REVIEW OF AERONAUTICAL FATIGUE INVESTIGATIONS IN THE UNITED KINGDOM DURING THE PERIOD MAY 2003 TO APRIL 2005

compiled by J E Moon

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2.1 INTRODUCTION

This review summarises aeronautical fatigue investigations carried out in the United Kingdom during the period May 2003 to April 2005. 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 and R A Adey, Computational Mechanics BEASY)

Background
The aim of the work is develop modelling tools that can predict the behaviour of cracks in complex structures under realistic loading conditions. Two years ago developments were reported which enable the behaviour of cracks to be predicted in 2D and 3D structures and components including multiple site damage. This work was based on a Boundary Element Model (BEM) of the crack which provided the user with details on stress intensity factors, crack growth rates and the crack path. Any shape crack can be represented and the load redistribution is fully accounted for in the model. This resulted in the release of BEASY version 9 in 2004.

Recent Developments
The aim of recent developments has been to:

- Enable existing finite element models to be used for crack growth
- Simplify the modelling of cracks
- Enhance the range of applications of the software

FE Models
The latest release of BEASY provides a powerful crack growth simulation tool that can used in conjunction with finite element models to predict stress intensity factors, determine crack growth path, and predict remaining service life. Recent enhancements in the software now also allow crack growth simulation in sub-models extracted from finite element (e.g. NASTRAN, ABAQUS , ANSYS, MSCPATRAN) data files.

In the past, the use of BEASY for crack analysis required a new model to be created despite the user having an existing finite element representation of the structure. In the past two years a new interface to finite element modelling has been developed which enables an existing finite element model to be used as the starting point for the BEASY analysis. A section of this model can then be selected, either using pre-defined groups of finite elements or using an area function to select a region close to the required crack. A model is then automatically created from the FE model including the loads and boundary conditions.

Using this approach large, complex finite element models can be routinely used to predict stress intensity and crack growth data while capturing the detailed geometry and the loading behaviour of the full structure. The developed interface works with many of the common finite element tools, such as NASTRAN, ABAQUS and ANSYS and this can also be used with any model that is available in PATRAN.

An example of this interface with a PATRAN model is shown in Figures 1, 2 and 3. A group of elements, at a region of high stress, has been defined in the PATRAN interface.

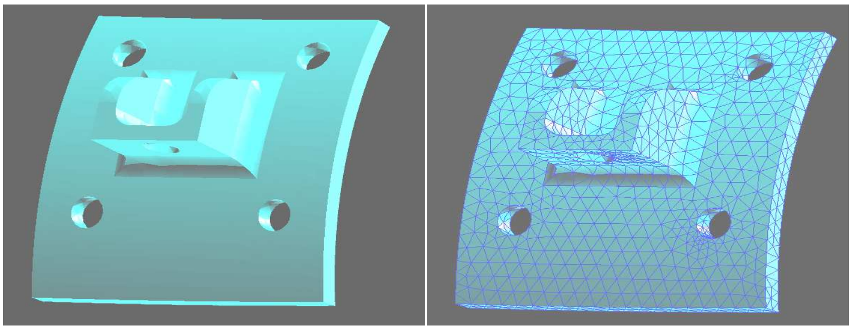


Figure 1 Bracket modelled and analysed using finite elements

The interface wizard generates a surface mesh from the solid Finite Elements and boundary conditions on the BEASY elements as required.

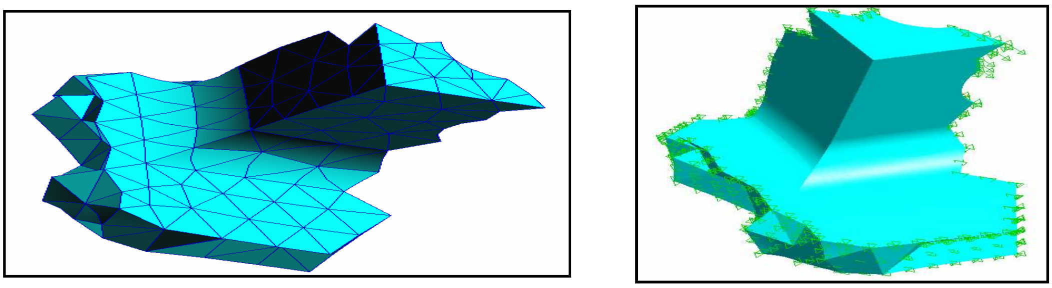


Figure 2 Extracted finite elements and BEASY sub model around required crack initiation site

This model is then suitable for crack growth, with the full load behaviour captured from the finite element results.

The use of boundary element analysis allows stress analysis to be performed using surface modelling only. In the area of crack growth this has a significant advantage as crack analysis is a surface feature and the re-meshing of the model as the crack grows is much simpler than in finite element analysis.

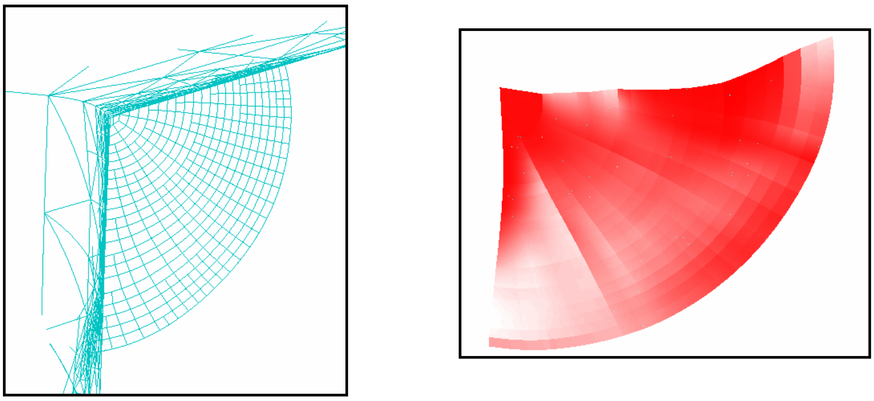


Figure 3 Grown crack surface in BEASY sub-model

Simplifying Crack Modelling

To reduce the complexity of performing a fracture analysis, customized "crack wizards" have been developed to guide users through the process of adding a crack into a model. Therefore the time consuming task of remeshing a model is no longer necessary as this is performed automatically by the software. It is now possible to compute accurate stress intensity factor data from realistic FEM or BEASY models in a much shorter time

Also a further function has now been created to interpolate a series of existing stress intensity factor result sets to give a new life prediction. This allows sensitivity studies to be carried out on a model without the need to fully re-analyze the structure.

Extended range of applications
Further work has been carried out on the analysis of multiple cracks. Both embedded and surface cracks can be represented in the models and their growth predicted automatically.

A common aerospace application is the growth of cracks near lugs. Recent extensions to the software enable the changes in the contact area and the load transfer to be simulated as the crack grows. Crack growth can now be performed fully automatically in these cases.

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

Data Item No. 97024 Derivation of endurance curves from fatigue test data, including run-outs:

ESDU Data Item, No. 97024, on the derivation of a mean S–N curve from fatigue test data, was issued in June 2003. The Data Item is 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. 04019 Endurance of high-strength steels:

ESDU Data Item No. 04019 presents endurance data for high-strength steels. The results of more than 10 000 constant-amplitude fatigue tests, on plain and notched specimens subjected to rotating-bending, plain-bending and axial loading, are presented in the form of curves of alternating stress versus endurance. In the Data Item, a high-strength steel is defined as a steel having a room-temperature 0.2 per cent proof stress greater than 800 MN/m^2 (116×10^3 lbf/in2). The data presented relate to the medium and high-cycle regimes, that is endurances of greater than 10 000 cycles. The data have been grouped according to type of steel based on the unified numbering system (UNS) codes and also according to alloy content, the tempering temperature applied to the specimen before testing, the test-piece shape, the surface condition of the specimen and the size of the notch, when present. Test temperature and mean stress are also considered. The text of the Data Item includes discussion of the data presented and of the influence of various factors on fatigue strength; the effects of tempering temperature, mean stress, test temperature and frequency, loading type, surface roughness and notches, and surface treatment are all discussed in detail with reference to the data in the Item. Figures that illustrate explicitly the effects of some of these variables are also presented.

Data Item No. 04022 An introduction to low-cycle fatigue phenomena:

ESDU Data Item No. 04022, 'An introduction to low-cycle fatigue phenomena', will be issued in May 2005 and will be followed by two related Data Items, No. 05007, 'Strain-life data for type 316 austenitic stainless steels at temperatures between -269°C and 816°C (-452 and 1501°F) ', and No. 05008, 'Cyclic stress-strain response of type 316 austenitic stainless steels at temperatures between -269°C and 800°C (-452 and 1472°F)'.

This item introduces the basic equations of low-cycle fatigue analysis, and some of the limitations of their usage. Both the strain-life relationship (otherwise known as the Coffin-Manson equation) and the cyclic stress-plastic strain equation are derived. The effects of mean stress on the strain-life relationship and the methods of correcting for these proposed by Morrow and Manson and Halford are discussed, as are the transition fatigue life and the Bauschinger effect.

The cyclic hardening and softening behaviour of metallic materials during cycling and their stress and strain response at different test temperatures (including the effects of oxidation, creep-fatigue interaction, dynamic strain ageing, cold creep or "time-dependent plasticity", and martensite transformation during cycling) are also considered.

The Data Item includes a section on low-cycle fatigue testing procedures and the uncertainties that arise during such testing. In addition, the Item discusses methods of estimating of low-cycle fatigue properties that can be used when limited data are available; two comprehensive examples which illustrate the use of these methods are also provided.

Rooke and Cartwright's Compendium of Stress Intensity Factors:

Work is nearing completion 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 mid-2005.

Cumulative damage programs:

Work on two Data Items and accompanying Fortran programs to perform cumulative damage calculations is ongoing. One program will perform cumulative damage calculations using the stress life approach, whilst the other will use the strain life approach. The programs will be based on methods presented in Data Items Nos 76014, "Estimation of endurance and construction of constant amplitude SN curves from related data corrected for notch and mean stress effects", and 95006, "Fatigue life estimation under variable amplitude loading using cumulative damage calculations".

2.3 FATIGUE LOADS MEASUREMENTS

2.3.1 Dominie TMK1 operational loads measurement programme (S C Reed, M J Duffield, M J Engelhardt and K A Lucas, QinetiQ Farnborough)

The Dominie TMk1 variant of the HS125 series of corporate jets is operated by the Royal Air Force (RAF) in the navigator training role. This necessitates operating the aircraft in diverse usage patterns including low-level turbulent conditions. However, the structural integrity inspections for the wing of the aircraft are based upon civil usage assumptions. Therefore, there was a requirement to capture a usage spectrum for the aircraft in order to validate the inspection criteria for the Dominie.

Hence, QinetiQ proposed a minimum wing Operational Loads Measurement installation to obtain the required flight data. The installation consists of 12 strain gauge full bridges on the wing lower surface, additional accelerometer and inclinometer sensors and a tap in to the aircraft ARINC 429 data stream and several other aircraft systems (such as weight-on-wheels indications). This initial design was developed and an installation kit was manufactured by QinetiQ. The equipment was installed on the aircraft under the RAF Special Trials Fit procedure (written by QinetiQ). The aircraft installation was largely fitted by the RAF maintenance contractors from SERCO at RAF Cranwell and the data acquisition equipment was programmed, fitted and tested by QinetiQ. The entire installation was undertaken while the aircraft was undergoing a major service without causing any delay to the original maintenance programme.

As at the mid-May 2005, the aircraft has flown 76 flights since the end of the major maintenance period in January 2005 and all flights have been captured. All 12 strain gauge bridges are producing high quality data and the flight parameters captured from the aircraft systems and additional OLM sensors are functioning satisfactorily. Processing of the data is being undertaken using software written by QinetiQ to support the programme. The overall aim of the programme is to capture 1 years' flying (approximately 200 flights).

Initially, it was intended to pass the results in the form of processed stress cycles to the Design Authority for comparison between RAF usage and the spectra used in the datum civil damage tolerance analysis. However, this work is also now being undertaken by QinetiQ. The team has 'reverse engineered' and optimised the damage tolerance analysis methodology to re-calculate fatigue crack growth rates, based upon RAF usage. With the agreement and cooperation of the Design Authority, the USAF-developed 'AFGROW' software is being used as the basis of this analysis. The results are to be compared with the civil datum as the basis for assessing the validity of the associated inspection schedules with respect to the specific usage patterns of the Dominie aircraft in RAF service.

2.3.2 Tornado ADV and IDS operational loads measurement (N Green, D Meadows, BAE SYSTEMS, Warton)

The fatigue life consumed by RAF Tornado ADV (or F3) and IDS (GR1 & GR4) aircraft is currently monitored by a fatigue meter and a number of Fatigue Meter Formulae (FMFs). Every aircraft carries a fatigue meter that is used to record variations of normal 'g'. This, together with other flight-related parameters, is used to calculate the aircraft FMF life.

The FMF life is refined for RAF aircraft by the use of the Structural Usage Monitoring System (SUMS) correction factors. The correction factors are based on continuous monitoring of a sample of ADV and IDS aircraft that have strain gauges installed at the various fatigue-critical FMF locations around the aircraft.

Over the past 20 years the SUMS system has been very successful. The data provided have been a key part of a wide range of fatigue analyses and usage investigations - for example, comparison of real in-service loading spectra to the various major fatigue test spectra, and the development of new FMFs based on measured SUMS data and statistical techniques (rather than earlier measures such as theoretical derivation, or coarser methods relying on the fatigue meter only). This has led to structural clearances and in-service monitoring being based on high quality information from actual in-service flying.

With the lengthy period of in-service use, during recent years the SUMS equipment became less reliable and prone to obsolescence owing to its age. Furthermore, it did not have the necessary capacity to respond to emerging requirements, such as the addition of life extension instrumentation. Consequently, at the end of March 2002 the SUMS equipment was switched off in preparation for a number of aircraft to be equipped with a new Operational Loads Monitoring (OLM) system. This upgrade will provide RAF Tornado aircraft with a more capable and modern system.

The upgraded OLM aircraft are currently operating at two standards: the OLM baseline fit and the life extension fit. The OLM baseline fit comprises up to 100 strain gauges, and recording of approximately 50 further parameters. It includes the SUMS baseline fit plus additional data from items such as control surfaces and locations highlighted as critical from the relevant ADV and/or IDS full-scale fatigue tests. The life extension fit supplements this with an extra 20 strain gauges monitoring components that require additional analysis to meet the needs of the life extension programme required for RAF aircraft.

