REVIEW OF AERONAUTICAL FATIGUE INVESTIGATIONS IN THE UNITED KINGDOM DURING THE PERIOD MAY 2005 TO APRIL 2007

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 2005 to April 2007. As in previous UK ICAF reviews, the individual items are grouped into topics which include developments in fatigue design tools, fatigue loads measurement, fatigue monitoring, full scale fatigue testing and damage tolerance. A list of references related to the various items is given at the end of the paper.

The compiler gratefully acknowledges 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 Developments in the fatigue analysis of undercarriage components (T Siddall, Messier-Dowty)

In the last Review, progress was reported on the strain-life approach to safe life fatigue analysis on new programmes, including full field analysis for all major structural items. Although the three major M-D sites have traditionally used their own approach to fatigue analysis, there is an on-going process to harmonise methods across all sites. It is interesting to review the reasons for this change.

The strain life (E-N) method is of course not new to Messier-Dowty; it has been used for specific applications where very high strain levels are seen during fatigue cycling and the S-N method breaks down (typically military aircraft, but also in assessing growth in civil aircraft weights.)

A typical E-N curve (when normalised to stress) is equivalent to an S-N curve in the high cycle region, so the method is applicable across all of the range of high and low cycle conditions seen by various landing gear configurations; in effect one method suits all, where S-N is not so relevant below 10000 cycles.

The E-N method has never previously been adopted as the standard approach as it is a more complex and computationally slower method to apply to fatigue analysis than a simple S-N approach, due to the method maintaining cycle history effects and using elastic-plastic conversions on the elastic stress data from FE analysis. In the past, the method has only been applied to loading/strains running in the low cycle region, with the S-N method used to cover the additional high cycle part of a spectrum (damages added together at the end of the analysis)

However, a few years ago, M-D began discussions with a fatigue software development house (nCode) that have a fatigue solver with the E-N method built in as standard. M-D has since taken on a programme of work to customise the software (traditionally based around automotive applications) to provide a fatigue analysis suite of tools for M-D landing gear simulation, whilst at the same time undertaking the characterisation in the E-N domain of all materials likely to be used, to complete a harmonised set of materials data for all of the sites to use.

However, the advances enabled by the use of this software combined with the use of full field FE based fatigue studies have raised some further questions. For example the effect of multi-axiality arises with more complex stress fields on FE models. It's an area not well covered by current fatigue methods and is one that M-D are working with the University of Waterloo (Canada) to solve for their analysis.

The validation of this new approach uses the new tool and materials data to analyse tests and in-service fatigue fractures. So far (although there are few fractures to work with) M-D have been able to predict real events with reasonable accuracy with the new approach, where traditional methods were either too optimistic or pessimistic.

The phase 1 material characterisation programme including 300M steel, 4340 (180ksi) steel, 7010 Aluminium, some ultra high strength stainless steels and Titanium alloys, under strain control, has been completed. Testing has included various R-ratios and several surface treatments and finishes to develop a comprehensive suite of tools for mean strain and surface abatement correction. Further testing is due in 2007 to cover phase 2 materials including lower strength stainless steels, plus further Aluminium and Titanium specifications.

2.2.2 Fuzzy variable and probabilistic methods applied to fatigue and crack propagation in ageing airframe structures (M Nash, QinetiQ, Farnborough)

Ageing platforms form an increasing part of the fleets of the UK armed forces. Each of these platforms has been designed for a particular life under a specific loading spectrum. In reality, each individual aircraft will be different due to variance in usage, loading, environmental degradation, material and manufacturing characteristics. The differences in past usage and structure of these platforms imply that a simple deterministic analysis cannot give an adequate assessment of the performance of the platform in its military role. To overcome some of the problems, frequent monitoring of the structure is usually a priority. An analysis which takes into account the structural variability would be most appropriate to consider the reliability of a whole fleet of vehicles, as this would identify the performance of not only the fleet average, but the worst examples in a statistical sense. Such an approach would also be more appropriate for setting inspection intervals, for example.

Probabilistic methods can establish the effect on structural performance of the variability in loading, material characteristics, environmental damage and manufacturing concessions. This paper, which is featured in this year's Symposium Poster session [1], describes the development and application of fuzzy variable and probabilistic methods for fatigue analysis using an S-N curve and for crack propagation. The latter approach uses a version of the USAF AFGROW code linked to reliability methods, response surface software and fuzzy variable membership function codes. This is possible since AFGROW has a COM (common object module) interface, which allows model properties to be changed from a remote process. This also means that all the standard solutions in AFGROW are available. The reliability analysis uses the DAKOTA code as part the process. The programmes are interfaced using the Phoenix Integration Model Center code. Use of properly configured response surfaces and well chosen limit state functions produces an approach that can rapidly assess the probability of fracture or determine the crack size spectrum after a selected number of stress cycles. Results of the analysis show either the possibility (fuzzy variable) or probability that the structure or component will not reach the desired life.

The paper is illustrated by applications to a simple structure and to a realistic airframe structure using load data from fatigue testing. It is concluded that probabilistic approaches provide a better assessment of risk, but require precise information on the variability of loading, material properties, etc. Fuzzy variables can link properties that are defined with different degrees of uncertainty, but cannot directly produce a risk of failure. They can, however, return similar sensitivity information to probabilistic methods. In general, they are more conservative than probabilistic approaches.

2.2.3 Analytical tool to determine the fatigue life of composite structures with impact damage (C Jones and F Ngah, QinetiQ, Farnborough)

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A simple, user friendly analytical tool was developed, which predicts the number of cycles to initiate delamination growth from a known damage size [2]. The interface was developed in Microsoft Excel, coupled with a database populated for specific materials, lay-ups, airframe geometries, in-plane strains and out-of-plane displacements. The database was populated using outputs from the FE code ABAQUS.

An FE model was developed with in-plane and out-of-plane loading applied to the damaged panel. The model was validated statically through experimental tests, with both in-plane and out-of plane loading applied.

Delamination growth from an impact damaged (35J) skin/stringer structure was modelled using principles of linear elastic fracture mechanics (vcct) and employing the ABAQUS FE analysis package. The model was developed using four-noded, linear, reduced integration quadrilateral shell elements.

Following static validation, data were extracted from the static computation and fracture mechanics data obtained from experimental mode I and mode II cyclic coupon data were used to determine the initiation values for various loads. A plot of GT (GI + GII) versus number of cycles to initiation (Ni) was produced. Then, from the static ABAQUS computation, the total strain energy release rate GT was calculated around the boundary; this will be a maximum at a certain location (usually on the defect boundary at 90° to the loading direction) and was plotted against the applied load. This gives information on the value of achievable GT for an applied load level, on a panel containing a certain defect size. From this output, by equating GT to a G curve from the experimental data, curves of load against number of cycles to initiation can be developed for a given initial defect size.

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

Data Item No. 05007 Strain-life data for type 316 austenitic stainless steels at temperatures between -269°C and 816°C (-452°F and 1501°F)

ESDU 05007, which is due to be issued in June 2007, presents strain-life data for type 316 austenitic stainless steels, including the low carbon and high nitrogen variants. The results of more than 300 tests performed at temperatures between -269°C and 816°C (-452°F and 1501°F) and at strain rates between 0.01×10^{-3} and 6.7×10^{-3} s^{-1} are given. The data are presented as plots of total, plastic and elastic strain amplitude versus the number of cycles to failure; the plots are accompanied by tables giving supplementary information regarding the material and test conditions. The data are discussed in detail and the results of a multiple linear regression analysis of the data are presented, culminating in an expression for predicted fatigue life as a function of total strain amplitude, temperature, strain rate and carbon and nitrogen content. A summary of the introduction to low-cycle fatigue phenomena presented in ESDU 04022 (described in the previous National Review) is given; this includes the derivation of the strain-life relationship and a discussion of the approximation of fatigue properties using monotonic data. A worked example illustrating the use of the data and of the expression resulting from the regression analysis is included. The effect on measured fatigue life of using alternative failure criteria is discussed in an appendix to the Data Item.

Data Item No. 05008 Cyclic stress-strain response of type 316 austenitic stainless steels during low-cycle fatigue at temperatures between -269°C and 800°C (-452°F and 1472°F)

ESDU 05008, which is also due to be issued in June 2007, presents stress-response data and cyclic stress and strain data for austenitic type 316 stainless steels at temperatures between -269°C (-452°F) and 800°C (1472°F). Tests are carried out on 316 stainless steels, and low carbon or high nitrogen 316 stainless steels. The low cycle fatigue data are examined using the methods described in ESDU 04022.

The evolution of stresses per cycle during testing, or the stress response, is discussed in terms of the initial and peak stress amplitude and the number of cycles to failure. The effect of temperature, applied strain amplitude and nitrogen content of the steels tested on the stress response is examined. The degree of hardening, which is calculated as the percentage difference between the initial and peak stress, for all data is considered with an examination of the hardening and softening behaviour of the steels.

This Item presents data for half-life cyclic stress-strain curves from tests carried out between room temperature and 800°C (1472°F) which are analysed to determine the cyclic strength coefficient, K'_{sc}, and the cyclic strain-hardening exponent, n'. An analysis of the effect of test conditions on K'_{sc} and n' is provided with methods to calculate these constants using fatigue strength and ductility constants (σ'_f, ε'_f, b and c) from ESDU 05007. The effect on fatigue life of temperature, applied strain amplitude, strain rate, and carbon and nitrogen content for stress-response data and strain-life data is analysed using a linear regression analysis presented in ESDU 05007.

A worked example using the linear regression equation and a comparison with actual data is provided. A second example of calculating the values K'_{sc} and n' is also given.

