the details of which are reported separately.

The application provides a K_t value from user-defined geometric parameters and also provides an indication of the sensitivity of K_t to changes in feature geometry via a reactive graphical display. In the first instance, this provided a quick and reproducible means for determining the stress concentration value for features covered in various published sources such as Peterson. However, the focus of the work now is on additional, unpublished features such as Kt determination for use with the strain-life approach to fatigue life estimation and interacting features. The data have been generated by Finite Element Analysis (FEA) in conjunction with photo-elastic stress analysis methods (where appropriate).

Some examples of common general features covered within SCONES are :-

- HOLES in Plates (Circular and Elliptical)
- NOTCHES in Plates (semicircular, U-shaped and rectangular)
- FILLETS in Plates (single and double shoulders)
- COUNTERSINKS (constrained and unconstrained)
- LUGS (round ended - parallel sided)
- INTERACTING FEATURES (e.g. satellite holes around a central hole)

Many of these features include different loading modes such as tension and bending. Figure 2 illustrates a number of windows from the SCONES application for different features.

The key focus of the last two years has been the incorporation of the strain life factors and interacting features. The strain life factor work allows the influence of other parameters such as fastener load and fit, thickness and surface finish to be taken into account via the K_t. The recent work has embodied existing data into the toolset but future work will involve further studies of the influence of some of these parameters.

The work on the interacting features involved the study of a number of parameters to assess the validity of some old "rules of thumb" with a view to developing a refined approach. In brief, whilst the rules of thumb were adequate for some situations, the approach was not infallible and there was no simple trend revealed from the examples studied. This lead to the interacting feature data that were generated being used directly to enable evaluation of $K_{t's}$ for those feature types within the application. Future work is planned in this area to extend the investigation of interacting features.

2.3 FATIGUE LOADS MEASUREMENT

2.3.1 Tornado Undercarriage Clearance - Operational Loads Measurement (OLM) Programme (T Siddall, Messier-Dowty and R T Jones, QinetiQ, Farnborough)

The number of original cleared landings for Tornado aircraft was based upon the results from Fatigue Tests of both the Main and Nose Landing Gear, and the Major Airframe Fatigue Tests on both IDS and ADV variants. All of these tests used the design spectrum developed in the late 1970s and early 1980s, and since then the operational aircraft have not only grown in mass but also their use has changed.

To quantify the effect of these changes an OLM programme is fundamental. This can range, for example, from a very simple counter on the Tornado Main Undercarriage Retraction Jack (where the jack pressure and hence the loading is known) to a comprehensive strain gauge and transducer package covering the whole undercarriage and its associated back-up structure.

Both the simple counter and the comprehensive undercarriage OLMs have been carried out on Tornado [1].

The simple counter has revealed that the Main Landing Gear System, of which the Retraction Jack Fittings are critical items, is experiencing many more cycles than originally envisaged. This is primarily due to more "in-air" and ground cycles. Many additional cycles were accumulated due to functional checks. Also, cycling the undercarriage while the aircraft was on jacks provided a great spectator attraction at Open Days etc. This action alone has now been stopped but was responsible for hundreds of additional cycles that were never budgeted for!

Two full undercarriage OLMs were carried out, one on the ADV variant in 1996 and one on the IDS, starting in 1997 and finishing in 1999. The IDS OLM was performed on both a German and a British Tornado aircraft. The purpose of selecting two aircraft was to investigate the effect of different operational usages and aircraft configurations. The German aircraft was operated at light mass in a training role, whereas the British aircraft was heavier and in a typical UK squadron role.

Both the Main and Nose Undercarriage Legs were fully instrumented in order to measure the axle loading in three orthogonal directions, the gyroscopic moments and the travel of the shock absorbers. In addition, there were a few strain gauges attached to the associated back-up structure of the Main Landing Gear.

An extensive collection of loading histories was measured which was followed by a detailed analysis. This has shown the need to conduct re-qualification by fatigue testing of both the aircraft back-up structure and the Undercarriage Legs. Subsequently, the need for testing the main and the nose undercarriage leg assemblies has reduced due mainly to revisions in the RAF landing gear requirements and the cost of procuring new parts versus testing. The Tornado IDS Nose Landing Gear assembly and associated Back-up structure is currently under test and it is hoped a revision of its clearance will result. The Main Undercarriage back-up structure is currently cleared by inspection.

2.3.2 Tucano Operational Loads Measurement Programme (L Murray, Bombardier Aerospace Shorts, Belfast)

An Operational Load Measurement (OLM) programme was undertaken on the Shorts Tucano T Mk 1, involving data collection from three instrumented aircraft over a period of 17 months.

Data was recorded from 22 aircraft parameters and 34 strain gauge bridges located on the wings, tailplane, fin, fuselage and landing gear.

QinetiQ (Farnborough) undertook the data analysis to Shorts defined requirements. Approximately 900 flight sorties have been analysed representing more than 3 aircraft years flying.

The aims of an OLM programme include:-

- Validation of pre-production Airframe Fatigue Test based life clearances
- Highlighting operational roles that cause high fatigue damage rates

- Validation of the Fleet wide Structural Monitoring System
- Provision of data to construct a test spectrum for the Production Airframe Fatigue Test

Before entry to service a comprehensive Load Calibration exercise was completed on each aircraft, to establish relationships between strain gauge output and structural loads. This calibrated data will be used to construct the fatigue test spectrum.

The life validation process was undertaken in accordance with DEF-STAN 970 Safe Life Approach, resulting in fleet management recommendations to the RAF Support Authority.

2.3.3 FiAF Mk51/51A Hawk Operational Loads Measurement Programme (M Gelder, BAE SYSTEMS, Brough)

In 1998 the Finnish Air Force (FiAF) contracted BAE SYSTEMS to undertake an Operational Loads Measurement (OLM) programme with two Mk51 Hawk aircraft. The programme was structured in two phases. Phase 1 scoped the programme, designed the OLM system and procured the Flight Test Instrumentation (FTI) equipment. Phase 2 covered the procurement and manufacture of modification kits; the installation, calibration and commissioning of the system; the development of analysis software; the provision of OLM equipment and data analysis training courses; the servicing of the OLM system; and the processing, analysis and reporting of flight data.

The OLM system, which was designed by BAE SYSTEMS, was largely based on the UK Royal Air Force (RAF) TMk1/1A OLM programme. Patria Finavitec Ltd (PFA) were contracted to prepare the two aircraft for receipt of the modification and were also responsible for installing the FTI equipment and preparing the aircraft for flight. The strain gauge installation, calibration and system commissioning was undertaken by VTT Manufacturing Technology (VTT). The processing and analysis of the flight data was reported by VTT with conclusions and recommendations based on these results reported by BAE SYSTEMS.

There were 52 strain gauges common to both aircraft and the flight programme comprised 200 sorties. The principal reasons for FiAF undertaking the OLM programme were to determine the structural component specific fatigue damage rates with respect to FiAF usage; to support the understanding of structural failures and aid the development of repair solutions; and to generate a database of appropriate information which would support a possible life extension programme. Since the completion of this initial programme in 2002 the FiAF have continued to collect flight data from the two aircraft.

2.3.4 RAAF Mk127 Hawk Flight Data Collection Programme (Mike Gelder, BAE SYSTEMS, Brough)

A Full Scale Fatigue Test (FSFT) of the Royal Australian Air Force (RAAF) Mk127 Hawk airframe is being designed and testing is scheduled to start in late 2004. The test specimen's load spectrum is being developed from a number of sources including the current aircraft in-service usage. The in-service spectrum is derived from the Fatigue Monitoring System (FMS) data which records either 7 or 21 strain gauge channels (Usage or OLM standard), accelerometers, and aircraft status and control surface status parameters. All Mk127 are fitted with an FMS.

Flight data has been collected since aircraft delivery to the RAAF in 2000 and will continue until the end of 2003 for the purposes of FSFT load spectrum generation.

2.3.5 C23B/B+ Aircraft Clearance for Operation on Gravel Runways (L Murray, Bombardier Aerospace Shorts, Belfast)

The task specified by the customer was to conduct a fatigue analysis of the airframe and critical components to determine the effect of operation from gravel runways. The analysis was to be supported with data secured through flight test (see Figure 3).

The test aircraft was fitted with a total of eighteen strain gauges and eight accelerometers at the pre determined components considered to be influenced by the ground loading part of the fatigue spectrum.

A total of four sorties were specified in the test plan with five runs to be performed within each sortie. Sortie 1 comprised of landing and taxy on a paved runway whilst sortie 2 was a repeat on the gravel runway. Similarly, sortie 3 included taxy and take off on a paved runway and sortie 4 was the comparable test on gravel.

Analysis of the data has shown that there is no reduction in the specified aircraft fatigue lives for operation on gravel strips.

2.4 FATIGUE OF METALLIC STRUCTURAL FEATURES, INCLUDING REPAIR

2.4.1 Measures to recover the effect of short edge distances in Hawk wing structure (J A Anderson, BAE SYSTEMS, Brough)

During manufacture of a Hawk wing, short edge distances were discovered; investigations indicated that several other wings had also been affected to a greater or lesser degree. The short edge distances (dimensions A and C in Figure 5) were restricted to two holes along the span of the front spar lower skin flange (edge of flange depicted as dotted line in Figure 5) in the region of a wing link pick up fitting, see Figure 4, a 3/8th and a 5/16th inch diameter hole for interference fit Hi-tigue fasteners. The worst condition revealed an interference fit Hi-tigue fastener of 3/8th diameter, having an edge distance of 7.05mm, giving an edge distance (e) to diameter (d) ratio of 0.88, compared to the nominal e/d = 1.75. An investigation was commenced into defining some kind of recovery action that could be set in place on the affected wings, which would recover the full specification life of the wings.

Several proposals were examined, including external patches, scalloping out the spar in the region of the short e/d ratio fasteners. These were ruled out for reasons outside the scope of this summary. Other investigations included determination of the benefits of cold working of the holes, polishing the hole bores, fitting interference fit fasteners into cold worked and non cold worked holes, and clamping effects.

A series of element tests were done, to support the investigations, with the specimen designed to replicate the through loading and load transfer that exists at the affected fasteners; Figure 6 shows a representation of the specimen. The results of the testing demonstrated the individual benefits of the cold working, clamping, polishing, and interference fits, and the compound of benefits that a combination of the aforementioned give, even at the low edge distances that were tested. The final solution for the wings with e/d greater than 0.88 was to fit interference fit fasteners and ensure that the torque tightening of the Hi-tigues was in line with the drawing requirements; for the aircraft with the e/d =0.88, the solution was a combined approach of cold working, fitting of an interference fit fastener and ensuring that the fastener was torque tightened to the drawing value. With these measures in place, the affected wings could be cleared to the relevant specification life.

2.4.2 Life enhancement of fatigue-aged fastener holes using cold expansion process (1999-2002) (X Zhang, J Gaerke and Z Wang, Aerospace Engineering Group, School of Engineering, Cranfield University)

In this work the benefits of cold expanding fastener holes of a low-load transfer joint specimen made of 2024-T351 at various stages of the fatigue life are examined. The specimens were pre-cycled to 25, 50, and 75% of the baseline fatigue life of a non-expanded specimen and then cold expanded prior to cycling to final failure. The experimental test was designed to provide a close comparison with standard maintenance practices for aircraft structures and used the FALSTAFF loading spectrum. Significant life improvements were obtained through cold expansion applied at all percentages of fatigue life tested in this work with the optimum stage being around 25% of the baseline life (see Figure 7). The major part of life extension was obtained through slower crack growth in the small crack stage. Additionally, tests on open-hole specimens of the same material and thickness were carried out with attempts to predict the crack growth lives of these specimens. The life improvement factors for the open-hole and joint specimens were comparable provided that the degree of cold expansion is the same. Crack growth life of the open-hole specimen was predicted by employing an analytical residual stress model and the *AFGROW* computer code. The prediction results showed good agreement with the experimental results.

The work was first published in 2000 [2]; significant additional work on open-hole specimens and FEA, and theoretical prediction was done in 2001-02, leading to the publication of two papers in a recent fatigue conference in 2002 [3, 4].

2.4.3 Improve and Assess Repair Capability of Aircraft Structures (IARCAS) (G E Shepherd, G R Brown, J Bowen, J Z Zhang and V Zitounis, Airbus UK, Bristol; P M Powell, QinetiQ, Farnborough)

The aim of the IARCAS research project, which is part of the European Union ‘Fifth Framework’ Programme and involves industrial, research and academic organisations from across Europe, is to improve and develop current repair techniques, and to extend allowable damages for primary metallic structure, thus reducing the burden on airlines regarding in-service damage. The successful completion of the project will advance repair technologies, bring new approaches closer to the market, provide confidence in the repairability of new materials or structure manufactured through new techniques, and demonstrate the efficiency of repair assessments developed within the European aerospace industry.

The objectives of Airbus UK within the IARCAS project consist of the experimental and analytical investigation of current and future repair procedures for Airbus wingbox structure. Specifically, these investigations have addressed the refurbishment of cold expanded holes, the repairability of welded aircraft structures (*i.e.* an assessment of the effect of placing fastener holes in or near the weld line, representative of the application of a conventional repair to welded structure), and an understanding of the effects of wing skin blends and dents on the fatigue and damage tolerance properties. Consideration has also been given to the development of a small portable friction stir welding tool for repairing cracks in wing skin structure, and the assessment of the fatigue and damage tolerance of cracked material repaired using such a tool.

In addition, Airbus UK are working with QinetiQ (see Section 2.4.4), who will carry out research to assess the effect of cold expansion/interference fit on small cracks at fastener holes, the use of enhanced stop hole drilling techniques and the use of bonded composite patches to repair friction stir welded structure.

2.4.4 Improve and assess repair capability of aircraft structures (P M Powell, K Brown, A Young and K W Man, QinetiQ)

QinetiQ is participating in the European Framework 5 project "Improve and Assess Repair Capability of Aircraft Structures (IARCAS)", which commenced in July 2001. This four year project entails collaboration with thirteen partners in the European aerospace industry, including Airbus UK (see also Section 2.4.3 for a report of Airbus' activities in this programme) and EADS. QinetiQ is investigating three areas: (a) the use of cold expansion/interference fit in techniques for repair of cracks at fastener holes, (b) the effectiveness of enhanced stop drill techniques, including cold expansion and interference fit and (c) the development of bonded composite patch repair technqiues for friction stir welded (FSW) structures.

During the first year of the project, the detailed test programme and material requirements were defined, in consultation with Airbus UK. Test pieces have been manufactured and the test programme is about to commence. For the repair of cracks at fastener holes, constant load amplitude testing will be carried out on symmetrical 2024-T351 aluminium alloy high load transfer joint specimens to investigate the influence of residual fatigue cracking on the effectiveness of cold expansion/interference fit repair techniques. For the investigation of stop-drill repair techniques, centre cracked 2024-T351 panels will be stop drill repaired in accordance with procedures described in the Airbus Structural Repair Manual to investigate the influence of cold expansion and interference fit techniques, under constant amplitude and lower wing spectrum loading. For the patch repair of FSW structures, butt welded 2024-T351 panels have been manufactured by Airbus UK and these will be pre-cracked and repaired using uni-directional carbon fibre patches. They will then be fatigue tested under constant amplitude (R= 0.1) loading to assess the effectiveness of the repair scheme and to validate theoretical predictions.

In addition to these experimental studies, modelling of crack growth for patch repaired friction stir welds and enhanced stop-drill holes is being carried out. A 3D boundary element/finite element model is being developed to account for residual stresses resulting from welding and various crack configurations with respect to the weld. A 2D boundary element model is being developed to take account of contact effects in the case of stop drilling plus the introduction of an interference fit fastener.

2.4.5 Investigation of damage tolerance behaviour of aluminium alloys (IDA) (V Strof and R A Collins, Airbus UK, Bristol; P M Powell, QinetiQ, Farnborough)

Damage tolerance is a design driver for the bottom wing covers and for areas of the top wing covers. The primary objective of the 'IDA' project, which is part of European Union 'Fifth Framework' Programme and involves industrial, research and academic organisations from across Europe, is to improve the basic understanding of the microstructural features influencing the crack growth and residual strength behaviour for established and recently-developed aerospace aluminium alloys, and to use this information to improve the performance of the next generation of aluminium alloys. The project is therefore of fundamental importance to Airbus UK, with the development of advanced damage tolerant alloys being a key aspect in the development of lightweight, cost-efficient structures.

Nevertheless, the participation of Airbus UK in the IDA project is limited to aspects such as establishing requirements and targets for improved aluminium alloys, including the definition of a generic fatigue load spectrum for large commercial transport aircraft wings. Airbus UK is also supporting the macroscopic crack growth and residual strength testing to be undertaken within the project. Any microscopic investigations, such as the identification of microstructural features and characterisation of the fracture surface morphology, are to be outsourced, since the specific resources and expertise required for such activities are not currently available within Airbus UK.

The partner selected to support the Airbus UK tasks within the IDA project is QinetiQ, which has world-class expertise in the necessary areas. QinetiQ is acting as a subcontractor to Airbus UK in the project and will be carrying out work in all five Work Packages within IDA.

2.4.6 Design of welded structures (WELDES) (P E Irving, J Lin, X Zhang, D Yapp and A Theos, Cranfield University; M Fitzpatrick, L Edwards and S Ganguly, Open University; I Sinclair and F Lefebvre, Southampton University; P M Powell, QinetiQ, Farnborough; S Richards, Alcoa Europe; R Maziarz, K Broad, G E Shepherd and M R Parry, Airbus UK, Bristol)

WELDES (Design of Welded Structures) is a collaborative research project involving industrial, research and academic organisations at the forefront of aerospace research in Britain. It is funded through the Defence & Aerospace Research Partnership (DARP) scheme designed to enhance the interaction between the industrial and research communities, supported by the Department of Trade and Industry. The primary aim of the project is to enhance the understanding of the way in which new and existing welding processes can be used in the design of aircraft structures. This is intended to lead to the development of methodologies for predicting fatigue crack growth behaviour of welded aluminium alloy aircraft structure.

The research project has focused on two welding processes, MIG (metal inert gas) and VPPA (variable polarity plasma arc). Tests and analyses have been conducted to assess the performance of the welded material/structure. These have included the initiation and growth of short fatigue cracks in and around the weld zone, including microstructural assessment, and the growth of long fatigue cracks transverse to the weld line under constant and variable amplitude. Residual stresses have been measured across the weld zone using neutron beam analysis techniques, and estimated through finite element simulation. Finally, a test specimen representative of a large wing skin panel with welded stringers has been designed and tested to determine fatigue crack initiation and crack growth under representative aircraft wing box spectrum loading.

2.4.7 Damage tolerance of weldable aluminium and titanium alloys for aircraft structures (M E Keeble, P M Powell, M W Squibb and J DG Sumpter, QinetiQ)

Present methods of airframe manufacture involving the use of mechanical fasteners are time consuming, labour intensive and costly. Recent analyses have indicated that a move to welded airframe structures could lead to cost savings in the region of 30% compared with conventional riveted structures. Welded structures should contribute to the prime objectives of reducing aircraft acquisition costs, reducing times to market and increasing flexibility in the design and manufacturing processes. Although the general use of welding within the aerospace industry is not new, welding of aluminium alloys is in the main not approved or proven for airworthiness, due to the historical inability to produce reliable defect free welded joints. There is clearly a need to understand the effects of these new manufacturing processes on the mechanical behaviour and damage tolerance of the materials. QinetiQ has participated in a collaborative project entitled “Cost Effective Manufacture: Welding Aerospace Materials (CEMWAM)” and has evaluated the damage tolerance of aluminium and titanium alloys to consider the influence of the laser, plasma and metal inert gas (MIG) welding processes in sheet and plate of thicknesses up to 12mm [5].

The tensile, fatigue crack growth and fracture resistance of 2024-T3 aluminium alloy sheet, titanium-6% aluminium-4% vanadium alloy plate, 2024-T351 aluminium alloy plate and 7150-W51 aluminium alloy plate (post-weld heat treated to the T651 condition) have been studied in butt welded test panels, manufactured by laser, plasma and MIG welding processes and using various filler alloys. Laser welding of 2024-T3 sheet led to significant reductions in tensile properties and in fracture toughness, K_{co} compared with the parent alloy. Further developments in the welding process are required to reduce levels of porosity. Both laser and plasma welding of the Ti-6Al-4V alloy led to relatively small reductions in tensile properties, but crack initiation toughness J_Q was significantly lower at both the edge of weld and at the weld centre than in the parent alloy. MIG welding of 2024-T351 and 7150-W51 plate led to a significant reduction in tensile properties for both alloys; crack initiation toughness J_Q was reduced more by welding in the 2024 alloy than in the post-weld heat treated 7150 alloy.

Fracture toughness measurements were made on the welded alloys using various techniques (R-curve, J integral and Kahn tear test methods) and the effect of test method on the ranking of toughness has been considered. The results demonstrate the value of the Kahn tear test for assessing the fracture resistance of ductile thin panel aircraft materials. It is possible to obtain a large amount of information on toughness and strength from a simple test, which uses only a very small sample of material (approximately 60 by 35mm).

The fatigue crack growth rates in the laser welded 2024-T3 sheet were only slightly faster than those in the parent sheet, despite the presence of marked porosity in these welds. In contrast, fatigue crack growth in the welded titanium and aluminium plate alloys was slower in the weld region than in the parent materials and the weld and edge of weld regions exhibited higher apparent threshold stress intensity factor ranges for fatigue crack growth than the parent alloys. An explanation for this behaviour, in terms of the effect of residual stress in the compact tension test piece, has been suggested.