The OLM system is automatically switched on and off via aircraft engine speeds. The system contains the following key components: a Heim D4 DAT recorder that receives data from up to 5 ACRA remote data acquisition units (1 master and up to 4 slaves). The instrumentation varies between each aircraft and consists of transducers, data bus, ADR, system tapping, strain gauges, linear accelerometers and a 3-axis gyro pack. The OLM system is capable of recording up to 45000 samples per second, but is optimised to record data at different rates for different channels as required.

The OLM system will maintain the benefits conferred by the long-term use of the SUMS, especially to calculate fatigue lives and to develop additional or refined FMFs based on actual flight data. In addition to this, the life extension specific data will be used to aid clearance of the aircraft to beyond their original design requirements (it is required to extend the life of the RAF IDS fleet from its original 4000 flying hours (FH) to 8000 FH, and of the ADV fleet from 4000 FH to 5500 FH). Finally, a further benefit that is expected from the OLM upgrade is the use of the data to improve the RAF Tornado maintenance schedule - in particular, to expand inspection periods, thus increasing aircraft availability and reducing the maintenance costs.

2.3.3 RAF TMk1/1A Hawk operational loads measurement (OLM) Programme (M Gelder and S Roberts, BAE SYSTEMS, Brough)

The Royal Air Force (RAF) Hawk Operational Loads Measurement (OLM) programme involves two aircraft, one which operates with the RAF Aerobatic Team (RAFAT) and the other with the Advanced Flying Training (AFT) unit. The programme with these aircraft began in 1996 to support the RAF's Hawk Life Extension Programme (LEP) and the results from the analyses of approximately 500 flights were reported in 1999.

Since then the OLM strain gauge configurations of both aircraft have been enhanced to investigate in more detail particular aspects of operational flying and changes to the aircraft structural standard. Since 2003 a further 200 flights have been captured from the AFT OLM aircraft in support of the Test Based Analytical Clearance (TBAC) programme to provide structural clearance for the RAF Hawk to its planned Out of Service Date (OSD).

The RAFAT aircraft is collecting flight data during 2005 to assess seasonal differences in the display sequence.

2.3.4 RAAF Mk127 Hawk flight data collection programme (M Gelder and S Roberts, 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 2005. The test specimen's load spectrum is being developed from a number of sources including the 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 Hawks are fitted with an FMS.

Flight data has been collected since aircraft delivery to the RAAF in 2000 and, for the purposes of FSFT load spectrum generation, continued to the end of 2003. FMS data has continued to be analysed from 2004 flying to assess changes and trends in the operation of the aircraft.

2.3.5 Hawk New Demonstrator Aircraft (HNDA) in-flight refuelling probe forward fuselage vibration trial (M Gelder and S Roberts, BAE SYSTEMS, Brough)

The Hawk New Demonstrator Aircraft (HNDA), ZJ951, is being used to investigate any impact to structural clearances of carrying the in-flight refuelling (IFR) probe. In particular the study is considering the forward fuselage vibration levels during high speed flight (higher than during normal air-to-air refuelling) with and without the IFR probe. This work began with data from a limited instrumentation fit being recorded during the Mk951 Rolls-Royce engine trials in South Africa in 2004.

The next trials phase is scheduled for the third quarter of 2005. The instrumentation fit comprises 7 tri-axial accelerometers and 12 strain gauges.

2.3.6 RAF Jaguar engine mountings OLM (Paul Johnson, Robin Trelfa, BAE SYSTEMS)

Current clearance for the Engine Mountings is based on calculation using a theoretical load spectrum as it was found that the original major airframe fatigue test was not representative of in-service loading. The calculated safe life is within the range 4,500FH to 6,000FH, there being significant difficulty in deriving engine loads by calculation. Many of the current active fleet have now exceeded this life and an inspection schedule has been introduced as mitigation against the risk of structural failure. Further to this, the RAF fleet has recently undergone a modification programme to an up-rated and heavier Adour Mk.106 engine (the clearance was based on a Mk.104 engine load spectrum). Therefore, an OLM programme has been performed in order to accurately determine engine mounting loads.

The calculation of loads at the engine mountings is made difficult due to the fact that only two of the three mountings can be strain gauged. The forward inboard mounting, which reacts load in three orthogonal axes (including the engine thrust load), cannot be strain gauged due to its geometry and construction. The solution to this was to derive a theoretical load for the forward inboard mounting based on aerodynamic effects (linear/angular accelerations and gyroscopic effects), aerodynamic loads on the engine tailpipe, engine spool speeds and engine thrust load. In order to achieve this, two dedicated software decks were produced; BAE Systems derived a deck to calculate the tailpipe loading based on various flight parameters, and Rolls-Royce derived a deck to calculate engine thrust based on measured flight and engine parameters. Similar theoretical loads were also derived at the strain gauged engine mountings to allow a check between measured and theoretical loads and hence provide confidence in the predicted loads at the forward inboard mounting.

Fatigue critical locations on the engine mountings and back-up structure were subsequently determined and stress equations developed, based on the measured/calculated loads at the mountings.

A flying programme of 18 sorties was completed in December 2004. Detailed analysis of the data is currently being performed by BAE Systems and a final report detailing safe lives is due to be issued in August 2005.

2.3.7 RAF Jaguar cabin pressurisation (P Johnson, R Trelfa, BAE SYSTEMS)

The Jaguar pressure cabin clearance is currently based on results from the major airframe fatigue test, completed in 1973, which completed 18,871 full pressurisation cycles without failure. A full pressurisation is defined in-service as an ascent through 18,000 feet, followed by a descent though 11,000 feet. However, concern regarding this clearance has been raised due to (i) the difficulty in manually recording the pressurisations accurately, and (ii) the significant number of partial pressurisation cycles which occur in flight.

Owing to these concerns, the Engine Mountings OLM aircraft was instrumented to provide cabin pressure data. Subsequent investigation of the data revealed higher levels of pressure in the cabin than was previously thought. These measured levels were found to be greater than the theoretical operating characteristics of the pressure system, and were primarily considered to occur due to a time lag in the operation of the mechanical valve system as altitude rapidly changed. A damage analysis was performed and concluded that an OLM Damage Factor of 1.88 should be applied to each full pressurisation count in order to correct for the actual damage accumulation that occurs per recorded count.

This factor shows that a significant amount of damage is induced into the pressure cabin by partial pressurisations and higher than expected pressure levels, which was previously unaccounted for by traditional methods of recording.

Due to the revised pressure cabin clearance being less than that required for the RAF Jaguar fleet, two cabin pressurisation fatigue tests (Strike & Trainer aircraft variants) are currently under way and are expected to be completed during the summer of 2005. The loading spectra for these tests have been derived using the OLM data.

2.4 FATIGUE OF METALLIC STRUCTURAL FEATURES, INCLUDING REPAIR

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

This section reports on a collaborative programme between Cranfield University, Open University, Southampton University, with QinetiQ, Alcoa and Airbus.

The programme developed an integrated prediction scheme for fatigue durability and damage tolerance in which the contribution of weld process variables was combined with design and service load data in a quantitative assessment of damage tolerance.

The project succeeded in creating a unique, completely integrated network of predictive capabilities which connected weld process and manufacture at one extreme, to welded structure design and prediction of damage tolerance performance at the other. A comprehensive study has been made of the microstructure and properties, fatigue initiation, fatigue crack propagation and fail safe design capability of MIG and VPPA welds in 2024 and 7150 aluminium; the former relevant to lower wing skin tension structures, the latter alloys used in compression in the upper wing. Properties have been established at three different size scales; 100mm coupons, 400mm centre cracked panels and 1.2m skin-stringer panels. See Figure 3 and Figure 4 for an example of the results. The Open University used a range of powerful non-destructive techniques to perform highly accurate 3D residual stress measurements on samples manufactured and tested in the project at all size scales: these samples have more extensive residual stress characterisation than any welds previously made. Knowledge of this key parameter determining structural integrity, together with extensive precision material property and structure data has permitted for the first time identification of individual contributions of residual stress, microstructure and local mechanical properties to overall weld properties, and has permitted development and validation of a range of performance prediction models.

Models have been developed to predict:

-Weld heat affected zone static strength properties from knowledge of weld heat inputs, utilising existing thermodynamic data for alloys and microstructure- dislocation interaction models;

-Fatigue crack initiation behaviour from knowledge of the weld process; the defect distributions in the weld, residual stresses and data on short crack growth rates in aluminium alloys; incorporating existing crack interaction models

Macroscopic fatigue crack growth behaviour from knowledge of the weld residual stress fields and their changes throughout the sample together with existing data on fatigue crack growth rates.

At all scales and in all properties new insights have been gained of the extent of individual contributions of microstructure, hardness, residual stress and imposed loading to the structural integrity of the component or structure. There is now the capability for specification of weld structure and–HAZ properties to optimise static strength and fatigue and damage tolerance capability.

A general account of the work can be found at [1] and, in addition, a paper will be presented at this year's ICAF Symposium [2] which focuses particularly on the residual stress aspect.

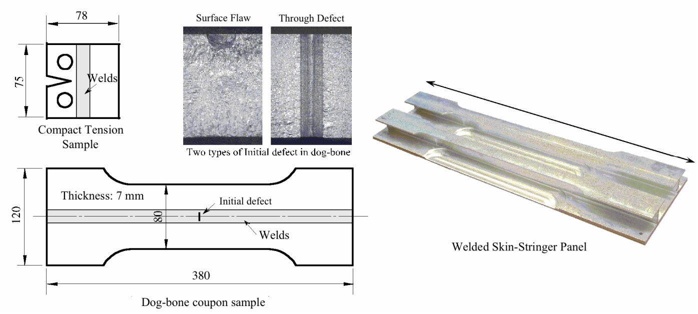


Figure 4 Welded samples used in WELDES investigations

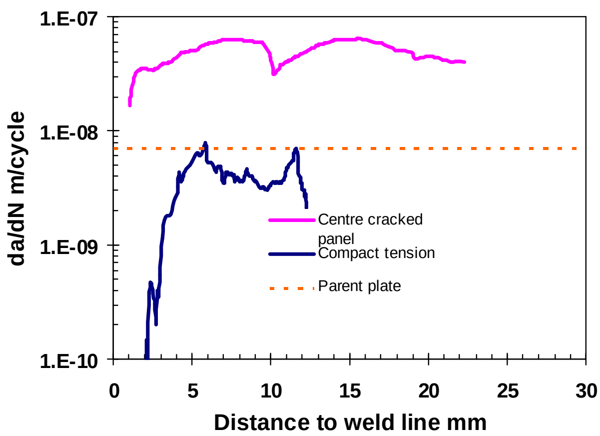


Figure 5 Constant ΔK tests of the same weld in different sample configurations- the role played by residual stress.

2.4.2 Fretting fatigue under variable amplitude loading (P E Irving, Cranfield University)

The project has investigated the influence of variable amplitude loading on fretting fatigue in aluminium 2618 contacting against cylindrical and non cylindrical steel contact pads under conditions of partial slip [3]. Finite element analysis was used to model the shear stress distribution at the contact patch and the influence of variable amplitude load sequences on it. Fretting fatigue lives were determined under complex spectrum loads and the influence of various components of the load spectrum on fretting fatigue life were determined.

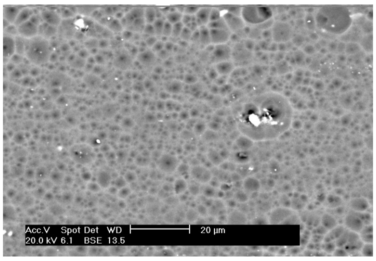
It was found that under variable amplitude loading, the position of the stick- slip boundary and the diameter of the slip region changes from cycle to cycle depending on the size of the current cycle, and the size of the largest previous cycle. The location of the maximum shear stress point will be continually moving, and is not always associated with the position of the stick - slip boundary as it is in constant amplitude loading. This implies there may be differences in crack initiation behaviour in the two cases.

Observations of crack morphology and density of crack development in constant and variable amplitude fretting fatigue, confirm that under variable amplitude loading, cracking occurs over a large proportion of the contact patch area; in constant amplitude loading cracking occurs at a specific location. Crack damage density is hence greater under constant amplitude loading. This implies that in variable amplitude fretting fatigue, predictions of Miners proportional damage rule will be conservative compared with experimental lives. This is confirmed experimentally in variable amplitude fretting fatigue tests.

Comparison of contact stress fields of a rounded punch with the cylindrical contact, shows that local stresses are greater under the punch, under the same remote loads, but the stressed volume is much reduced and fretting fatigue lives are longer than would be expected on the basis of the contact stresses when evaluated using a range of damage parameters such as Brown- Miller and Ruiz. Once again this is confirmed via experimental testing.

2.4.3 Improving the fatigue resistance of aircraft panels using ultrasonic impact treatment (C A Rodopoulos Sheffield Hallam University and L Tehinni, Applied Ultrasonics, Birmingham Alabama, USA)

The properties of a metal are strongly influenced by the grain size of the metal. On the basis of experimental results obtained in microcrystalline (mc) metals with grain size typically above 1μm, it is widely recognized that an increase in grain size generally results in a reduction in the fatigue endurance limit. Recent research further indicated that grain refinement to the ultra fine crystalline (ufc) metals (grain size typical in the 100nm to 1μm region) and nanocrystalline(nc) metals (grain size typically less than 100nm) can have a substantial effect on the total life under stress- controlled fatigue and on fatigue crack growth [4]. Ultrasonic Impact Treatment (UIT) is a technique that directly deforms the surface of materials using ultrasonic vibrations. The development of these methods dates back to 1950s which were characterized by employing continuous ultrasonic vibrations at the ultrasonic transducer output end strengthened with hard materials (carbide-containing alloys, artificial diamonds etc.) and being in direct contact with the treated surface. As a result, a relatively thin surface layer of the treated material was plastically deformed, producing modifications of the surface microstructure (fine grain size) (Figure 6) and redistribution of residual stresses in this layer which has been wider confirmed to efficiently improve the fatigue property of the materials. The development of the method resulted in the ***Esonix*** technology that supported a firm position of the ultrasonic impact treatment among well-known techniques of improving fatigue resistance, corrosion resistance and neutralisation of heat affected zones from friction stir welds. An example in the life improvement possible with this treatment is shown in Figure 7, and a demonstration of improved corrosion resistance in Figure 8.