Data Item No. 69023 Average gust frequencies. Subsonic transport aircraft

Work will begin shortly on a comprehensive update to ESDU 69023. The data currently presented were obtained from accelerometer VG and VGH records mainly from subsonic transport aircraft in scheduled operation. Curves present the average distance flown by aircraft, with and without cloud warning radar, at various altitudes up to 55 000 ft, before experiencing a positive or negative gust of 10 ft/s. Details of all the aircraft flight histories are tabulated. Also presented, are curves of relative frequencies of gusts of various amplitudes at different altitudes (the equations for these curves are given) and the gust alleviating factor as a function of aircraft mass parameter and aspect ratio. A simple method outlines how the user may compare the flexibility of the aircraft wing with that from which the data have been obtained (the expressions used are fully discussed). An example illustrates how the curves may be employed to compile cumulative frequency gust spectra for an aircraft wing. The update will consist of the addition of a considerable number of new data that are known to exist and of details of the methods of analysis that are currently used.

Cyclic stress-strain and strain-life data for aerospace materials

Local strain-based fatigue analysis has been in use since the 1970s; its applications being initially rooted in the military aircraft and automotive industries of the US where much of the developmental work was based. The technique tracks conditions of local stress and strain at a notch root and in order to estimate life two sets of data are required for each material - cyclic stress-strain data and strain-life data. Most of the large number of data initially produced are held privately by industry, and relatively few data entered the public domain. The technique is now widely accepted throughout the aerospace industry, yet the extent of published data (whilst increasing) remains limited and much of the testing continues to be done on a private basis. In many cases this can involve duplicating tests carried out by another company on the same material.

ESDU began work in 2006 on the collection and collation of the raw cyclic stress-strain and strain-life data points for aerospace metallic materials. Beside the data available in the literature and other readily-accessible sources, efforts are being made to encourage organisations with their own data to contribute to a database from which they would then benefit. Considerable support and enthusiasm have been expressed for this project and generous donations and promises of data have been received from a number of companies. Such generosity is clearly vital to a project of this nature and fundamental to its success and the subsequent benefit derived by the aerospace community as a whole. With this in mind, any contributions of data would be warmly welcomed; please contact Adam Quilter via Adam.Quilter@esdu.com or via +44 207 566 5704.

2.2.5 Fatigue crack growth predictions using the BEASY code (R Adey and S Mellings, BEASY Inc)

Background

Development has continued on computational tools to simulate the behaviour of cracks and damage in structures. This has resulted in the release of version 10 of BEASY containing many new capabilities and features. The major developments are summarised below. Note that there is a paper in this year's Poster Session covering some of these features [3].

Improved stress intensity factors

3D J integral has been added to the methods available for stress intensity factor calculation. The J integral has been incorporated with the automatic crack growth capabilities to provide improved accuracy in the model results for stress intensity factors, crack growth rate and crack path. The J integral is less sensitive to the model mesh than other methods of stress intensity factor evaluation and requires fewer elements to provide high accuracy.

Crack Wizard

The crack wizard is a tool which guides the user through the process of performing a fracture analysis, automating many of the tasks and performs all the meshing operations. It has recently been redesigned to capture typical processes an engineer may wish to perform when assessing crack behaviour or damage tolerance. For example users now simply select the process (e.g. compute the SIF values for a crack located at a certain position in the model as its size varies from some initial value to a maximum value, alternatively the process may be to perform a fatigue crack growth study where the initial crack is specified and the software automatically predicts the crack path and shape as it grows).

Riveted Panel Connections

Often, particularly in aircraft structural modelling, it is necessary to analyse the connection between two panels in an assembly. An example of this type of modelling is where two or more panels overlap for structural stability and the connections are attached by rivets. The rivets cause complex point loads into the panels of the model and often there are stress concentration points around the rivet attachment points. In these assemblies the stress field can be highly complex, which in general would lead to a complex crack growth path.

Recent developments enable this type of situation to be modelled including the contact between the fastener and the panel. The model shown below consists of two overlapping panels.

Figure 1 shows the full model which consists of two overlapping parts, a loaded panel with a central hole and a cover. The panels are connected together with a series of rivets, shown by the circles in the pictures. The lower panel is loaded using a bi-axial tension loading.

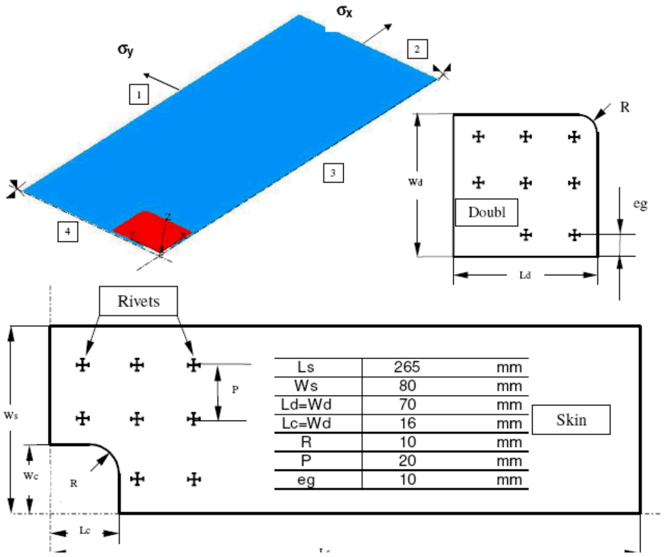


Figure 1 Geometry of overlapping panels

In this application, the rivets are modelled using two “special fastener elements” which act in the same way as a rivet to connect the two panels. It provides a connection between the loaded base panel and the overlapped, doubler panel. These elements allow the transfer of direct loads but do not transfer moments.

A crack has been initiated from the edge of the panel and the predicted crack path and crack growth rate is shown in Figure 2 and Figure 3.

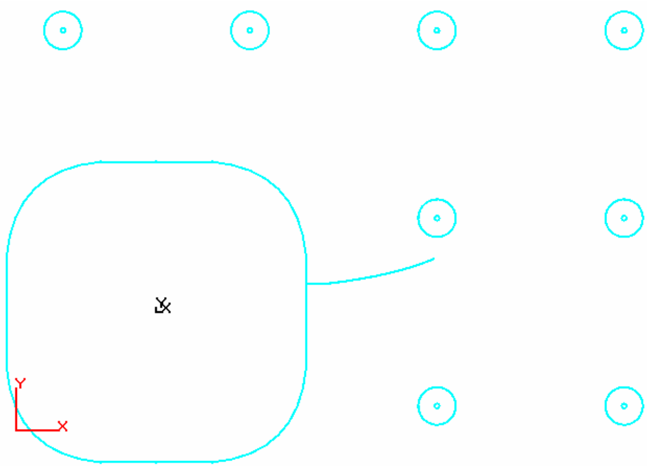


Figure 2 Crack growth path with offset initiation position

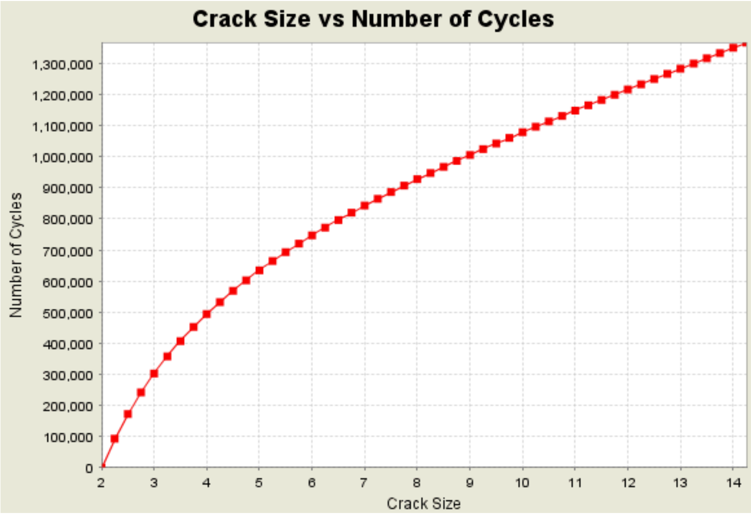


Figure 3 Predicted crack growth rate

Crack Growth in Shells and Plate Structures

Recent research has developed a hybrid Finite Element (FEM)/Boundary Element Method (BEM) for predicting the behaviour of cracks in shell type structures. The approach uses a FEM model to represent the overall structure and a BEM model to represent the local region near the crack. As the crack grows the redistribution of the load and stresses is captured by the overall FEM model while the BEM model represents the complex stress fields near the crack.

This method enables cracks and crack growth to be simulated in large scale aerospace type structures. An example is shown in Figure 4.

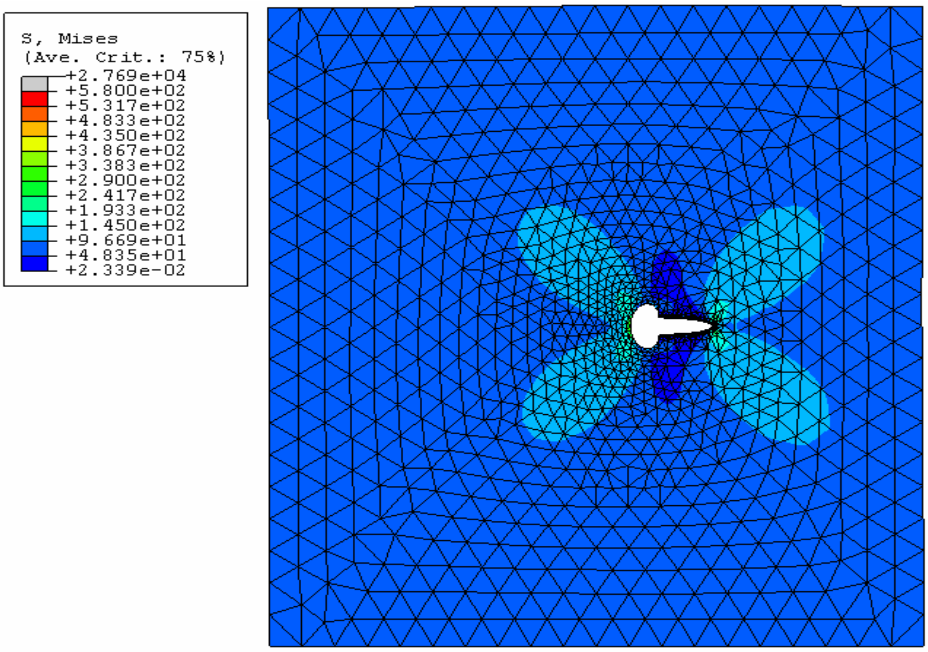


Figure 4 Example crack growth simulation in FEM shell structure

Corrosion in Aerospace Structures

Modelling tools have been developed to simulate corrosion in aerospace structures and testing has been performed against experimental data. This work has now developed into a larger EU funded project (SICOM) which aims to develop a package of simulation tools capable of studying a wide range of corrosion types.