Although welding the aluminium and titanium alloys did not adversely affect their fatigue crack growth resistance in this programme, significant reductions in tensile properties and in fracture resistance were apparent. These reductions must be considered by the aircraft manufacturers in the context of design targets for structural applications. If welded structures are to be adopted for airframe applications, then operating stresses will have to be adjusted to accommodate the static and fracture resistance properties of the welded alloys and target properties for as-welded materials will need to be defined and be routinely achievable.

2.4.8 Fatigue resistance and damage tolerance of welded aircraft wing skin panels (X Zhang, A Theos and S Mielow, Aerospace Engineering Group, School of Engineering, Cranfield University)

This is an EPSRC funded project with support from QinetiQ, Airbus UK, and Alcoa; the research consortium also includes Southampton and Open Universities. The main objective for this small group is to design and analyse welded aircraft stringer panels in order to improve the damage tolerance. The skin panel was made of aluminium alloy 2024-T351 and manufactured by welding the stiffeners to a slightly extruded wing skin panel. The fatigue load is applied in the longitudinal direction of the weld joint. Two distinct but related tasks have been performed. Firstly, FE analyses of fatigue crack growth behaviour in the welding-induced residual stress field. The results show that the residual stresses in the heat-affected-zone (HAZ) will decrease the crack opening stresses, hence accelerating the crack growth rate in the HAZ. The relaxation of residual stresses due to crack extension and cyclic loading and its effect on crack growth rate are also examined. Secondly, computer simulations are currently being performed for stringer panels for two scenarios: crack from weld joint and crack in mid-bay between two stringers (see Figure 8). For these cases, detailed finite element models are being employed to determine the *effective* stress intensity factors. The *AFGROW* package is then used to calculate the crack growth rate under variable amplitude loading. Another Cranfield Group will carry out fatigue tests on the welded stiffened panels shortly. The numerical predictions will be validated against the test results in terms of failure modes and crack growth lives.

2.4.9 Optimisation of the fatigue resistance of aluminium alloys using surface engineering (E R de los Rios and C A Rodopoulos, University of Sheffield; P Peyre, CLFA; A Levers, Airbus UK)

Saturation shot peening is believed to have a beneficial effect on the structural integrity of aerospace components, giving improved fatigue endurance and increased resistance to stress corrosion cracking. However, the process represents a significant cost and time to the project, and there is continual pressure from manufacturing to remove the requirement. Furthermore, there is conflicting evidence as to the effect of saturation shot peening on the integrity of mechanically fastened joints.

A number of linked projects was instigated, to study the effect of surface-engineering practices on the structural integrity of aerospace components; to develop microstructural fracture mechanics models for the characterisation of fatigue damage on engineered manufactured surfaces; to develop fatigue evaluation methods, based on these microstructural models, relevant to the designer and the maintenance engineer who have to perform safe-life calculations or damage tolerance predictions; and to keep a watching brief on international developments on surface-engineering research.

The project has established the qualitative benefits of shot peening, both in terms of confirming existing results known from the literature, as well as understanding the practical application of the process used by Airbus UK. Activity has focused on process optimisation using factorial experiments, understanding the effect of peening on the corrosion fatigue behaviour of alloys, and the development of fatigue modelling capability (including fretting fatigue) as a consequence of the process. There has also been a small amount of work to explore the potential benefits of laser shock peening, as compared to conventional processing.

Between 2001 and 2003 a number of individual projects have been undertaken at the University of Sheffield as part of this programme, to investigate the effect of surface engineering (shot and laser shock peening) on:
a) the fatigue resistance of as-received 2024-T351 and 7150-T651;
b) the corrosion fatigue of 2024-T351;
c) the fretting behaviour of 2024-T351 and
d) the effect of laser peening on peen formed components.

Results of this research are presented in [6 - 12].

In brief, the experimental data showed that:
After shot peening - a) excessive shot peening in order to achieve deep residual stresses can cause ductility loss. This is illustrated in the fractograph shown in Figure 9.; b) residual stresses from shot peening are unstable and tend to relax with loading history; c) the pit growth rate of shot peened surfaces is 10% lower than that of the as-received material; d) shot peening can improve the fretting life of aluminium on aluminium by 35%.

After laser shock peening - a) the residual stress are stable for stress levels up to $0.8\sigma^{y}$; b) laser shock peening can improve the as-received surface roughness by 22%; c) laser shock peening can improve the fatigue life of 2024-T351 by 60% and d) similar life improvement should be anticipated when laser shock peening is used to treat peen formed components. Figure 10 illustrates a number of the effects that laser shock peening has on the fatigue resistance of aluminium 2025-T351 alloy; for example, the life may be improved by up to 60%.

2.4.10 Cold expansion in the repair of aircraft structures - enhanced stop-drilling techniques (K Brown, K A Greedus, K W Man and A Young, QinetiQ)

Combined experimental and theoretical research was carried out at QinetiQ, under MoD Defence Logistics Organisation Structures Support Group (Aircraft Structural Integrity) funding, to investigate repair methods involving cold expansion of fastener and stop-drill holes. The research programme was in two parts. The first was a study of cold expansion of fastener holes and the second part was a study of enhanced stop-drill repair methods.

Single hole 7050-T76 aluminium alloy specimens were tested under variable amplitude FALSTAFF loading. Similar fatigue lives were obtained for specimens with open holes or clearance fit pins installed. Fatigue lives were increased by a factor of about three with interference fit pins installed. Cold expansion gave further increases in fatigue lives by a factor of about four for specimens with open holes or clearance fit pins and by about 50% for specimens with interference fit pins. Crack growth data demonstrated that the effect of cold expansion on fatigue life was due to delayed crack initiation and retardation of early crack growth. When small (up to 1mm) corner cracks were present, the benefit of employing cold expansion in repairs was substantial for specimens with open holes and holes containing clearance fit pins, but was negligible for specimens with interference fit pins.

Experimental and theoretical work was undertaken to develop and assess enhanced stop-drill techniques for the repair of fatigue cracks; the results of the research have been used to prepare a draft Work Instruction for enhanced stop-drill repair. Variable amplitude fatigue tests showed that the lives of thin 2024-T3 aluminium alloy and titanium TA2 centre cracked panels were not increased by the introduction of open stop-drill holes. Since open hole stop drilling (without enhancement) gives little, if any, benefit and may even be detrimental, it is recommended that the future use of this technique on military aircraft should be reviewed.

The fatigue life of aluminium alloy test panels was not increased significantly by either cold expansion of stop-drill holes or installation of clearance fit rivets. However, fatigue lives of thin aluminium alloy and titanium test panels were increased significantly by the installation of interference fit rivets in stop-drill holes. The life extensions (up to ten times) were greatest for repairs where rivet tail diameters were relatively large. It was concluded that this was largely due to frictional forces allowing load to bypass the hole through the rivet head and tail. Additional improvements in fatigue life were obtained when stop-drill holes were cold expanded prior to installation of interference fit rivets.

A two-dimensional boundary element model was used to study the effects of stop-drill holes, with and without interference fit pins, on stress concentrations at holes and stress intensity factors for cracks growing from the stop-drill holes. The model predicted that open stop-drill holes would reduce stress concentrations at holes and stress intensity factors for cracks shorter than 1mm, such that only a very small beneficial effect on overall crack growth was expected. In addition, the model predicted that interference fit pins would increase the mean stress intensity factor but reduce greatly the stress intensity factor range, such that significant retardation of crack growth was expected. An optimum level of interference was predicted, beyond which no further improvement in fatigue crack retardation would be achieved.

The model was also used to study whether improvements over a standard open hole stop-drill repair are possible by using multiple holes or non-circular holes. An elongated hole (i.e. two holes drilled either side of the crack tip and joined by saw cuts) was predicted to be the most effective of the alternative stop-drill scenarios.

2.4.11 Bonded composite patch repair of thick sections - effect of adhesive/prepreg co-cure and service temperature (P Poole, K Brown and A Young, QinetiQ; A D Armstrong, SSG RAF Wyton)

QinetiQ contributed to the BRITE-EURAM project known as COMPRES by carrying out theoretical and experimental research on bonded patch repair of thick sections.

Constant amplitude fatigue testing demonstrated that effective retardation of crack growth was achieved when 20 ply woven carbon/epoxy patches (manufactured from Cycom 753-42%-3KHS-P-199-1520 prepreg) were bonded to both sides of 12mm thick aluminium alloy test pieces containing central through-thickness cracks. For both $R = 0.1$ and $R = -1$ loading, patches applied by co-cure of adhesive and prepreg were as effective as bonded precured patches in retarding the growth of fatigue cracks. As expected, 10 ply patches were less effective than 20 ply patches. In almost all of the tests, no significant debonding or delamination of patches was detected during fatigue testing. Thus, in general, it is concluded that effective repairs can be achieved when woven carbon/epoxy patches up to 20 plies thick are applied by co-cure of adhesive and prepreg under vacuum pressure only.

For specimens repaired with 10 or 20 ply precured patches, small increases in fatigue crack growth rate occurred when the test temperature was increased from 24ºC to 70ºC and from 70ºC to 82ºC.

A 3D BE/FE model for analysing bonded patch repairs was developed to enable the effect of non-linear adhesive stress-strain behaviour to be taken into account. In general, slightly slower rates of fatigue crack growth were predicted if linear rather than non-linear adhesive behaviour was assumed. Furthermore, fatigue crack growth rates predicted from an analytical 1D model were significantly different from those predicted by the non-linear 3D model.

In general, the fatigue crack growth rates predicted by the non-linear 3D model were in very good agreement with corresponding experimental data for patched specimens. The predicted increases in fatigue crack growth rate with increasing test temperature were in excellent agreement with test data, as illustrated in Figure 11. The small increases in crack growth rate observed when the test temperature was increased from 24ºC to 70ºC or from 70ºC to 82ºC were explained in terms of the opposing effects of reduced residual stresses and increased adhesive strain as test temperature increased. The non-linear 3D model predicted that a reduction in cure temperature from 120ºC to 95ºC would result in only very small reductions in fatigue crack growth rate, due to reduced residual stresses. Since the experimental fatigue crack growth data for patched specimens exhibited significant scatter, it is not surprising that there was no evidence of a small beneficial effect due to the lower initial cure temperature used in the case of patches applied by co-cure of adhesive and prepreg.

The research summarised above will be described in detail in a paper to be presented in the poster session at the ICAF 2003 Symposium.

2.5 FULL SCALE FATIGUE TESTING

2.5.1 Global Express Horizontal Stabilizer Metallic Centre Box Subcomponent DADT Test (L Murray, Bombardier Aerospace Shorts, Belfast)

The GX Horizontal Stabilizer Metallic Centre Box Subcomponent test, designated EX01 and shown in Figure 12, is required to complete two lifetimes of durability and one lifetime of damage tolerance testing; one lifetime for the Global Express being 15,000 flights. To date it has completed the required two lifetimes of durability testing and has now entered the damage tolerance phase. For this phase artificial damage has been introduced to the structure at several locations and to date none of these artificial damages has prompted fatigue cracking. If this situation remains these artificial damages will be increased in size in an attempt to promote fatigue cracking, therefore enabling correlation with the corresponding theoretical crack growth predictions.

2.5.2 Lear 45 Fatigue Testing (L Murray, Bombardier Aerospace Shorts, Belfast)

The Learjet 45 wing and fuselage fatigue and damage tolerance test (test article designation TA05, shown in Figure 13) has completed 40,000 flights (2 design lifetimes) of durability testing, and is 7,000 flights into a 20,000 damage tolerance phase with artificial damage installed. The loading applied represents gust, manoeuvre, and pressurisation loads to an altitude of 51,000ft. A separate test article for the empennage and tailcone (test article TA04) has completed 40,000 flights of durability and 10,000 flights of a 20,000 damage tolerance phase with artificial damage installed. The loading spectrum represents gusts, manoeuvres, and thrust reverser deployments.

2.5.3 Tucano Full Scale Fatigue Test (L Murray, Bombardier Aerospace Shorts, Belfast)

A Full Scale Fatigue Test (FSFT) on a RAF Tucano production airframe is under development. The test spectrum is being constructed from data collected during an Operational Loads Measurement (OLM) programme (see Section 2.3.2). The spectrum is expected to include up to 100 flight types constructed from a library of 1000 balanced load conditions, representing typical ground and flight loading events.

The test airframe will include the fuselage, wings, tailplane and fin. The landing gear, flaps, elevators, airbrake and engine can be represented by dummy components.

Loads will be applied to the airframe using 42 hydraulic actuators acting, where appropriate, through distributed linkages. The airframe will be grounded to provide translational and rotational load balance. Calibrated dual bridge load cells will be fitted to all actuators and reaction points. Actuator demand and applied loads will be monitored continuously throughout the test. At present the test rig has been designed and manufacture is at an advanced stage.

The test airframe is fitted with an extensive strain gauge installation. The strain gauges will be sampled at regular intervals throughout the test.

2.5.4 Typhoon Main Landing Gear (T Siddall, Messier-Dowty)

Messier-Dowty Ltd (MDL) is responsible for the design and qualification of the Main Landing Gear (MLG),which has a safe life qualification requirement of 6000FH (8000 landings). The fatigue test is being conducted under sub-contract at INTA Spain, and is currently 80% complete. The test rig has the capability to dynamically retract and lower the landing gear and apply ground loading via a dummy wheel. The test is carried out in blocks of retractions, followed by ground loads and finally a strip and NDT of the structural components, before re-assembly and continuation of testing. The test spectrum incorporates various mission profiles which include taxy out, turning, take-off, landing impact, braking and taxy back sequences for combinations of take-off and landing weight. A life extension test for the MLG is under review, but it is envisaged that the testing will continue until failure of one of the main structural items. Fatigue analyses have also been carried out to assess the impact of operation under new mission profiles, using heavier weapons configurations.

2.5.5 Airbus Landing Gears (T Siddall, Messier-Dowty)

In June 2002 MDL Gloucester opened the largest landing gear test facility in the world. The plant, shown in Figure 14 was expanded by building a new superstructure over the old test house and then demolishing the old test house from within.

There are several fatigue tests in progress covering the growing Airbus family of aircraft.

In the case of the A330/340 "Growth" standard Main Landing Gear, the fatigue test is currently developing the fatigue life of the landing gear based on service loads which incorporates taxying, landing braking, retraction, turning and pushback.

The Main, Nose and Centre Landing Gear fatigue tests for the A340-500/600 programmes are completed to certification level. These tests are now continuing to develop the fatigue life of these units. The A340 fatigue test is shown in Figure 15. This rig weighs 250Tonnes, with a loading capacity of up to 1000Tonnes force.

Additional element fatigue tests are being carried out to qualify ancillary items such as the shortening mechanism.

In all of the above cases, extensive photoelastic and strain gauge techniques have been employed to gather data to support validation of theoretical analyses.

2.5.6 Eurofighter Typhoon Production Major Airframe Fatigue Test (M Greenhalgh, BAE SYSTEMS, Warton)

The Eurofighter development aircraft Major Airframe Fatigue Test (MAFT) was performed between 1993 and 1999 on a single-seat airframe and loading. It successfully reached 18000 test hours with few significant damage incidents. The airframe has been redesigned for production and includes changes due to these damages. The aim of the Production Aircraft MAFT is to test this revised design standard using a twin-seat airframe and twin-seat loading

The test rig will apply fatigue load cases representing symmetric and asymmetric manoeuvres as a flight by flight spectrum of 1000 hours for a total of 18000 test hours. Fin buffet will be applied by dynamically exciting the fin and will be applied in blocks after each 1000-hour manoeuvre sequence. Further loading actions applied to the test include gust loading on the outer wing region, ejector seat inertia loads applied directly in the cockpit (as they are a major internal loading action), cockpit, intake and fuel tank pressurisation. The undercarriage loads applied to the PMAFT are a condensed set of the loading sequence used for the undercarriage fatigue tests.

The test rig has been designed to allow complete access around above and below the test article. The airframe is 3m above floor level, which allows complete access to all parts of the structure for inspections.

A second test is also being performed on a single-seat front and centre fuselage specimen to qualify the different structure using single seat loading. This test will also include the effects of In-Flight Refuelling probe loading on the front fuselage (including the effect of drogue impacts applied dynamically with a pendulum).

As the Production Aircraft MAFT needs to qualify the common structure to both single and twin-seat loads, the differences in loading need to be identified. Test loading is calculated for a full single seat aircraft in the same way as for the PMAFT and test load cases are simulated on full airframe finite element models of each version. A comparison of the internal loads within the common structure is performed and if there is any shortfall in the required damage to qualify the common structure for the single seat airframe, then additional loading will be applied at the completion of the twin-seat fatigue test.

Further details of this work may be found in [13], to be presented at the Symposium**.**

2.6 DEVELOPMENTS IN FATIGUE MONITORING

2.6.1 Development of a Parametric Aircraft Fatigue Monitoring System Using Artificial Neural Networks (Steve Reed and David Cole, QinetiQ, Farnborough)

QinetiQ has been tasked, by the UK MOD Tri-Service Aircraft Structural Integrity Branch (SSG ASI), with investigating the use of advanced mathematical methods in aircraft fatigue monitoring systems. The purpose of this task is to produce draft regulatory material for certification and qualification of artificial neural network-based aircraft fatigue monitoring systems. However, in order to achieve this, a deeper knowledge of how these techniques work, how they must be trained and the strengths and weaknesses of the systems used needs to be developed.

During the first phases of this task, a Multi-Layer Perceptron (MLP) artificial neural network has been used to predict strains from flight parameters. Data from the Tucano Operational Loads Measurement (OLM) programme were collated for training and testing the artificial neural network. Domain knowledge was applied to the input parameters and training set optimisation methods were developed. Subsequently, strains at the inner wing bending bridge and the outer wing bending bridge were predicted, using the OLM parametric data. The final models produced were tested using data from within and outside of the training set and from an alternative OLM aircraft, not used in the training set.

Results were encouraging and the actual to predicted strain correlation factor for the inner wing, for flights outside the training data, (and largely from the alternative aircraft) was 0.995. Additionally, the correlation between actual and predicted strain for the outer wing was 0.997. A typical cross plot between the actual and predicted strain, for an individual flight, for the outer wing is in Figure 16.

In fatigue damage terms, equally encouraging results were obtained. The correlation between the actual fatigue damage and predicted fatigue damage for the inner wing for flights, outside the training data, (and largely from the alternative aircraft) was 0.990 (Figure 17). Additionally, the correlation between actual and predicted fatigue damage for the outer wing was 0.997 (Figure 18).

The next phases of work will investigate the prediction of tailplane, fin and undercarriage strains, using the Tucano OLM data, before migrating the models to a combat aircraft.

Further details can be found in [14], to be presented at the Symposium.

2.6.2 FUMS^{TM} Technologies for affordable Prognostic Health Management (PHM) of gas turbines, helicopters and aeroplanes (H Azzam, Smiths Aerospace Electronic Systems - Southampton [ES-S[, J Cook, Aircraft Integrity Monitoring [MOD] and S Driver, Propulsion Support Group [MOD])

Introduction

PHM is a logical step following from the Ministry of Defence (MOD) continual efforts to improve aircraft Management, Affordability, Availability, Airworthiness and Performance (MAAAP). MOD has fitted aircraft with data collection equipment such as Flight Data Recorders (FDR), Engine Usage Monitoring Systems (EUMS) and Health and Usage Monitoring Systems (HUMS). It is mandatory that all the UK registered civil helicopters carrying more than 9 passengers are fitted with HUMS. MOD is committed to meeting or exceeding the civil requirements for all helicopters subject to sufficient remaining life. Whilst the MOD maintenance/logistics systems contain a wealth of information including the huge volume of data downloaded from aircraft, there is a growing interest in analysing and automatically trending, fusing, and mining this information. Combining this with MOD and Design Authority (DA) experiences will allow the development of advanced diagnostics/prognostics and the establishment of enhanced maintenance procedures. Advanced model-based/statistical tools, powerful user-friendly interfaces and intelligent data management tools are required to address this need. Smiths ES-S has worked closely with MOD and evolved a Flight and Usage Management Software (FUMS^{TM}) to address this need. The following developments/studies have been selected from the total capability to summarise those features particularly related to fatigue:

Diagnostics, Prognostics and Life Management Techniques
Smiths have used their mathematical models to establish diagnostic techniques; Figure 19 shows the results of the Dirac Delta model and the vibration induced by adjustable faults. The figure indicates the validity of these techniques.

The tools have successfully synthesised from flight parameters loads, fatigue and all-up mass as shown in Figure 20. Proactive fatigue management should not only evaluate loads and fatigue but also indicate the impact of usage on the condition and life of aircraft components. Therefore, Usage Indices (UIs) have been developed to provide concise summaries of recorded flight data and, at the same time, indicate the impact of usage on component condition and life. Smiths have also developed a suite of Artificial Intelligence (AI) models and signal-processing techniques to aid prognostic and diagnostics developments. Vibration indices have been derived and used to quantify the effects of rotor system faults on fatigue damage.