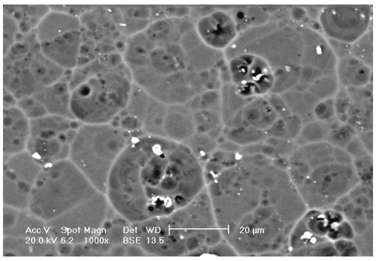


Figure 6 Left the effect of UIT on 2024-T351 at a depth of 2.6µm from the surface, right the microstructure of the untreated material at the same depth

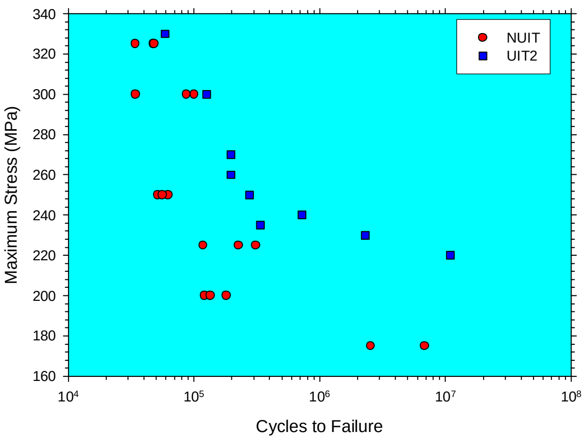


Figure 7 The effect of UIT on the fatigue life of 2024-T351 in push-push cyclic at R=0.1

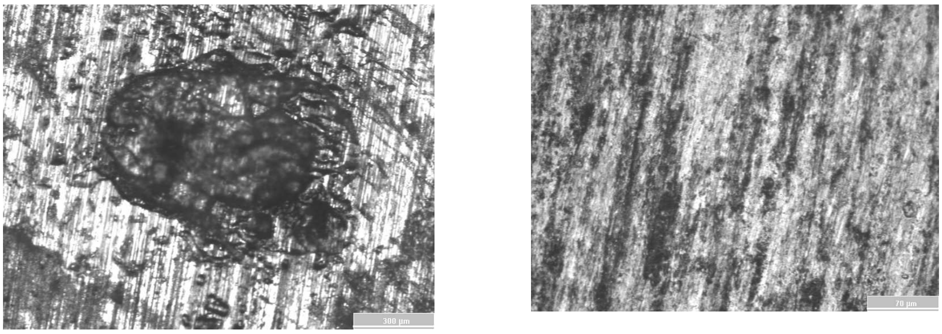


Figure 8 The effect of 48hours in ASTM exfoliation solution on bare (left) and UIT treated (right) 2024-T351

2.4.4 Developments in the fatigue analysis of undercarriage components (T Siddall, Messier-Dowty)

Messier-Dowty are developing their fatigue analysis methods and adopting the strain-life approach. As part of this, there is an on-going programme to characterise all the structural materials that they employ, or anticipate using, via strain life testing. The programme includes all the existing materials used, such as 300M, 4340 steel, 15-5PH stainless steel, 7010 and 7175 aluminium alloys and titanium 6-6-2 and 10-2-3 and also new development materials, such as high strength stainless steels and titanium alloys. In addition to testing being carried out on plain samples at several stress ratios, there is a programme of supplementary testing of various machining finishes, platings and treatments and stress concentrations.

The materials data will be incorporated into a newly developed fatigue solver package that can integrate full field FEA results and provide a complete fatigue profile of a component. This latter approach (full field fatigue analysis) has been introduced slowly over the past two years, but the integration with the strain life analysis is a significant step forward.

2.4.5 Improve and Assess Repair Capability of Aircraft Structures (P M Powell, K Brown, A Young, 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 and EADS. QinetiQ is investigating three areas: (a) the use of cold expansion/interference fit techniques for repair of cracks at fastener holes, (b) the effectiveness of enhanced stop drill techniques, including cold expansion and interference fit plugs and (c) the development of bonded composite patch repair techniques for friction stir welded (FSW) structures.

Experimental test programmes have made good progress and are now close to completion. A series of fatigue tests on high load transfer joint test pieces, both with and without residual cracking following repair to fastener holes, is in progress. An experimental study has been completed on the application of enhanced stop-drill repair techniques to 2024-T351 aluminium alloy plate and to the FSW alloy. This included comparison of the effectiveness of stop-drill repairs of the parent alloy under constant amplitude (CA) and wing spectrum loading, and comparison of stop-drill repairs in the parent alloy and the FSW alloy under CA loading.

A fatigue test programme has also been carried out to investigate the effectiveness of bonded composite patch repair for FSW 2024-T351 plate. Fatigue tests have been carried out on unpatched and patched panels, either with a single starter crack at the weld centre line or double starter cracks at the edges of the weld. The effectiveness of bonded composite patches is also being assessed for the repair of corrosion damage in FSW 2024-T351 plate. Panels have been exposed to aqueous 3.5% NaCl in the weld region on one face and fatigue tests have been carried out on unrepaired, corroded panels. A further two corroded panels have subsequently been patch repaired on the corroded face and are to be tested as a back-to back assembly to indicate the life improvement which can be achieved by the repair scheme.

Analytical models have been developed for the prediction of fatigue crack growth in the presence of bonded composite patch repair to FSW aluminium alloy structures and in the presence of enhanced stop-drill repair schemes. A combined boundary element/finite element model has been developed for the three-dimensional analysis of composite patch repairs to cracked friction stir welded plate. A two-dimensional linear elastic boundary element model has been applied to stop-drill repair schemes with and without an IF pin. A milestone report on the development of the models has been completed [5] and the models are currently being validated against the results of the test programme.

2.4.6 Feasibility study of the optimisation of stop-drill holes and assessment of applicability (P M Powell, K Brown, K Man, A Young, QinetiQ)

There is a requirement for improved repair schemes that allow fatigue cracks in military aircraft to be repaired more cost effectively, without compromising structural integrity. Ideally the repairs should be cheap and quick, in order to minimise both costs and aircraft downtime. The stop-drill method can be used as a short-term repair technique, but in order to provide effective retardation of fatigue cracks, it is advisable to apply interference fit plugs (IF) and/or cold expansion (CX) to the stop-drill hole. However, the use of CX/IF may be impractical for some crack problems because of geometry restrictions imposed by the structure, and the CX/IF processes add to the time and cost of the repair. Previous work at QinetiQ and AMRL has shown that fatigue life may be extended significantly if multiple stop-drill holes or “elongated” holes are used instead of a single stop-drill hole at each crack tip. Consequently a theoretical study of the possible improvements from the use of these types of stop-drill repair, without the need for CX/IF, has been carried out at QinetiQ [6].

The 2D boundary element method (BEM) was used to investigate the improvements offered by multiple or elongated stop-drill holes. Initially, BEM results were obtained for the standard stop-drill repair geometry, from which an approximate (±0.5%) formula for stress concentration factors was derived. Four scenarios involving multiple or elongated stop-drill holes, illustrated in Figures 9 to 12, were then investigated, namely (A) two circular holes, (B) three circular holes, (C) two circular holes joined by a saw cut to form an elongated hole and (D) an elliptical hole. The stress concentration factors for these different cases, as functions of crack length and hole dimensions R and L (defined in Figures 9 to 12), were compared with the factors for single stop-drill holes of equivalent area. The study also considered the influence of bi-axial loading and edge proximity on the effectiveness of the repairs.

The BEM analyses indicate that each of the four alternative stop-drill repair scenarios can be used to reduce the stress concentration, when compared to that for a conventional, single circular stop-drill hole of equivalent area. Repairs of types A, B and C give stress concentration factors up to 10% lower than that for a circular hole. Whilst it would be the most difficult to produce practically, the elliptical hole offers the greatest reduction in stress concentration factor, with a hole of aspect ratio L/R=2 giving at least a 30% reduction when compared to a circular hole of equivalent area. It is recommended that a programme of experimental and theoretical work should be undertaken to determine the improvements in fatigue life that these reductions in stress concentration factors can provide under typical service stress levels, and to address the practical details of applying the alternative scenarios, such as machining elliptical cut-outs.

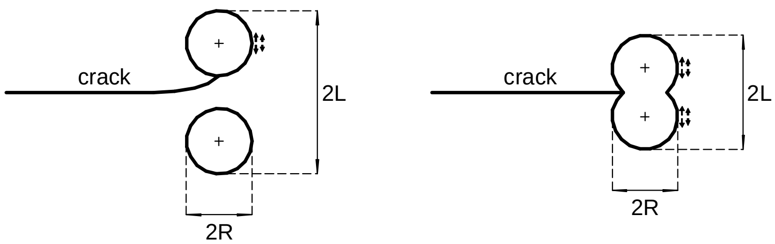


Figure 9 Stop-drill scenario involving two holes (Repair type A)

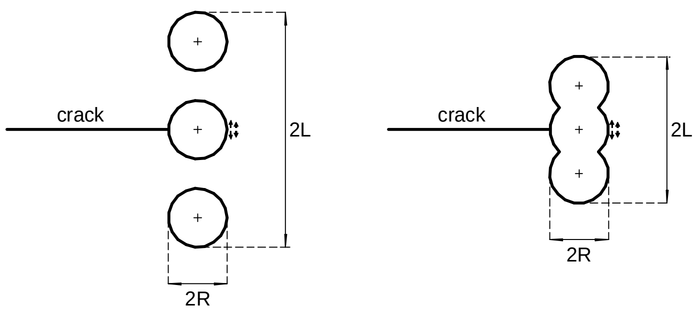


Figure 10 Stop-drill scenario involving three holes (Repair type B)

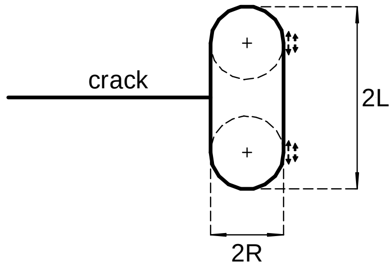


Figure 11 Stop-drill scenario involving two holes joined by saw-cuts (Repair type C)

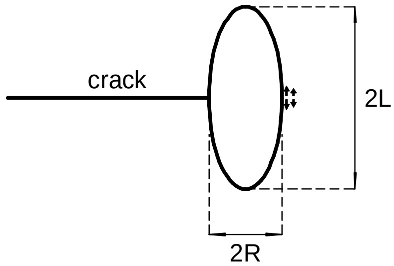


Figure 12 Stop-drill scenario involving elliptical hole (Repair type D)

2.4.7 Effect of loading variables on the debonding and efficiency of bonded patch repairs (K Brown, K Greedus, P Poole, A Young and P M Powell, QinetiQ)

Adhesive-bonded composite patches can be very effective in the repair of cracked metallic structures subjected to fatigue loading, but the efficiency of the repair is reduced by patch debonding. Work is therefore being carried out to improve understanding of the influence of debonding on the efficiency of the patch repair, and to develop models or empirical rules for predicting the development of debonding under fatigue loading [7].

A literature review showed that there were only limited data available on patch debonding and that these were insufficient to enable criteria for initiation and growth of debonds during fatigue to be investigated and identified. A fatigue test programme was therefore undertaken with the specific aim of generating appropriate patch debond data. In order to design a suitable test specimen, a parametric study was carried out of shear stress and shear strain in the adhesive layer, using a simple one-dimensional model. The study indicated that a wide range of adhesive shear stress and strain values could be achieved by using film adhesive to bond 1.6mm thick unidirectional carbon fibre patches to both sides of 12mm thick centre-cracked 2024-T351 aluminium alloy plates. Fatigue crack growth tests on such specimens under a range of constant amplitude and variable amplitude (FALSTAFF) loading did not give rise to patch debonding. Stress analysis of the patched panels using a three-dimensional combined boundary element/finite element (BE/FE) model developed by QinetiQ confirmed the predictions of high levels of maximum shear stress/strain and of hysteresis in the adhesive layer. The results of the fatigue tests and modelling indicated that there is unlikely to be a straightforward relationship between debonding and the maximum shear stress/shear strain or hysteresis in the adhesive layer, as originally anticipated.

In view of the lack of debond behaviour in these initial tests, several alternative specimen configurations were investigated, viz. reduced patch thickness, single-sided patching and boron fibre patch repair on a single edge notch test piece. The last configuration, as in a previous investigation, produced debonding during constant amplitude loading. Based on the present and previous data, it is considered that debonding may depend on the mechanical properties of the resin system in the patch.

The results of the fatigue tests showed that debonding will not occur if suitable patch designs, patch materials, adhesives and process controls are employed. However, further investigation of the debonding mechanism is clearly needed to allow accurate prediction/prevention of debonding in composite patch repairs of metallic structure. It is recommended that further work should be carried out to examine the influence of the patch resin system on debond behaviour. Thus would include development of a model to predict debonding or empirical rules which will allow conservative prediction of debonding, and the incorporation of the models or rules into the current 3-D BE/FE method.

2.4.8 Tornado life extension (I Moody and A Beatts, BAE SYSTEMS, Warton)

Flying beyond the original design life, deviation from the original performance and design requirements and increases in flying hour rates have together led to a situation where certain RAF Tornado aircraft have reached (or are approaching) their current known fatigue life clearances on various parts of the aircraft. In most cases, these life exceedances are associated with unmonitored structure - that is, structure that does not benefit from monitoring by fatigue meter.

In the medium term, improved clearances may be achieved by a range of measures: use of data from a new Operational Loads Monitoring (OLM) system (described elsewhere in this National Review), by further fatigue testing and, if necessary, by modification. In the short term, the current status of the fleet has led to a necessity for other actions to provide support for continued flying by the RAF. The RAF has requested that the manufacturers provide advice to mitigate airworthiness risk whilst the actual clearances are established. This advice has taken the form of a series of inspections to be applied to aircraft before they breach the known clearances and, in some instances, item replacement.

The philosophy for determining inspection and replacements was to review the static strength summary, fatigue type record, check stress and fatigue calculations, assess principal load paths for the loading actions requiring clearances and establish areas of high stresses and/or high stress concentrations. From this data locations most susceptible to fatigue failure of a critical nature were established and inspections, generally using assisted-visual, eddy current or ultrasonic techniques were devised. For certain areas where inspection was deemed to be impractical or not economically viable a replacement policy was adopted.

Whilst this suite of inspections/replacements was considered sufficient for the initial advice, additional investigation is ongoing to enable continued flying before the delivery of information from other programmes such as OLM. Clearly, assessing repeat inspection periodicity is a large part of this ongoing work, but care is also being taken to ensure that the initial inspections are not treated as ‘structural fuses’ since effects such as material scatter, different loading actions or combinations of loading actions and the replacement of items may potentially lead to other features becoming critical.