2.3 FATIGUE LOADS MEASUREMENT AND FATIGUE MONITORING

2.3.1 The RAF Hawk Operational Loads Measurement Programme: Supporting an Aging Aircraft Fleet (M Gelder, S Roberts, BAE Systems, Brough, S Reed, QinetiQ, Farnborough)

This is the subject of a paper to be presented at this year's ICAF Symposium [4], and the following summarises its content and the background to the programme.

The BAE Systems Hawk TMk1 aircraft entered service in 1976 and has now accrued more than a million flying hours in Royal Air Force (RAF) service. It was originally intended that the life of the aircraft would be 6,000 flying hours and 15.000 landings over a 20 year operating period within a Flying Training School (FTS) role. However, almost as soon as the aircraft entered service the RAF expanded its role to include Tactical Weapons Unit (TWU) flying, and in 1980 the RAF Aerobatics Team (RAFAT), better known as the Red Arrows, replaced their Folland Gnats with the Hawk.

For many years the aircraft have also been used by 100 Squadron for target facilities training, Fast Jet Navigator Training (FJNT) and to support the Joint Forward Air Control Training and Standardisation Unit (JFACTSU); by the Fleet Requirements and Air Direction Unit (FRADU) in support of the Royal Navy's training programme; and by the RAF Centre of Aviation Medicine (RAFCAM) for the resolution of clinical issues, aircrew equipment trials and aircrew training.

Due to the successful in-service operation of the aircraft the RAF initiated a Life Extension Programme (LEP) in the late 1980s. The aim of the LEP was to extend the Out of Service Date to 2010 with 10,200 operational flying hours and 25,000 landings. The main components of the programme were the continued running of the full scale fatigue test (FSFT) with a new wing standard; the analytical clearance of items not fitted to the FSFT; the assessment of all build concessions and in-service repairs and modifications; and the development of new inspection and modification requirements. A central element to achieving the aims of the LEP was a thorough understanding of the in-service operation of the aircraft and the development of component level usage spectra.

This was achieved through an Operational Loads Measurement (OLM) programme involving two aircraft, XX260 and XX319. XX260 operated in the RAFAT and XX319 in the main fleet, principally at RAF Valley in an FTS/TWU role. Data collection began in 1997 with over 500 good flights being reported on by 2000. At the start of the programme the intention had been to only collect data for two years in support of the LEP, but the value and versatility of the OLM system on the two aircraft was recognised and data collection continued until 2006.

The original aims of the OLM programme (assessing the adequacy of the structural monitors, the FSFT derived clearances, the dynamic response of the airframe, and identification of particularly damaging flight regimes) were continually assessed throughout the ten year data collection period. Throughout this time though, the emphasis and reasons for data collection have changed. Examples of these are:

- periodic revision of OLM derived sortie profile specific coefficients for the fin and tailplane fatigue formulae
- support to the clearance activities associated with Hawk Fuselage Replacement Programme (FRP)
- support to the Test Based Analytical Clearance (TBAC) activities which were undertaken when the running of the Hawk FSFT was stopped in preparation for teardown
- confirmation that the aft false spar cracking of the tailplane FSFT specimen was test induced
- support to the redefinition of the spectrum of buffet loading applied to the dedicated tailplane FSFT
- confirmation that a programme of relatively simple inspections was adequate for ensuring the structural integrity of the rear fuselage strakes, instead of a complex and expensive replacement programme

At the time of writing the official Hawk TMk1/1A out of service date (OSD) remains at 2010. The RAF are in the process of acquiring the latest Hawk variant, the Mark 128, which replace a number of the RAF Valley based aircraft. However, it is expected that a further extension to Hawk TMk1/1A service life will be sought for many of the operational units. This would include the RAFAT and any such additional clearance will involve the resurrection of the OLM programme.

The Symposium paper describes the instrumentation installations of the two OLM aircraft and the various phases of data gathering since 1997. It presents the principal objectives and findings from each phase, and how these impacted and influenced the structural integrity management of the fleet. Additionally, future OLM requirements for the Hawk TMk1/1A are discussed briefly. Finally, the paper compares the Hawk TMk1/1A OLM system with the Fatigue

Monitoring System (FMS), which is fitted to all new build Hawks, including the Hawk Mk128 which is expected to enter RAF service in 2008.

2.3.2 Guidance for Aircraft operational loads measurement programmes (S Reed, D M Holford, QinetiQ, Farnborough)

Aircraft Operational Loads Measurement (OLM) programmes are now firmly mandated in UK MoD policy and there is general acceptance within the military aviation community that these programmes need to be undertaken. However, OLM is technically complex, costly and often takes many years to complete. This means that, certainly from the UK MoD's side, it is highly unlikely that an OLM programme will be completed on any one person's tour of duty; this brings great inefficiency. From an industry perspective, much hard won experience is frequently concentrated in a few key personnel. Furthermore, where programmes are cyclic rather than continuous, knowledge gained in industry is often lost in the intervening period between programmes. Additionally, in-depth technical and management problems have occurred in many OLM programmes; these have incurred additional cost, time delays and, in some cases, undesirable compromises have had to be made to make progress. Therefore, there is a very real need for more detailed guidance to be made available for those involved in OLM programmes.

Therefore, under the auspices of the Military Aircraft Structures Airworthiness Advisory Group (MASAAG) and with funding from the UK MoD Structural Integrity Branch, a paper has been written to guide those involved in OLM programmes through the planning, installation, data capture and analysis, and reporting phases of the programme. The reasons behind each of the steps in each phase are explained in generic terms and the key activities are captured and have been concatenated into a generic Statement of Requirement for use as a starting point in OLM planning.

This draft MASAAG Paper has been circulated to MASAAG members and OLM practitioners for their consideration and comments. A workshop to discuss the draft MASAAG paper was held at QinetiQ Farnborough on 22 March 2007. OLM practitioners from various projects within BAE Systems, Bombardier Aerospace, Marshall Aerospace and the UK MoD attended the workshop and provided valuable input and comment on the draft paper. The final publication will be released as MASAAG Paper 109 [5] before the end of May 2007.

2.3.3 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 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 into 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.

The OLM data capture exercise began on 14 Jan 05, with the post-major maintenance flight test. The aim of the programme was to capture 300 sorties of data and by the 8 May 06, when the programme was terminated, 319 sorties of data had been recorded. During the period of the programme, the aircraft flew 348 sorties. Hence a total data capture rate of 92% was achieved, due largely to the excellent support of the SERCO Team at RAFC Cranwell. The lost sorties occurred between 28 Jul 05 and 1 Sep 05, when an apparent transient voltage in the aircraft power supply damaged one of the data acquisition cards and prevented recording until a replacement card could be sourced.

The 12 strain gauge channels remained serviceable throughout the programme and all 319 sorties of strain data were processed and considered satisfactory for use in the damage tolerance analysis. There were a few minor problems with the normal acceleration (NZ) channel but these were considered secondary and did not affect the core data needed for the programme.

Initially, it had been 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. In the event, however, this work was also undertaken by QinetiQ. The team '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 was used as the basis of this analysis. The results were 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.

The "AFGROW" analysis showed that intense flying at low-level and the use of the aircraft for pilot continuation training are two particular aspects of Dominie usage that have a considerable effect on the crack propagation predictions compared with the civil datum. The very large numbers of relatively small stress cycles generated by low-level, high-speed flying were observed to be significant and the effects of roller landings were also determined; both of these aspects of military flying were entirely absent from the original civil stress spectrum, resulting in a considerable divergence in the calculated crack propagation rates. This is a clear demonstration of the value of OLM which, in this case, has enabled these aspects to be identified and quantified where, previously, there were no reliable data available.

This paper is featured in this year's ICAF Symposium Poster Session [6].

2.3.4 Structural monitoring systems using non-adaptive prediction methods - Proposed Defence Standard (S Reed, QinetiQ Farnborough)

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 supplements 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, and 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 has 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, ESS 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 Defence Standard, issued as MASAAG Paper 107A [7] has now been endorsed by the UK Joint Airworthiness Committee and shall be issued within the next revision of Defence Standard 00-970.

2.3.5 RAF VC10 fatigue monitoring by NzW analysis (M J Duffield and K A Lucas, QinetiQ, Farnborough)

The VC10 has been in service with the Royal Air Force for 40 years, initially in the strategic air transport (AT) role, which continues to this day alongside the air-to-air refuelling (AAR) role for which all of the fleet are now equipped. There remain three distinct versions of the VC10 in service, namely the CMk1K tanker/transport and the KMk3 and KMk4 tankers. Fatigue monitoring of the KMk3 and KMk4 variants is accomplished by the use of the well-proven Fatigue Meter Formula methodology which relies on the flight-by-flight recording of 'g' spectra and various aircraft parameters. These data are processed by an empirical formula to generate estimates of fatigue life consumption for each airframe and are also used to monitor usage severity relative to the datum that underpins the structural maintenance regime.