A Study into the Feasibility of Performing Usage Spectra Analysis at 3rd Lines

At present, the fatigue lives of helicopter critical components are calculated by the DA in terms of flying hours after which the components should be replaced. The bases of fatigue life computations are measured component loads at various flight conditions and design/intended usage, which is expressed as the likely times spent at the various flight conditions. Since the actual usage can be at a great variance with the intended usage, it is mandatory to periodically assess the actual usage. At present, the MOD and DA assess the actual fleet usage by monitoring a small sample of helicopters. The monitoring methods employed are Manual Data Recording Exercise (MDRE) and Operational Data Recording (ODR). The former method requires a member of aircrew to record selected flight parameters over a range of flight conditions and, hence, MDRE impacts the service use of the aircraft and is subject to human error. The latter method is performed every three to five years where the fleet sample is fitted with sensors and recording equipment. Due to the complexity of the system fitted, the ODR programmes are often very costly. Both methods do not provide accurate usage for each individual helicopter. With the envisaged growth in using MOD HUMS flight data, it would be possible to monitor the usage spectra of MOD helicopters through FUMS^{TM}: The actual usage spectrum of an individual helicopter has been calculated and continuously updated using the samples of flight data that have been downloaded and comparisons have been made with intended/design usage spectra. Flight Condition Recognition (FCR) algorithms have been developed to identify flight conditions in accordance with DA methods. A thorough literature survey has highlighted an inherent FCR weakness: identical flight conditions/manoeuvres have been found to induce loads/damages that differ in values by order of magnitudes. Therefore, advanced usage algorithms have been developed to identify flight conditions and, at the same time, evaluate the severity of them. Whilst helicopter applications have recently been considered, the algorithms have been generically developed for future engine/aeroplane applications. The FUMS^{TM} usage capabilities would allow operational recommendations to be made to adhere with the intended usage agreed with the DA. If particular operations require deviations from the agreed intended usage, FUMS^{TM} would provide the MOD and DA with the information required to evaluate the impact of these particular operations.

2.6.3 The evolution of an affordable airframe Structural Prognostic Health Management (SPHM) system (H Azzam, Smiths Aerospace Electronic Systems–Southampton [ES-S], Louis Gill, BAE SYSTEMS, Warton)

Introduction

There is a growing interest in developing affordable SPHM systems that use existing flight parameters to track how each individual aircraft is used and quantify the damaging effects of usage. Over the past four years, Smiths ES-S and BAE SYSTEMS have launched collaborative efforts to evolve a practical SPHM system. The work has built on the BAE SYSTEMS experience of operational load monitoring that has spanned more than 30 years, combined with the unique experience of Smiths ES-S over the past 20 years examples of which were given in the previous section. The collaborative work has concentrated on the SPHM areas described in the following paragraphs:

Anomaly Detection/Correction

Operating on data from a large number of flights, ES-S demonstrated that their anomaly detection/correction tools could automatically identify corrupt data and correct data corruption by interpolation or by reconstruction from other flight parameters wherever possible. The types of data corruption that were automatically identified and corrected included spikes, steps, data out of range, DC signals and more complex corruption patterns.

Fatigue From Flight Parameters

Using BAE SYSTEMS' engineering knowledge, ES-S configured mathematical networks to synthesise fatigue from flight parameters. Fatigue was synthesised at Eurofighter starboard and port wing-to-fuselage brackets. Data from 30 flights selected at random were used for training and 14 flights were used to blind test the networks. Across the 44 flights, the errors in accumulated fatigue for the two locations were less than 0.52%. A more challenging component such as the fin is sensitive to rare buffeting events that induce high dynamic loads. Significant effort is required to ensure that the characteristics of such a component is fully understood and the dynamic events are adequately modelled. Therefore, ES-S developed a Eurofighter finite element model using public-domain data, configured a dynamic rare event model, embedded the model into a mathematical network and trained the network to compute the fin fatigue. Figure 21 shows the network fatigue over 104 blind test flights that were not used for training. Whilst the figure indicates that the network results are better than a strain gauge system with 5% error, the network and its models are being optimised to ensure good generalisation capabilities.

Stresses/Strains from Flight Parameters

Strains/stresses synthesised from flight parameters would allow health assessments to be made using a safe-life approach (fatigue) or a damage-tolerant approach (crack characteristics). Synthetic strains/stresses would also provide information to identify over-load events. Using BAE SYSTEMS' engineering knowledge, Smiths ES-S trained their mathematical network, which combined dynamics, aerodynamics and static analysis, to synthesise the strains from parameters at wing to fuselage brackets. Training data were compiled from 45 Eurofighter flights. Figure 22 shows the high degree of accuracy of the network; a correlation between actual and synthetic signals of 0.995 was achieved in blind testing.

For more challenging components such as the fin and foreplane, a quick look investigation indicated that the mathematical networks were superior to conventional regression and neural network approaches. For example, using linear regression, there was a poor correlation of 0.593 between recorded and synthetic foreplane stresses; the neural network improved the correlation to 0.603 and the mathematical network showed a substantial increase in correlation to 0.928. Figure 23 illustrates the excessive prediction errors during a dynamic event caused by airbrake applications. A worldwide literature survey indicated that all prediction methods reported to date would fail and produce similar errors. Preliminary configuration of a mathematical network that used a rare event dynamic model rectified this problem and produced the results shown in Figure 23. The error in fatigue computed from the network synthetic stress was 5.7%. The above encouraging preliminary results have justified further effort to optimise the mathematical network.

Qualifying SPHM Systems that Use AI Techniques

Smiths ES-S and BAE SYSTEMS have been carrying out extensive investigations that would lead to establishing and agreeing a route for qualifying/certifying AI based systems. This is essential at this stage, to avoid a delay in introducing such systems into service. The adequacy and quality of the data required to train/configure an AI method have been assessed using methods that have examined billions of strain/flight data points combined with model-based and statistical analysis. The Regulatory Authority is concerned that some operations/configurations may contain novel data outside that gathered to train an AI method. This concern is being addressed. Guidelines for management approaches based on Individual Aircraft Tracking have been also investigated.

2.7 FATIGUE IN COMPOSITE MATERIALS AND STRUCTURES

2.7.1 Modelling the fatigue life of polymer-matrix fibre-composite components (O Attia, A J Kinloch, F L Matthews, Imperial College, London)

A methodology has been proposed for predicting the fatigue life of fibre-composite components and structures, which combines relatively short-term fracture mechanics data, obtained from experimental measurements, with a finite-element analysis of the component or structure. The finite element model is used to deduce strain-energy release rates for the growth in area of damage. The fracture mechanics data are measured under cyclic fatigue loading.

In one study, the approach was used to predict the growth of damage, and the cyclic fatigue life of I-beams [15], containing a circular notch in the web. Previous experimental work had shown that the development of significant damage was confined to the region of material in the web around the notch. The damage was observed to proceed from early matrix micro-cracking, leading to limited delamination. These two damage growth mechanisms eventually led to fibre fracture, which was the final cause of structural failure. Hence, the modelling was aimed at these types of damage mechanism.

The agreement between the results from the theoretical model and the experiments was good, especially when it was considered that there are no ‘adjustable factors’ involved in the modelling studies. For example, the number of cycles to failure of the I beam predicted by the model was 4.9×10^6 cycles, compared with an experimentally measured value of 4.78×10^6 cycles.

This methodology was extended to predict the growth of impact damage in carbon-fibre reinforced-polymeric skin/stringer structures, when subjected to cyclic fatigue loading [16]. In this case, the delaminations caused by impact damage in the structure were modelled using a ‘rigid link’ technique. Dents in the surface of the panel have been modelled by considering a perturbed shape. It was demonstrated that the amplitude of this dent had an important role in determining the number of fatigue cycles to cause the onset of damage growth, for a given maximum applied strain. Figure 24 shows the number of fatigue cycles to failure, predicted using the model, which combines the FE analysis with experimental results from the fracture mechanics tests. The number of cycles to failure is very sensitive to the depth of the perturbation caused by the impact, as well as by the maximum value of applied strain in the fatigue cycle. While these parameters can be measured from the damaged panel, it would be advantageous if these factors could be predicted by an impact damage model.

It can also be seen that some of the strain levels imposed in the FE modelling studies are above the buckling strain of the panel, and thus correspond to the post-buckled regime. To represent this response accurately, non-linear analysis will be required to account for local buckling in the impacted region and the associated delamination growth.

A further publication is being prepared, describing a comparison of these theoretical predictions with experimental results.

2.7.2 Fatigue life prediction of polymer composite materials (P T Curtis, Dstl, and A J Davies, QinetiQ, Farnborough; H Carroll and T J Matthams, University of Cambridge)

QinetiQ and the University of Cambridge have collaborated on research over the last 3 years to develop improved models for predicting the fatigue life of polymer composite materials. Existing fatigue life prediction models were reviewed [17, 18] and a residual strength model proposed by Yang and Shani [19] was selected for further study. The model was incorporated into a Microsoft Excel spreadsheet for ease of data manipulation. The spreadsheet was fully populated with both static and residual strength data. From these data, Weibull parameters were obtained and these were used in the predictive equations. The output of the model was the probability of a specimen (which had been subjected to a set constant amplitude loading regime) surviving a given number of cycles. The model was verified experimentally and performed well.

The drawback of using purely statistical models, however, is that by their nature they tend to be somewhat conservative in their output. To understand more fully what was happening mechanistically, work was undertaken at the University of Cambridge, funded by QinetiQ. The research work included the manufacture and testing of axially loaded fatigue specimens and the subsequent investigation of various interfaces using both static and cyclic mixed mode bending

(MMB) specimens [20, 21]. The growth of interfacial cracking and strain energy release rates under these loading conditions was found. Further work was done to validate these data, by testing fatigue specimens containing PTFE embedded defects; the crack growth and subsequent increase in delamination area was monitored using non-destructive evaluation techniques. It was found that although a very wide coupon was tested, the damage growth from the insert was masked by damage growing in from the specimen edges. The knowledge gained from this package of work is currently being applied to structural elements such as tapered stringer run-out specimens.

2.7.3 COMPASS modelling fatigue-driven delamination (P Robinson, U Galvanetto and G Bellucci, Department of Aeronautics - Imperial College London)

The aim of the research project is to simulate fatigue-driven crack growth numerically, which could either be delamination growth in laminated composites, or crack propagation in adhesively bonded joints, by using interface elements. The formulation of the interface element stems from a combination of fracture mechanics theory and damage mechanics and has been used extensively to simulate both the initiation and the propagation phases of a crack under the application of a static load. Since cyclic loadings are a common working condition for many structures, the numerical model is being developed to simulate damage accumulation due to repeated loads. The interface elements developed by Professor Crisfield [22] have been adopted in conjunction with a damage model of high-cycle fatigue.

A numerical model for the crack growth due to cyclic application of a load (fracture Mode I) has been developed. The numerical model is able to evaluate the damage due to monotonic loading and due to cyclic application of a load. The damage rate due to fatigue, in the case of a brittle material, can be considered as function of the damage itself and the strain state of the material:

$$\frac{dD}{dt} = g\left(D, \tilde{\varepsilon}\right)\dot{\tilde{\varepsilon}}$$

where g is a function and $\tilde{\varepsilon}$ is an equivalent strain measure. For the interface element, the equivalent strain has been taken as the relative displacement, δ, non-dimensionalised by dividing by the Failure Limit δ_C.

Two expressions for the function g have been considered, one proposed by Paas **[23]** and the other by de Borst **[24]**, which yield the following expressions for damage rate:

(de Borst's law – exponential law) $$\frac{dD}{dt} = C \cdot e^{\alpha D} \cdot \left(\frac{\delta}{\delta_C}\right)^{\beta} \cdot \left(\frac{\dot{\delta}}{\delta_C}\right)$$

(Paas' law – power law) $$\frac{dD}{dt} = B \cdot D^{\varphi} \cdot \left(\frac{\delta}{\delta_C}\right)^{\gamma} \cdot \left(\frac{\dot{\delta}}{\delta_C}\right)$$

in which:

δ	the relative displacement
$C, B, \beta, \gamma, \varphi$ and α	material parameters to be evaluated experimentally

Typical graphs for the Paris' law have been plotted using the output from the numerical analysis; they simulate likely crack growth behaviour. Several numerical simulations have been carried out and a methodology has been identified to evaluate the parameters present in the numerical model from experimental data.

The Finite Element model has to be validated on the basis of experimental data. Thus, it is important to carry out a series of fatigue tests with different values of cyclic load, in terms of cyclic amplitude and stress ratio. So far only fracture Mode I has been investigated. In order to simulate the general case of crack propagation under fatigue it is important to apply the present Finite Element approach to Mode II and Mixed Mode I/ II fatigue-driven crack growth. To develop a reliable numerical model, it will be necessary to investigate other laws for interface degradation due to fatigue and their influence on the macroscopic behaviour of the cracked component.

2.7.4 Delamination of multi-directional carbon composite laminates under static and fatigue loading (M J Hiley, QinetiQ)

Mode I, Mode II and Mixed-mode (I+II) tests were done on continuous fibre reinforced composites, between plies of different orientations. Tests were conducted statically between 0º/0º, 0º/45º and 0/90º interfaces and in fatigue between

0º/45º interfaces. Three materials, all with the same Hexcel 8552 resin, but different fibre types (T800, IM7 and AS4) were evaluated to assess the influence of fibre type on delamination propagation. Strain energy release rates G_{IC} and G_{IIC} were determined statically, with fatigue tests being used to assess crack growth rates. In addition to determining physical parameters, a brief study of the fracture surfaces (fractography) of the specimens was done, using scanning electron microscopy (SEM) [25].

Tests showed that the orientation of the plies and the test specimen, as well as the fibre type, were important in controlling delamination growth. Studies on the 0º/90º and 0º/45º ply interfaces showed that delamination tests could be conducted using conventional Mode II and mixed-mode bending specimens (as used to test unidirectional laminates), provided the 0º ply was positioned on the upper (compressive) half of the laminate during testing. This was because shear stresses present in the laminates tended to drive microcracks forming ahead of the crack tip upwards to the 0º ply, through which they could not migrate any further. If the specimens were reversed, so that the angle ply was on the compressive surface, the delamination quickly migrated through the plies to other ply interfaces, thereby changing the toughness of the specimen. Such behaviour was observed in the Mode I specimens. Fatigue tests conducted in Mode I and Mode II gave Paris plots with very steep gradients, which meant that their use for predictive purposes may be limited, the determination of fatigue thresholds being potentially more useful. Fractographic assessment of the fracture surfaces revealed the presence of striations within matrix and fibre imprints. The nature of the features observed, however, varied depending on the fibre type used.

2.7.5 Fractographic assessment of fatigue Failures in complex laminate and structures (M J Hiley QinetiQ)

In a combination of in-house research, and research within the GARTEUR European collaboration, the mechanism controlling fatigue delamination growth in laminated composites has been investigated. Efforts have been directed at the interpretation of interlaminar fractures generated under mode I, mode II and mixed-mode (I+II) loading. Initial studies within GARTEUR AG20, (focusing on fatigue failure in unidirectional laminates) revealed a range of fractographic features by which fatigue failures could be differentiated from equivalent static fractures. Features identified included striations within the resin and fibre imprints (see Figures 25 & 26), as well as matrix rollers. General relationships between these fractographic features and the direction and rate of crack propagation were also established in some instances. Following the conclusion of AG20 [26], a new activity (AG27) has been set up to investigate fatigue failure in more complex laminates based on multidirectional plies, woven and non-crimped fabrics (NCFs). This action group involves participants from QinetiQ, BAE SYSTEMS, CETIM, CSM Materialteknik, EADS (Germany & France), INTA, NLR, FOI and SICOMP. In initial studies, the read-across between fatigue failures in unidirectional (U.D) laminates and those containing angled plies has been investigated. It was found that many of the features observed in U.D laminates were also present in multidirectional laminates, although the features were often prone to modification, depending on the ply interface at which they occurred. In current work, failure in laminates containing woven fabrics with different weave styles is being examined.

2.8 FRACTURE MECHANICS AND DAMAGE TOLERANCE

2.8.1 Advanced design concepts and maintenance by integrated risk evaluation for aerostructures (ADMIRE) (M R Parry, Airbus UK, Bristol)

The ‘initial flaw’ concept has been the subject of considerable attention in recent years. Originally introduced by the United States Air Force, the concept is now widely used throughout the US aerospace industry in the determination of threshold inspection intervals for structural maintenance programmes. However, within Europe the inspection thresholds are typically set by conventional damage accumulation methods, in which different hole and bolting standards are accounted for by specific fatigue endurance data. In the absence of validated initial flaw distributions based on European practice, the introduction of initial flaw methodologies may only be possible by accepting unduly conservative assumptions, with a subsequent impact on design efficiency.

Investigation of the initial flaw concept is included in the ADMIRE research project, part of the European Union ‘Fifth Framework’ Programme, which involves industrial, research and academic organisations from across Europe. In particular, the main focus of the work carried out by Airbus UK within the ADMIRE project includes the development of methodologies to derive equivalent initial flaw sizes from fatigue endurance data; coupon testing to determine the effect of manufacturing induced damage on fatigue life, in relation to the definition of an equivalent ‘rogue flaw’; the specification of initial flaw sizes, representative of both standard quality and ‘rogue flaws’ for all joint types, bolting standards and materials of interest; and the enhancement of prediction methods for crack growth under aircraft spectrum loading.

2.8.2 Development of equivalent initial flaw size approach to the lifing of aircraft components (K. Brown, QinetiQ, Farnborough; M R Parry and R A Collins, Airbus UK, Filton, Bristol)

The use of the Equivalent Initial Flaw Size (EIFS) approach to determine fatigue endurance, and consequently inspection thresholds, is new to the European aerospace industry. It requires the development of new models to include short crack effects, and probabilistic approaches to examine variability in EIFS and fatigue endurance data. In this project, research has been carried out under a GARTEUR agreement in collaboration with European aerospace partners to understand and develop the EIFS approach (see also Section 2.8.3 for a report of work done by QinetiQ under this aegis). Programs have been developed to determine the EIFS from fatigue endurance data, and have been used to determine the EIFS for a range of simple geometries and materials, with an emphasis on the effect of 'rogue flaws' (e.g. manufacturing defects such as scoring in the bores of holes, swarf entrapment, and insufficient cooling during drilling of fastener holes). The sensitivity of the EIFS to model variables has been examined, and the key parameters identified.

The work of the primary investigator - QinetiQ - has been supported by the experience gained by Airbus UK from other complementary research projects related to the initial flaw concept, such as the Fifth Framework Programme 'ADMIRE' (see Section 2.8.1). Financial support for this project has been obtained from the METEOR Programme, which is joint-funded by Airbus UK and the Department of Trade and Industry.

2.8.3 Development of an equivalent initial flaw size (EIFS) approach to the lifing of aircraft components (K Brown, A Young and P M Powell, QinetiQ)

QinetiQ is participating in the GARTEUR Action Group AG26 entitled "The Initial Flaw Concept and Short Crack Growth". Research is being carried out to improve understanding of the EIFS approach and computer models have been developed to calculate EIFS distributions rapidly. EIFS values have been derived for four test cases: 2024-T351 open hole specimens, 204-T351 cold worked open hole specimens, 7010 open hole specimens and 2024-T351 high load transfer joints with interference fit bolts. The dependence of calculated EIFS values on alternating stress, mean stress and specimen geometry has been evaluated and conservative values of EIFS have been identified for these test pieces.

Work has started, under the METEOR collaborative programme with Airbus UK (see the summary in the previous Section 2.8.2), to investigate the initial flaw distributions associated with "rogue flaws" arising from exceptional manufacturing defects. High load transfer joint specimens are being manufactured in 2024-T351 fitted with interference fit fasteners in non-cold worked holes. The rogue flaws introduced into these test pieces will consist of various combinations of (a) a scratch of controlled depth down the bore of one hole, (b) no tension in the bolts and (c) a scratch of controlled depth across the surface of one of the side plates, on the face that will be in contact with the centre plate.

2.8.4 Short crack approach to damage tolerance assessments (P Bowen, University of Birmingham; P M Powell, QinetiQ, Farnborough; M R Parry and R A Collins, Airbus UK, Filton, Bristol)

The short crack approach to damage tolerance assessments has been proposed by several investigators and preliminary investigations have commenced in the United States. However, within Europe the application to civil transport wings is entirely new. The objectives of this project are the experimental study of short crack behaviour for a range of materials, environments and loading conditions; the development of improved models for predicting the growth of short crack; and the compilation of a database of short crack growth data, which is of importance in the development of the Equivalent Initial Flaw Size approach to fatigue life prediction.

In this project, short crack growth observations have been made using a test facility (unique in Europe) at Birmingham University. These observations use a novel potential drop technique, and crack closure studies using a SEM with an integral loading stage. Financial support for the project has been obtained from the METEOR Programme, which is joint-funded by Airbus UK and the Department of Trade and Industry.

2.8.5 Enhanced life prediction for three-dimensional cracks (S Simandjuntak, H Alizadeh, M J Pavier and D J Smith, University of Bristol; R A Collins, Airbus UK, Bristol; D Nowell and D Hills, Oxford University)

This project is a joint research programme between the University of Bristol and the University of Oxford, with funding from Engineering and Physical Sciences Research Council (EPSRC) and industrial sponsorship from Airbus UK and Rolls-Royce. The intent of the project is to develop improved models for predicting the growth of three-dimensional fatigue cracks. Although the project is to concentrate on conditions appropriate to the loading of aircraft wing structure, the results of the research should have applications in the Fatigue & Damage Tolerance community across all of the Airbus Company.

The primary objectives of the project are to obtain fatigue crack growth data and closure information for representative three-dimensional crack geometries under a range of load histories; to use finite element methods to model crack closure and to investigate the three-dimensional nature of the problem; to develop and validate simplified crack closure models appropriate to three-dimensional crack problems; and to develop a crack life prediction methodology for components subjected to non-uniform loading using the simplified model developed above. The main focus of the work at the University of Bristol is the investigation of the growth of three-dimensional cracks in 2024-T351 aluminium alloy. The key specimen geometry is a plate containing a central hole, with starter cracks machined at the edges of the hole using the EDM technique. Surface measurements of fatigue crack growth rates have been made under constant amplitude and variable amplitude loading, and finite element analyses have been conducted to improve understanding of how various physical processes influence fatigue crack growth rate.