2.4.9 Tornado wing modification (I Moody and A Beatts, BAE SYSTEMS, Warton)

Certain RAF Tornado wings are due for extensive modification action, involving the over-sizing and cold working of numerous holes in the skin/wing spar interfaces, to enable them to reach their full cleared life. However, current RAF fleet management predictions indicate that these particular wings are only required for a limited time (in terms of FI) beyond their current life, meaning that the modification will be expensive in relation to the benefit achieved. To avoid the need for modification, BAE Systems is currently investigating the feasibility of an inspection method. This investigation is taking the form of fatigue calculations to determine the most significant areas for inspection, fracture mechanics calculations to determine critical crack lengths and associated crack growth rates, and (together with other UK agencies) the development of an ultrasonic inspection processes to enable crack detection at the fastener bores in both the skin and spar with minimal maintenance penalty (i.e. without removal of the fasteners and when viewed from the outer surface only).

2.4.10 The effect of corrosion damage on the fatigue performance of cold expanded fastener holes and interference fit fasteners (P Wagstaff, A Nesterov, Kingston University; K Brown and P Powell, QinetiQ)

Previous work at Kingston University on open hole specimens manufactured from 7050-T76 aluminium alloy showed that surface corrosion significantly reduced the fatigue life improvement which would be expected from cold expansion. This reduction was greater if the corrosion occurred prior to, instead of after, cold expansion. Corrosion pits on the surface of the plate acted as crack initiation sites in a region remote from the hole edge where the residual stresses arising from the cold expansion become tensile in nature and hence increase the effective mean stress.

The aim of the current investigation is to assess the effects of surface corrosion on the fatigue performance of 2024-T3 aluminium alloy containing cold expanded fastener holes of 6.35mm nominal diameter. In addition the effects of the use of interference fit titanium alloy fasteners on both plain and cold expanded fastener holes is being studied. For this test programme 4% cold expansion and 1.3% interference fit were used. Fatigue tests were carried out on specimens with plain or cold worked central holes, with and without interference fit fasteners. The effects of corrosion prior to and after cold expansion of open holes were also assessed.

All specimens were tested under constant amplitude sinusoidal loading of 210MPa net maximum stress, $R = 0.1$, at a frequency of 10 Hz. The results obtained thus far are presented below.

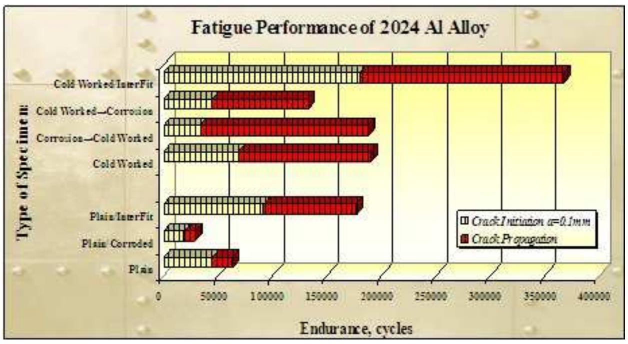


Figure 13 Fatigue lives of 2024-T3 specimens with and without corrosion

The data shown above indicates that:

- Cold expansion increases total life by more than 200% compared with plain specimens
- The use of an interference fit fastener in a plain hole increases total life by about 180%
- The combination of cold expansion with an interference fit fastener increases total life by approximately 500%.
- Interference fit increases the number of cycles to initiation (a =0.1mm) to a greater extent than cold expansion; cold expansion extends the crack propagation stage more than interference fit does
- Corrosion reduced the total life of plain specimens by over 50%.
- Corrosion prior to cold expansion appears to have a negligible effect on total life compared to cold expanded specimens with no corrosion (180% improved life compared to 200%)
- Corrosion after cold expansion limited the total life improvement which would be expected from cold expansion to 110%

The diagram below shows crack growth rates for the six types of specimens tested

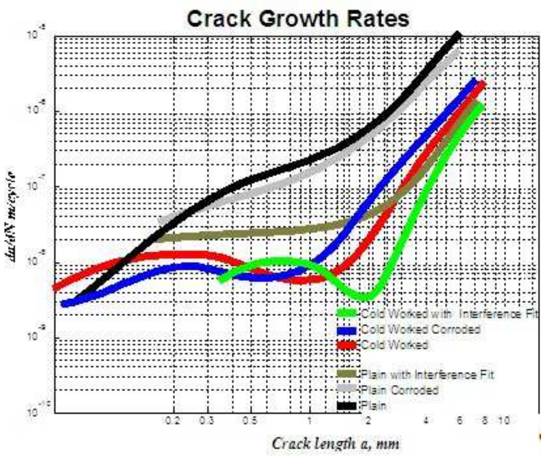


Figure 14 Fatigue crack growth rate data for 2024-T3 specimens
with and without corrosion

These results show that:

- The main effect of both interference fit and cold expansion is to reduce the crack growth rates in the initial stages (up to 3mm crack length).
- Cold expansion reduces the crack growth rate more than interference fit up to 2mm crack length.
- The combination of cold expansion plus interference fit produced the lowest crack growth rate up to crack lengths of 4mm.
- Crack growth rates in plain and plain corroded specimens were similar after 0.2mm indicating that the corrosion damage affects the crack initiation rather than growth.
- Crack growth rates were also similar in cold worked specimens with and without corrosion.

Further tests are in progress to evaluate the effect of prior and post corrosion exposure of cold expanded specimens containing interference fit fasteners.

2.5 FULL SCALE FATIGUE TESTING

2.5.1 Undercarriage fatigue testing (T Siddall, Messier-Dowty)

Fatigue testing of the production Eurofighter Typhoon Main Landing Gear is nearing completion. Further work is now underway to extend the clearance to Tranche II conditions with heavier all-up-weight. The Nose landing Gear of the Harrier GR9 is undergoing life extension testing. On the civil Airbus side, the nose landing gear for Airbus A380 is under test, due for completion in early December.

2.5.2 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, 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 also completed the damage tolerance phase. For this phase artificial damage was introduced to the structure at several locations. Several of the damages were increased in size for the final 7500 cycles of damage tolerance testing.

Figure 16

There was no crack growth from any of the saw cut locations. Residual strength testing to limit load, has now been successfully completed, again with no damage extension. To conclude the testing, the loading was increased to the jack capacity. For the critical down bending test crack extension occurred at one location at approx 120% of limit load.

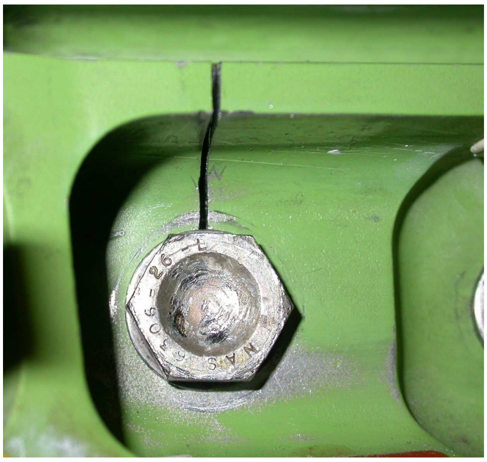


Figure 17 Saw cut Damage prior to limit load

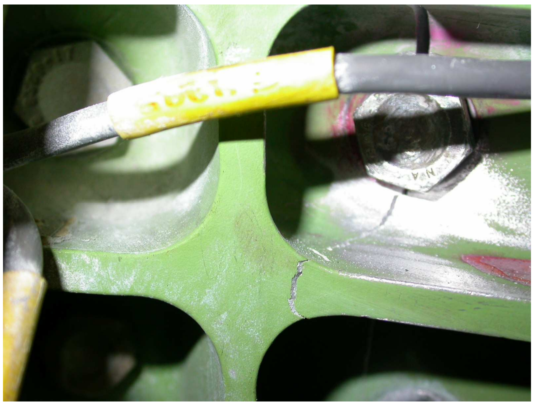


Figure 18 Damage extension at 120% limit load

2.5.3 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) has completed 40,000 flights (2 design lifetimes) of durability testing, and is 15,000 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 15,000 flights of a 20,000 damage tolerance phase with artificial damage installed. The loading spectrum represents gusts, manoeuvres, and thrust reverser deployments.

Figure 19 TA05 Test Article

2.5.4 Tucano full scale fatigue test (L Murray, Bombardier Aerospace, Shorts Belfast)

The Tucano T.Mk.1 2nd Full Scale Fatigue Test commenced fatigue testing in January 2005 and has completed the first 500 test flights. The test article is the last RAF production airframe (designated T132) and includes the wings, fuselage, tailplane and fin.

The test spectrum has been developed primarily from Operational Load Measurement data recorded on 3 instrumented aircraft over a 17 month period. The spectrum is applied in a repeatable block that represents 1000 flying hours. This spectrum block is constructed from 75 flight types, and 755 balanced load conditions representing typical ground and flight loading events.

The loads are applied to the test article through 42 active load channels using hydraulic actuators, and 7 grounded and monitored reaction points. The test article is fitted with an extensive strain gauge installation, which will be sampled at regular intervals throughout the test.

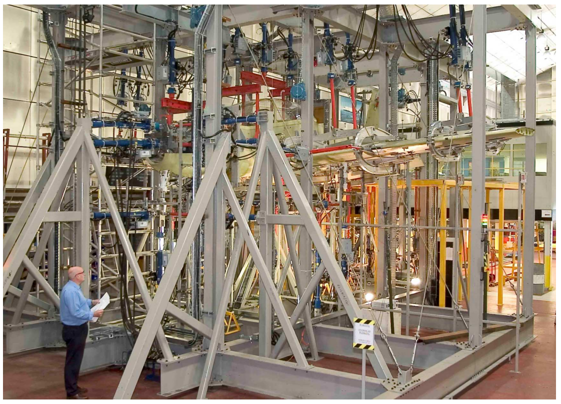


Figure 20 Tucano

2.5.5 Hawk Mk.127 Lead-In Fighter full scale fatigue test (R M Aaron, C B Benstead, G L Dove, G F Duck, M Henningsen, and R W Young, BAE SYSTEMS, Brough)

A Full Scale Fatigue Test (FSFT) of the Hawk Mk.127 Lead-In Fighter is in the advanced stages of development. The Mk.127 is the Hawk variant supplied to the Royal Australian Air Force (RAAF).

The test spectrum and loading is being developed based on Hawk Mk.127 Usage and Operational Loads Measurement data, collected over a two-year period. The loading spectrum also takes into account potential future mass growth in the aircraft and increases in severity of the spectrum.

The test airframe comprises of fuselage, wing and fin, with the tailplane, flaps, aileron, landing gear and engine being represented by dummy items.

Loading will be applied to the test specimen via a total of 83 hydraulic actuators applying loads to the structure either through the dummy items or via loading linkages. In addition, a compressed air system is to be used to apply cockpit and fuel tank pressurisation. In order to represent the effects of the varying fuel head the fuselage fuel tank has been divided into two volumes separated by a splitter plate, which is earthed back to the test frame. In addition the wing fuel tank has been divided into three sections in order to represent the rolling effects on the wing fuel tank pressures.

As a result of contractual arrangements, BAE SYSTEMS and the Commonwealth of Australia are jointly funding the test. The test is to be undertaken at the Defence Science and Technology Organisation (DSTO) Platform Sciences Laboratory (PSL) in Melbourne, Australia, acting as a subcontractor to BAE SYSTEMS, with the load and spectrum development activity being undertaken by BAE SYSTEMS at Brough, UK.

Currently the test rig is completed and the test article is installed in the rig. The test loads and spectrum are in an advanced stage of development, and the test is due to commence cycling in September 2005 and run for 50000 test hours.

The test airframe is fitted with an extensive strain gauge installation, which will be sampled regularly throughout the test.

2.6 DEVELOPMENTS IN FATIGUE MONITORING

2.6.1 A parametric-based empennage fatigue monitoring system using artificial neural networks (S Reed, QinetiQ)

With the ever-increasing capital cost of military aircraft fleets maximising the aircraft's reliability and its fatigue life are paramount. However, combat aircraft in particular operate in a fatigue-damaging environment from necessity. For the current generation of UK military aircraft, fatigue monitoring is really only undertaken for the wing and those areas of the structure where fatigue damage is driven by normal acceleration of the aircraft. Areas of the empennage (fin, tailplane and rear fuselage) are lifed in terms of flying hours because no other monitoring systems have been proven to be sufficiently accurate or affordable. However, UK Operational Loads Measurement (OLM) programmes have identified that flying hours bear little correlation to fatigue damage and that buffet loading in the empennage region can consume a disproportionate amount of fatigue life in very short time periods. Consequently, most combat aircraft fleets have suffered significant costs of ownership due to major empennage inspections, component changes, modifications or repairs and associated reductions in fleet availability and in some cases unforeseen structural failures. Whilst a fatigue monitor would not negate the accrual of fatigue damage, it would allow increased safe-lives, delayed initial inspections and more efficient directed inspections thereafter. This would reduce unscheduled down time for repairs and replacements and thereby improve the overall reliability and availability of the platform considerably.

A fleet-wide strain-based system would offer a solution, and such a system will be introduced for the UK Eurofighter-Typhoon fleet. However, from current experience, these systems are costly to develop and support. Therefore, alternative solutions should be investigated. Among these, artificial neural network based systems are worthy of consideration on cost grounds alone.

Hence, an artificial neural network based parametric fatigue monitor has been developed for the empennage (fin and tailplane region) of combat/combat trainer aircraft, to account for both low-frequency damage and high-frequency (buffet) damage. The system uses the relatively low bandwidth aircraft flight parameters available to modern military aircraft through Mil-Std 1553 or equivalent data bus systems. The monitoring system has been trained and tested using Operational Loads Measurement (OLM) data and the system architecture could be transferable from one aircraft type to another without significant modification.

Such a method of quantifying the fatigue damage accrual of individual combat aircraft in the fleet for the empennage region could significantly improve the safety, reliability and availability of the fleet. Any elements of safe life structure should have a significant life improvement because they would be monitored (estimate up to around 30% increase in Safe life). Furthermore, any structure cleared by an inspection-based regime would have more accurately defined inspections with delayed first inspection and increased periodicity of inspection (up to around 30%) and consequent improved fleet availability and reduced cost of ownership. This work is featured in this year's Symposium Poster session [8].

2.6.2 Structural Monitoring systems using non-adaptive prediction methods - proposed Defence Standard (S Reed, UK MOD Military Aircraft Structures Airworthiness Advisory Group (MASAAG), QinetiQ)

Proposed Defence Standard 00-970 regulations and guidance have been produced for the use of non-adaptive prediction methods in aircraft structural monitoring systems. This material is intended to supplement the existing Defence Standard regulations and guidance for aircraft monitoring systems and addresses the specific issues related to non-adaptive prediction methods.