There is no Fatigue Meter Formula available for the CMk1K and, historically, structural usage was monitored on the basis of flying hours and landings accrued. It has been recognised that, in the face of evolving usage patterns, the overall age of the CMk1K fleet and the adoption of damage tolerance criteria, this approach may not be adequate to fully assure structural airworthiness up to the expected retirement date.

Consequently, an alternative method has been developed by QinetiQ at Farnborough that makes use of the same data as routinely recorded to enable an equivalent level of structural usage monitoring to be achieved for the CMk1K. The method combines the key parameters of aircraft mass, sortie profile and 'g' spectra with the results from the finite element models used for the damage tolerance qualification of the primary structure. By working in close cooperation with the Design Authority for the VC10 (BAE SYSTEMS at Chadderton), the resulting analysis process was reduced to a set of simple algorithms that can be used in a spreadsheet application interfacing directly with the database of flight records. This ensures that the method does not introduce any additional data capture requirements and the analysis can be undertaken at any convenient time according to the demands of the Airworthiness Authority for the aircraft.

The principal elements of the calculation process are the sortie profile code (SPC), the total mass of the aircraft at start (typically referred to as the Ramp Weight) and the maximum ‘g’ level (or normal acceleration, Nz) recorded for the flight. The SPC provides the link back to the finite element model results which are the source of a defined set of stresses for each sortie type forming the basis of the corresponding fatigue calculations. These calculations are performed according to a set of algorithms which combine the Ramp Weight and Nz with the appropriate fatigue endurance (s-N) curve; for this reason, the method is generally referred to as NzW and a typical set of algorithms are plotted below in Figure 5 for illustrative purposes:

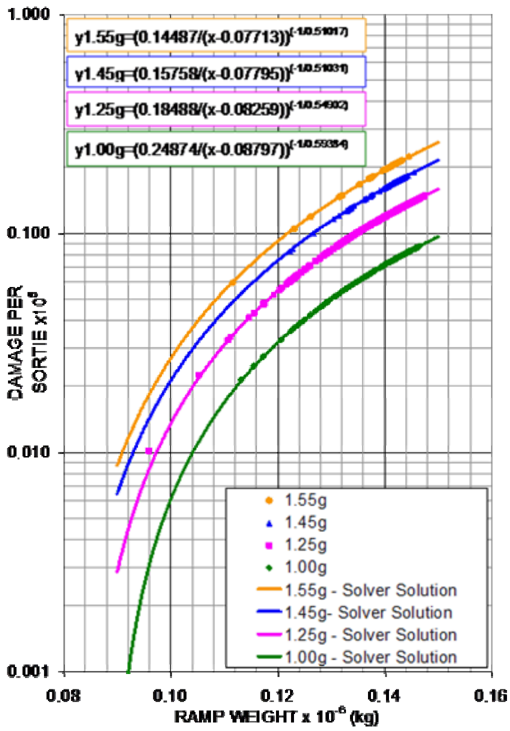


Figure 5 Typical plot of damage against ramp weight for different g levels

Various corrections are applied to the results to account for additional sources of damage - such as roller landings - or sorties flown outside of the set of reference SPCs and for which specific stress data are not available.

The s-N curve has been validated by reference to the results from the teardown analysis of several high-time VC10 airframes and so the results from the NzW analysis are considered to be equivalent to the traditional Fatigue Meter Formula for the purposes of monitoring structural usage of the VC10 CMk1K fleet. The principles underpinning NzW have also been applied retrospectively to enable the whole service history of individual aircraft to be assessed to a degree not previously possible due to the lack of comprehensive flight records prior to the mid-1990s. The results have shown that there are important structural considerations to be recognised in the management of the fleet in order to ensure that the planned out-of-service dates may be met without erosion of airworthiness standards or loss of operational capacity

2.4 FULL SCALE FATIGUE TESTING

2.4.1 Major fatigue testing activities at Bombardier Aerospace (L Murray, Bombardier Aerospace, Belfast)

Most of the on site activity in Belfast has centred around the full scale fatigue testing of the Learjet 45 business jet airframe, designated test article TA05, the second full scale fatigue test on the RAF Shorts Tucano training aircraft and fatigue testing of two RTM aileron specimens for the Bombardier CRJ Regional Jet family. These are described below.

The Stress Department in Belfast has also provided technical support to five major test rigs at the Bombardier site in Montreal. Belfast has responsibility for fuselage structure on CRJ 200/700/900 Regional Jets and the Challenger 300 business jet.

The CRJ 700 forward and aft Fuselage tests, designated 0004 and 0006 respectively, have completed 160,000 cycles of fatigue testing (2 lifetimes) and are in preparation for Residual Strength Testing. The Centre Fuselage Test, designated 0005, has completed 121,000 cycles.

The CRJ 900 Centre Fuselage Test, designated 15995, has completed 123,000 cycles.

The Challenger 300 has completed 37500 cycles of fatigue testing (2.5 lifetimes) and is currently in the Residual Strength Test programme phase.

Learjet 45 Fatigue Testing

The Learjet 45 Wing and Fuselage fatigue and damage tolerance test (test article designation TA05) shown in Figure 6 has completed all Fatigue Testing: 59,200 flights (3 design lifetimes), 2 of durability testing and one lifetime of damage tolerance testing with artificial damage introduced. The loading applied represents gust, manoeuvre, and pressurisation loads to an altitude of 51,000ft.

Eight residual strength tests (as noted below) have been successfully completed.

1) Condition 1a & 1b
Engine beam up load without/with thrust
2) Condition 2a & 2b
Engine beam down load without/with thrust
3) Condition 6
1 G wing up bending/fuselage down bending with 1.15 x 9.9psi (11.39psi), going to 105 % (11.96psi) limit cabin pressure.
4) Condition 3
Fuselage up bending - aft shear without/with 9.9psi cabin pressure
5) Condition 4
Fuselage down bending - aft moment without/with 9.9psi cabin pressure
6) Condition 7
Wing down bending/fuselage up bending - wing shear moment torque - fwd fuselage moment without/with 9.9 psi cabin pressure
7) Condition 5
Fuselage up bending - aft fuselage moment fwd fuselage shear without/with 9.9psi cabin pressure
8) Condition 8
Wing up bending/fuselage down bending - wing shear moment torque - fwd fuselage shear moment aft fuselage shear without/ with 9.9psi cabin pressure

Figure 6 TA05 TestArticle

Tucano Full Scale Fatigue Test

The Tucano T.Mk.1 2nd Full Scale Fatigue Test commenced in January 2005. The test article shown in Figure 7 is the last RAF production airframe (designated T132) and includes the wings, fuselage, tail-plane and fin. The airframe also includes the fatigue modifications arising out of first full scale fatigue test.

The test spectrum has been developed primarily from Operational Load Measurement data recorded on three 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 are being sampled at regular intervals throughout the test. These have proved useful in both monitoring the performance of the test and in highlighting fatigue cracks local to the gauges.

At present the test has completed 10 spectrum blocks representing 10000 flying hours. A number of damages have been recorded on the test, some resulting in the introduction of fleet inspections. A GLARE repair patch is being substantiated on the test, for possible introduction to the fleet as a preventative repair. Considerable work has also been undertaken in demonstrating the running accuracy of the test.

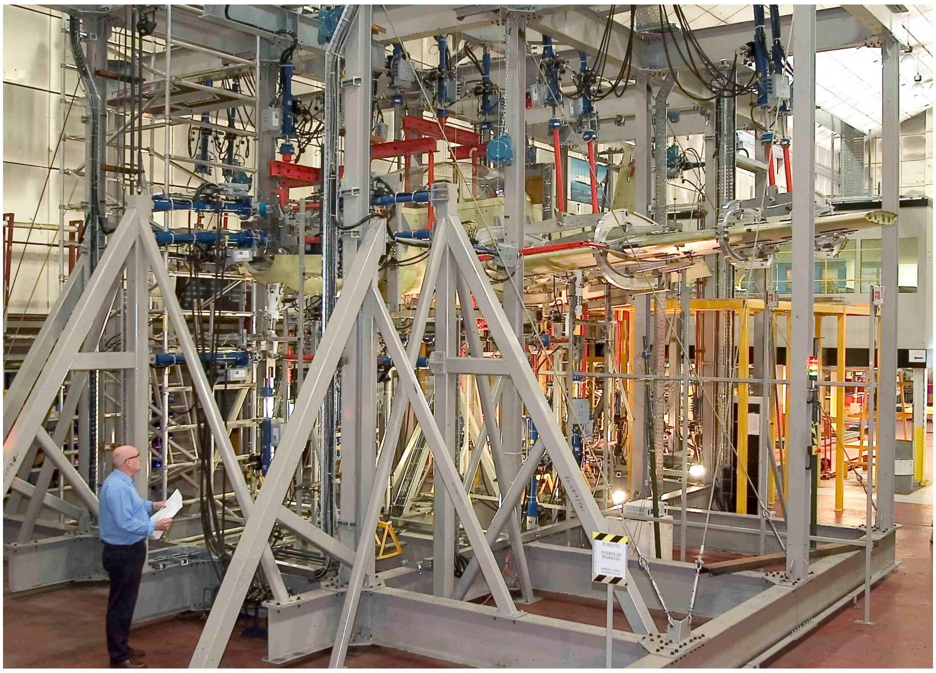


Figure 7 Tucano Full Scale Fatigue Test Specimen

Fatigue Testing of RTM Ailerons for the Bombardier Regional Jet Family

One specimen has been tested for Composite Fatigue and one specimen tested for the metal components in the ailerons.