In a further part of the EPSRC project Oxford University Department of Engineering are collaborating with University of Bristol. Oxford are using a Moire technique to measure surface closure on titanium beams and extending their 1D plane stress closure model to plane strain. Their work has additional support from Rolls Royce.

2.8.6 Toughness characterisation by energy dissipation rate (J D G Sumpter and P M Powell, QinetiQ)

Work has continued on the characterisation of tearing in aluminium aircraft alloys by the energy dissipation rate approach [27]. Application of the method to thin (1.6mm) alloy and weld was described in the 1999 to 2001 review. A fuller description of the updated rationale and methodology is contained in a paper recently accepted for open literature publication [28]. In the past twelve months the emphasis has switched to the toughness evaluation of thicker material (15mm). A range of specimen sizes from 500mm wide centre cracked tension down to 30mm wide three point bend have been tested. Five different alloys have been evaluated to give a spread of toughness and yield strength characteristics.

There has traditionally been a split between the way that tearing resistance is evaluated in aluminium alloys (K_R curve by ASTM E561 using large centre cracked tension 'plane stress' specimens) and in steels (J_R curve by ASTM E1820 using small 'plane strain' bend specimens). Analysis using energy dissipation rate provides an opportunity to understand and link both approaches.

Obtaining a K_R curve in thick alloys raises various difficulties. The first is the practical problem of machine load capacity. This limits the width of panel and hence the maximum K that can be achieved before the panel yields. 'Pop-ins' are common in less toughness alloys and these confuse the concept of a K_R curve. At intermediate toughness, major instabilities can occur which do not appear to be well predicted by the K_R curve tangency criterion.

It takes several panel thicknesses for 'plane stress' shear lips to develop. This is not a problem in 1.5mm thick panels, but represents a complicating factor when the panel is 15mm thick. In two of the alloys tested here, the shear lips were so large that they effectively diverted the tear path. The initial ligament dimensions in the test piece also influence shear lip development. If the initial ligament is square, shear lip development is suppressed, and K_{1c} type behaviour occurs in the less toughness alloys. Conversely, an initial ligament several times the plating thickness encourages shear lip development. Pop-ins are thought to be due to the transition from flat to shear fracture.

A good ranking of K_R curve behaviour in the 500mm K_R curve panels was obtained by measuring energy dissipation rate in W = 70mm bend specimens. It can be shown by analysis [28] that the K_R curve shape (decreasing dK_R/da with crack growth), and the instability tangency criterion, are consistent with crack propagation at constant energy dissipation rate. However, this is probably an oversimplification in thick panels. Also energy dissipation rate is a function of plastic zone length and hence test piece geometry. A method of predicting energy dissipation rate in K control (small scale yielding) from fully yielded test pieces is described in [28]. This was used on the present data but was found to under-predict the failure loads in the 500mm wide K_R curve panels.

The same method was used to predict instability in much larger widths (too large to test). These predictions indicate a significant advantage in using tougher alloys. The critical defect length of C433-T351 was predicted to be five times that of 2024-T351.

At its current stage of development the energy dissipation rate method provides a reliable way of ranking the tearing resistance of aircraft alloys and welds at all thicknesses. It can be demonstrated that there is a clear link to the K_R curve concept, and that the instability criterion is self consistent between the two methods. The methodology to scale the energy dissipation rate from a small fully yielded specimen to a structure needs further development and verification; but, on current evidence, the method is conservative compared to K_R curve technology.

2.8.7 Boundary element methods for crack growth under aircraft loading (M H Aliabadi, Queen Mary and Westfield College, University of London; P M Powell, QinetiQ, Farnborough; R A Collins, Airbus UK, Bristol)

This project is a continuation of collaborative work with Queen Mary and Westfield College to develop the boundary element method for shear-deformable plates and shells to allow numerical stress analysis of metallic honeycomb structures. The work contributes to the maintenance of excellence in modelling and design analysis capabilities, through the development of a multi-domain boundary element formulation for the analysis of crack growth in multi-layered structures. The initial application is to the prediction and understanding of the damage tolerance of new lightweight titanium SPFDB (superplastically formed and diffusion bonded) aerospace components, with modelling of the growth of bondline flaws. The formulations will be tested and evaluated against fatigue tests on SPFDB components. The method will then be applied to other aerospace structures, such as lap joints.

Financial support for this project has been obtained from the METEOR Programme, which is joint-funded by Airbus UK and the Department of Trade and Industry.

2.8.8 Design tools for welded aircraft structures (P E Irving, Cranfield University)

A collaborative project with the Universities of Cranfield, Southampton, Open University, together with Airbus, QinetiQ and Alcoa, is investigating fatigue initiation and fatigue crack propagation in welds of high strength aluminium alloys, 2024 T 351 and 7150. The work is funded by EPSRC, with contributions from Airbus and Alcoa.

The weld processes studied are MIG and VPPA (variable polarity plasma arc). For the majority of the work welded coupon samples have been studied, but at the end of the project, welded skin stringer panels will be manufactured and tested.

A feature of the project is that the fatigue crack initiation and propagation behaviour will be related to details of the microstructure, hardness level and residual stress fields. Residual stresses are being measured by the Open University using neutron diffraction. It has been found that residual stresses in the two alloys on the weld line are about 140 MPa, achieving a peak of 180 MPa, at 5mm from the weld centreline in 7150 and 12mm from the weld centreline in 2024, before declining to zero and into compression at distance of about 25mm from the weld centreline. Local hardness measurements in the weld show qualitatively similar behaviour, with a softened zone on the weld line, a rapid increase to a peak at 2-5mm from the weld centre line, a further softened zone around 10mm, and a gradual increase in hardness up to parent plate in regions beyond 20mm from the weld centre line.

Constant ΔK tests on cracks growing from the weld centre line across the weld show an initial rapid increase, more marked in 7150 reaching a peak at 5-7mm followed by a decline to trough at 8-10mm, a rise to a peak at 15mm. In these materials, fatigue crack growth rates appear to be controlled by both tensile residual stresses and the local hardness values. Elimination of the residual stresses by application of a 2% prestrain, results in a great reduction in growth rates. However, in regions of virtually constant residual stress the growth rate changes match hardness fluctuations

In the early stages of crack initiation the University of Southampton have investigated the role of local defects such as pores and interdendritic cavities, as well as the role of hardness. Multiple crack initiation occurs at the defects and crack advance takes place by growth and by crack coalescence. VPPA welds are more resistance to fatigue initiation than MIG by virtue of their lower porosity levels. A model has been developed to predict the development of fatigue damage in the small crack regime [29, 30].

The figures show some examples of residual stress measurements (Figure 27), crack development from pores (Figure 28) and fatigue crack growth rates (Figure 29).

2.8.9 Development of models to predict the fatigue performance of bolted joints (P M Powell, QinetiQ, Farnborough; W Zhu, Airbus UK, Bristol)

This project is a continuation of research into the development of models and methods to predict the fatigue behaviour of mechanically fastened joints. The aim is to produce a set of improved analysis procedures to optimise designs and repairs for maximum fatigue endurance. In addition, the procedures will be used to determine the increases in fatigue endurance resulting from the use of improved fastener systems accurately, and the residual lives and inspection intervals following the repair of fatigue damage. These improvements will result in reduced life cycle costs, increased service lives and increased aircraft availability.

At QinetiQ, three approaches have been compared, *viz.* the boundary element method, the weight function method, and a semi-analytical method (this is described in detail in the following contribution, Section 2.8.10). Although these have been found to be reasonably consistent in predicting fatigue crack growth from interference fit fastener holes, the semi-analytical method is the most promising as a simple, computationally efficient design tool. Work has therefore concentrated on extending this further to consider clearance fit fasteners, the effect of friction on contact stresses at the hole, plasticity effects and cold working. At Airbus UK, a model to predict the accumulation of fatigue damage in bolted joints has been developed, using a maximum shear stress criterion. This model has been compared with those studied at QinetiQ, so as to validate the different approaches.

Financial support for this project has been obtained from the METEOR Programme, which is joint-funded by Airbus UK and the Department of Trade and Industry.

2.8.10 Development of models to predict the fatigue performance of bolted joints (K Brown, K W Man, D B Rayaprolu, A Young and P M Powell, QinetiQ)

Research has been carried out to develop simple, easy to use, models allowing rapid assessment of the performance of mechanically fastened joints, including cold expansion and/or interference fit fasteners [31]. The basis of the development is a semi-analytical model (SAM), which uses an approximation to the contact stresses around the hole circumference from an idealised model of a fastener in an uncracked plate. The SAM has been developed for predicting fatigue crack growth from a high load transfer interference fit fastener hole in a uniformly in-plane loaded hole, including plasticity effects due to hole cold expansion. The SAM results may be obtained very rapidly on a PC.

The SAM calculates stress intensity factors over a range of crack lengths, from which crack growth rates may be estimated using a specified crack growth law. These crack growth rates have been compared with results from experiment and from a two-dimensional boundary element analysis; fatigue lives have also been compared where possible. The agreement with experiment is better for high load transfer specimens than for low load transfer specimens. Cold expansion of the fastener holes results in crack growth through a compressive residual stress field to crack lengths of around 2mm. The lack of a suitable crack growth law prevented comparison of the SAM results with the experimental results in this region, but at longer crack lengths agreement was satisfactory between the SAM and experiment. The SAM in its present form is unsuitable for clearance fit fasteners, as it does not take account of separation between the pin and the plate. Further developments of the SAM could include consideration of secondary bending, clamping forces, friction and clearance fit fasteners. The model could be developed to calculate equivalent initial flaw size for joints and its speed makes it particularly suitable for theoretical parametric studies.

2.8.11 Investigation of the K_{pr} model for load interaction effects in steels and aluminium alloys (P E Irving, Cranfield University)

The K_{PR} concept is a new approach to the modelling of load interaction effects during fatigue crack growth of metallic materials. K_{PR} is the minimum stress intensity above which the crack propagates and may be defined as

$$\Delta K_{eff} = K_{\max} - K_{PR} - \Delta K_T$$

where K_{max} is the maximum stress in the load cycle and ΔK_T is the intrinsic threshold for fatigue crack propagation at a high R ratio. (0.8-0.9). The approach is thus similar to the closure based ΔK_{eff} approaches to load interaction, but is more generic, as it can incorporate modifications to ΔK_{eff} caused by compressive residual stresses ahead of the crack tip, as well as crack closure [see 32, 33, 34].

The value of K_{PR} is dependent on previous loading history of the crack. There are two fundamentally different forms of load transition in any variable amplitude loading history (shown in Figure 30). Cycle dependent transitions are low-high transitions and the value of K_{PR} in the current cycle depends on its size in relation to the previous load cycle. Crack growth dependent transitions are high- low transitions, and K_{PR} depends on the extent of crack growth since the most recent low- high transition.

K_{PR} can be determined experimentally for a range of situations governing these two types of cycle for individual materials. Once measured, the K_{PR} master curves can be used in conjunction with conventional constant amplitude crack growth data for a range of R values in a predictive model which assesses the type of cycle in the load sequence, assigns the appropriate value of K_{PR}, calculates the ΔK_{eff} and adds the relevant growth rate increment da/dN from the growth rate data.

In the work at Cranfield, performed in conjunction with EADS in Munich, K_{PR} master curves were determined for 7010 aluminium and 4340 steel. A computer based model for life prediction was constructed by EADS to incorporate the K_{PR} data, together with constant amplitude da/dN data. Validation testing under the helicopter spectrum Rotarix, and the fixed wing spectrum Falstaff was performed on the two materials, and compared with K_{PR} predictions and also with predictions of AFGROW and FASTRAN models. Agreement of the K_{PR} model on the 4 variants of the Rotarix spectrum with 7010 aluminium was excellent, with errors of 2-20% on life. Other models had larger non-conservative errors. Predictions under Falstaff on this material were less accurate with 50% errors; K_{PR} was still the most accurate. Prediction accuracy on the 4340 steel was less accurate with errors of a factor of two for Rotarix and a factor of 3 for Falstaff. A comprehensive description of the work is contained in [35].

2.8.12 Round robin problem to benchmark fatigue crack growth life prediction capability in the helicopter community (P E Irving, Cranfield University)

A selection of 13 organisations comprising helicopter and fixed wing aircraft manufacturers, operators, research organisations and software houses, participated in a round robin exercise to benchmark fatigue crack growth prediction capability in the helicopter community. The sample geometry is shown in Figure 31. Participants were required to calculate the flight hours to grow a crack from a 2mm quarter circular flaw on the corner of the central lightening hole to a length of 25mm. Material fatigue crack growth data for aluminium 7010 T73651, the relation between crack length and stress intensity, and the service loading spectrum (derived from measurements on a helicopter lift frame) were all provided. Participants could use any software package they wished to calculate the life. A total of 8 were used producing a total of 27 predicted lives using different options within the software packages.

To validate the predicted lives, two samples of identical geometry and material to those set in the problem were tested under the same loading spectrum.

It was found that predicted lives varied from 250 hours to over 12,000 hours. Real experimental lives were 420 and 440 flight hours. Over 60% of the participants were within a factor of 2 of the experimental life. However, more than half of the participants had non-conservative predictions, and in many cases, calculated inspection intervals were longer than the actual component life. In an analysis of the predicted lives it was established that significant contributors to the scatter and error were differences in the curve fits produced by the packages to the supplied data, together with different models for the load interaction effects. See [36] for further details.

2.8.13 Robust models for fatigue crack growth in helicopters (P E Irving, Cranfield University)

Research has continued to establish the reasons for the fatigue crack acceleration effects observed when fatigue cracks are subjected to helicopter spectra. Many helicopter spectra consist of large numbers of small cycles of high mean stress or R ratio, interspersed with a much smaller number of under loads to zero or small tensile loads. The latter arise from manoeuvre loads; the former from rotor cycles. Such spectra are typically found in the lift frame, rotor components and rotor head on helicopters. Under this type of loading, crack acceleration relative to a non-load interaction prediction of a factor of 2-3 was found in titanium 10-2-3 alloy. There has been further work on this material and on [37, 38] aluminium 8090 alloy subjected to simplified two level load spectra consisting of constant amplitude high R ratio cycles with periodic under loads to zero load. This has demonstrated that the size of acceleration effect increases with increasing frequency of under load, reaching a peak of a factor of almost 10 before declining. Investigation of closure behaviour during the cycling has demonstrated that in 8090 crack opening levels in the large low R ratio under loads are reduced by the presence of the small high R cycles, causing an increased ΔK_{eff} and increased growth rate. This effect is most marked in 8090 which at low r ratio constant amplitude loading has a substantial extent of microstructure induced crack closure. In the presence of the small high R ratio cycles this closure is greatly reduced, leading to increased growth rates. Factors controlling the smaller accelerations found in Ti 10-2-3 and 7010 aluminium alloy are continuing.

2.8.14 Damage Tolerance in helicopter structure and dynamic components (B H E Perrett, QinetiQ)

The 'Felix28' international standard helicopter load spectrum has been used to compare Safe Life and Damage Tolerant design methodologies for helicopter components and structure in a QinetiQ research project funded by the MOD under the Applied Research Package. The sensitivity of safe life linear cumulative damage calculations and fracture mechanics based estimations of crack growth life and inspection intervals have been studied with respect to the mathematical description of raw data, the influence of changes in usage and changes in load spectrum shape that represent the application of vibration suppression in helicopter airframe structure. The influence of short crack growth anomalies have been modelled and discussed with reference to flaw tolerant S-N data.

The work highlights inherent difficulties associated with the application of damage tolerance for structure in which damage accumulates under load spectra representing helicopter dynamic components. The work shows that these load spectra produce crack growth estimations having early slow growth periods that are sensitive to the occurrence of low frequency events (and thereby changes in service usage) and abrupt transitions to rapid growth rates that are impossible to support economically by inspection. This is contrasted with crack growth under the FALSTAFF combat aircraft load sequence which, due to the nature of the spectrum shape, is generally less sensitive to usage parameters and produces manageable crack growth rates and inspection regimes.

Further work simulating active vibration suppression technology indicated that this may allow the safe application of damage tolerant design methodology in helicopter airframe structure.

Figure 32 illustrates the sensitivity of crack growth estimation processes to spectrum shape and the accuracy of definition of material crack growth threshold. Crack growth is predicted for a best fit in the threshold region and also for a fit at a level 10% greater - though still within the scatter of data. It can be seen that Spectrum shapes such as those found in fixed wing aircraft (FALSTAFF) are far less sensitive to the description of raw data and far more amenable to supporting inspection based damage control than are helicopter spectra. Figure 33 illustrates the influence that vibration suppression may have. It shows normalised crack growth under FALSTAFF and Felix28 with and without progressively effective vibration suppression applied to Felix28. It can be seen that vibration suppression corresponding to 4 times attenuation can produce crack growth similar in nature to that for fixed wing aircraft.

Recommendations are made [39] to investigate other forms of crack growth modelling and to compare the behaviour described in this work with that produced by laboratory measurement of crack growth under helicopter spectra.

2.8.15 Crack paths (L P Pook, University College London)

As is well known, many engineering structures and components contain cracks or crack-like flaws. It is widely recognised that crack growth under static and fatigue loadings must be considered both in engineering design and in the analysis of service failures. Methods of determining whether or not a crack will grow, and for a fatigue loading, the rate of crack growth, are well established. These methods assume that the crack path is known. At the present state of the art the factors controlling the path taken by a propagating crack are not completely understood. In general, crack paths are difficult to predict, and in practice crack paths in structures are often determined by large scale structural tests. There is a large amount of published information on crack paths, but this is scattered and often incomplete. This book [40] is a state of the art survey, with major themes presented in as unified a manner as is at present possible.

2.8.16 The concept of the Fatigue Damage Map (C A Rodopoulos and E R de los Rios, University of Sheffield, J-H Choi, Hyundai Motor Company, A Levers, Airbus UK/The Royal Academy of Engineering/The Engineering and Physical Science Research Council)

Work [41-46] has focused on the development of a technique able to map the different damage mechanisms that operate during fatigue loading in monolithic materials. Based on the principles of microstructural fracture mechanics, the five different areas that governed fatigue damage from initiation to failure have been identified, i.e. crack arrest; short crack growth (Stage I); stable crack growth (Stage II); unstable crack growth (Stage III); toughness failure and general yielding, Figure 34. The accuracy of the what is known as the Fatigue Damage Map (FDM) has been experimentally proven for a variety of aerospace materials including aluminium alloys, titanium alloys and stainless steels, Figure 35. In the last three years research collaboration between the Hyundai Motor Company and the University of Sheffield modified the FDM to predict stress ratio effects. The benefits from the application of the FDM are: a) material selection using readily available data; b) easy integration with FE code; c) prediction of error by the application of long crack theories to characterise Stage I; d) NDI scheduling; e) minimum requirements for computational power (use of palm-held computers on site); f) crack growth rates at transition stages, etc.

2.8.17 Modelling combined high and low cycle fatigue crack growth in Ti6Al4V alloy (J Byrne, R F Hall and J Ding, University of Portsmouth)

Fatigue crack growth rates have been studied in Ti-6Al-4V aero-engine disc material under the conjoint action of major and minor cycles at room temperature. Overloads have been introduced into the major cycle component of the test sequence, which has been based on ratios of minor to major cycles of 1000 and 10000:1. Systematic increases in the overload, applied prior to the commencement of the minor cycles, clearly demonstrated a diminution of the effect of the minor cycles on crack growth rates. Accompanying the reduction in crack growth rates is an increase in the stress intensity range at which the minor cycles commence to contribute to the crack growth rate. For three minor cycle stress ratios ($R = 0.7$, 0.8 & 0.9) the size of the overload required to negate the effect of the minor cycles has been established. Additionally, threshold values of the same stress ratios have been found and used to predict the onset of minor cycle contribution to crack growth. The result of multiple overloads, and increasing the minor-to-major cycle ratio has been investigated. Modelling to predict crack growth rates and the onset of minor cycle damage has been undertaken using both the Wheeler model and FASTRAN software.

2.9 OTHER ASPECTS OF FATIGUE

2.9.1 Hawk New Development Aircraft engine and structural vibration trial (Mike Gelder, BAE SYSTEMS, Brough)

The Hawk New Development Aircraft (HNDA), ZJ951, is being used to support the clearance trials of the Rolls-Royce Mk951 Adour engine. This engine standard will enter service with the Mk120 Lead In Fighter Trainer (LIFT) Hawk for the South African Air Force (SAAF). In addition to the instrumentation related to the engine trials twelve triaxial accelerometers have been installed to measure structural vibration levels. Nine of the accelerometers measure the vibration levels in the immediate or close vicinity of the engine and the other three measure the vibration at locations close to avionic equipment.

The trial comprises two phases, a UK based flight programme with an interim standard Mk951 engine and a South Africa (SA) based flight programme with the full Mk951 engine standard. The UK phase was undertaken between August and December 2002 and the SA phase will run from April 2003 to the end of the year.

2.9.2 Inclusion of buffet cycles into the Hawk manoeuvre spectrum for fatigue assessment of the wing tip structure (John Hook and Graham Duck, BAE SYSTEMS, Brough).

Traditionally fatigue analyses of the Hawk wing structure have not included the effects of store-induced vibration-driven load cycles. A series of flight trials was conducted on Hawk demonstrator aircraft ZA101 to measure vibration levels at each store/weapon station covering a range of representative aircraft store/weapon configurations. Following a detailed examination of the flight test data, an approach has been developed that allows the quantification and inclusion in the manoeuvre fatigue spectrum of high-energy buffet load cycles.