Non-adaptive prediction methods are techniques used to predict a target value (e.g. stress, strain, load or fatigue damage) from input parameters (such as flight parameters e.g. normal acceleration, roll rate). These techniques use a range of mathematical or statistical methods that may include artificial neural networks, model-based analysis, linear or non-linear regression, clustering algorithms. The models are subject to a training phase in which the relationships between the input and target values are captured. Once the training phase is complete and had been tested and validated, the relationships between input and target parameters are fixed until the commencement of any further training.

The initial work to support this programme was funded by the UK MOD Tri-Service Structural Integrity Branch and undertaken by QinetiQ. Thereafter, a working group from the UK MOD Military Aircraft Structures Airworthiness Advisory Group (MASAAG) was formed with a team from QinetiQ, BAE Systems, Smiths, UK MOD Tri-Service Structural Integrity Branch and the UK MOD Aircraft Integrity Monitoring Branch. The team brought together the work undertaken by QinetiQ and independent development work carried out by BAE Systems and Smiths into a regulatory framework. This proposed Defence Standard is soon to begin the formal review process.

2.7 FATIGUE IN COMPOSITE MATERIALS AND STRUCTURES

2.7.1 The prediction of fatigue damage growth in impact-damaged composite skin/stringer structures: theoretical modelling studies (A J Kinloch, Imperial College London)

A methodology has been proposed for predicting the number of fatigue cycles, N_g, needed to cause a specified damage growth in CFRP skin/stringer structures after impact. The method combines an experimental relationship between the strain-energy release-rate, G, and the number of fatigue cycles obtained from direct measurements, with a linear finite element (FE) analysis of the component. Essentially, the approach uses the FE model to deduce the strain energy release rate, G, of a skin/stringer panel after a growth of approximately 5% of its impact-damage area. This value of G is then used to ascertain the value of number of fatigue cycles, N_g, for the onset of damage growth using the experimental fracture-mechanics data. It is noteworthy that the impact on the CFRP skin leads to a dent in the skin and this has been modelled by considering a perturbed shape for the dent. The amplitude of this dent has been varied and shown to have an important role in determining the number of fatigue cycles, N_g, to cause the onset of damage growth for a given maximum applied strain in the fatigue cycle.

Current work with QinetiQ (Dr Andrew Davies) has experimentally confirmed that the results from the model are very accurate and tests are continuing. This work will be published by QinetiQ and Imperial when the test are completed in the near future.

2.7.2 Fatigue life prediction of polymer composite structures (A J Davies, M Grassi, F Ngah and A B Clarke, QinetiQ)

The current design philosophy widely adopted by many aircraft manufactures for polymer composite structures is that of a no damage growth criteria, this approach has traditionally led to safe, statically designed components, which are arguably heavier than actually necessary for safe operation. Recent work at QinetiQ funded by the MOD has continued to research the effect of in-plane compressive cyclic loading on impact damaged end loaded I - beams. Traditionally composite coupons have been tested as the basis of predictive lifing methodologies; however, these are too simplistic in nature and not representative of real structures. Previous work undertaken has shown that without an external influence to create a defect such as a delamination, delaminations will not form in such structures under normal cyclic loading conditions. Specimens were impacted and cycled at strains higher than traditionally seen in such structures in fatigue (4500 μE); even at these higher strains the structures were found to be resilient to in-plane fatigue loading with little or no damage growth occurring until a large number of cycles (> 1E06) had been accumulated. Work is currently looking at ways of combining in-plane and out of plane loading on to representative stiffened structures, as out of plane loading is likely to have more of an effect on damage growth and subsequent failure of a component.

2.7.3 Prediction of damage growth in sandwich structures (S J Lord, R Barnet, M J Hiley and C E Jones, QinetiQ)

Work carried out for the MOD Materials Technology Section of the Sea Technology Group (STGMT) aims to predict damage growth in defective GRP composite sandwich structures under fatigue loading. Although currently aimed at naval structures, the work is equally applicable to airframes. To date, the modelling has identified the DEBUGS code [9] to carry out the work, which has been proven for modelling similar structures under static compression [10] and [11]. The code predicts local buckling and crack growth of a defect in a skin/core interface under static compression loads. Within this programme, the code is being modified to predict initiation of growth and subsequent growth due to fatigue loads.

2.7.4 Dent depth relaxation in composite aerospace structures (A J Davies and R Goddard, QinetiQ)

Laminated composite structures do, in general, exhibit low through-thickness strength compared to their in-plane properties and as such are vulnerable to impact damage which may arise from a variety of sources including dropped tools and runway stones. An experimental test programme was undertaken by QinetiQ for the MOD under the Applied Research Programme to look at dent depth relaxation under the effect of external influences such as cyclic fatigue loading, temperature and time. A number of specimens were tested under a full (tensile and compressive) FALSTAFF (Fighter Aircraft Loading STAndard for Fatigue) spectrum to establish the effect of differing numbers of applied FALSTAFF cycles on the depth of a dent. Once the specimens had undergone their required number of applied cycles the dents were again measured and the panels destructively tested in tension to establish their residual strength; this was compared to their impacted but un-fatigued strength, to establish the effect of load cycling on the specimen strength. The specimens were subjected to a range of sequences from 1 sequence which was equivalent to 35,966 cycles up to 80 sequences which was equivalent to 2,877,280 cycles. From the experimental programme undertaken a number of trends were noted; with both time, temperature and under the influence of mechanical loading the initial dent depth relaxed to leave a shallower dent. A general trend established was that shallower dents (i.e. dents caused by lower energies) had a greater percentage relaxation than for deeper dents - for a given material the percentage relaxation was not constant but energy related.

2.7.5 Structural evaluation of composite vane root sections under static, dynamic and fatigue loading (M Grassi, A Clarke, C Meeks and M L Hiley, QinetiQ)

Under a MoD funded research programme, Rolls-Royce (RR) and QinetiQ are looking at alternative light-weight materials for the next generation of engine vanes for aerospace platforms. To reduce the weight of the fan system, carbon composite stator vanes are being considered. The projected cost and weight of the composite vane is between 90% and 70% of the titanium alternative, depending on the stators considered.

Due to the vane complex geometry, under tensile and bending loads through-thickness stresses can develop within the vane root. Through-thickness stresses can cause delamination, which significantly reduce the performance of the component. Structural evaluation of composite T-shaped specimens under static, dynamic impact and fatigue loads was carried out. Although a T-shaped specimen was thought to be a simplistic representation of a real vane root, its geometry was considered to be complex enough to study the through-thickness properties of the corner radii and deltoid region which are common geometric features of real vane roots. Fatigue testing was carried out on both intact T-specimens and Foreign Object Damage (FOD) impacted T-specimens and also including environmental conditioned specimens (70^0 C, 85% Relative Humidity).

Under high cycle fatigue, composite specimens did develop different levels of damage. The structural mechanisms of damage accumulation, including intra-lamina micro-cracking and macro delaminations at critical ply interfaces were studied in detail. Fatigue failure peculiarities were clearly identified, and compared with the static failure mechanisms. It was demonstrated that most of the specimens cycling between 10^5 and 10^6 cycles suffered fatigue damage, which then affected their residual strength. The amount of accumulated damage was a function of the percentage of the static failure strength at which fatigue testing was carried out, as well as specimen initial status (FOD damage, Hot/Wet). Future work will look at modelling of the fatigue damage accumulation failure process.

2.7.6 Analysis of composite laminates with damage introduced under static or fatigue tensile loading (C Soutis, The University of Sheffield)

The failure process of composite laminates under quasi-static or fatigue loading involves sequential accumulation of intra- and interlaminar damage. Matrix cracking parallel to the fibres in the off-axis plies is the first intralaminar damage mode observed. These cracks are either arrested at the interface or cause interlaminar damage (delamination) due to high interlaminar stresses at the ply interface. In recent work theoretical modelling was developed by the author and his co-workers on stiffness property degradation and mechanical behaviour of general symmetric laminates with off-axis ply cracks and crack-induced delaminations. Closed-form analytical expressions were derived for Mode I, Mode II and the total strain energy release rates associated with these damage modes [12-15]. Dependence of strain energy release rates on crack density, delamination area and ply orientation angle in balanced and unbalanced symmetric laminates was examined and discussed. Also, stiffness degradation due to various types of damage was predicted and analysed.

2.8 FRACTURE MECHANICS AND DAMAGE TOLERANCE

2.8.1 Effects of stress ratio and temperature on fatigue crack growth in a Ti-6Al-4V alloy (J Ding, R Hall and J Byrne, University of Portsmouth)

Fatigue thresholds and fatigue crack growth (FCG) rates in corner notched specimens of a forged Ti-6Al-4V aero-engine disk material have been investigated at room temperature and 350°C. The threshold stress intensity range, ΔK_{th}, was determined by a method involving a step change in stress ratio (the "jump in" method). It was found that for three high stress ratios (R = 0.7, 0.8, 0.9), where crack closure effects are widely accepted to be negligible, there were similar ΔK_{th} values at room temperature and 350 °C under the same R. For a given temperature, ΔK_{th} was observed to decrease from 3.1 MPa√m to 2.1 MPa√m with R increasing from 0.7 to 0.9. The fatigue crack growth rate was influenced by increasing temperature. For high stress ratios, FCG rate at 350°C was higher than that at room temperature under the same ΔK. For a low stress ratio (R = 0.01), higher temperature led to higher FCG rates in the near-threshold regime, but showed almost no effect at higher ΔK. The influence of stress ratio and temperature on threshold and FCG rates was analysed in terms of a K_{max} effect and the implication of this effect, or related mechanisms, are discussed. In light of this, an equation incorporating the effects of the K_{max} and fatigue threshold, is proposed to describe FCG rates in the near-threshold and Paris regimes for both temperatures. The predictions compare favourably with experimental data.

2.8.2 Foreign object damage (FOD) initiated small crack growth behaviour in Ti-6Al-4V alloy under combined LCF + HCF loading. (J Byrne, R Hall, J Ding, J Tong, University of Portsmouth; D Rugg, Rolls-Royce plc; P Tranter, QinetiQ; M Winstone, Dstl)

The study aims to understand small crack growth behaviour in Ti-6Al-4V after foreign object damage for combined LCF/HCF loading. The combined LCF/HCF loading studied represents a simplest load sequence experienced by rotating aerofoils, each block of which include a single LCF cycle and 1,000 HCF cycles with a high stress ratio. The FOD events were simulated by firing a 3.2 mm cube onto targeted specimens with an impact velocity of 200 m/s, leading to a blunt "V" notch on the flat surface. Microstructural features associated with the FOD damage, e.g. microcracks, were characterized by SEM. Those microcracks were found to be preferred sites for crack propagation. The growths of these FOD-initiated small cracks under combined LCF/HCF loading were dramatically different from large-crack results obtained on corner cracked (CC) specimens. Stress relief annealing was conducted on several FOD specimens to remove residual stresses. The consequent fatigue results indicated that these residual stresses affected not only the fatigue life, but also the development of crack shape. A general and non-linear finite element code, ABAQUS, is being employed to model FOD, which determines the geometry of the damaged site and the estimated residual stress field. A fatigue crack growth model is proposed to predict FCG rates and thereby fatigue lives of FOD-initiated small cracks.

2.8.3 Influence of surface damage on the fatigue behaviour of an advanced nickel base superalloy at elevated temperature (J. Byrne, A. Burgess, University of Portsmouth; J. Penny, Rolls-Royce plc)

The influence of surface damage on fatigue crack initiation and propagation life has been studied for a nickel base Superalloy. Fatigue tests were performed at elevated temperature for scratch damage on both as machined and shot peened surfaces. Tests were designed to simulate an aspect of gas turbine disc geometry operating under extreme conditions. The scratch damage was designed to lie within the layer of compressive stress created by the shot peening. It was found that any improvement in fatigue life gained from shot peening was negated by the presence of a scratch. This was seen as a marked reduction in the number of cycles required to initiate a crack and was attributed to the scratch being deeper than the peak compressive stress imparted by the shot peening.

2.8.4 DBEM for residual strength calculation of curved stiffened panels undergoing nonlinear deformation (M H Aliabadi, Department of Aeronuatics, Imperial College London, T Dirgantara, Department of Engineering Queen Mary, University of London)

This work was carried out as part of the European Project **A**dvanced **D**esign **C**oncepts and **M**aintenance by **I**ntegrated **R**isk **E**valuation for Aerostructures (ADMIRE) in collaboration with ALENIA, AIRBUSUK, CASA, DCABS, ECOPF, SNLR, INASCO, IDMEC, TUBRAUN, ISTRAM, UNAP and UPSA.

In this project a Dual Boundary Element Method (DBEM) was developed for analysing fracture mechanic problems in curved reinforced shells (fuselage) taking into consideration the effects of large deformation and plasticity. The formulation developed is based on shear deformable shell theory and can accurately model the stress field in the vicinity of a crack. The DBEM method developed originally in 1992 by Aliabadi and co-workers had in the past been shown to be efficient and robust for modelling cracks in two- and three-dimensional configurations. The work in ADMIRE demonstrated that DBEM can equally well be extended to thin reinforced shells. The simplicity of modelling processes and ability to perform crack growth simulation makes this method very attractive.

2.8.5 Boundary element analysis of super-plastically formed and diffusion bonded aircraft structures (M H Aliabadi, Imperial College London, C. DiPisa, Queen Mary, University of London, A Young and P. Powell QinetiQ, Farnborough)

The aim of this work item was to develop a boundary element method (BEM) for the computational stress analysis of superplastically formed and diffusion bonded (SPF/DB) titanium alloy x-core and cellular structures. This has been pursued through extra-mural research with Queen Mary University of London (QMUL) [16].

In this work a new boundary element method is developed for stress analysis of multi-layered panels and structures and which has particular application to super-plastically formed and diffusion bonded (SPF/DB) aircraft structures. The new method, which is applicable to loading cases resulting in small and large deflections, was developed to enable analysis of cracks in the skin, webs as well as bond-line defects. Associated fracture parameters are evaluated automatically along the crack front using an energy release method and J-integral. Fatigue crack growth can be simulated automatically without the need to remesh the entire structure, since cracks can be extended simply by adding extra boundary elements to the tips of each crack. The direction and rate of growth of each crack is determined from the fracture parameters, and it is not limited by predefined possible directions of growth common to the finite element applications. Figure 21 shows the normal stresses contours of a bond-line defect in an X-core section. Notice the mesh is limited only to the boundary of the different plate sections and no elements are required in the interior.