The composite fatigue specimen was tested in an environmental chamber at 40°C and 82%RH as shown in Figure 8. The Spectrum Load Enhancement Factor was 1.15 and only high strain load cycles (with truncation at 30% Limit Load) were applied. Following testing to 2 lifetimes, a Residual Static Test to Ultimate Load was conducted.

The metal fatigue specimen was also tested to 2 lifetimes (160,000 cycles) using a load spectrum developed for the metallic components. Following fatigue testing, a successful Residual Static Strength Test was conducted.

Figure 8 RTM Aileron in Environmental Chamber

2.4.2 Lead-In Fighter Hawk Mk.127 Full Scale Fatigue Test (FSFT) (R W Young, R M Aaron and G F Duck, BAE SYSTEMS, Brough)

This is the subject of a paper to be presented at this year's Symposium Poster Session [8], which describes the background and test definition of the Lead-In Fighter (LIF) Hawk Mk.127 Full Scale Fatigue Test. This aircraft is in service with the Royal Australian Air Force (RAAF). The LIF Hawk test is the third full scale fatigue test of a Hawk variant, therefore many aspects of the test are similar to previous tests.

The FSFT is the responsibility of BAE SYSTEMS, who have carried out the development work on loads and spectrum. DSTO are carrying out the test in their facility at Fisherman's Bend, Melbourne.

The test spectrum and loading is based on the in-service usage of the LIF Hawk fleet, taking due account of the anticipated increase in severity of future operations. The test spectrum and loading has been developed in 3 key phases.

The first phase in the development of the spectrum and loading was the Theoretical Test Spectrum (TTS). The TTS defined the test spectrum in terms of the key aircraft parameters that exercise the aircraft structure. The Theoretical Test Spectrum was derived using the Check Stress spectra, in-service spectra information obtained from Usage Monitoring (UM) and Operational Loads Measurement (OLM) supplied by the CoA, and other pertinent Hawk data. It incorporated a degree of robustness to allow for future anticipated changes in the LIF configuration and usage.

The second phase in the development of the spectrum and loading was the Detailed Theoretical Test Spectrum (DTTS). The DTTS takes the data contained in the TTS and defines the spectrum in terms of a series of standard manoeuvre types. The standard manoeuvre types used in the DTTS are based on an analysis of a subset of the OLM data used to define the TTS. An aircraft FEM is utilised with fully balanced loading cases representing the required Fatigue Point in the Sky conditions. Distributed Aerodynamic and inertia loads are applied to this FEM, with some correlation work being conducted on the LIF Hawk Instrumented Aircraft. This combined spectra and loading data, when utilised in conjunction with BAE SYSTEMS in-house developed spectrum generation software and FE analysis defines the target external loading and damage levels required within the structure.

The final phase in the development of the spectrum and loading was the Test Jack Load Spectrum (TJLS). The TJLS uses the fully distributed external loading defined in the DTTS to calculate target shear force bending moment and torsion values at key locations. These target loads are then used to determine the jack loading to be applied to the test. The final step in the development of the TJLS, and hence the test spectrum, was to truncate the spectrum to a size appropriate to maintain the required test running rate. The spectrum as defined by the DTTS contained in excess of 20 million loading conditions in a 250 hour block. The truncation was carried out by first calculating target damage values based on the DTTS and the TJLS in its un-truncated form at a series of control points. The spectrum was then truncated by a process of filtering and removing non damaging events until the target of approximately 87500 turning points in a 250 hour block was achieved.

The specimen comprises a LIF Hawk production standard Fuselage, Wing and Fin.

A production standard Windscreen and Canopy are also fitted to enable cockpit pressurisation to be applied. Some structural items have been, or are being, cleared by an alternative route, and therefore are not fitted to the test specimen. Other items are present in dummy form, which means that the item is not being tested but that it is used to introduce load into the test specimen. Any items, which are regarded as non-structural such as pipework, wiring, avionics boxes, etc, are omitted.

The loads are to be applied to the airframe by 83 hydraulic jacks, each one fitted with relief valves to prevent overloading. Compressed air systems are installed to provide cockpit pressurisation and to simulate fuel tank pressures.

All of the loading devices are controlled by an electronic control system that ensures the applied loads are of the correct magnitude and in the required sequence. The system simultaneously controls 90 items (83 jacks, 6 independent pressure systems, and one channel for the operation of the flap control actuator).

The LIF Hawk FSFT commenced cycling on 16th December 2005 with the application of a small loading block to simulate Production Flight Acceptance Testing. Continuous test running commenced on 23rd February 2006 and at the time of writing the test has accumulated approximately 1100EFH of testing.

2.4.3 Undercarriage Fatigue Testing (T Siddall, Messier-Dowty)

The Harrier II main landing gear, refer to Figure 9, has completed 90% of its qualification test cycles to clear increased Maximum Take-Off Weight.

The Eurofighter Main Landing Gear has been strengthened for Tranche II fatigue spectrum loading, with heavier stores loads. The analysis is complete and achieves full life, with testing now underway.

The A321 landing gear has been re-qualified by fatigue analysis to cover extreme cold temperature operation on Russian runways (take-off and landing). Full life achieved

The A400M Nose Landing Gear fatigue analysis has been completed for Pre-production aircraft, and fatigue testing is due to start in 2007.

Figure 9 Harrier II main landing gear fatigue test

2.5 FRACTURE MECHANICS AND DAMAGE TOLERANCE

2.5.1 Fatigue crack growth from foreign object damage under combined low and high cycle loading (J Ding, R F Hall, J Byrne and J Tong, University of Portsmouth)

The experimental studies were described in the last National Review, and are repeated here to serve as an introduction to the subsequent analytical work.

Part 1: Experimental Studies

The study [9] aims to understand small crack growth behaviour in Ti–6Al–4V plate specimens after foreign object damage (FOD), under combined low and high cycle fatigue (LCF/HCF) loading conditions. The loading block studied represents one of the simplest loading sequences experienced by aero-engines and each block includes a single LCF cycle at a load ratio (R) of 0.01 with 1000 HCF cycles at R of 0.7 superimposed at peak load.

In Part I of this paper, the experimental studies are described. FOD was simulated by firing a 3mm hardened steel cube onto targeted specimens with an impact velocity of 200 m/s, leading to a blunt "V" notch in a flat surface. Microstructural features associated with the FOD damage, e.g. microcracks, were characterized by optical and scanning electronic microscopy. The microcracks were found to be preferred sites for subsequent crack propagation. The growths of these FOD-initiated small cracks under combined LCF/HCF loading were drastically different from the long-crack results obtained from corner-cracked specimens. Stress relief annealing was conducted on selected specimens to remove the residual stresses due to FOD. The results indicated that residual stresses significantly affected both the development of crack front shape and the fatigue life.

Part II: A two-parameter predictive approach

An approach based on a two-parameter fatigue theory has been developed to predict FOD-induced small crack growth under combined low and high cycle fatigue loading conditions [10]. In this approach, both ΔK and K_{max} are recognized as being important to the fatigue crack growth process and therefore included in a crack growth formulation. An elastic stress intensity factor is not appropriate for the small cracks studied here, particularly due to the influences of residual stresses and stress concentration produced at FOD indentations. The El Haddad approach was used as a correction factor, providing a modified stress intensity range ΔK_{mod} and maximum stress intensity $K_{mod,max}$. The levels of stress concentration and residual stresses were calculated by a general finite element code ABAQUS. The results show that small crack growth data compare well with those of long cracks when correlated with a new two-parameter crack driving force $(\Delta K_{mod})^{n'}(K_{mod,max})^{p'}(1-\Delta K_{mod,th}/\Delta K_{mod})^{0.25}$, where n' and p' are material constants derived from long-crack fatigue tests. This means that FOD-induced small crack behaviour may be predicted using long-crack experimental results.

2.5.2 Effect of bonded straps on fatigue crack growth resistance of aluminium substrates (P E Irving and X Zhang, Cranfield University)

A number of investigations have taken place to develop experimental data and predictive models for fatigue crack growth rates in the presence of a bonded strap. Such straps are in many ways similar to the action of bonded repair patches in aircraft which act to stop or slow down a pre-existing fatigue crack. They differ from patches in that the straps are an intrinsic part of the aircraft design, present at the start of service, and would be part of the initial fatigue and damage tolerance substantiation, whereas the patches are applied ad hoc- where an inadvertent fatigue crack has developed. In this sense, these straps act like "sub-stringers". Note that some of this work is being presented in this year's Symposium [11].

2.5.2.1 Use of adhesively bonded CFRP straps as fatigue crack growth retarders on AA2024-T3 (P E Irving and X Zhang, Cranfield University)

Unidirectional carbon fibre-polymer matrix straps, 25mm wide were bonded to centre cracked 1.6mm thick 2024 aluminium panels. Referring to Figure 10, the straps successfully reduced the fatigue crack growth rates in the panel by up to a factor of 3.5, provided that the cure for the bonding was performed at room temperature. Elevated temperature cure at 70°C, generated tensile residual stresses causing panel distortion. Bonding with a single ply of CFRP followed by curing at 70°C actually increased fatigue crack growth rates and reduced the overall fatigue life. It was notable that the beneficial effects of the straps occurred even before the crack tip had attained the edge of the strap as well as during the traverse of the tip past the strap and into the region when the tip was completely out of the strap region. A delamination in the bond line between the strap and the substrate was observed to form as the crack tip encountered the edge of the strap and continued to grow as the tip traversed the strap. Finite element models of the substrate-adhesive-strap system, incorporating the delamination have been constructed to calculate the crack tip stress intensity factor ΔK operating in the strap region. Agreement between predicted and experimentally determined growth rates in the reinforced panels was excellent in the region of the strap but the improvements in the growth rate measured outside the strap region were not predicted [12].