At the wing tip location, buffet has been characterised using spectral analysis of the in-flight vibration environment measured at three locations within a tip-mounted Sidewinder missile. Buffet cycles are included for the first three major wing modes by calculating equivalent accelerations at the missile. The buffet vibration spectrum is superimposed onto typical buffet manoeuvre conditions. The analysis includes an assessment of the time spent in the buffet environment for specific aircraft usage patterns.

The analysis has been achieved using in-house mainframe-based software to calculate the buffet vibration load spectrum that interfaces with the Hawk FIST (FEM Integrated Spectrum Tool). The Hawk FIST is used to provide the fatigue loading spectrum for any location within the aircraft FEM, using a concatenation of manoeuvre time history files in conjunction with unit applied FEM loading cases. The resulting fatigue loading spectrum can then be analysed using the BAE SYSTEMS local strain tracking software.

2.9.3 Developments in the prediction of acoustic fatigue (M Nash and M Harper-Bourne, QinetiQ Farnborough)

Acoustic fatigue is a phenomenon that has affected many military aircraft since the development of the modern jet engine and is continuing to cause damage to in-service military aircraft today.

Complementary programmes are underway at QinetiQ, aimed at the integrated development and evaluation of appropriate prediction schemes for near-field noise, the structural response to such loading and the resulting fatigue of the structure.

The acoustic modelling has Focused on predicting the intensity and spectra of the jet noise close to the airframe using a semi-empirical model that marries aero-acoustic theory with source location data. This methodology has now been extended to predict the coherence of the noise field. The method has been validated through small scale jet measurements and measurements on a full scale Spey engine.

A new approach to the analysis of the structural response to this loading, which extends the well-known modal analysis method into the non-linear domain, is under development. The extended approach develops modal stiffness terms, which are functions of the mode amplitude. In the case of multiple modes, the nonlinear stiffness terms link the modal equations. These stiffness terms are derived from a series of nonlinear finite element static analyses. The approach has been applied to the analysis of simple structures, from which it is concluded that it shows a good potential for the analysis of acoustic fatigue. The method is currently being extended to improve its capability to model curved and hot structures and to add a loading model representative of the pressure fluctuations produced by a high-speed jet running close to the structure. A loading model for structures placed in a progressive wave tube is already available. The response of the structure, in terms of displacement or stresses, is returned as a time series, from which fatigue usage can be assessed.

The analysis method is being developed as a practical method to rapidly assess structural components without the need for testing and will enable many more design configurations to be considered at an early stage, leading to better-optimised structure, which should be able to meet the desired life. In addition, it could be used to assess modification options to address in-service arisings.

2.9.4 Teardown examination of the VC10 tanker/transport airframe (C Hoyle, BAE SYSTEMS, Chadderton, M J Duffield, QinetiQ)

At the 2001 ICAF National Review, an outline description of the structural teardown programme for the VC10 tanker/transport aircraft as operated by the Royal Air Force (RAF) was presented. Since then the scope of the programme has increased as availed by the rationalisation of the fleet, which rendered a total of seven airframes redundant. All of these airframes will be used to provide samples of particular structural features for detailed examination within the teardown facility operated by QinetiQ at Farnborough. Furthermore, three of them are being targeted for extensive trials work to validate the structural inspection programme applicable to the aircraft remaining in service.

These parallel and complementary activities will provide a substantive basis for the long-term assurance of structural integrity for the subject aircraft. An example of this was considered in the previous ICAF submission wherein the work underway to validate the fatigue life of the major joint at the wing root was described. That work has been successfully completed and the results are summarised in the following paragraphs.

<u>Teardown of Rib ‘O’ Joint</u>

Using sections of the chordwise joint at the so-called Rib ‘O’ at the wing root position from three different aircraft a total of 1106 bolt holes were subject to bolt extraction and inspection using rotating head high frequency eddy current (RHFEC) probes. In many cases damage had been caused to the hole bores by the installation and removal of the bolts in the form of scratches or scores. However, this type of damage was easily distinguishable from defects due to fatigue. As a result only ONE hole was found to contain a defect that could not be attributed to such damage. This defect was found in the outer joint plate at the bolt head end of the hole.

The analysis of this defect confirmed that it was a crack that had propagated under the influence of fatigue loading, having reached a size of 1mm x 2mm. A similar crack (albeit much smaller and undetected by RHFEC) was discovered in an adjacent hole. In both cases the initiation of the cracks may have been influenced by the presence of corrosion. Furthermore, both holes revealed evidence of very small multiple cracks within the bores and, again, these appeared to be associated with surface corrosion effects.

The general condition of the affected joint plate was no different from any of the other samples obtained from the three aircraft. Consequently, it was suggested that the type of cracks evident in the hole bores might be indicative of a widespread phenomenon. To investigate this possibility several more holes were examined for evidence of similar deterioration especially within the main skin planks. This additional work established that there was no evidence of the type of cracking observed in the joint plate present in the critical elements of the joint.

These results were sufficient to validate the fatigue life predictions for the main lower skin joint at the Rib ‘O’ position. Plans for the precautionary refurbishment of the joints by replacing the bolts by oversize equivalents in reworked holes (688 steel taper bolts per aircraft) were rendered unnecessary with consequent savings in maintenance costs and aircraft downtime.

<u>On-Going Teardown Activities</u>

The management of the VC10 teardown programme is being achieved by dividing the whole task into a number of work packages, each of which is dedicated to a particular feature such as the Rib ‘O’ joint discussed above. Of the 36 work packages, TWO have been successfully completed and SEVEN are presently underway. A further FOUR items have been deleted from the programme as the teardown objectives have been satisfied by other means. The remaining items have been prioritised such that the overall timescale targets for each work package can be achieved. These targets are subject to constant review as results emerge and the demands on the programme continue to evolve. Consequently, it must be emphasised that the VC10 teardown programme remains a joint effort between the programme sponsors (namely the VC10 Integrated Project Team of the RAF), the design authority for the aircraft (BAE SYSTEMS at Chadderton) and QinetiQ at Farnborough.

2.9.5 Teardown of ageing aircraft (D M S Taylor and A B Mew, QinetiQ, Farnborough)

Over the last two years teardown inspections have continued to be used by the MOD to help substantiate Aging Aircraft Audits. These audits, carried out on aircraft platforms that have reached at least 50% of one of the design lifing parameters (landings, flying hours fatigue index, etc) are an important part in underwriting the airworthiness and structural integrity of these mature aircraft.

In co-operation with the Design Authority, teardowns have been performed on Hawk, Jaguar Canberra and Harrier, as well as targeted teardown inspections of VC10. The teardowns have been done under funding from the MOD at QinetiQ. These inspections have enabled substantiation of fatigue testing and design analysis. In many cases, the teardown has been able to confirm that damage being seen on the fatigue test is also present in service aircraft

Another factor that teardown is able to provide more than any other activity is how paint finish and sealant are standing up to service life. Corrosion damage and interfaying surface breakdown are difficult to determine while an aircraft remains in service, until it manifests itself by appearing at a joint or panel surface. With teardown, the degradation of joints and surfaces can be accurately observed and hot spots identified for in-service inspections. For example, many of the interfaying surfaces in the Jaguar showed evidence of movement. However, once paint and primer surfaces were removed very little damage was found to the underlying metal surface.

Further corrosion problems found on Jaguar have concerned nitrogen embrittlement of titanium surface caused through the fretting of cadmium plated bolts in close tolerance holes. Further work is under-way to understand the mechanism causing this to occur.

Teardown also gives the opportunity to test NDT techniques. The in-service technique is applied before teardown commences and the results recorded. The item can then be dismantled in detail and the results of the visual examinations compared to the NDT results.

Teardown continues to be seen as seen as a costly exercise. This is understandable. If nothing is found, then a lot of money has been spent for, in the eyes of some, not a lot in return. If something is found, then again expense is incurred in inspecting the fleet and may lead onto costly repair and modification. Trying to drive costs down on teardown is a constant battle. Removal of sealant from fuel tanks is probably the most time consuming, followed by the removal of close tolerance fasteners. Techniques are always being reviewed and other methods tried. However, sometimes it still boils down to the tedious task of manual stripping of sealant and a bigger hammer with fasteners!

Finally, teardown gives one an appreciation of quality of manufacture. Some of the techniques allowed on the older platforms would certainly be frowned upon today. However, some of the “modern methods” observed certainly do not inspire confidence in our quality control systems or in the longevity of these less mature platforms.

The experience and best practice developed over past few years at QinetiQ have been compiled into a report. This report [47], produced under funding from the MOD’s Applied Research Programme, and peer reviewed by the UK’s Military Aircraft Structural Airworthiness Advisory Group, is intended to support the implementation of the MOD’s structural integrity management policy with respect to ageing aircraft.

2.9.6 Automated Fatigue Testing System (Dr E R de los Rios and Dr C A Rodopoulos-University of Sheffield / Servocom Ltd / TBG Solutions Ltd)

Based on the increasing demand for reliable and fast experimental data, especially in the field of fatigue and damage tolerance, the project aims to develop the first fully automated fatigue testing system. The system utilises an advanced crack imaging system able to interrogate a live video feed up to 100 times per second. Crack detection is achieved using a digital image technique known as image masking. According to the technique the initial area of the specimen is masked with a particular greyscale (representative to the material). With the onset of the loading stage, continuous images are then compared to that initial image. The comparison allows the detection of a crack of minimum size 2μm. The software is then able to perform a number of real time measurements including: da/dN, ΔK, crack angle, crack area, crack opening displacement, crack closure, etc. In addition, the data are continuously fed into the waveform generator and servo-controller, thus allowing the programming of load shedding tests for ΔK_{th}. The first commercial unit is expected to come out in 2005.

2.9.7 Thermoelastic evaluation of crack closure (F Diaz, J R Yates and E A Patterson, University of Sheffield)

Thermoelastic Stress Analysis (TSA) is a non-contacting technique which provides full-field stress maps from the surface of structural components by measuring small temperature changes that happen at the surface of the structure as a result of cyclic loading (Figure 36). Moreover, the technique has great potential in the evaluation of crack closure since the stresses are inferred directly from the temperature changes that occur at the component rather than the applied load range. In recent years, the use of TSA in fracture mechanics has been improved with the development of new infrared array detectors. These detectors make it possible to collect data simultaneously from the whole scan area. Consequently, the data acquisition time can be reduced to just a few seconds, ensuring that during the data collection period the crack is not growing.

Work, [48] has been conducted on steel specimens with a central weld (see Figure 37) using TSA. The purpose was to study the different crack closure mechanisms and the influence of residual stresses due to welding during the fatigue life. The equipment employed for this research (DeltaTherm 1500) consisted of an infrared camera based on an In-Sb infrared array detector cooled by a Stirling engine. This system makes it possible to collect thermoelastic data simultaneously form 81,920 points with a maximum thermal resolution of 2 mK and a maximum spatial resolution of 50 μm

Fatigue experiments were conducted at different R-ratios. The test specimens employed were single edge notched specimens with an initial 4mm notch located at the central part of the weld (see Figure 37 for details). The tests were performed at a frequency of 10 Hz. During the fatigue tests thermoelastic pictures were captured at different crack lengths and numbers of cycles. At the same time, the crack length was monitored from the rear part of the specimen using a Vernier microscope. Subsequently, thermoelastic images were processed for the evaluation of the Stress Intensity Factor (SIF). Experimental data points were collected from the region dominated by the crack and fitted to a mathematical model describing the stress field in the vicinity of the crack tip. The stress intensity factor was inferred for the resultant fitted expression. Results show that the technique is able not only to detect the crack closure effect for tests performed at different R-ratio but also the influence on fatigue life of residual stresses due to welding. Work is also in progress for monitoring the fatigue crack path directly from thermoelastic images [49].

2.9.8 Investigation into the influence of Pitting Corrosion on the fatigue strength of 7010 aluminium alloy and an evaluation of Equivalent Initial Flaw Sizes (S H Spence BAE SYSTEMS, Warton)

With reducing defence budgets leading to increasing demands for extended lives and in-service dates of military airframes, corrosion is becoming an increasing cost burden and threat to structural integrity. BAE SYSTEMS is undertaking studies to provide data and methods to enable enhanced corrosion and fatigue management programmes including a move away from a rigid find-it-fix-it approach to corrosion to a more flexible detect-and-manage approach. Part of this programme of work includes an investigation into the influence of environment on crack growth rates that will be reported further by QinetiQ elsewhere within the UK National Review (see Section 2.9.9). Another part of the programme, that is the subject of this report, is the influence that prior pitting corrosion damage has on fatigue. This programme of work is a collaborative programme, funded by BAE SYSTEMS and includes partners from: BAE SYSTEMS in Australia, DSTO and CSIRO (also both in Australia) and the University of Wales Swansea in the UK.

The objectives of this programme are to characterise base line fatigue behaviour in the aluminium alloy, 7010 T7651, in the as-machined and anodised and primed condition (representative of initial build conditions) and then contrast with test data from coupons containing corrosion pits. In addition to the testing of dog bone fatigue samples, tensile, toughness and crack growth properties were also characterised.

A pitting protocol was developed by CSIRO and DSTO in order to be able to generate controlled pitting corrosion on a small area on one surface of dog bone coupons in an accelerated manner. This allowed in excess of 300 test coupons to be exposed to generate pitting corrosion. The objective of the programme was not just to compare the fatigue endurance for the pitted and non-pitted coupons but to provide a comprehensive study to assess whether or not the Equivalent Initial Flaw Size (EIFS) approach can be extended to account for corrosion pitting damage. One of the key things that sets this study aside from others in this area is the large number of coupons that were tested under the same conditions. To characterise the influence of parameters such as load ratio and stress level on the value and distribution of the EIFS, 4 stress levels were evaluated at each of 3 load ratios and 25 repeats were tested at each condition. All testing was conducted at laboratory temperature and a relative humidity level of at least 90% in order to remove environmental acceleration on crack growth rates as a variable.

The programme is still under way and the EIFS values are yet to be calculated and studied. However, the majority of testing is complete. The testing also includes some variable amplitude loading, FALSTAFF, in order to study the effects of variable amplitude on EIFS values and evaluate the effectiveness of our modelling. Pit characteristics are being measured and cross related to EIFS values in order to identify any correlation of pit dimensions with EIFS.

Crack growth rates measured from thumbnail cracks in dog bone coupons where marker band loading has been applied compare favourably with crack growth rates from through cracks in centre crack tension coupons. Fatigue lives for anodised coupons compared well with those from as machined coupons whilst lives for the pitted coupons were significantly lower, as expected. Cross laboratory comparisons have also been made and the correlation was excellent.

The programme is now in the final analysis stage to determine EIFS values and undertake the statistical analyses and data interpretation required.

2.9.9 Corrosion fatigue in 7000 series alloys (M W Squibb, P Poole, M Sumner, QinetiQ; S Spence, BAE SYSTEMS)

A joint QinetiQ/BAE SYSTEMS test programme has been carried out, to investigate the effect of a corrosive environment on the fatigue crack growth rate in aluminium 7010 alloy. Squibb, *et al.*, describe the results of this investigation in [50], in which the fatigue crack growth data obtained in the second phase of the test programme are described and discussed. Constant amplitude fatigue test data are presented which show that, for 7010-T6 and 7010-T7651 alloys, the rate of fatigue crack growth may be up to an order of magnitude higher in 3.5%NaCl solution than in 25^{o}C/30%RH. The magnitude of this effect was reduced when the load cycle frequency was increased from 1Hz to 10Hz. Slightly higher rates of crack growth were observed for tests carried out in 25^{o}C/95%RH rather than 25^{o}C/30%RH. A limited number of tests under FALSTAFF loading indicated that when testing was carried out in 3.5%NaCl solution rather than 25^{o}C/30%RH, the fatigue life of 7010-T7651 was reduced by a factor of about 3, compared with a factor of about 4 for 7010-T6.In general, the effect of environment on the rate of fatigue crack growth was similar for 7010-T6 and 7010-T7651 alloys. There was no evidence that stress corrosion cracking contributed to environmentally enhanced crack growth in either alloy. If the resistance to stress corrosion cracking of 7010-T7651 alloy is assumed to be superior to that of 7010-T6 alloy, then the fatigue crack growth data obtained in the present work invalidates the proposal that the effect of alloy type on environmental enhancement of fatigue crack growth may be estimated from a generic fatigue crack growth law for by applying factors based on resistance to stress corrosion cracking. In contrast, it should be possible to estimate the effect of different environments on fatigue crack growth by applying factors based on the severity of the environment to a fatigue crack growth law (derived from data obtained by testing in an ambient laboratory environment).

2.9.10 Investigation into the influence of UV degradation on Stretched Acrylic (S H Spence BAE SYSTEMS, Warton)

It has become evident in recent years that transparencies, such as a/c canopies, can remain in service for extended periods of time. During this time, the canopy is exposed to moisture and ultraviolet (UV) radiation. It is known that UV radiation can degrade both the optical and mechanical properties of polymeric materials. BAE SYSTEMS has been conducting a study to characterise the influence of exposure to UV radiation to allow any degradation effects to be taken into account for design and support activities.

The study is nearly complete and further work is planned to determine the differences for stretched and cast materials. The programme involved fatigue endurance, crack growth and fracture toughness testing for a stretched acrylic, Rohm 249 in three conditions and at three temperatures. As-received sheet was employed as the control condition and a QUV chamber was employed to condition coupons, in an accelerated manner, to simulate five and ten years of in-service exposure. Since early tests revealed no observable influence of five years simulated exposure, efforts were focused on the ten year condition. Temperatures of –30, 21 and 75°C were employed and moisture content was measured before and after the fatigue and crack growth tests. The fatigue endurance tests were conducted on standard dog bone coupon but a scratch was introduced across half the gauge length in order to identify any increased notch sensitivity from the exposure to UV radiation.

Fracture toughness increases with increasing temperature but no influence of UV exposure was observed. This is, perhaps, not unexpected since it is anticipated that UV degradation would be a surface dominated effect and the toughness tests were for through cracks. Crack growth rates were virtually indistinguishable for the –30 and 20°C conditions and no influence of UV exposure was observed for the through crack configuration. At high stress intensity factor ranges, the crack growth rates at 75°C are lower than for the lower temperatures studied, probably due to the elevated toughness levels and increased plasticity levels. A corner crack configuration is currently under final investigation, where the cracks are initiated on the exposed side of the coupon. Good resistance to scratches has been observed for the dog bone coupons tested to date. Fatigue crack growth behaviour was in excellent agreement with data from earlier work on stretched Polycast 76 material.

2.9.11 Safety factors and risk in fatigue substantiation of helicopter components (P E Irving, Cranfield University)

A study has been made of the variability in helicopter usage, and the variability in fatigue damage of military helicopters. This has been combined with an analysis and experimental investigation of the errors and assumptions of design safety factors during fatigue substantiation of helicopter components. Major conclusions of the work are that, while use of the Goodman mean stress correction results in conservative predictions of the influence of mean stress, the assumption that small rotor cycles are non-damaging and can be neglected is non conservative. For the spectra investigated so far, the two errors approximately cancel. However, whether this remains true will be dependent on details of the load spectrum.

The usage and fatigue damage variability investigation has revealed that, whilst there is substantial variability in damage for individual manoeuvres and the incidence of those manoeuvres, the effect that this has on the damage variability depends on how often the usage type is changed. Frequent changes of usage will lead to reduced variability of damage in the helicopter fleet. General usage and damage levels are less than those assumed in design, leading to the situation that maintenance credits are possible with usage monitoring.

2.9.12 Manual One Way Assembly (M Stalley and M R Parry, Airbus UK, Bristol)

A significant proportion of the production time of wing assembly is attributed to the breaking down of the wing structure during the wing assembly cycle, to allow for the removal of hole burrs and swarf, application of sealant, and to assist inspection. The breakdown of the wing structure disrupts the build cycle and is time-consuming. If it can be shown that the structure can be drilled off to a standard that meets the design requirements without being broken down, then significant cost benefit can be gained.

This research programme is being carried out to investigate whether a manual one-way assembly of certain wing areas is feasible, thus removing the necessity to deburr and to clean out swarf between sealed parts. This is being undertaken in part though an assessment of the fatigue lives of high load transfer joint specimens manufactured by a one way assembly process in comparison to those of conventionally assembled high load transfer joints. Interim results suggest that controlling the gap between the various structural elements during drilling is critical. If the gap is equal to the maximum design allowable for conventional joints then a detrimental effect in terms of fatigue life is seen, resulting from swarf ingress at the interface. If the interface gap is restricted to 50% of the maximum design allowable for conventional joints, no swarf ingress at the joint interface is seen and no effect on the fatigue life of interference fit joints of the manual one way assembly process is seen.

The manual one way assembly process is being further validated through the manufacture and test of sub-element coupons designed to be more representative of real aircraft structure, as well as through the implementation on the 'development wing' of the A380 full scale fatigue test.

2.9.13 Bonded Composite Patch Repair of Corrosion Damage (P Poole, K Brown, K A Greedus and A Young, QinetiQ; H G Hutchinson, SSG RAF Wyton)

Theoretical and experimental research has been carried out to assess the effectiveness of bonded composite patches for the repair of corroded aluminium alloy structures. The research involved the following types of specimen:

Single edge notch (SEN) specimens, machined from 3.25mm thick, 7475-T7651 aluminium alloy sheet. Rectangular test panels, machined from 3.19mm thick 2024-T351 and 7075-T651 aluminium alloy sheets. Some of the panels contained a central open hole, others did not.