Collaboration has taken place with BAE SYSTEMS, Warton. Details of the Typhoon fore-plane were made available and used to test the code for x-core structure. BAE SYSTEMS has also provided details of fatigue test specimen geometries for cellular structural elements. These structural elements are manufactured from Ti-6%Al-4%V sheet using the SPF/DB process. The BEM model has been developed using these representative geometries and predictions of fatigue lives for spandrel-T specimens can be compared with the results of the BAE SYSTEMS tests, when these become available. It will be feasible to use the model to predict the growth behaviour of bond-line flaws in cellular structures. Collaboration also involved Airbus UK, which has an interest in the application of the BEM to wing box sections. This collaboration took place under the DTI-funded METEOR programme (Metallic Technologies Research Programme for Commercial Aircraft).

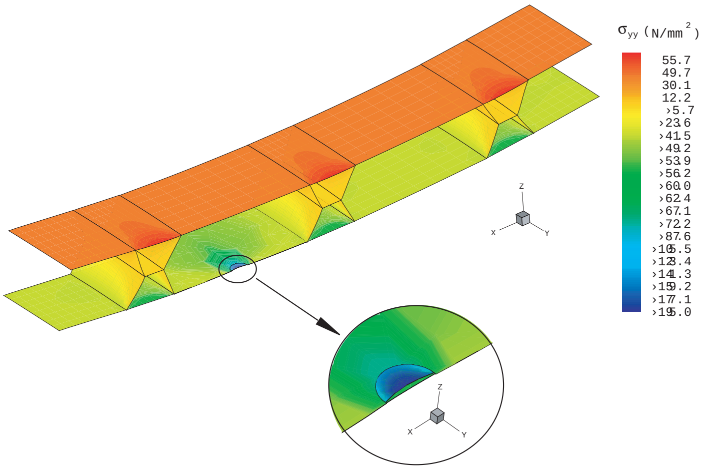


Figure 21

2.8.6 Damage detection and damage tolerance in rotating mechanical systems (P E Irving, Cranfield University)

Shearography is a laser based optical strain measurement technique, capable of mapping the distribution of strains or displacements in loaded components over selected surface areas. Cracks and similar defects which threaten the structural integrity of the component, could thus be revealed by detection of the strain distributions which surround them. Using shearography, the project has demonstrated detection and quantitative assessment of defect size and severity of loading in titanium samples. Quantitative measurement of strain using the shearography technique has been demonstrated. The work has additionally demonstrated detection and quantitative assessment of cracks in components covered with a protective polymer coating. This would be impossible using traditional visual NDT. Examples of images of cracks in a titanium component without a protective coating and with the protective coating are shown in Figures 22 and 23. Cracks as small as 0.7 mm may be detected without the coating and 2 mm with the coating.

Factors controlling fatigue crack initiation and early crack growth from defects in a high strength titanium alloy used for helicopter rotorheads have been studied. The data produced has been used in conjunction with data on shearography crack detection capability, in a fracture mechanics analysis to determine the damage tolerance performance of a helicopter flying with the shearography system. It is found that use of the shearography system in this way would allow defects to be detected with 15 hours flying time remaining before failure. This performance is sufficient for a retirement for cause approach to damage tolerance on the rotorhead-or could provide a safety net for the current safe life. This is a notable advance on the current safe life approach. The combination of defect sensing and non contact strain measurement could permit development of a prognostic system for future life prediction.

In a parallel investigation, initiation from mechanical defects and fatigue crack growth rates in Ti 10-2-3 have been studied Short crack growth rates were identical to those of long cracks in this material for cracks as short as 30 μm (Figure 24). Initiation behaviour from mechanical defects was found to be controlled by residual stresses induced by plastic deformation, rather than by defect size. Larger defects were associated with larger deformation fields and increased lives to achievement of a fully developed crack (Figure 25), [17].

Figure 22 5 mm crack imaged under *du/dy, dv/dy and dw/dy*

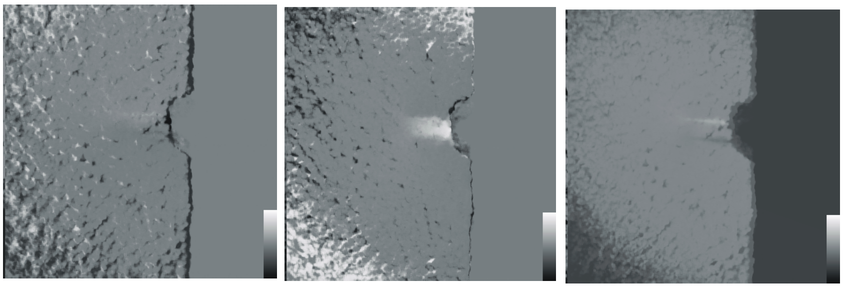


Figure 23 as for Figure 22, but with the component covered with opaque protective coating

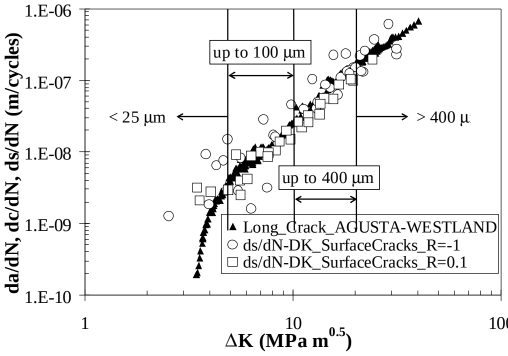


Figure 24 Comparison of long crack and short crack data for Ti 10-2-3

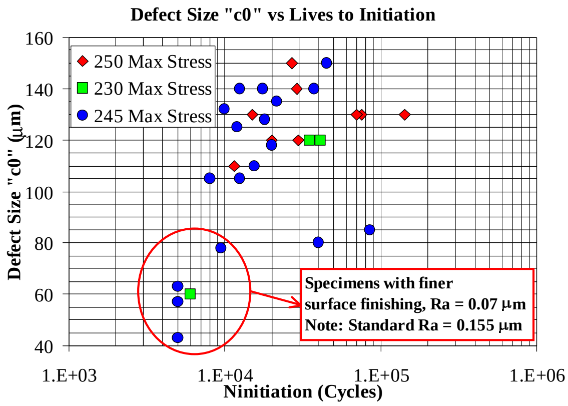


Figure 25 Defect size plotted against life to initiate a fatigue crack

2.8.7 Understanding the material’s fatigue behaviour under cyclic loading using the a fatigue damage map (C A Rodopoulos, Sheffield Hallam University, J R Yates, University of Sheffield)

Based on the principles of microstructural fracture mechanics, the concept of the fatigue damage map (FDM) demonstrates the physical relationship between the size of crack plasticity and the five stages of crack growth from catastrophic crack initiation to failure. The relationship allows the continuous evaluation and distinction of the five stages of fatigue damage in the form of a map, an example of which is shown in Figure 26. Crack arrest is modelled as a condition where crack tip plasticity tends to a minimum, approximately one Burger’s vector,; microstructurally short crack growth considers the accumulated damage (crack length and crack tip plastic zone) to be confined within one grain,; transition from physically short crack to long crack growth is considered to take place when the crack tip plastic zone obtains size equal to two grains,; and the transition from long crack or Stage II to Stage III crack growth takes place when the crack tip opening displacement over normalised by the size of crack tip plasticity is equal to the true elongation to failure.

The concept of the fatigue damage map is being developed to deal with sparse populations of data and the influence of uncertainty on the quality of the outputs from the damage map [18-22].

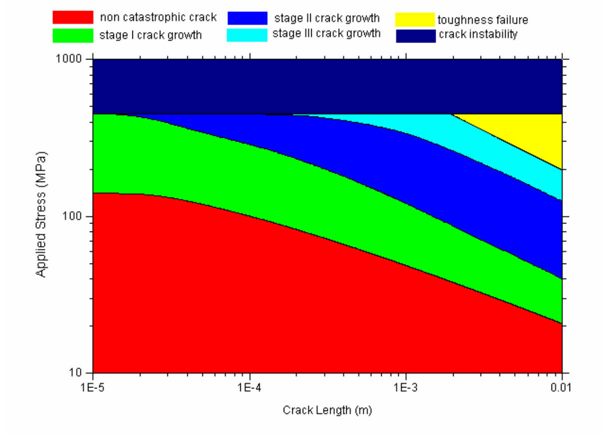


Figure 26 The FDM for 2024-T351 at R=0.1.

2.8.8 Understanding the effect of block overloading on the fatigue behaviour of 2024-T351 aluminium alloy using the fatigue damage map (C A Rodopoulos, Sheffield Hallam University and Alexis Th Kermanidis, University of Patras)

The aim of this work is to present extensive experimental evidence that would further reinforce the fact that block overloading can produce phenomena of either crack growth acceleration or retardation depending on its magnitude, duration and the fatigue damage stage characterising its onset. Compared to the majority of similar experimental works in the field, the investigation is based on the classification of identical overloading conditions applying to four distinct crack lengths namely, microstructurally short, physically short, long crack and very long crack. The above is done in order to deliver a better understanding towards the capacity of the crack length to affect the post-overloading response, which in many cases and in many works has been hidden from the use of ΔK oriented experiments. The work is based on the popular 2024-T351 aluminium alloy specifically chosen to allow comparison with similar works. The work concludes that the response of the material to block overloading is complex and is represented by a mixture of acceleration or retardation mechanisms, the strength of which depends on the aforementioned parameters and the corresponding activation of load interaction mechanisms, Figure 27. The Fatigue Damage Map is used to incorporate the 4 distinguished load interaction mechanisms such as crack tip shielding effect, strain hardening, plasticity induced crack closure and flow resistance degradation which prevail at crack growth stages ranging from microstructurally short to unsteady long crack growth. The boundary conditions indicating the onset of such load interaction mechanisms have been determined and incorporated into the FDM (Figure 28). The above methodology allows the mapping and therefore the discrimination of the overloading conditions in terms of crack growth retardation or acceleration and hence life gain or loss. The methodology allows damage tolerant engineers to identify stress envelopes that will maximise fatigue life [23-24].

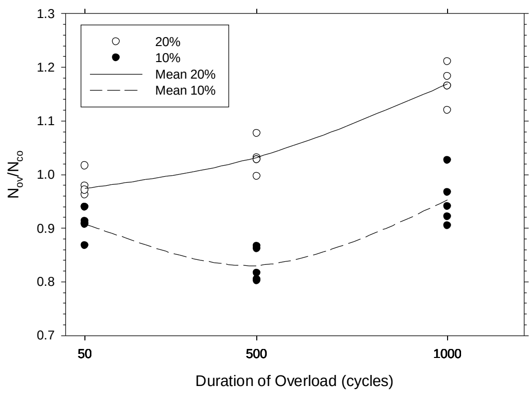


Figure 27 The effect of overloading a microstructurally short crack (23µm) with base line stress 300 MPa and R=0.1 on final life as a function of the duration of the overload and overloading range

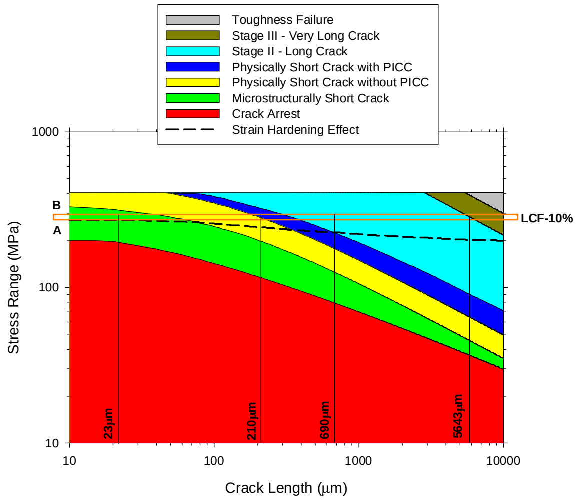


Figure 28 Integration of the testing conditions for LCF and 10% overload with the FDM

2.8.9 Development of a technique for direct estimation of testing toughness of aerospace aluminium alloys (S H Hashemi, R Gay, I C Howard, Yee Han Tai and J R Yates, The University of Sheffield)

Recent experimental work and computational studies on the ductile failure behaviour of high-strength high-toughness materials like aerospace aluminium alloys and gas pipeline steels have shown that Crack Tip Opening Angle (CTOA) can be effectively used to characterise the material fracture resistance in the case of large amounts of stable crack propagation.

One of the main difficulties currently limiting the more extensive use of the CTOA fracture toughness parameter is its practical evaluation, either in the real structure or in a laboratory-scale experiment. Although there are a number of test methods available for CTOA calculation, their implementation requires a combined test and finite element analysis and the use of experimental load-deflection data for tuning processes.

The authors have recently developed a novel laboratory test technique for direct measurement of the critical CTOA. Fracture tests are conducted on a modified double cantilever beam like specimen. The test samples have a long ligament to allow extensive crack growth and flat side-grooves of large dimension to facilitate photography of the progressing crack tip. A fine mesh is scribed on the side-groove on both sides of the specimen for accurate study of the crack opening profile. Optical imaging is used to register the uniform deformation of the reference mesh during the fracture experiment. The variation of the slope of the deformed gridlines near the progressing crack tip is measured from captured images. Its value is a representative of the material CTOA.

So far the method has been successfully used to determine the steady state CTOA of 2024 and 6005 aluminium alloys and gas pipeline steels of grade API X80 and X100. In each case the test approach was able to generate highly consistent CTOA data even from a single specimen. The extensive data set allowed the statistical analysis of the variance of the measured CTOA.

Stable values of CTOA have been measured with a standard deviation of less than 1°. Values determined are consistent with those found with other, more complex, methods [25-29].

2.8.10 Fatigue damage assessment using thermoelastic stress analysis (J R Yates, The University of Sheffield)

This items provides an update on the last UK review, but, for completeness in this review, the background to the development is reported here as well.

Accurate crack detection and assessment of the fatigue damage in industrial components has been a major area of interest and research over the last decades. In this sense, the ability to make reliable stress measurements at the crack tip is an essential part in the understanding of the fatigue process.

In recent years, the advances on infrared thermography together with the development of infrared starring array radiometer detectors have made it possible to apply all this technology to fatigue damage assessment. Such an example is Thermoelastic Stress Analysis (TSA). This experimental technique is non-contacting and non-invasive, making it possible to infer the in-plane stresses on a solid structure by computing the small temperature changes induced as a result of a repetitive loading. The fundamentals of this technique are based on the principle of the thermoelastic effect.