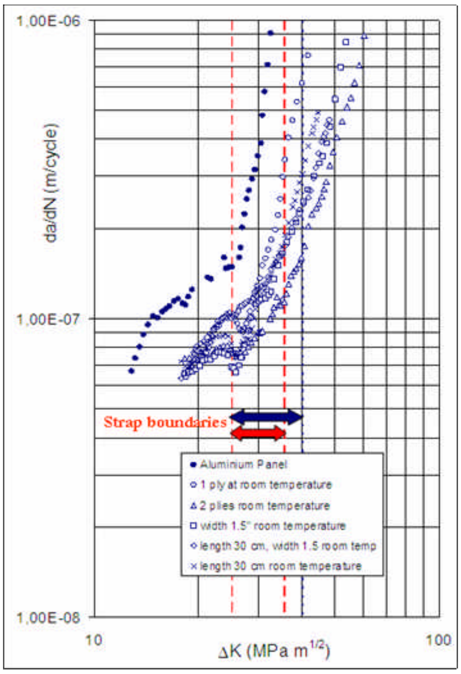


Figure 10 Effect of bonded CFRP straps on centre-cracked aluminium panels

2.5.2.2 Fail-safe integral structures using bonded crack retarders (P E Irving and X Zhang, Cranfield University)

This project is investigating the effectiveness and design of bonded straps as crack retarders. The crack retardation behaviour of four strap materials is being investigated. Straps are made of aluminium/glass fibre metal laminate - GLARE, unidirectional carbon fibre/polymer composite, unidirectional glass fibre polymer composite, and titanium alloys.

Research so far has demonstrated that there are number of distinct contributions to the effect which the straps have on the fatigue crack growth rate. Which of these is dominant depends on where the crack tip is located in relation to the strap as the crack approaches, traverses and emerges from the bonded strap region. For cracks approaching the strap, the effect of the tensile residual stress field caused by elevated temperature curing of the strap bonding agent is important. As the crack traverses the strap width then both residual stresses and the additional stiffness of the strap, which act together to reduce the applied ΔK at the crack tip, interact in determining the growth rate. When the crack tip emerges from the strap, the strap stiffness continues to retard the crack by reducing the *ΔK*, However, delaminations developing between the strap and the substrate progressively grow under the influence of fatigue loading and gradually reduce the effective strap compliance.

A 2D finite element model of strap-substrate interaction has successfully predicted the effect of the strap on the observed “crack length - cycles” curves for the bonded samples. Residual stress fields produced by the curing process have been measured directly and also calculated, again by using FE analysis. The tensile residual stress fields are being incorporated into the fatigue life predictive mode [13-15].

2.5.3 Effect of mechanical surface defects on the fatigue performance of clad aluminium alloys (P E Irving and X Zhang, Cranfield University)

Investigations have explored the influence of mechanical surface defects from 50 - 200 µm deep on the fatigue life of 2mm thick clad 2024 and 7075 aluminium sheet. Fatigue cracks developing in this orientation from defects of this size are too small to be valid in LEFM terms; they are growing in the through thickness direction of a thin sheet and there is little or no data for crack growth in this direction. The defects can have sharp root radii. Systematic studies of the effect of notch depth and root radii on fatigue lives to failure of samples with simulated defects have shown that defects with the tip located in the cladding, can initiate fatigue cracks which can grow into the substrate and cause failure. Defects with the tip in the substrate material with a depth just greater than the clad thickness produce non-propagating cracks. Defects of depth much greater than the clad depth initiate cracks which can propagate to failure. Refer to Figure 11. Life predictions using models of the failure process based on macroscopic fracture mechanics do not provide accurate predictions of the life to failure. [16-17].

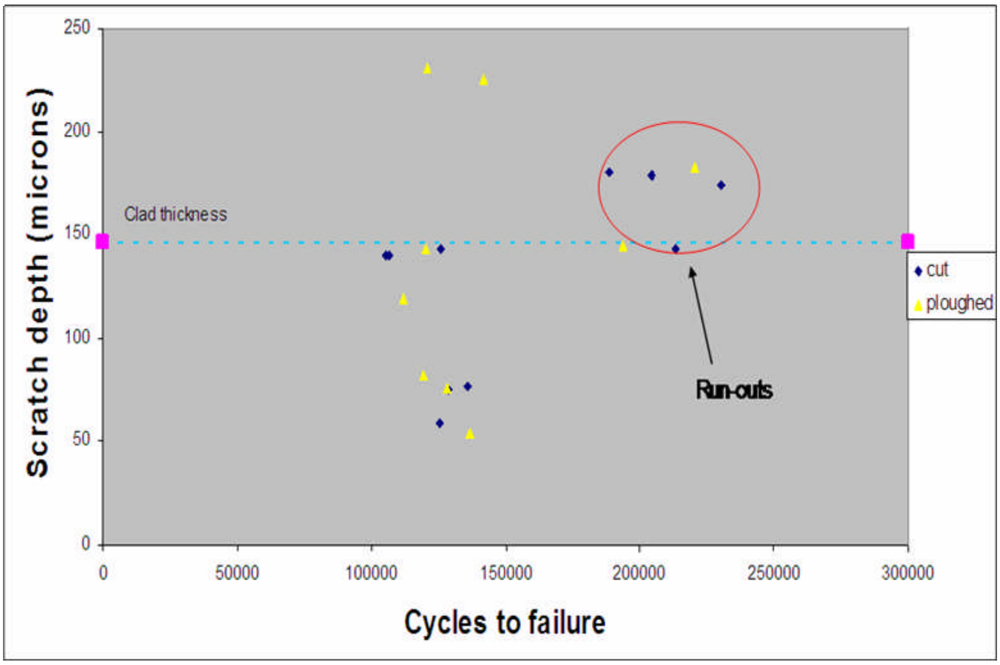


Figure 11 Effect of surface defects on fatigue life of clad aluminium alloys

2.5.4 Influence of residual stress on crack growth in aerospace structures (Mike Fitzpatrick, Lyndon Edwards, Jeffrey Tan and David Liljedahl, The Open University)

The OU Structural Integrity group applies advanced experimental and analytical methods to the study of residual stresses, particularly for problems and applications in the aerospace industry. The group routinely uses neutron and synchrotron X-ray diffraction techniques, which can measure residual strains several centimetres inside metallic components.

Recent examples include the determination of residual stresses around cold-worked holes, and comparing the FTI and StressWave processes as shown in Figure 12. These results were obtained from a combination of neutron and synchrotron X-ray diffraction, allowing a stress map to be produced non-destructively. This has the particular advantage that subsequent testing - e.g, fatigue testing - can be performed with accurate knowledge of the pre-existing residual stresses in the sample.

These measurements show that the StressWave process - involving dimpling of the sample and then drilling a hole - provides a more uniform through-thickness stress field than the FTI process, which draws an oversized mandrel through the hole. Additionally, the StressWave process provides the peak compressive stress at the hole bore itself, rather than at a small distance away, which is seen for the FTI process as a consequence of reverse yielding at the bore.

The group has also performed measurements on candidate weld processes for aircraft construction, and around bonded straps for crack retardation. A recent success has been to demonstrate that a crack growing through a weld residual stress field relaxes the stress field as predicted by FE modelling (Figure 13 shows measured and predicted stress profiles ahead of a crack grown 17mm into the residual stress field of a VPPA-welded 2024 aluminium sample). The results indicate that incorporation of some plasticity into the FE model is essential in order to predict the position of the relaxed stress profile. This result is significant because it demonstrates clearly that FE modelling can be used to predict the form and magnitude of a relaxed residual stress field during crack growth; but also it indicates the importance of the plasticity associated with the growing fatigue crack in determining the final residual stress profile. Elastic assumptions are an oversimplification.

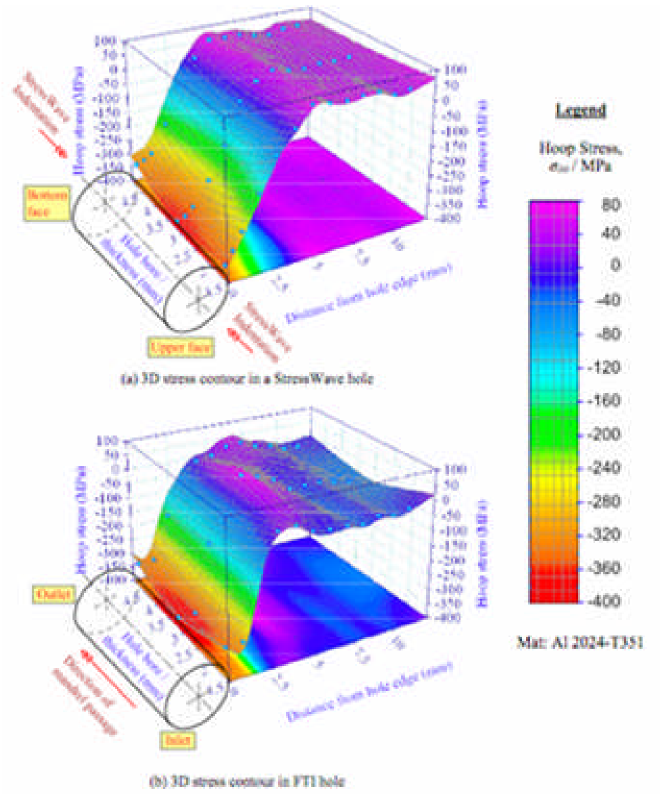


Figure 12 Residual stresses around cold worked holes

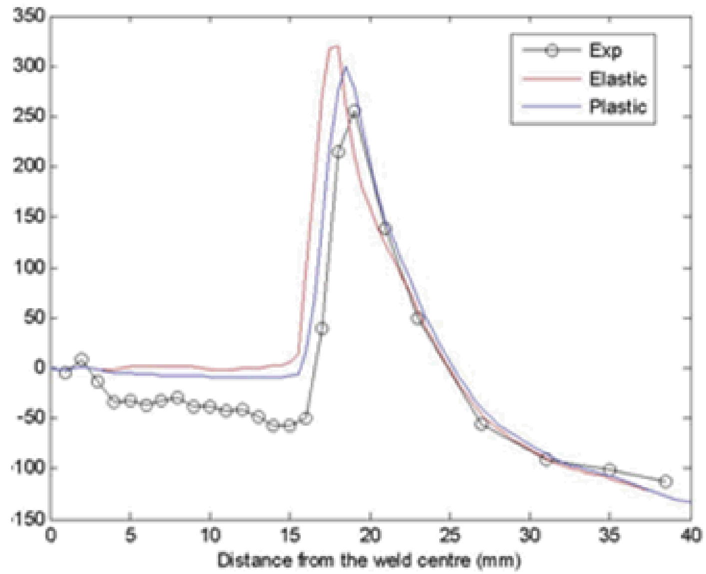


Figure 13 Stress profiles ahead of a crack in a VPPA-welded aluminium sample

2.5.5 Development of closed-form SIF solution for cracks in a residual stress field: Generic & Extended Applications (J M L Tan, Airbus UK Filton, M E Fitzpatrick, L Edwards, The Open University)