The fatigue life of SEN specimens was reduced by up to 61% due to pitting corrosion induced at the notch by exposure to neutral salt spray. When carbon/epoxy composite patches were bonded to the outer surfaces of corroded, two-sheet specimens, fatigue lives were increased well beyond those of uncorroded, unpatched specimens. The life improvement factor due to patching was greater than 24. The rate of fatigue crack growth was reduced by a factor of at least 10 due to patching; the magnitude of this effect increased as crack length increased. A three-dimensional boundary element/finite element model was used to predict the effect of patching on the stress intensity factor range ΔK and the R-ratio. For patched specimens, the rate of fatigue crack growth was determined from theoretical values of ΔK^P and R^P and a crack growth rate law derived from experimental data. The predicted reduction in crack growth rate due to patching became more pronounced as crack length increased and maximum stress (for R=0.1 loading) increased. The predicted crack growth rates were in very good agreement with experimental data. The number of cycles to initiate a small crack (e.g. 0.127mm) was calculated from the measured number of cycles to failure and the predicted number of cycles to grow a crack from 0.127mm to failure. It was shown that the number of cycles to initiate a 0.127mm crack was much greater for patched than unpatched specimens, and that the local stress range was significantly lower for patched specimens. Thus, the effectiveness of bonded patches in increasing the life of corroded SEN specimens was explained by a combination of an increase in the number of cycles to initiate fatigue cracks and a decrease in the rate of crack growth.

The fatigue lives of 2024-T351 and 7075-T651 open hole specimens were reduced by a factor of about 4 due to corrosion damage resulting from exposure to EXCO solution for up to 190 hours. Single side patching increased the lives of corroded open hole specimens by factors of approximately 11 and 14 for 2024-T351 and 7075-T651, respectively. When a corroded 7075-T651 specimen was blended and filled (with an aluminium loaded epoxy resin) before patching, fatigue life was increased by a factor of 18, compared with 14 in the absence of blending and filling. In the case of plain specimens with central corrosion damage, bonded patches (with/without blending) resulted in significant increases in fatigue life. The increases in fatigue life resulting from patching of corroded open hole specimens were explained in terms of increased number of cycles to initiate cracking and decreased rates of crack growth. Numerical results for repairs involving blending, filling and patching, showed that if the depth of the blend is increased, then the number of cycles to initiate fatigue cracking is reduced and the rate of crack growth is increased. Thus, for optimum repair effectiveness, minimum depth of blending should be used. The numerical model predicted that the use of alternative fillers with higher values of Young's modulus should result in additional improvements in repair effectiveness.

The research demonstrated that, in general, adhesively bonded composite patches offer an extremely effective technique for the repair of corrosion damaged aluminium alloy structures and the restoration of fatigue performance. For some repairs, patching directly over corrosion damage may suffice but, for other repairs, blending, filling and patching may be appropriate.

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FIGURE 1 *Concept Analyst* screen shot showing interacting stress concentrations

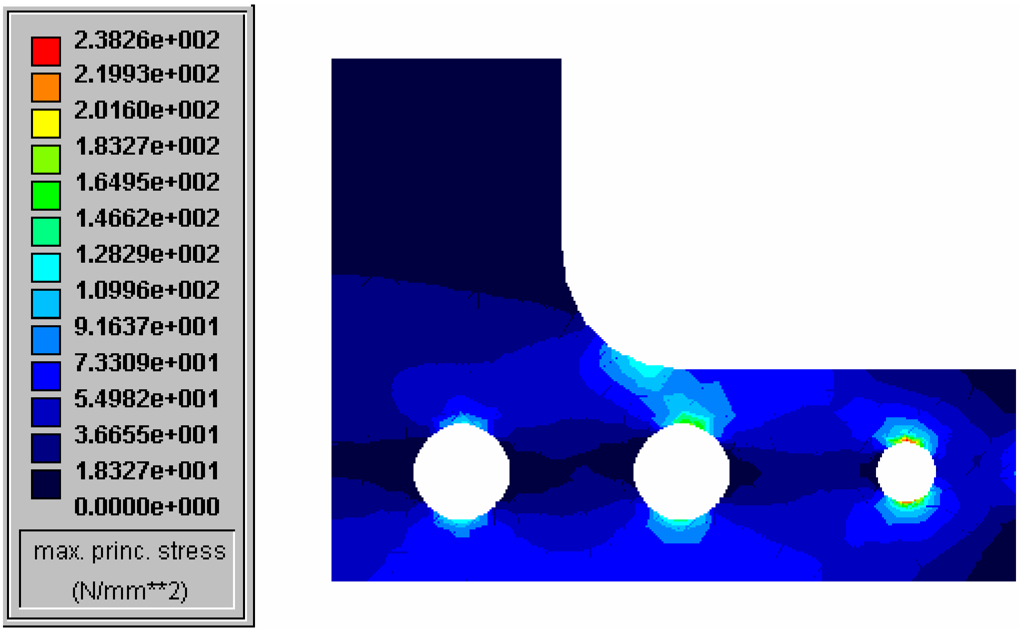


FIGURE 2 Example interface windows for the SCONES application

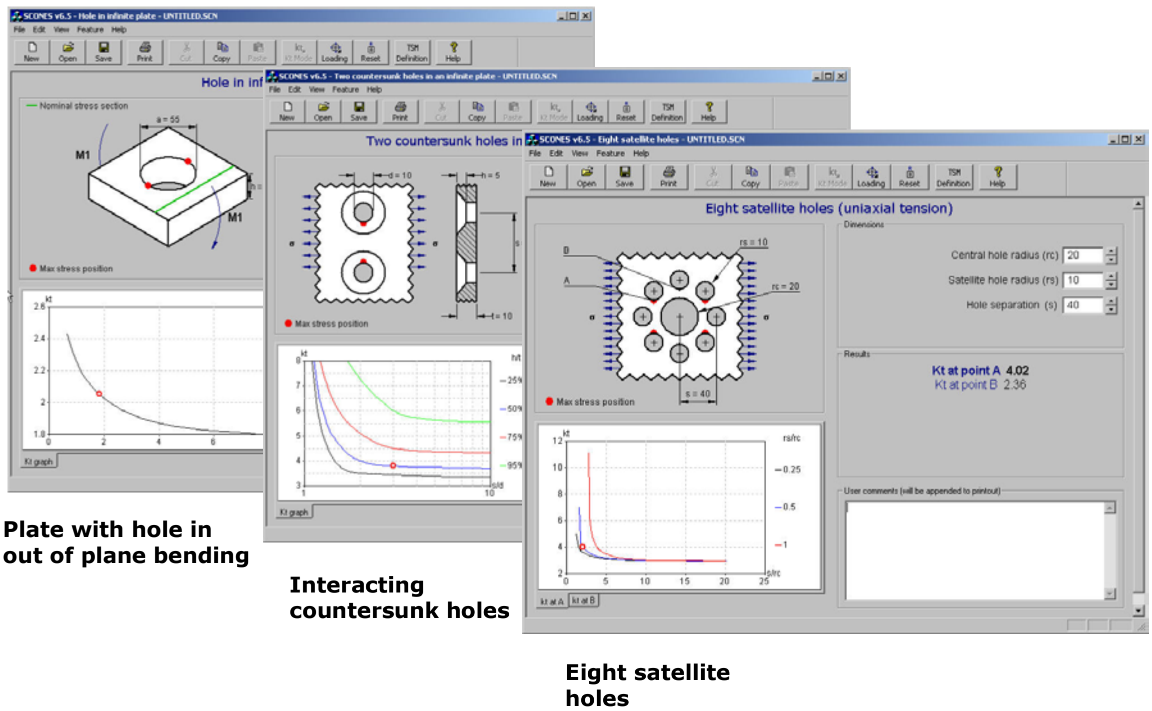


FIGURE 3 C23B/B+ Aircraft Clearance for Operation on Gravel Runways

FIGURE 4 View of front spar wing link area

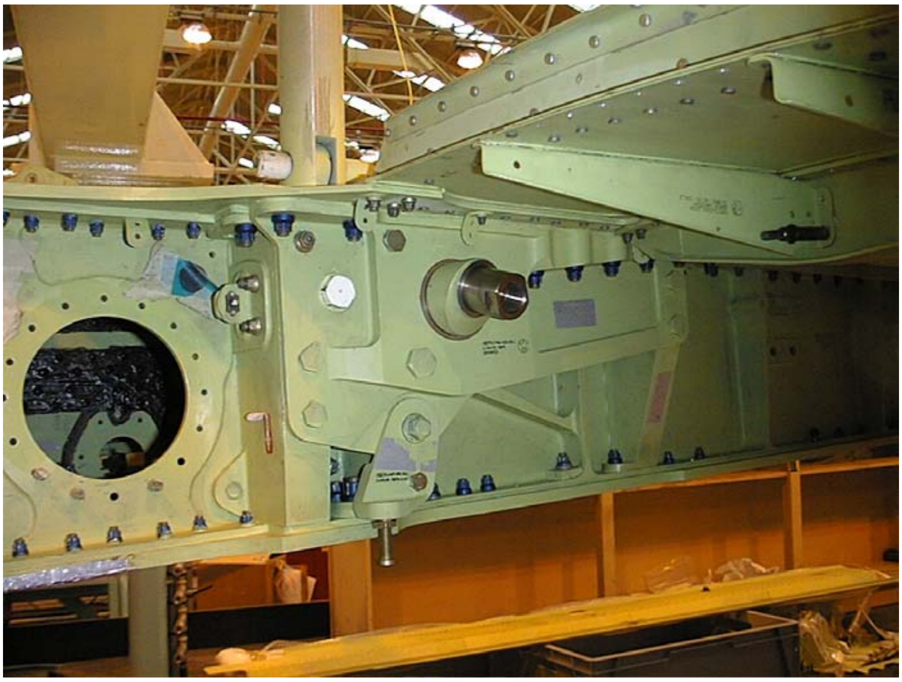


FIGURE 5 View on under side of wing at wing link fitting

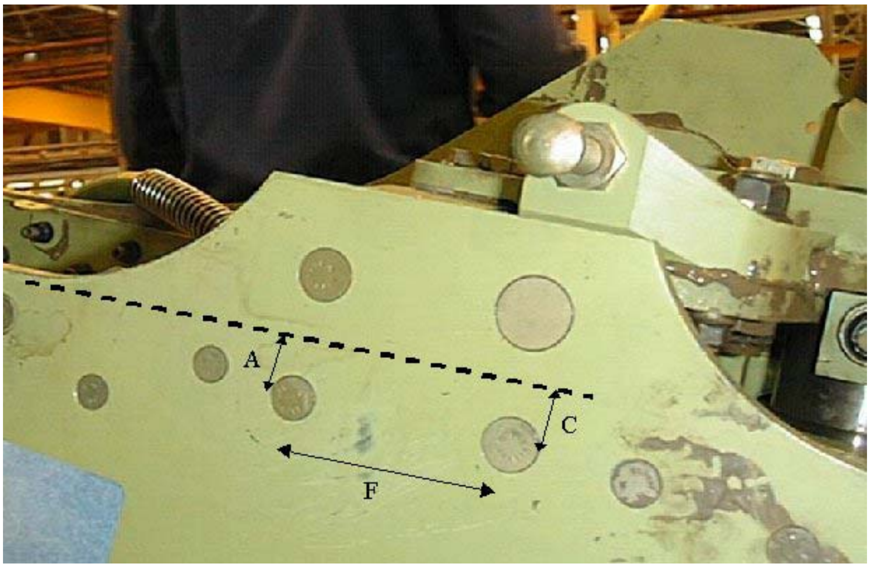


FIGURE 6 View of short edge distance test specimen

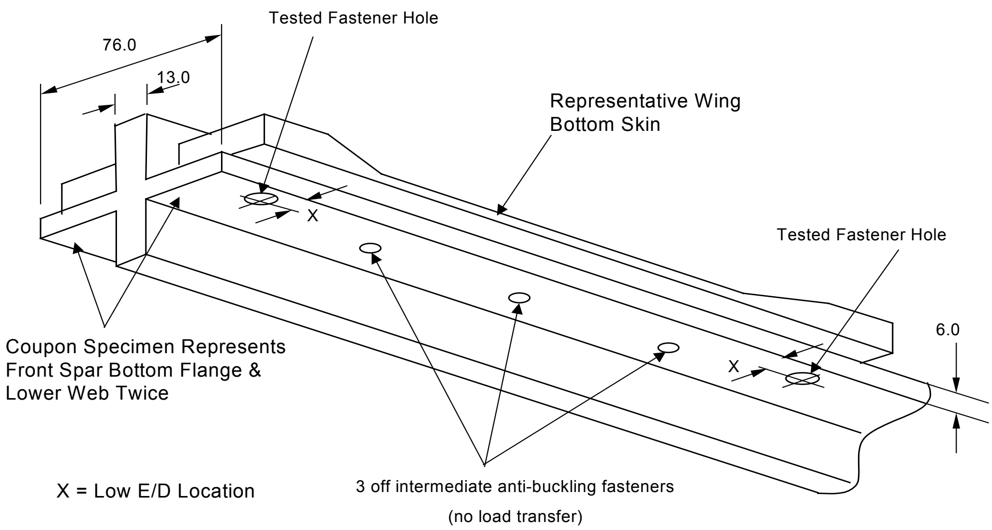


FIGURE 7 Fatigue lives of part-life cold expanded fastener holes: Low load transfer joint specimen under FALSTAFF spectrum (LIF = Life Improvement Factor, Cx = Cold Expansion)

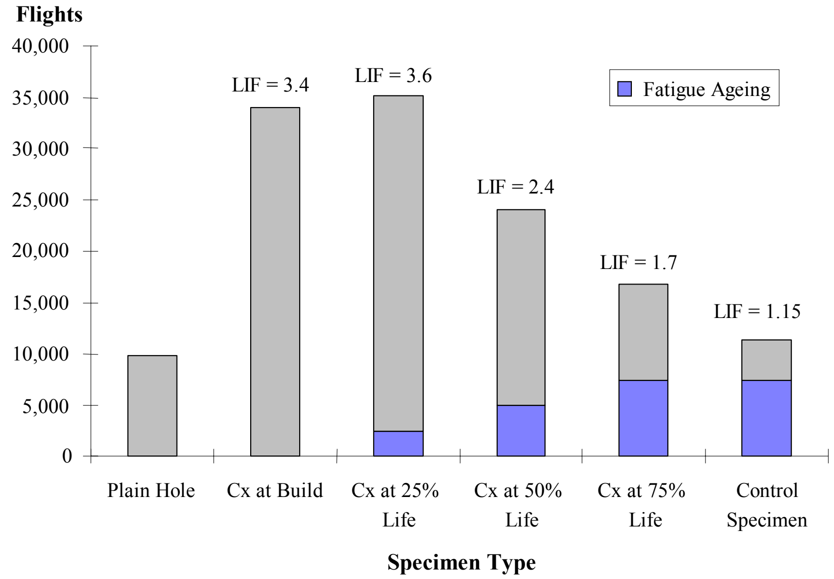


FIGURE 8 Crack scenarios considered in current investigation

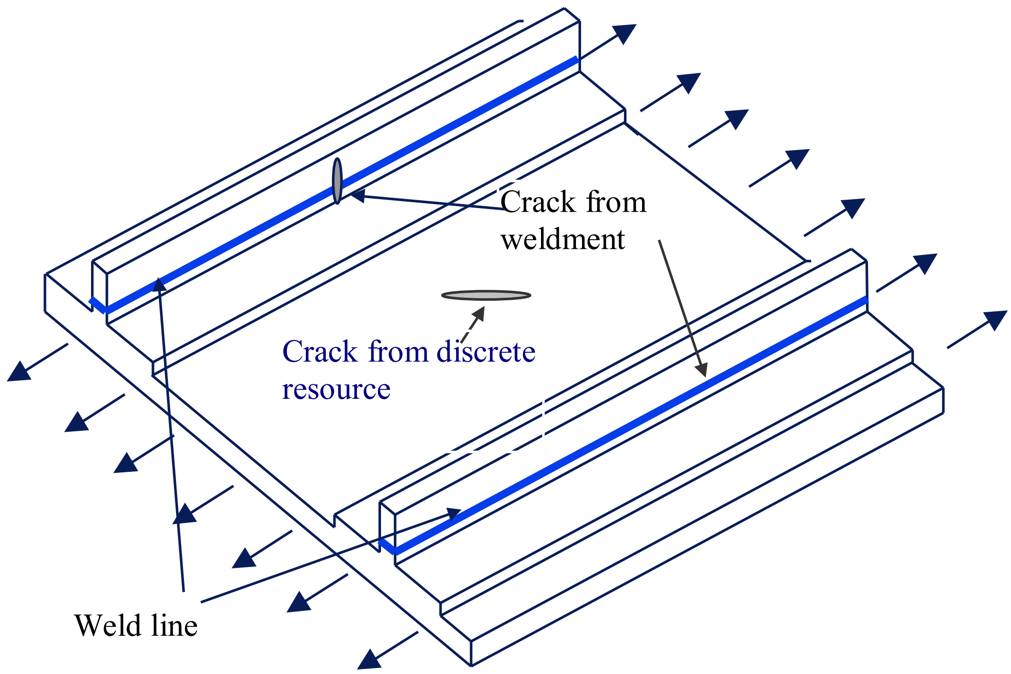


FIGURE 9 Evidence of ductility loss close to the peened surface in the case of 2024-T351 peened at 4A intensity

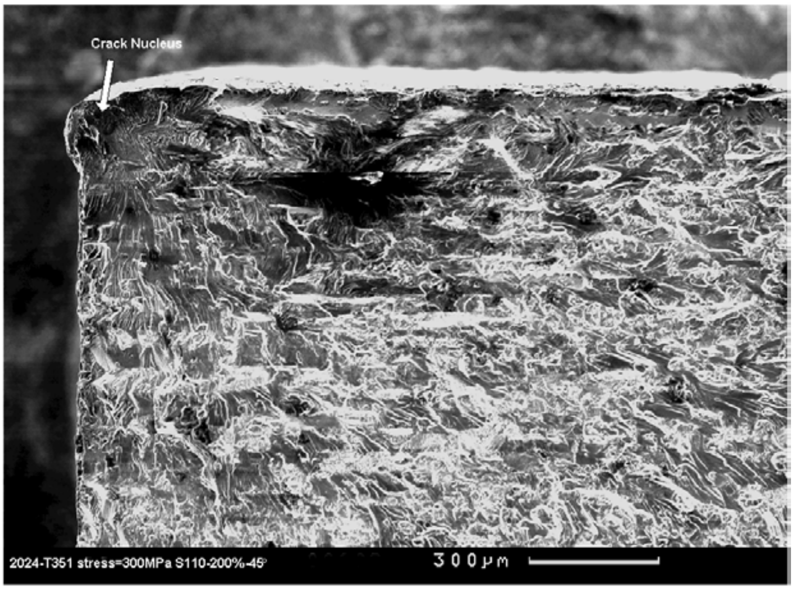


FIGURE 10 S-N curves for 2024-T351 with 6 different surface finishes: Mirror effect; EDM; Shot Peened 4A; Laser shock Peened at 10GW/cm^2 2 passes; Laser shock Peened at 10GW/cm^2 3 passes and dual treatment

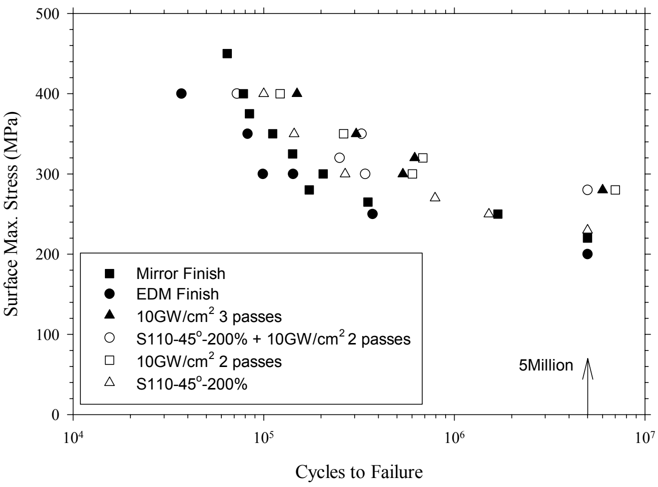


FIGURE 11 Effect of test temperature on rate of fatigue crack growth for specimens repaired with 20 ply, pre-cured patches (R=0.1, S_{max}=65MPa loading)

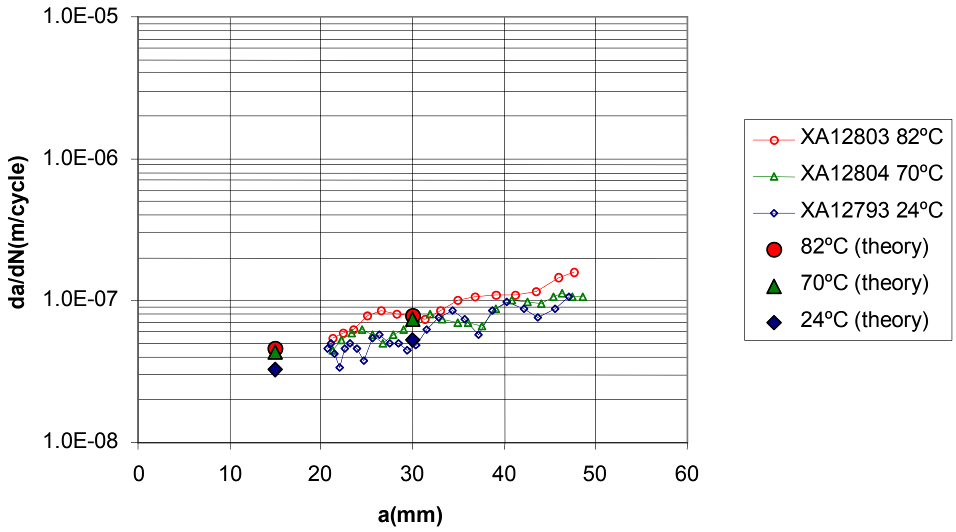


FIGURE 12 Global Express Horizontal Stabilizer Metallic Centre Box Subcomponent DADT Test