From the fatigue point of view, TSA constitutes a breakthrough over other conventional techniques for the experimental calculation of the stress intensity factor (SIF). With TSA the stress intensity factor is directly obtained by computing the cyclic stress field ahead of the crack tip, which make it possible to look at the actual crack driving force for the fatigue advance.

In the current work a novel approach for the calculation of the SIF range from thermoelastic images corresponding to fatigue cracks is presented. The new approach is based on the Multi-Point Over-Deterministic (MPOD) method developed by Sanford and Dally. From a thermoelastic image, a set of data points from the region surrounding the crack tip are collected. Subsequently, a mathematical expression describing the crack tip stress field based on Muskhelishvili's complex potentials in series form is fitted to thermoelastic data. As a result of the fitting process, the SIF range can be inferred. Additionally, the proposed method makes it also possible to include the crack tip location during the fitting process, making potentially feasible to monitor the fatigue crack path from the thermoelastic images. In order to asses the previous methodology, a set of fatigue tests have been conducted using steel Single Edge Notched (SEN) specimens. Results show the ability of the technique to accurately predict the SIF range and locate the crack tip (Figure 29) [30-33].

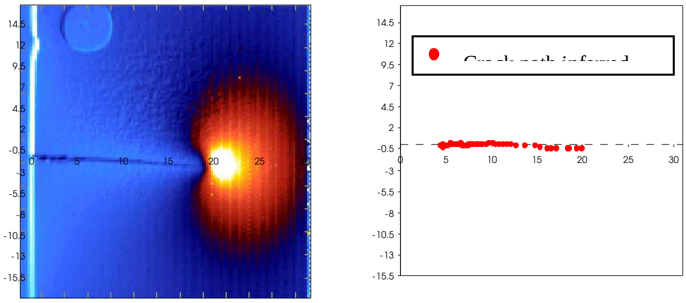


Figure 29 Fatigue crack path inferred by processing thermoelastic images captured during a fatigue test performed at R = 0.1

2.8.11 Fail safety and damage tolerance of welded integral stringer panels (X Zhang, Y Li and A Theos, Cranfield University)

The project investigates fatigue crack growth behaviour and fail safety of integral stringer panels fabricated by advanced welding techniques. It is part of an EPSRC funded project with support and input from the Airbus UK, QinetiQ and Alcoa; the research consortium also includes the Southampton and Open Universities. The main objective of this task is to design and analyse welded aircraft wing panels in order to improve the damage tolerance capability. The stringer panel is made of aluminium alloy 2024-T351 and fabricated by the Variable Polarity Plasma Arc (VPPA) welding process. The sample simulates a part of aircraft lower wing skin structures. Based on linear elastic fracture mechanics, numerical simulations are performed for two configurations, two-stringer and nine-stringer panels, and three damage scenarios, i.e. mid-bay cracking, failed central stringer and crack growth in the welded stringer web due to welding defect. Welding-induced longitudinal residual stresses are taken into account in all analyses. A typical load spectrum for large transport aircraft is employed for the analysis. For the two-stringer panel life predictions have a reasonably good correlation with the test results. Based on this validation, large-scale nine-stringer panels with three manufacture options, i.e. riveted, integrally machined, and welded integral, are simulated for a skin crack under a broken central stringer propagating to two-bay length. Useful comparisons are made among the three variants. Finally remedies for improving damage tolerance and fail safety of integral stringer panels are explored. The incorporation of crack retarder straps bonded to the inner surface of an integral panel has greatly improved the fail safety behaviour of the component with dramatically increased crack growth live. A paper describing this study is accepted for publication by the AIAA Journal in 2005 [34].

Another Cranfield Group (under Professor Irving's guidance) has carried out fatigue tests on the welded stringer panels. The aforementioned numerical predictions have been validated against the test results in terms of failure modes and crack growth lives. See section 2.4.1.

2.8.12 Fatigue tolerant design in airframes and the influence of vibration suppression on crack growth in helicopters (B H E Perrett, QinetiQ)

This study of the application of flaw tolerant design methodology to rotary and fixed wing aircraft is being reported in this year's ICAF Symposium, [35]. Flaw tolerant design is compared to safe life design with reference to the requirements of FAR29 and the significance of load spectrum shape is used to highlight fundamental differences between rotary and fixed wing applications. Data are presented to show how the active suppression of vibration in helicopter airframe structure could be used to overcome problems associated with load spectrum shape and restore the ability to apply flaw tolerance based on fracture mechanics calculations.

Crack growth predictions are compared under load spectra representing helicopter airframe features, helicopter dynamic components and fixed wing structure of combat aircraft. By recording damage rate distributions within the load spectra, the work illustrates fundamental differences in the nature of growth predictions for long cracks and short cracks.

The work concludes that flaw tolerant design and on-condition monitoring for components influenced by loads from rotating mechanisms are likely to produce unsupportable inspection requirements and an increased risk of service failure. This is contrasted with fixed wing structure where it can be demonstrated that the loading environment is suited to the flaw tolerant approach with manageable inspection programmes.

Laboratory measurement of load attenuation in a helicopter airframe, achieved using Active Control of Structural Response, is used to illustrate the potential value of vibration suppression. The attenuation of high frequency low amplitude loads is shown to increase fatigue life and, more importantly, enable inspection periodicities similar to those achievable in fixed wing structure.

2.8.13 Development of an Equivalent Initial Flaw Size (EIFS) approach to the lifing of aircraft components (K Brown, A Young, K Greedus, P M Powell, QinetiQ)

In the DTI-funded project METEOR (Metallic Technologies Research Programme for Commercial Aircraft), QinetiQ has collaborated with Airbus UK to investigate the equivalent initial flaw size (EIFS) concept and its application in establishing inspection thresholdsb [36]. The incorporation of short crack growth behaviour in an EIFS model was investigated. An approach was developed which attempts to account for the closure behaviour of small cracks, in order to determine if this would reduce the stress-dependence of calculated EIFS values. The work studied the feasibility of adjusting the Elber long crack growth law to account for the recent history of the crack tip plastic wake. The resulting modified Elber crack growth law was shown to reflect the higher growth rates (compared with long cracks) for short cracks in 2024-T3 plate. However, using the modified law did not solve the problem of stress-dependence of EIFS values for the 2024-T3 open-hole specimen case, previously examined in the METEOR project. EIFS calculations were also performed for high load transfer joint specimens containing various “rogue flaws”. The EIFS values derived for the rogue flaw specimens were, as anticipated, significantly larger than those for the “quality flaw” specimens analysed in the first phase of the METEOR project.

2.8.14 Fatigue crack growth under wing spectrum loading and the application of predictive modelling (K Brown, K Man, D B Rayaprolu, K Greedus and P M Powell, QinetiQ)

In the DTI-funded project METEOR (Metallic Technologies Research Programme for Commercial Aircraft), QinetiQ has collaborated with Airbus UK to identify a suitable analytical tool for fatigue crack growth prediction under wing spectrum loading. Airbus UK carried out a re-evaluation of two long crack models, namely PREFFAS and the Newman model, and QinetiQ has carried out complementary experimental and modelling work [37].

Fatigue crack growth testing was carried out on 12mm thick 2024-T351 plate specimens to compare the behaviour under a new, generic short duration lower wing spectrum with that under an established, longer duration Airbus lower wing spectrum. Crack opening displacement (COD) measurements were made during the spectrum loading tests with the aim of determining crack opening and closure stresses. The measured value of these stresses were shown to be highly dependent on the analysis procedure applied to the load-COD data.

The effectiveness of a rapid, approximate tool for estimating fatigue crack growth under variable amplitude loading was investigated. The tool is based on the Elber equivalent constant amplitude method. The method was applied to the fatigue crack growth in 2024-T351 plate under the two lower wing spectra. The method was shown to be promising when applied to these spectra. It was shown that the predictions were critically dependent upon the magnitude of the crack opening stress, S_{op}. The values of S_{op} giving the best correlation with experimental results for both spectra were determined and their significance considered. Further work is required to assess the applicability of the method to other wing spectra and the significance of the value of S_{op} applied in the model.

2.8.15 A Round Robin to compare the performance of computer software packages in predicting service lives of helicopter components (P E Irving, Cranfield University)

The original round robin programme was summarized in the last National Review, having just been reported [38]. Additional analysis has now been carried out, enabling the conclusions to be more definitive [39]. For the sake of completeness, the background and initial findings are reported here, together with a summary of the more recent work.

13 organisations, comprising Helicopter manufacturers, users, software developers and R & T organizations collaborated to benchmark the capability of a range of 8 different software packages to predict damage tolerant lives and inspection intervals of a helicopter component. The component geometry is shown in the Figure 30 and is representative of a helicopter lift frame. The starting defect consisted of a quarter circular corner crack at the edge of the central lightening hole. Participants were requested to calculate the service life in hours to grow the crack from the starting defect size to a length of 25 mm. Participants were supplied with a service loading spectrum ASTERIX which was developed from service load data gathered on the Merlin Helicopter lift frame. They were also supplied with the Beta function for the stress intensity calculation for the range of crack lengths of interest, and a set of constant amplitude crack growth data at R values of 0.1, 0.4, 0.7 and 0.9 from threshold to final failure, for the material of the component, 7010 73651 aluminium. This data was to be used as input to the computer package of their choice. All predictions of the life were made and submitted prior to two validation tests on identical samples of identical material, subjected to the same loading spectrum and peak stress values.

The predicted results totalling 28 in all ranged from 250 flight hours to almost 12,000 flight hours- a factor of between 40 and 50. The validation tests yielded 420 and 440 flight hours. A histogram of the data is shown in the Figure 31.

Analysis of the predicted life calculations showed that there were a number of contributing factors to this unacceptable level of scatter in results. For Helicopter spectra, unlike fixed wing, the majority of the load cycles produced stress intensity ranges in the near threshold region, throughout the important early growth stage of the calculation. The gradients of the da/dN data are steep in this region and small errors or differences in curve fit can change the final life greatly. Similar considerations apply to the curve fits of the complex Beta function for the calculation of stress intensity. In addition to these errors, recent analysis of the Beta function itself has refined the original calculation in the early stages of growth, and some of the computer packages now give very accurate predictions- (and others less accurate predictions) however, this is a scaling error which does nothing to eliminate the factor of 50 variation in predicted lives

The final conclusions are that while damage tolerance crack growth prediction of fixed wing aircraft lives may be considered robust, similar prediction of helicopter components is non robust. Accurate life calculations under helicopter spectra require more accurate values of stress intensity factors, material crack growth data and load spectra than has been observed with fixed wing aircraft. There remains a significant challenge to refine the material data, the load spectrum and the stress intensity calibration to a satisfactory accuracy for damage tolerant helicopters.

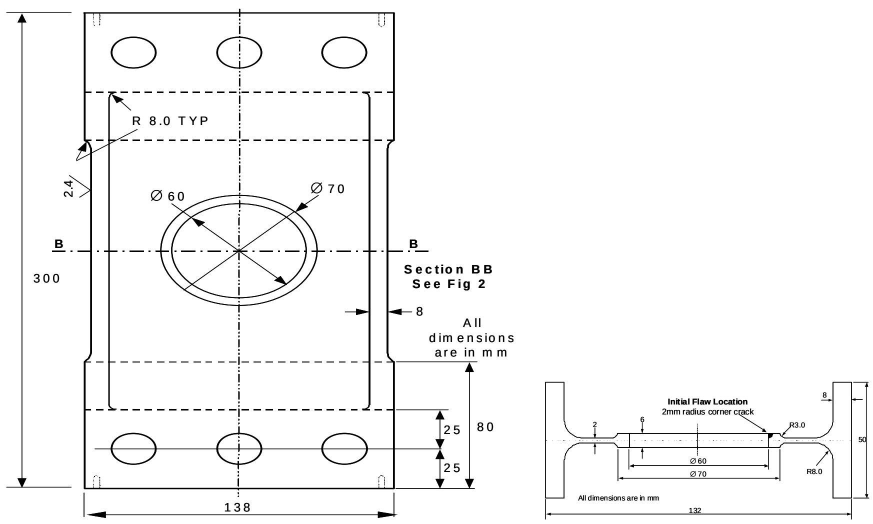


Figure 30 The round robin component

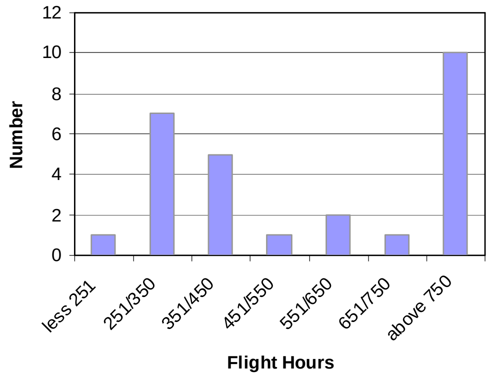


Figure 31 Distribution of predicted lives

2.9 OTHER ASPECTS OF FATIGUE

2.9.1 Testing of aged cast acrylic coupons in support of the structural integrity of Canberra canopies (C Weaver, A Simmons and R Trelfa, BAE SYSTEMS, Warton)

The Canberra aircraft operated by the Royal Air Force have all been in service for over 40 years. Consequently, ageing issues are of primary concern in managing the structural integrity of the remaining small fleet.

It is known that the canopies on the aircraft (which were manufactured from cast acrylic) may be up to 30 years of age, with the consequent possibility of a significant amount of environmental and ageing degradation having occurred. The available records of in-service usage are inadequate to accurately quantify the degree of exposure and degradation of any individual canopy.

One of a range of measures implemented to reduce the risk of failure of the canopies is to inspect the edges using a visual prism technique. A test programme has been undertaken to generate acceptance rules for the technique, and to provide data to define the inspection period.

The scale of the test programme was limited, owing to the fact that any aged specimens had to be manufactured from retired canopies. However, the programme included static testing, fracture toughness testing under increasing load and crack propagation rate testing under cyclic loading, using both new and aged coupons. The conclusions were as follows:

- As expected, tensile properties reduced with age. For the Canberra fleet, it was found that the strength may be estimated using the following equation:

$$\text{UTS (MPa)} = -0.21 * \text{AGE (in years)} + 75.1$$

- The fracture toughness reduced with age.
- Most importantly, the rate of crack propagation showed a significant increase with age.

These results have led to a recommendation that the inspection of the canopy edges using the visual prism technique should take place every 6 months.