Generic exact closed-form stress intensity factor (SIF) solutions have been developed for mode-I, II and III through-thickness cracks in an infinite plate. Centre-crack problems have been analysed comprehensively in the literature, but the focus has been on the effect of simple loading about the crack centre. In the current work, the formula of Sih-Paris-Erdogan was extended to consider the difference in SIF on the left and right crack tips under an asymmetric stress field [18]. Mathematical manipulations were performed to derive exact stress magnification factors for SIF computations and simultaneously circumvent the problem of crack-tip stress singularity. The solutions so obtained are applied to derive the residual SIFs that would act on a crack growing under the influence of the residual stress fields associated with VPPA (variable polarity plasma arc) and friction stir welds, using experimental stress data from contour method and neutron diffraction [18-19].

Based on the above generic representation, the basic formulation has been further extended for single [20, 21] and double cracks [21] problems in fatigue-aged holes treated with the StressWave cold-indentation method. The cold working process by StressWave indentation [20-21] is known to induce substantial compressive residual stresses around fastener holes to enhance fatigue life. Coupled with the generic approach, analytical geometry functions to take account of the interplay of crack(s) with the internal hole and external geometry have been similarly expressed in an exact manner. The closed-form nature of the generic and extended solution is shown to be an efficient method of representing SIF for a given stress field, σ (x), and crack size, c, without resort to numerical integration. This permits the SIF computation to be either carried out manually, or implemented directly into a spreadsheet environment [18-21].

The SIF solution for single-cracked holes, calculated according to the extended formulation, was then used in fatigue crack growth (FCG) life prediction [20-21]. The FCG assessment was carried out in an analytical framework that systematically correlates the local variation of SIF at the crack tip due to the interaction between the residual stress and external loading [21, 22]. Incorporating the SIF data of residual stress profiles measured by the X-ray diffraction method, results in the predicted FCG life curves being in good agreement with those of experiment. An additional parametric FCG life study suggests both the SIF solution and residual stress data are equally critical to the predictive reliability of FCG life for mechanical components operating under the influence of primary and secondary stresses. Recent experimental evidence indicated that the redistribution of the original residual stress field in components will lead to the possible need of reconsideration of the use of original fracture *K*-parameters, and this might further complicates the fatigue and damage tolerance analysis of critical parts [22].

2.5.6 Use of corrosion metrics from SEM and NDE techniques in the prediction of fatigue lives of 2024-T3 aluminium alloy specimens with pitting and crevice corrosion (A Birt, L Jones, L Nelson, R A Smith, A Young and K Brown, QinetiQ, Farnborough)

Future corrosion control management on aircraft is planning to move from a ‘Find-and-Fix’ approach to an ‘Identify-and-Manage’ approach, which requires the development of suitable approaches for predicting the effects of corrosion on structural integrity. The aims of this study [23] were to correlate corrosion metrics measured from scanning electron microscope (SEM) and NDE data with fatigue life and to assess the use of these corrosion metrics in predicting remaining fatigue life.

Forty flat dog-bone specimens (L-T orientation) were manufactured from 1.6mm thick, unclad 2024-T3 aluminium alloy. Eight specimens were not corroded. Accelerated corrosion damage was produced in the other thirty-two specimens, in an area 25 x 25mm square at the centre of one face of each specimen, by placing them in a 5% neutral salt fog chamber for times of 1, 7 or 21 days. Half of the corroded specimens had pitting corrosion; the other half had crevice corrosion produced by placing a 25 x 25mm square piece of the same 2024-T3 alloy as used in the test specimens against the central area of each specimen, with filter paper soaked in 5% sodium chloride solution between the two surfaces.

The corroded region of each specimen was characterised using ultrasonic and eddy-current NDE techniques prior to fatigue testing. Constant amplitude axial fatigue loading at 20Hz, with 220 MPa maximum stress and R=0.1, was carried out on all specimens, in the ambient laboratory environment, until 10,000,000 cycles was reached or specimen failure occurred. During some fatigue tests oblique-incidence ultrasonic measurements were made at intervals to assess the effectiveness of this technique in measuring the growth of fatigue cracks before specimen failure. The fracture surfaces of the failed specimens were examined visually and in an SEM to characterise the origin of the fatigue crack which had caused failure, and the data obtained were used in simple life-prediction models to calculate fatigue life. Correlations were sought between the SEM and NDE corrosion metrics obtained before fatigue testing and the fatigue life, to determine which are the most important in the prediction of remnant fatigue life. NDE corrosion metrics were also used as inputs to the life-prediction model to compare the predicted lives with those from the SEM data and the measured lives in the tests.

Uncorroded test specimens achieved 10,000,000 cycles without failure, but fatigue lives were significantly reduced by the presence of pitting or crevice corrosion. After only 1 day’s corrosion, the geometric mean life of specimens with pitting corrosion was reduced to 264,000 cycles. The fatigue life reduced further as the exposure time in the salt fog chamber was increased. There was a tendency for the specimens with crevice corrosion to have longer fatigue lives than those with pitting corrosion for the shorter exposure times (1 and 7 days), but after 21 days of exposure in the salt fog chamber the two types of corrosion had similar fatigue lives, of between 90,000 and 170,000 cycles.

In both the pitting and crevice corrosion specimens the crack that caused failure had initiated from a corrosion pit. There was a general correlation of increasing pit dimension from the SEM (pit depth, pit half-width and the calculated pit area assuming a semi-elliptical pit) with decreasing fatigue life. The strongest correlation was observed between the calculated pit area of the pitting specimens and the fatigue life.

Simple fatigue crack-growth models were used to calculate the fatigue lives of corroded specimens. An initial semi-elliptical crack with dimensions equal to those of a corrosion pit was assumed to grow, with no constraints on the shape, under the applied loading; stress-intensity factor (SIF) solutions of Newman & Raju and assumed crack-growth laws were used to calculate the crack growth until the crack became a through-crack, after which an SIF solution of Feddersen was used. Figure 14 compares the calculated fatigue lives for four crack growth laws (with different short crack growth behaviour) with test data (measured fatigue lives and calculated pit areas using SEM data). The figure suggests that different laws may be required for different types of corrosion, reflecting differences in the early stages of fatigue crack-growth, such as the transition from corrosion pit to fatigue crack.

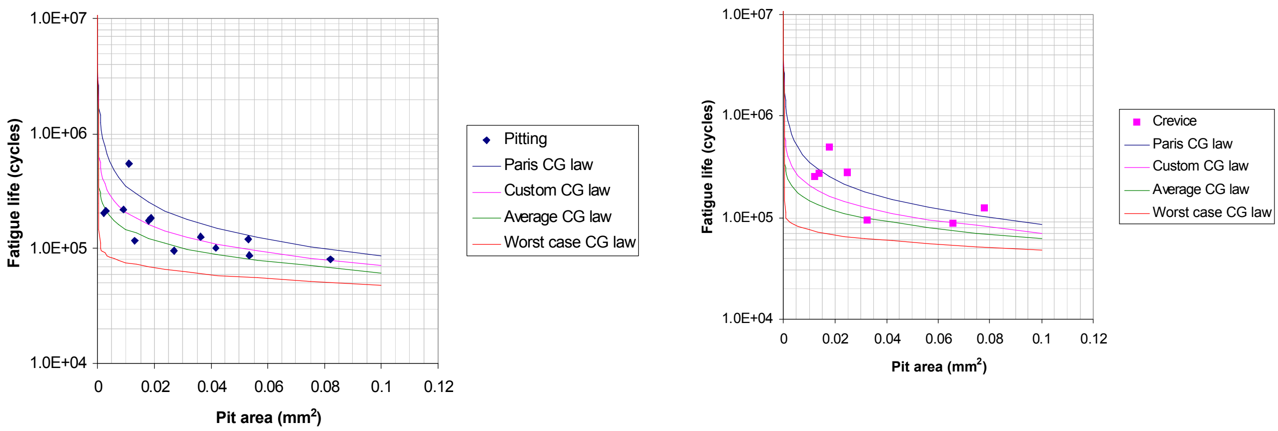
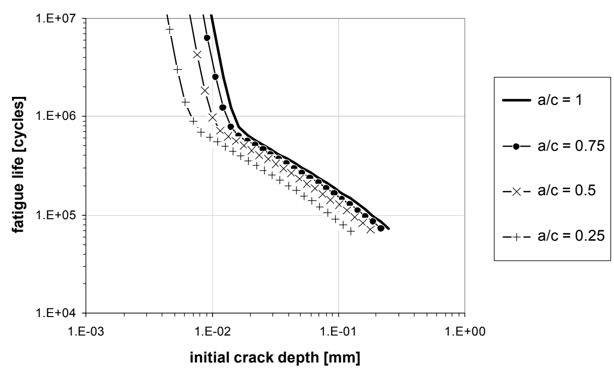
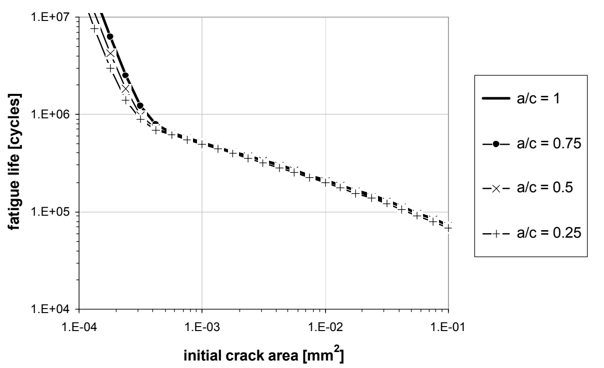


Figure 14 Measured (points) and calculated (curves) fatigue lives plotted against pit area for pitting corrosion (left) and crevice corrosion (right)

A theoretical parametric study using the life-prediction model showed that the calculated fatigue life for semi-elliptical cracks was dependent on both the width and depth of the initial flaw, as illustrated in Figure 15a. However, the fatigue life was determined largely by the area of the initial crack and was almost independent of its aspect ratio, as shown in Figure 15b.