FIGURE 13 Learjet 45 wing and fuselage test article (designated TA05)

FIGURE 14 New landing gear test facility at Messier-Dowty Gloucester

FIGURE 15 A340 landing gear fatigue test in the new test facility

FIGURE 16 Cross plot actual against predicted strain – outer wing – outside training set – alternative aircraft

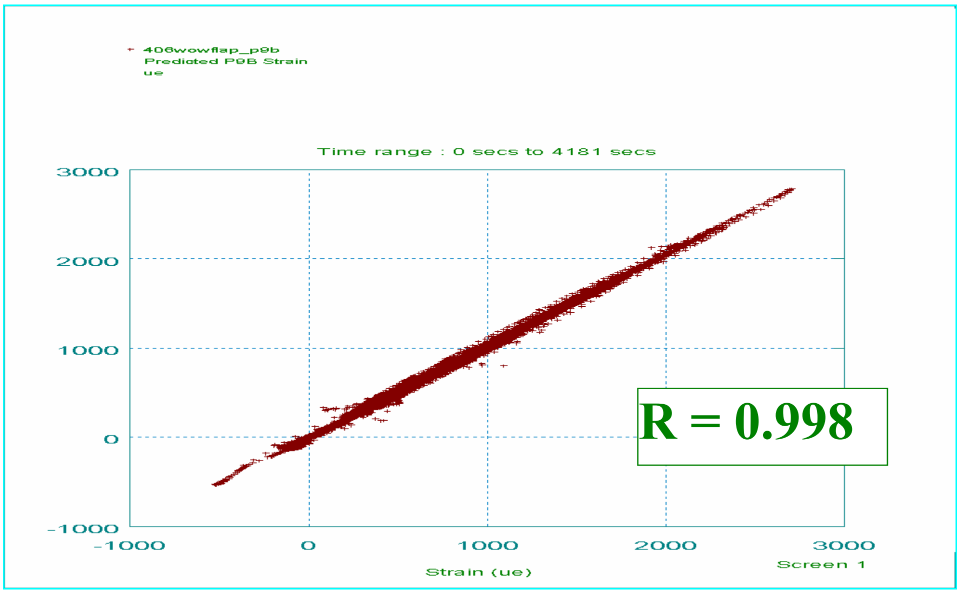


FIGURE 17 Comparison between actual and predicted fatigue damage by flight for the inner wing

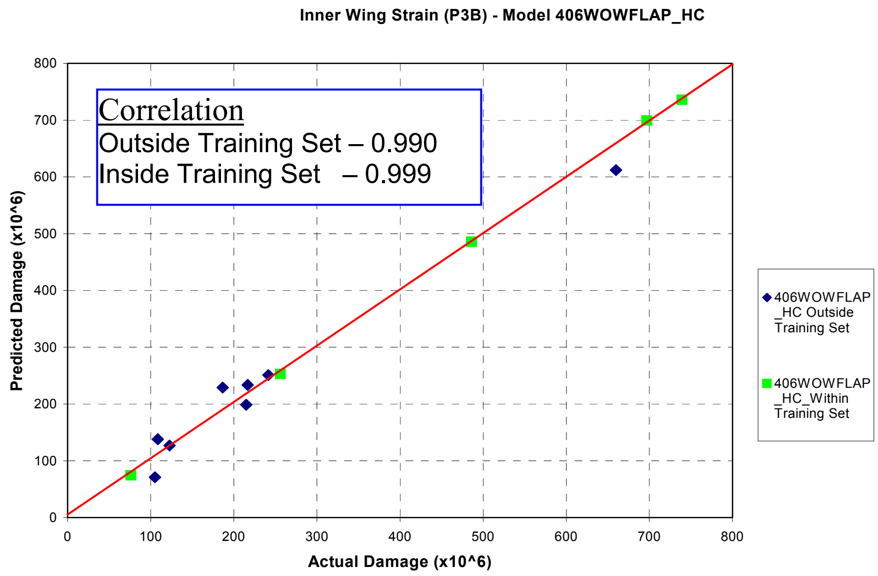


FIGURE 18 Comparison between actual and predicted fatigue damage by flight for the outer wing

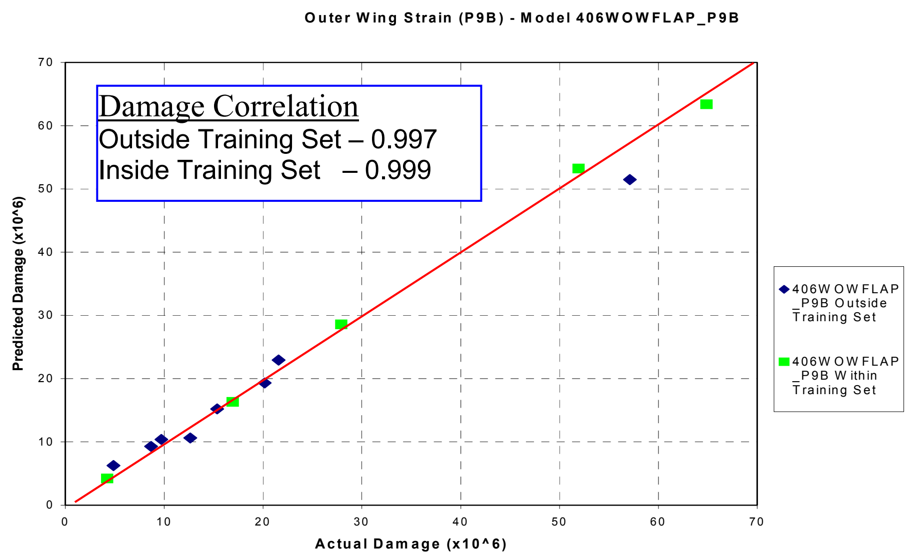


FIGURE 19 Examples of Smiths' Diagnostic Models

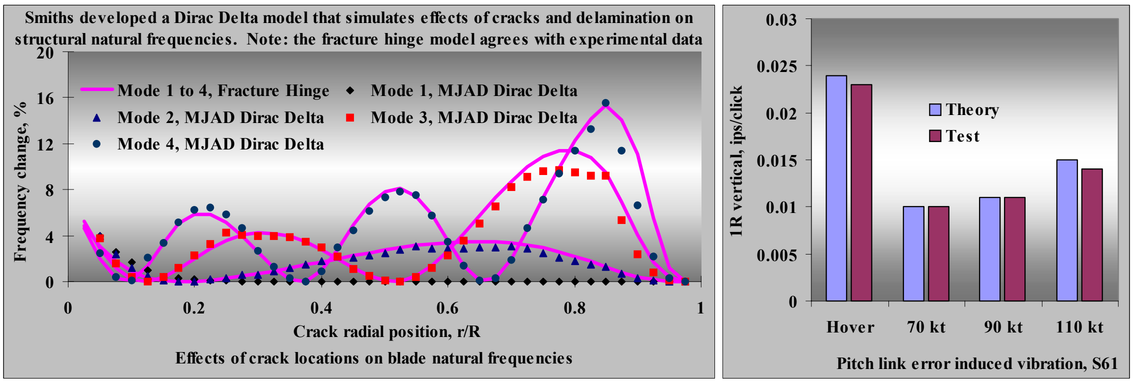


FIGURE 20 Diagnostic/Prognostic Information synthesised from flight data

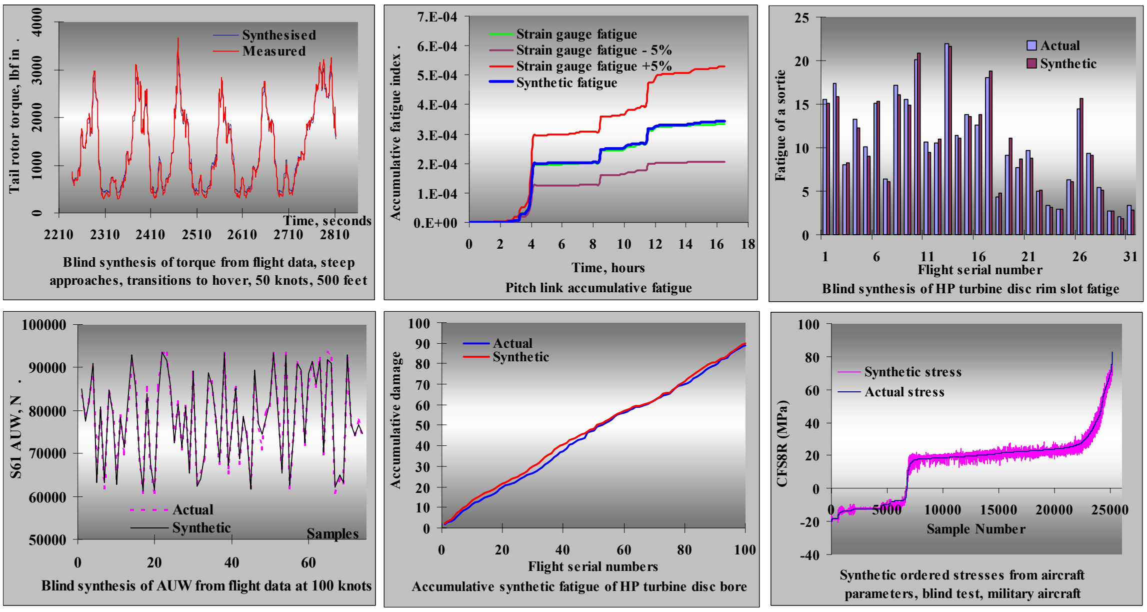


FIGURE 21 Synthetic Fatigue at the Wing-to-Fuselage Bracket and Fin Locations

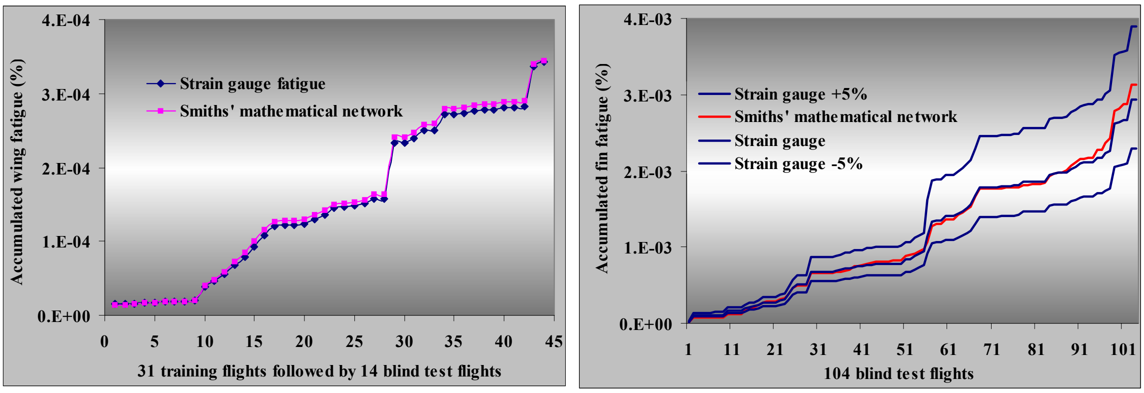


FIGURE 22 Synthetic Wing and Fin Strains

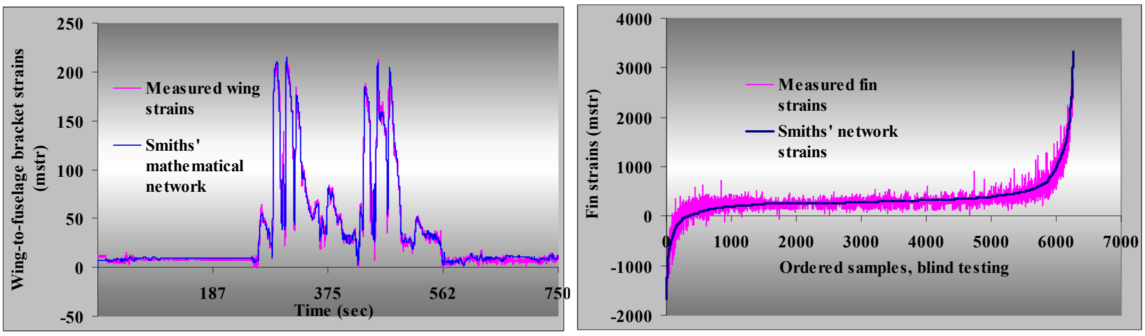


FIGURE 23 Effects of Dynamic Rare Events on Fin Strains

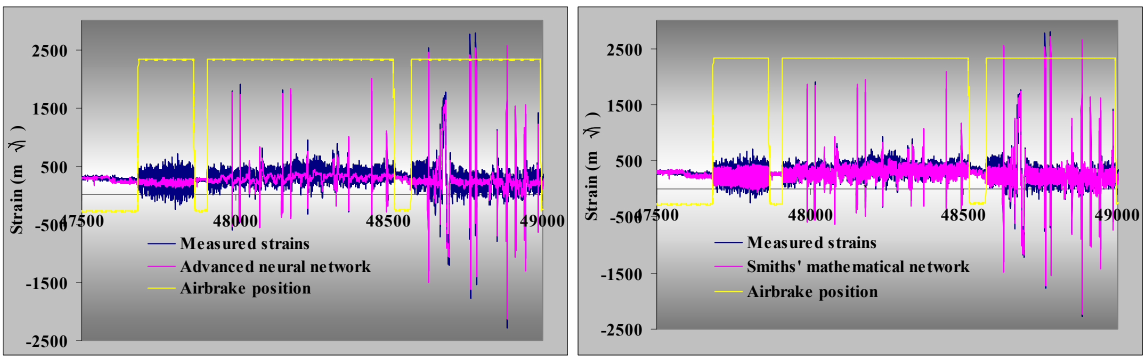


FIGURE 24 Fatigue cycles as a function of applied strain for IM7/8552 panel

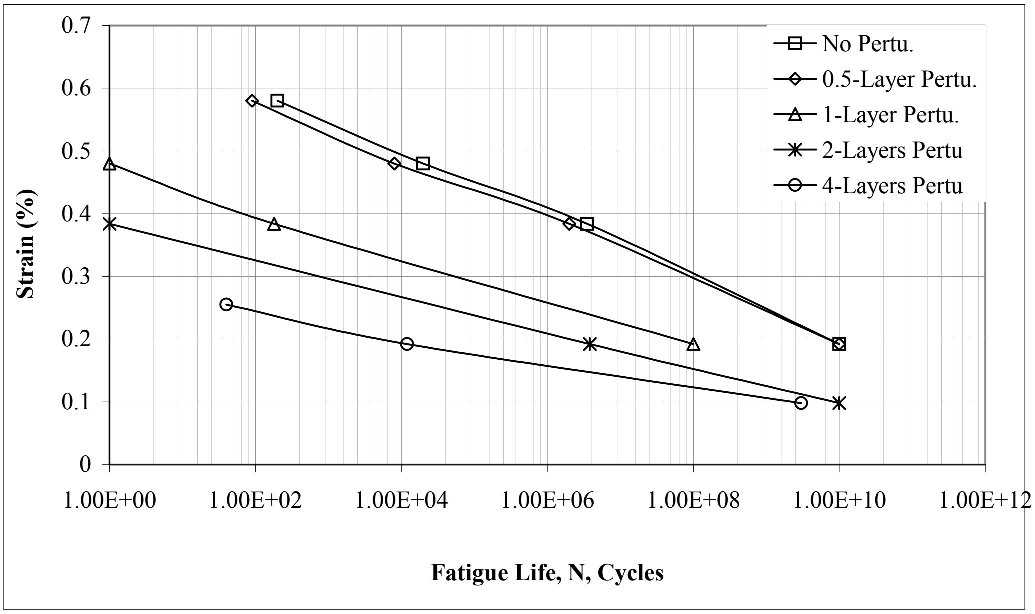


FIGURE 25 (AG20) Micrograph illustrating variation in striation spacing between adjacent fibre imprints T800/5245 (mode I + II) x3100

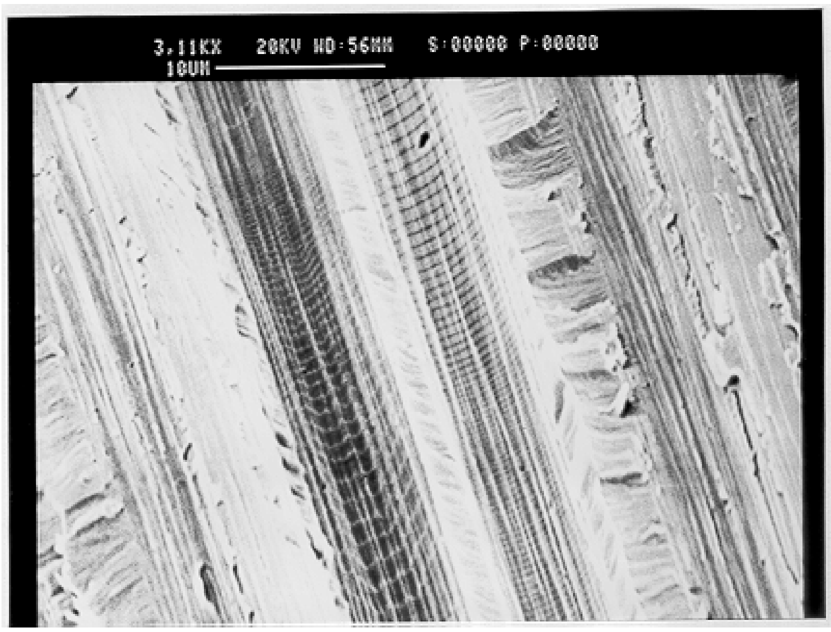


FIGURE 26 (AG20) Striations within the resin between fibres T800/5245 (mode I + II) x9K

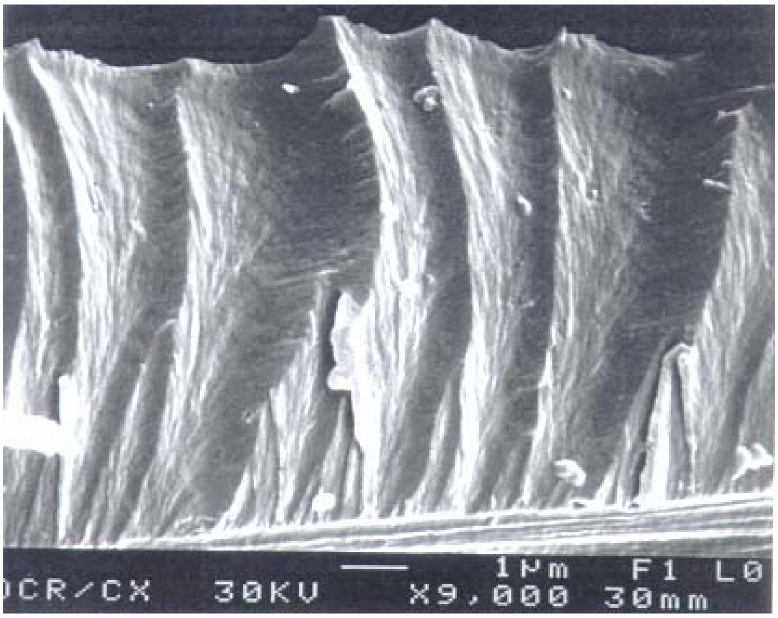
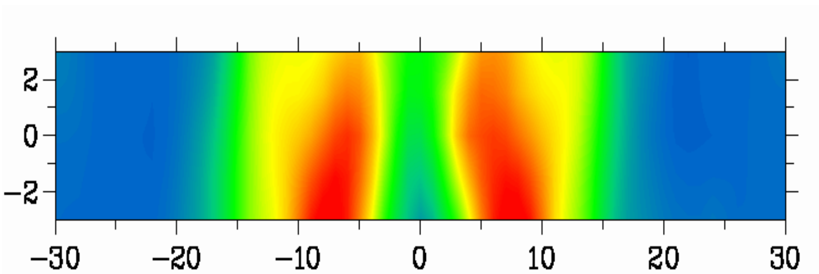


FIGURE 27 Stress field in MIG welded 2024 aluminium alloy measured by Open University using Neutron diffraction
Fehler!



Fehler!