2.9.2 Ageing airframe teardown programmes (D Taylor, QinetiQ)

The MOD has continued to use structural teardown as one of the methods of ensuring continued airworthiness of ageing aircraft platforms that remain in service. QinetiQ has been one of the key providers of this service and has now completed programmes on Jaguar, Harrier, Hawk, Canberra and VC10. These programmes have contributed to the Ageing Aircraft Structural Audit required by MOD regulation on aircraft reaching a life threshold of 15 years. The results of these inspection programmes have been reported back to the MOD customer and the Designer who between them have assessed the significance of the findings and made recommendations including, for example, special inspections to quantify the extent of problems that may exist on the in-service fleet. The teardown has also been able to quantify just how difficult it would be to gain access to certain structural elements if inspection programmes were required. In the case of the Canberra tailplane, for example, it was demonstrated that it would be impossible to design an inspection programme that would adequately address inspection needs, this led to the component having to be refurbished to underwrite its continued airworthiness.

QinetiQ has also been involved in 2 programmes of work on the teardown of fatigue test specimens. This is a relatively new role as this work has been traditionally done “in house” by the Designer. The hard won expertise in tearing down ex-service aircraft translates well into the dismantling of fatigue test specimens that in some ways are less complicated, as they come without any systems being fitted or the build up of grime and general soiling found on retired aircraft.

2.9.3 Teardown examination of the VC10 tanker/transport airframe (M J Duffield, QinetiQ)

As reported at the 2003 ICAF National Review, the teardown programme on the VC10 airframe continues as a joint effort between the UK MOD, BAE SYSTEMS and QinetiQ. The aim of the activity remains focussed on providing tangible evidence of the condition of the airframe structure that can be used to qualify the long-term structural integrity of the RAF fleet of tanker/transport aircraft. The combined fleet of CMk1K, KMk3 and KMk4 aircraft represent a mixture of specific airframe configurations with varying service histories whilst many of the aircraft are up to 35 years old.

The scope of the teardown programme was initially confined to the obsolescent KMk2 variant when these aircraft were withdrawn from service. However, the subsequent availability of a KMk4 and a CMk1K was used to address the teardown of structural features specific to these variants that are different or absent from the KMk2. The large freight door on the main deck of the CMk1K was one such feature and was the subject of a complete teardown assessment during 2004. The results indicated that the structure remained fundamentally sound with no instances of previously unknown fatigue-, corrosion- or usage-related defects being detected.

Other aspects of the programme recently completed include the teardown of several examples of fuselage skin panel joints. Here the emphasis was centred on the detection of multiple-site fatigue damage at the rivet holes in a number of lap joints and butt joints and also a check for corrosion affecting the interfacing surfaces of the joint elements. The samples for this work were obtained from one of the oldest airframes in both calendar and service life terms; however, no significant defects were detected during the comprehensive evaluation of the joints by both visual and NDT methods.

A number of features remain to pass through the teardown process but so far the results show that the primary structure of the VC10 remains robust and largely free from age-related defects. With the emphasis on the examination of features that cannot be dismantled on a routine basis, the teardown programme continues to underpin the structural maintenance programme for the VC10 aircraft, each example of which will be 40+ years old at retirement.

2.9.4 Repair Assessment Programme for ageing military transport aircraft (M J Duffield, QinetiQ)

During 2004 the Military Aircraft Structural Airworthiness Advisory Group (MASAAG) undertook a review of the emerging requirements for the long-term structural airworthiness of ageing transport aircraft. In particular, the release of the UK Civil Aviation Authority Airworthiness Notice No. 89 (AN89) at Issue 4 caused attention to be focussed on the read-across to the UK military fleet of transport-type aircraft of these requirements with respect to repairs to the primary structure. This aspect of the civil requirements has become known as the repair assessment programme (RAP).

Within AN89 the RAP is defined as the requirement to “assess the adequacy of structural repairs and their influence on inspection intervals” whilst the applicability of the requirements is classified as “all ageing transport aeroplanes used for commercial operation and certified to JAA, USA or UK requirements … Ageing aeroplanes are considered to be those that have exceeded half their published design life goal or 15 years since manufacture …” Several transport types operated by the UK military fall into the category of “ageing aeroplanes” and the certification basis of most of these types is based on UK or USA civil standards. MASAAG therefore concluded that the same concerns as discussed within AN89 are equally applicable to military aircraft.

MASAAG went on to consider the application of the RAP requirement and identified a number of aspects of the civil recommendations that would be difficult to implement within the UK military airworthiness framework. For this reason, the adoption of the RAP requirement is being tailored to each aircraft type, taking into account such factors as the original design philosophy and certification standards. Furthermore, the initial phase of implementation of the requirement is being focussed on the external pressure shell of the fuselage only; the reasons for this are totally in accord with guidance from the civil world.

2.9.5 An unusual fatigue failure (L P Pook, University College London)

About 40 years ago cracking was found in engine nacelles of two aircraft after 2000 hours flying. The cracked areas were in areas affected by jet noise. The nacelles were made from 0.56 mm thick (24 SWG) DTD 710A clad aluminium 4% Cu alloy. Two areas of cracking, one in each aircraft, were unusual in that they were remote from any stress raiser. The cracking was unexpected so a failure analysis was carried out to ascertain the cause. It was suspected by the nacelle designer that metallurgical defects might be responsible for the cracking. Although this work was carried out some time ago, at which time there was a great deal of debate on the cause of the cracking, the author has re-visited the original investigation and produced a more thorough report for a recent conference at [40]. This work also serves as a useful introduction to the next item, which describes work aimed at improving design tools to combat acoustic fatigue.

The investigation revealed a few areas of fine striations; this confirmed that the cracks were fatigue cracks. Located near the centre of each of the cracked areas, on both sides of the sheet, was a network of surface cracks that had a crazed appearance. The overall shape of the cracked areas, together with the presence of the crazed regions, suggested that failure had been caused by biaxial bending fatigue loading.

The failures were probably produced by a membrane mode of resonance excited by jet noise, and so would be the result of a very large number of cycles (of the order of 10^{10}) applied in biaxial bending at a very low stress level.

2.9.6 Acoustic fatigue (M Nash, M Harper-Bourne and F Khumbah, QinetiQ; J R Wright, University of Manchester)

Acoustic fatigue results from the pressure fluctuations on a structure caused by the passage of a jet efflux in close proximity. This effect can occur in many military aircraft, but STOVL aircraft and those with buried engines are likely to be particularly susceptible to acoustic fatigue. The level of the pressure fluctuations can be so large that the structural response is non-linear. This usually results in the rapid accumulation of fatigue damage, requiring expensive repair or replacement of the affected structure.

Two main aspects of acoustic fatigue are being studied in this programme. These are the development of improved tools for prediction of the acoustic loading environment, and for the development of more efficient ways of predicting the response of the structure to this loading. This will have the benefit of both optimising the number of experimental noise measurements to be made, and reducing the cost of acoustic fatigue analysis.

Structural analysis of acoustic fatigue is being analysed using a novel method, which extends the application of modal analysis into the regime where geometric nonlinearity is important. In the non-linear region, the modal equations of motion are coupled, so that the modal stiffness terms are polynomials containing quadratic and cubic combinations of all the mode amplitudes. These polynomial terms can be found from a controlled set of non-linear static analyses. The modal equations of motion for the structure subject to various types of acoustic loading can then be solved by direct integration in the time domain. This approach has proved to be about 3 orders of magnitude faster than conventional finite element analysis, making it possible to consider many more palliatives when trying to eliminate fatigue of the structure.

In the last year, new capabilities have been added to the specially developed non-linear modal analysis code, which is linked to the NASTRAN finite element code. The code has been modified to make sure that it can properly represent composite structures. Also, the ability to predict strains, as well as stresses has been added. Signal processing tools have been added so that 1/3 octave averaged responses can be easily calculated, for example. Excitation such as that provided by a progressive wave tube can also be easily generated. The code is in the process of being linked with one which generates the pressure fluctuations from a jet efflux, reproducing both the spectrum and the spatial correlation correctly.
The majority of the recent work has been concerned with validation studies and with application. The validation study was centred on a simple plate structure which was designed and manufactured at Manchester University. Manchester University are working in collaboration with BAE SYSTEMS at Warton. The structure consisted of a thin aluminium panel with heavy surrounding structure. The structure was vibration tested at Manchester University in order to measure the baseline mode shapes. The structure was then subjected to high level noise loading by placing it in the acoustic test facility at Warton. The data from this test was made available to QinetiQ. The structure has been modelled and the response predicted for a similar acoustic loading. One complication was the way that structure was placed in the acoustic field, as it was not placed in the wall of the progressive wave tube, but suspended within the acoustics building, which was exposed to acoustic excitation from the siren. The overall noise level was 143 dB OASPL. The excitation spectrum was measured by a microphone.

Predicted strain levels on the panel were compared with actual measurements made by strain gauges. There was a close correspondence between predicted and measured levels, as shown in Figure 32 , thus showing the efficacy of the new code.

The new code was also used to predict stress levels on the wing of a missile placed on the inner wing pylon of a Harrier aircraft. QinetiQ had carried out tests for the UK MoD to measure the excitation on the missile wing, which was found to be subject to cracking. The measured excitation spectrum was used, together with a finite element model of the wing to predict the stresses. A modified wing model, with an improved support configuration was also modelled and the stresses predicted for this. The result showed that the modifications to the support resulted in greatly reduced stresses, which would substantially increase the life. Although this was an exercise using a measured excitation spectrum, the new methods to predict the loading in proximity to a jet efflux which will be incorporated into the code will enable such exercises to be done using solely computational methods.

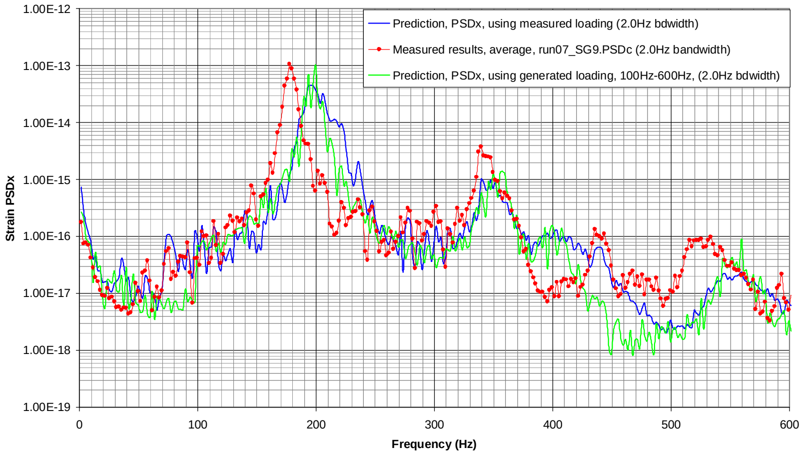


Figure 32 Comparison between measured and predicted strains

Progress has also been made in the mathematical synthesis of the pressure time history of the jet noise impinging on the airframe. This capability is needed for the new nonlinear structural analysis tool described above, which requires the actual pressure fluctuation history as an input parameter when predicting the structural response under high acoustic loading. For the synthesis of the input pressure fluctuation history a new technique, involving Finite Impulse Response (FIR) theory, has been devised and this uses the mathematical models of the jet noise sources and their spatial distribution, developed under the assignment. Thus, the pressure fluctuations around, as yet to be built, combat aircraft can be predicted this way. The technique ensures that the relevant statistical properties (spectrum, level and skewness) of the random pressure fluctuations are replicated at each point on the airframe and stores. A second, equally important, accomplishment is that the method also ensures that the spatial coherence will be correct between different points in the noise field. Noise measurements made on Eurofighter and Harrier are used in the validation process.

The new tools which are under development should allow many problems to be removed in the design stage, will aid in the development of palliative solutions where problems exist, and should reduce the need to test in expensive progressive wave tube facilities.

2.9.7 Improving the fatigue life of missile attachment structure (D Crouch, MBDA)

Introduction

The fatigue challenges within MBDA primarily relate to the air carriage environment that occurs with guided weapons on modern high performance aircraft. This environment is particularly challenging at the missile to launcher (or aircraft attachment) interface. These interfaces are controlled by STANAGS, and because of compatibility with other stores and inter-operability with NATO partners, are of relatively old designs that cannot easily be changed. However, since these launchers were designed, aircraft performance has increased with consequentially higher manoeuvre forces and vibration levels. Simultaneously with these performance improvements, the customer is demanding longer fatigue lives for his operational and training missiles. In the past, the materials used at these interfaces have been optimised, to ensure that those materials selected give optimum lives without sacrificing strength or toughness. As previously mentioned, the design of the interfaces with the aircraft is very severely constrained, but this is the major area of concern, therefore efforts are being concentrated on achieving a greater fatigue life for the weapon by obtaining a better understanding of the environment and the response of the weapon system to that environment. Therefore, research is being concentrated in the following areas.

Load Reconstruction

In 2002, as part of the applied research into vibration fatigue, a technique was demonstrated for constructing a set of ‘virtual loads’ which, when applied to a finite element model of a structure, reproduced acceleration responses measured on the vibrating structure. In addition, the ‘virtual loads’ allowed the prediction of strain responses at locations that were not included in the original analysis, and therefore offered the opportunity to predict response at locations where transducers could not be located.

The approach is based upon the development of a transfer function matrix from the finite element model of the structure and the Fourier transformation of acceleration response time histories. Predicted stress/strain spectra are inverse Fourier transformed into the time domain for fatigue processing. This technique, with some slight modification, was subsequently applied in the assessment of the ASRAAM fatigue environment.

A time domain approach to 'virtual load' reconstruction has been investigated. Although good correlations were found for the dominant frequency modes, some deficiencies were found. These include difficulties in numerically integrating measured acceleration response; and the relationship between the number of measurement transducers and the number of modes of vibration retained in the analysis. These aspects will require further investigation to increase confidence in the approach.

Proposed programmes, to address these problems, are aimed at exploring:

- The form of the FE model, i.e. the full system model, missile only or something between?
- The 'optimal' number and location of virtual forces.
- Model up dating from flight data.
- To undertake a test programme for comparison with the finite element predictions.

Frequency Based Damage Assessment

The background to this applied research programme is the availability of commercial software, now used within MBDA, to provide a capability for performing frequency domain fatigue analysis and the need to validate the approach when used in conjunction with finite element models.

The initial objectives of the programme were:-

a) To develop a finite element model of a simple component exhibiting multiple modes of vibration in the frequency range of interest.
b) To explore the sensitivity of fatigue life predictions to various model parameters; ability to predict stress concentration, damping, frequency resolution, etc.
c) To incorporate the findings of the sensitivity study into the finite element model and predict the fatigue life of the component to various vibration spectra.
d) To undertake a test programme for comparison with the finite element predictions.

The first three parts of the programme have been met, and some testing was done. However, further test data are required to complete the programme.

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