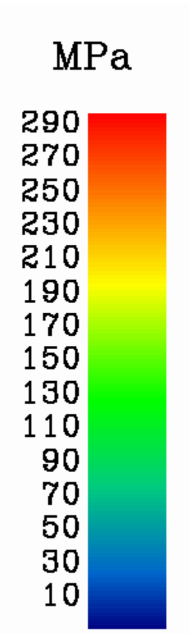
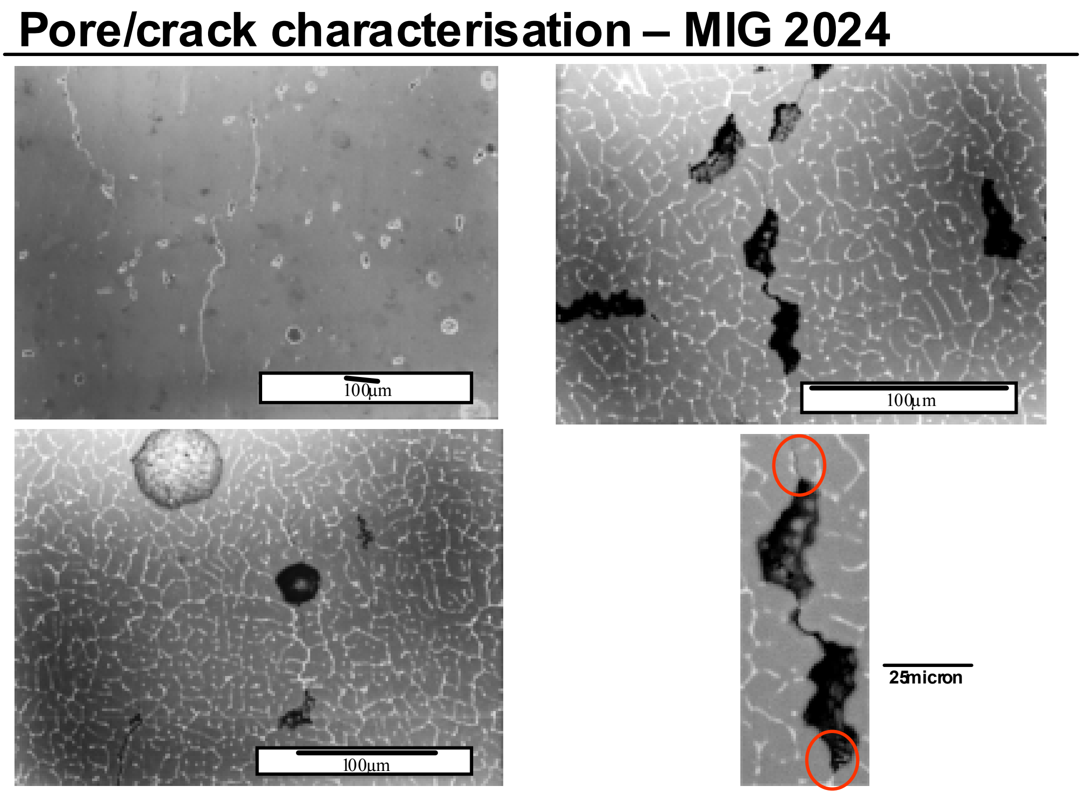


FIGURE 28 Fatigue crack initiators in MIG welded 2024 T 351, measured by Southampton University
Fehler!



Fehler!

FIGURE 29 Fatigue crack growth in welds at long and short crack sizes, measured by Southampton University

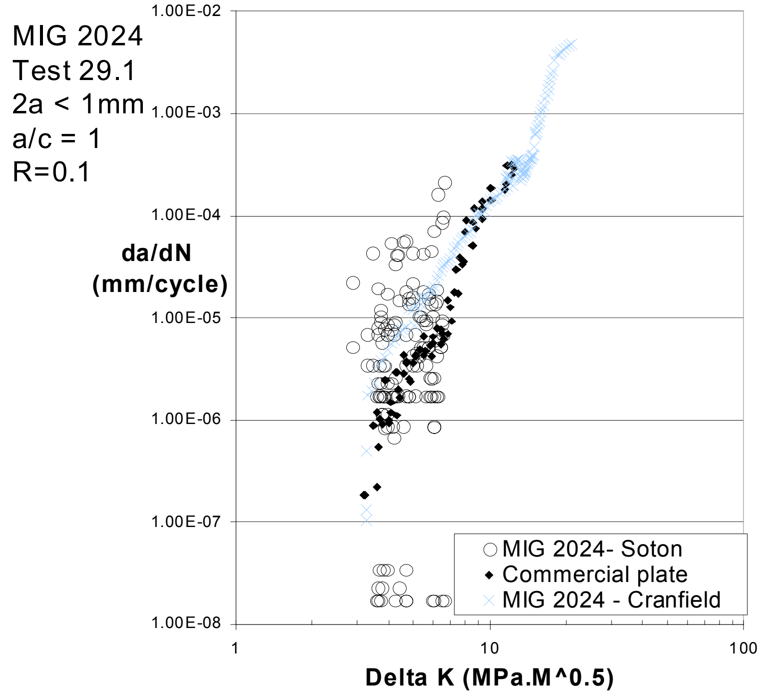


FIGURE 30 The two types of load transition in the K_{PR} model

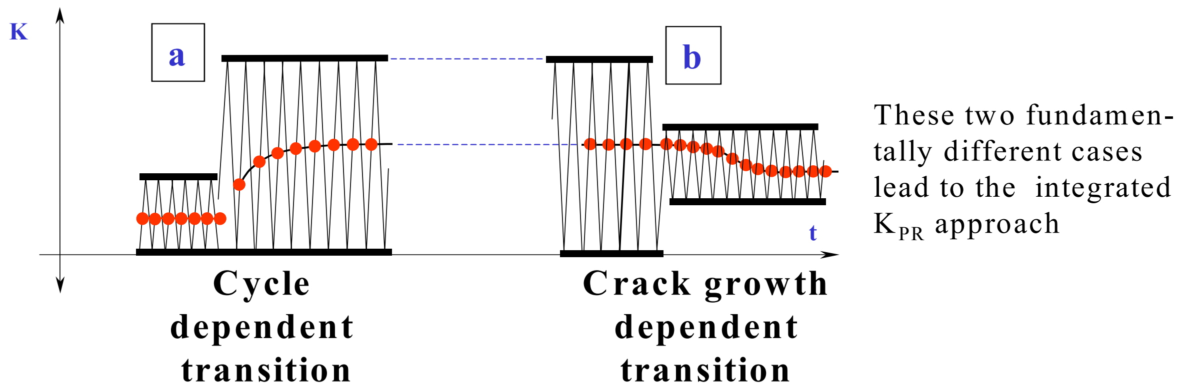


FIGURE 31 Geometry of the round robin test sale. Defect was located on the corner of the central lightening hole

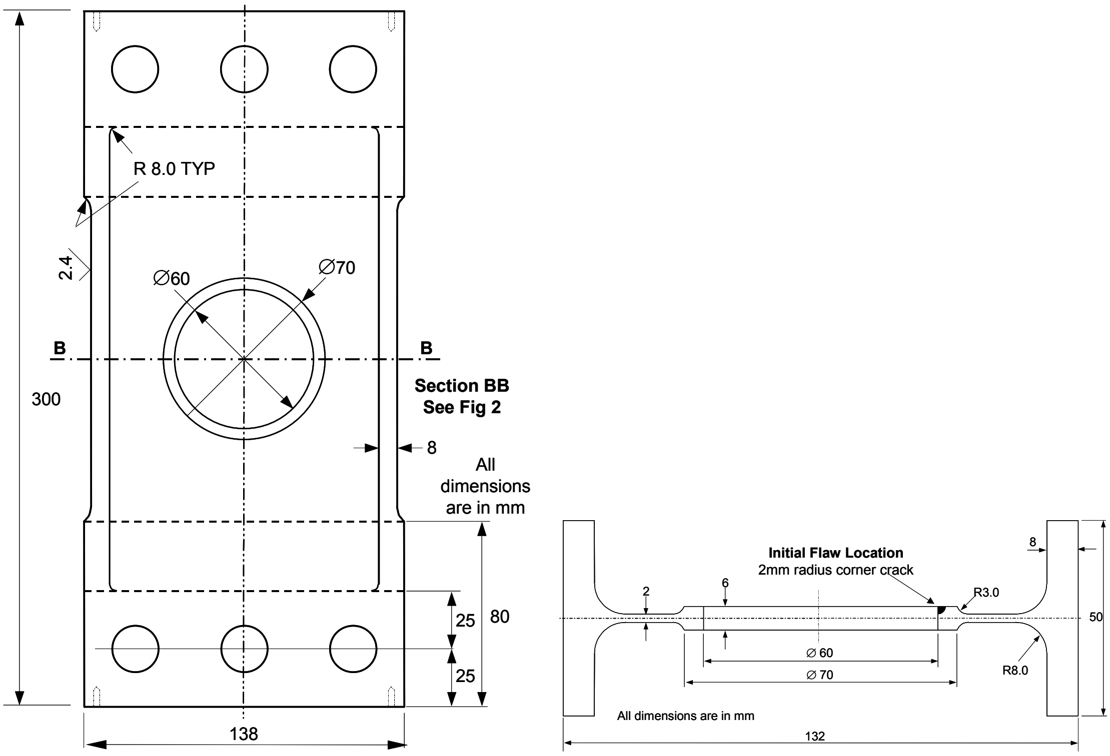


FIGURE 32 Crack growth predictions for Fixed wing (FALSTAFF) and helicopter spectra (Felix) and their sensitivity to threshold placement

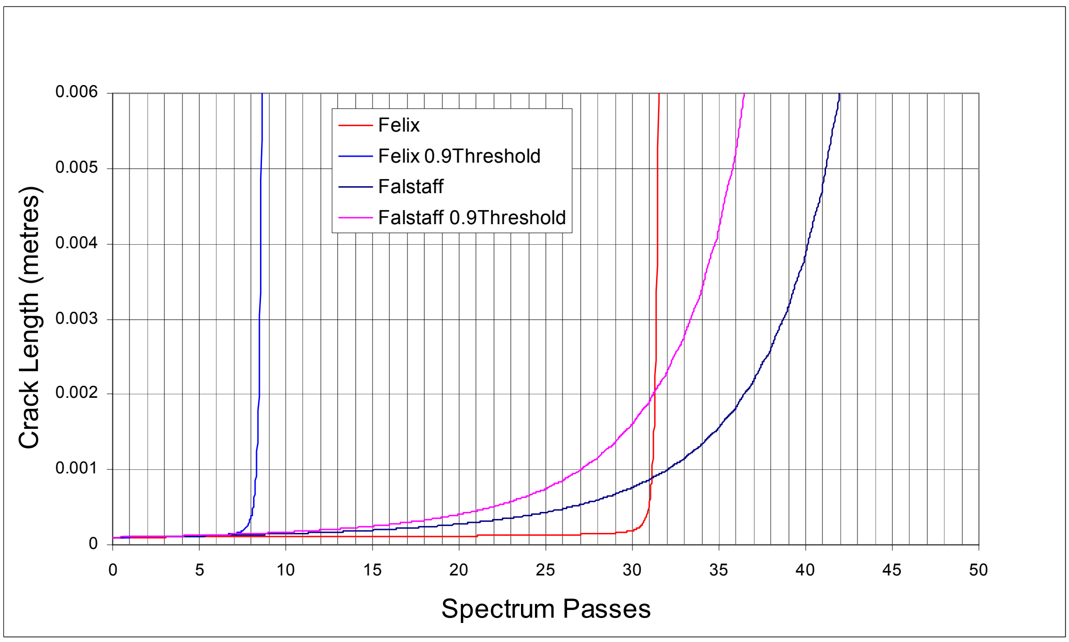


FIGURE 33 The influence of vibration suppression on crack growth prediction from helicopter spectra

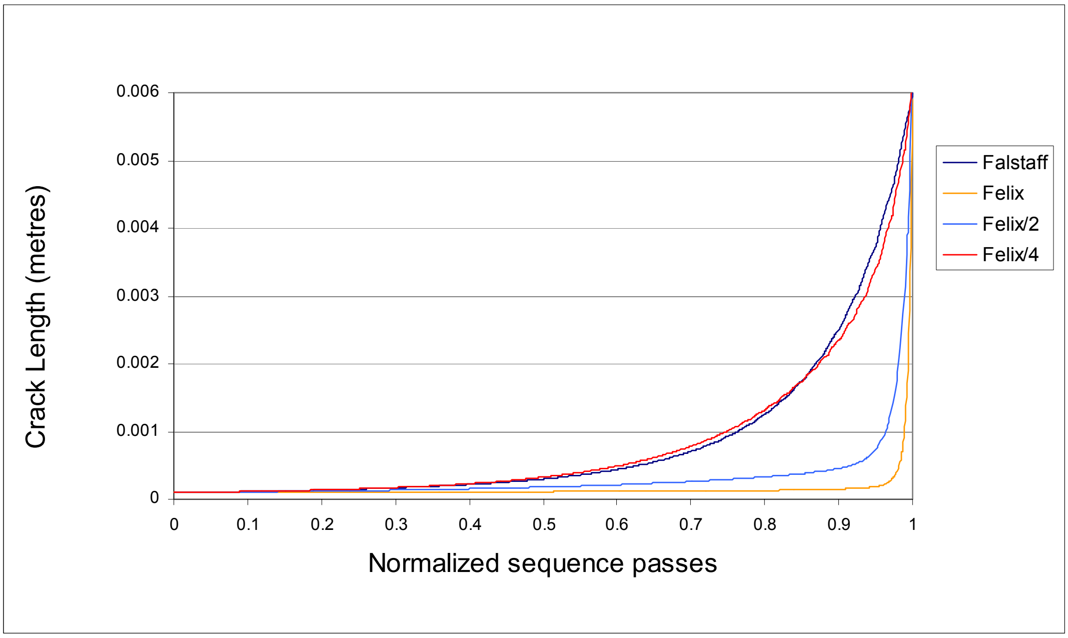


FIGURE 34 The FDM for 2024-T351 at R=0

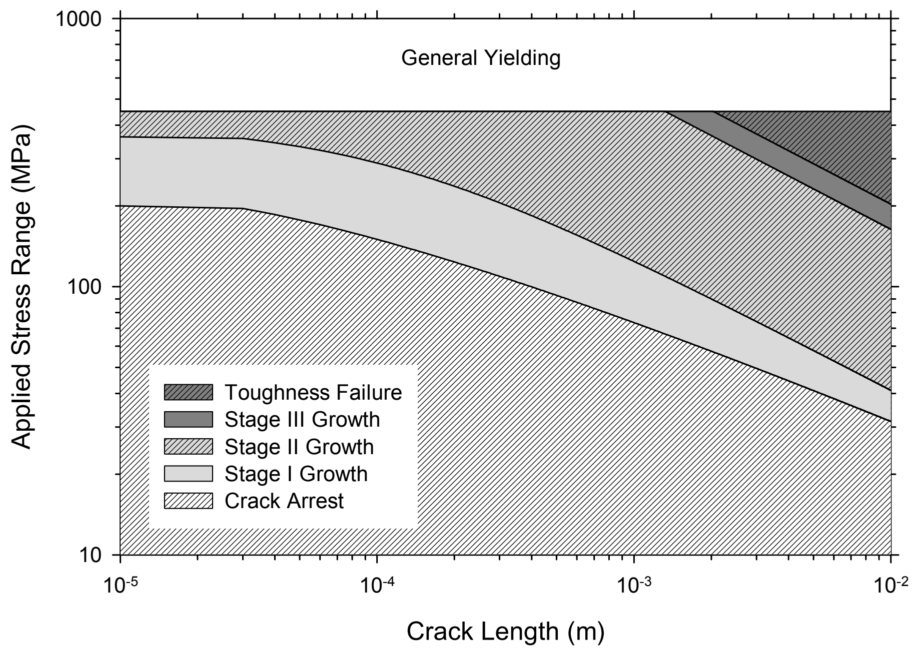


FIGURE 35 Verification of the transition from Stage I to Stage II using fractography for 3 different stress ratios

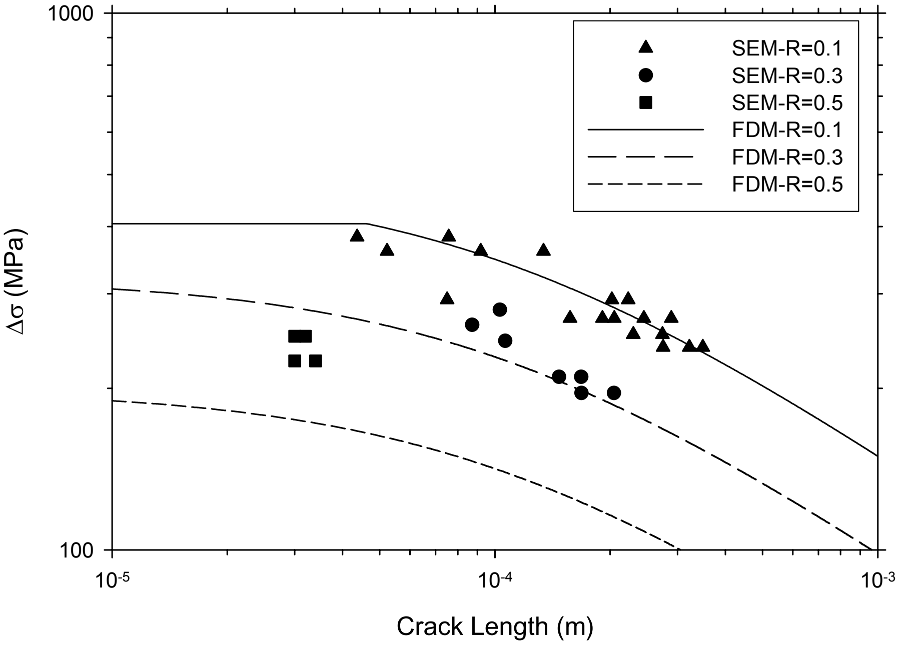


FIGURE 36 A) Typical thermoelastic image of a 18mm crack captured during a fatigue. B) Distribution of the sum of principal stresses surrounding the crack tip derived from a thermoelastic image

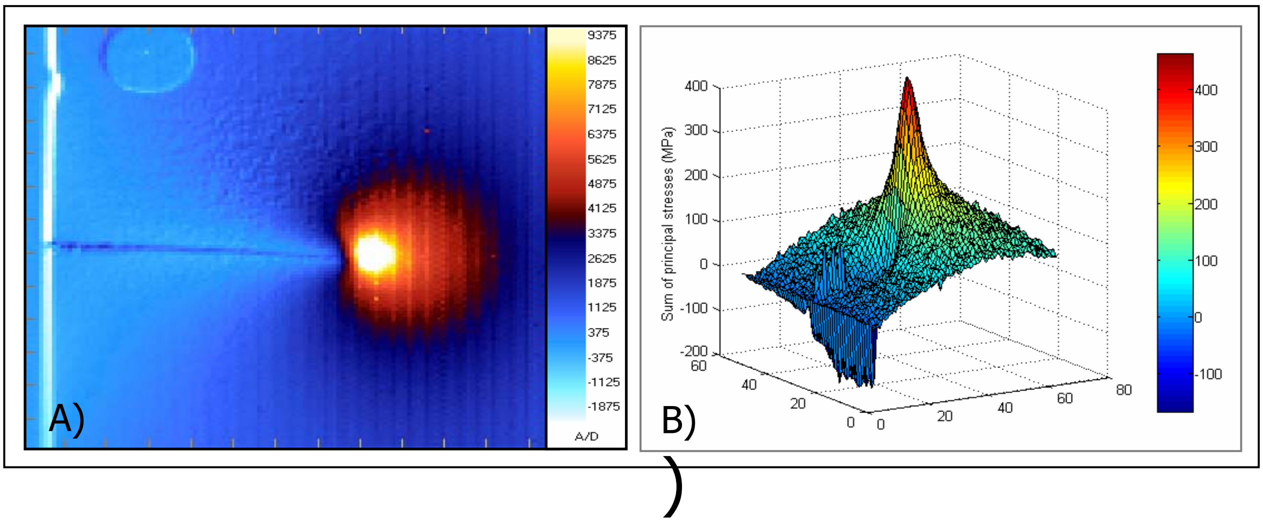
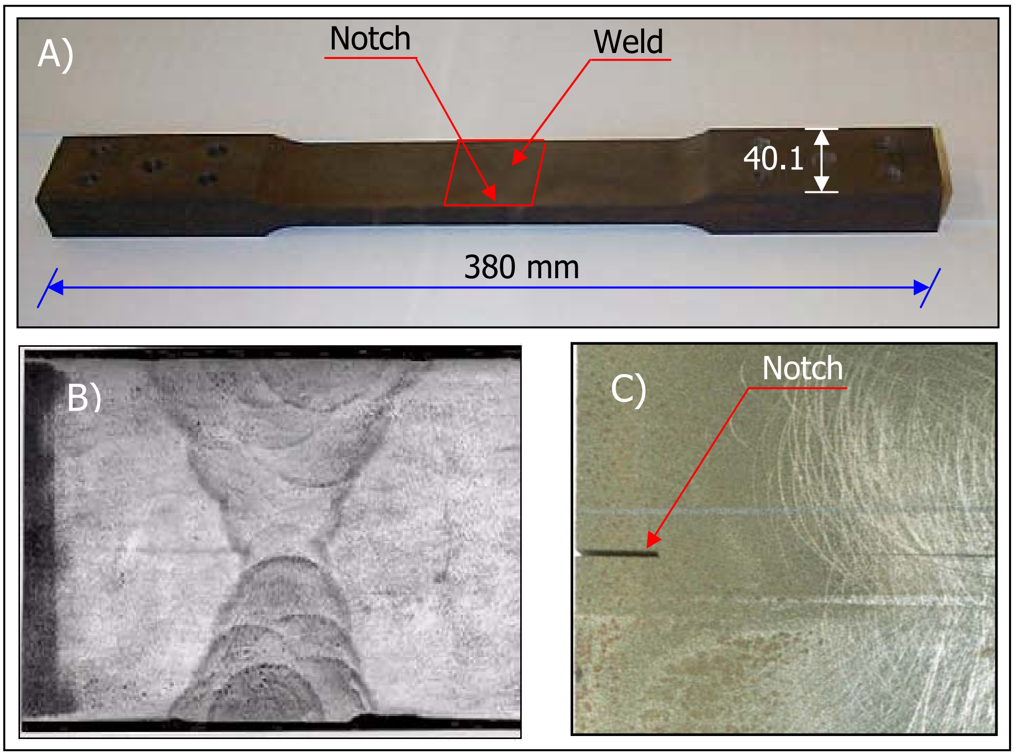


FIGURE 37 A) Description of the test specimen dimensions. B) Detail of the welded region. C) Detail of the notch



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