(a) Fatigue life versus initial depth



(b) Fatigue life versus initial area

Figure 15 Correlation of calculated fatigue lives with initial depth and area for cracks with different initial aspect ratios (a=depth, c=half-width).

The results of the ultrasonic and eddy current NDE carried out before fatigue testing were used to establish whether correlations of measured parameters with fatigue life could be made. The references given below assess the strengths and weaknesses of the various techniques in this application and present the findings of this study concluding that pit width and depth should both be determined to obtain the maximum accuracy in the calculated fatigue life. In addition, recommendations are made of the most suitable NDE techniques for the provision of corrosion metrics for use in the prediction of remaining fatigue life.

2.6 OTHER ASPECTS OF FATIGUE

2.6.1 Repair assessment programme for military aircraft (C Hoyle, BAE Systems, Chadderton)

As part of the continuing development of policy to ensure the structural integrity of aging military aircraft, the UK has produced guidance on Repair Assessment Programmes (RAPs), which is now being implemented. A specialist sub-group of the Military Aircraft Structural Airworthiness Advisory Group (MASAAG) reviewed the civil aircraft policy [24] and adapted it taking into consideration the military operational context. The resulting MASAAG Paper [25] recommends that the pressure cabins of large civil-based aircraft should be addressed first. The VC10 and Nimrod aircraft have been used to pilot the guidance material so that lessons learned may be applied to other platforms and the guidance material developed. It is intended to extend the coverage of the RAP to other parts of the airframe and then consider what needs to be done on trainer and combat aircraft.

One of the concerns associated with conducting a RAP was whether the survey of existing repairs was feasible from an economic and aircraft availability viewpoint. It was recognised that the VC10 might be a challenge because the airframes were old, the fleet contained examples that had been in commercial service with various operators before being converted to military use and repair records were not comprehensive. It was thus a good choice to gain experience.

The VC10 surveys, which started in 2004, have been conducted on an opportunity basis during scheduled maintenance periods with a full survey during “Majors” and part surveys during “Minors”. About half the fleet have new been fully surveyed and the remainder will have had the initial survey by mid-2007. The work categorising the different types of repair is now well underway.

The major lessons derived are:

- With careful planning, the survey work is feasible without excessive disruption of the scheduled maintenance programmes.

- It has been found that a significant proportion of repairs do require additional analysis to qualify them, but very few require actual replacement with an upgraded repair. Thus the effort to achieve a reduction in risk, although high in survey and assessment terms, is unlikely to be excessive in actual airframe rework.

The RAP programme has also been applied to other airframes; for example, 12 Nimrod fuselages have been assessed and the RAP is being extended to the whole airframe for the long-term RMk1 fleet; the other aircraft will be phased out of service before these.

2.6.2 HUMS related research (B H E Perrett, QinetiQ, Farnborough)

Work has been carried out by QinetiQ, Farnborough on behalf of the UK MoD to study the risks associated with embodying HUMS capabilities and introducing Condition Based Maintenance in Military Helicopters [26]. The work has also assessed the level of engineering support and associated costs required to mitigate these risks.

The study has been carried out by considering the application of diagnostic and prognostic technologies to those mechanical systems of the helicopter whose performance is determined by wear-out mechanisms and by fatigue damage mechanisms. Systems dominated by wear-out can be generalised as bearings and meshing surfaces found predominately in transmissions (gearboxes, drive-shafts, support-bearings, etc). Although fatigue sensitive structural features can potentially be found in any helicopter system, those that are more critical are found in rotor systems, transmission systems, dynamic controls and airframes. Risk has been assessed by comparing relationships published in the literature and analyses carried out ‘in-house’.

Figure 16 assesses risk of failure using descriptions of the way in which damage accumulates and the way that indicator outputs relate to the level of damage for diagnostic and prognostic analysis. The figure shows probability of failure on the vertical axis plotted against the normalised proportion of log-mean life on the horizontal axis for mechanisms whose failure life lies within a log-normal distribution. The red curve shows the probability of failure for a component that is operated without monitoring. The component might fail at any point on this curve and, as the life increases towards the mean, the probability of failure increases to 50% at the mean life (normalised to unity). The other curves assume that a monitor system produces an output that is proportional to the damage mechanism that leads to failure and have been produced by normalising typical relationships found in the literature and produced by 'in-house' analysis. It is assumed that the component can perform adequately up to the mean life and that, as the monitor output increases, so it becomes more certain that failure is not imminent but can be predicted at some point in the future. The probability of failure is then calculated by subtracting the monitor output from the unmonitored feature risk.

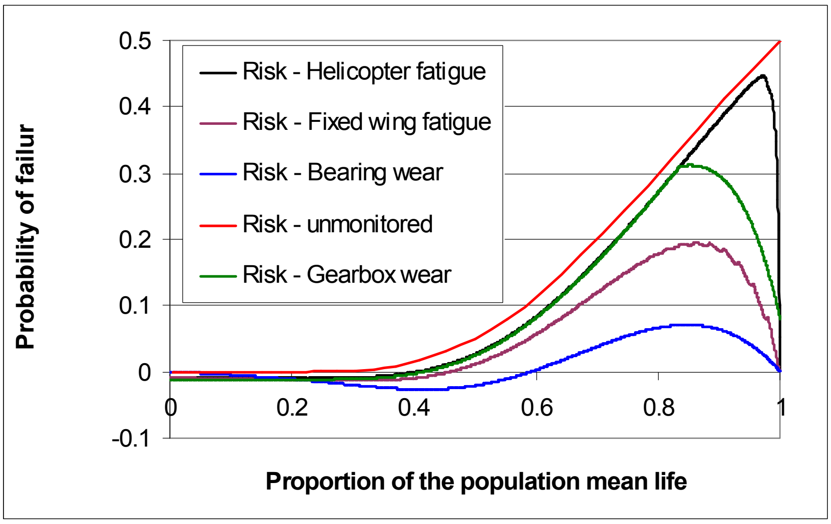


Figure 16 Probability of failure for various mechanisms

The blue curve shows the risk associated with operating a typical roller bearing up to its mean life (not failure but the point at which performance has degraded to an unacceptable level). Here it can be seen that risk does not rise above 10% due to an early condition related output at about 20% of the life. The green curve shows wear related output associated with a gearbox meshing surface. Here it can be seen that the relatively late condition indication (about 85% of the life) leads to risk rising to around 30%. The brown curve shows the risk associated with monitoring a fatigue crack in a fixed wing aircraft and it can be seen that here the risk rises to around 20% following an indication that begins at about 40% of the life. Finally, the black curve shows the risk associated with monitoring a fatigue crack in a helicopter component. Here it can be seen that, due to the very late crack indication, the risk remains at that of the unmonitored component for at least 95% of the life.

Whilst it is not proposed that this is a mathematically rigorous process, it does characterise the relative risks associated with pursuing condition based maintenance for the different systems of the helicopter. Whilst the risk is relatively low for bearing wear, the risk increases for meshing surfaces and becomes very high for the fatigue sensitive components of the helicopter that are traditionally lifed using a safe life design philosophy.

This work can be used to estimate the level and nature of research required to mitigate the risks.

2.6.3 Crack closure under spectrum loading at cold-worked holes containing interference-fit fasteners and its effect on ultrasonic measurements of crack length (K. Brown, A. Young, L. J. Nelson, L. D. Jones and R. A. Smith, QinetiQ, Farnborough)

Closure of a fatigue crack at a cold-worked fastener hole may occur in the compressive residual stress region around the hole produced by the cold-working, or at the crack tip from the plastic zone developed under variable amplitude loading. The crack closure causes ultrasonically measured cracks to be undersized or missed altogether; a static tensile force may need to be applied to overcome the compressive force acting on the crack faces and allow accurate crack length measurement. A study [27] has been carried out to investigate the crack-closure effects and to determine the dependence of two ultrasonic parameters (amplitude of reflected ultrasound and size of defect indication) on the loading level (i.e. applied tensile load) of a simulated structure. The aim was to determine the amount of crack closure that occurs at different applied load levels and hence to allow recommended mark-up values for the minimum reliably detected crack size to be made for each load level.

A 6mm thick 2024-T3 aluminium alloy specimen containing eleven 6.35mm (nominal) diameter cold-worked holes fitted with interference-fit fasteners was subjected to FALSTAFF spectrum loading (truncated to remove compressive loads and prevent specimen buckling) to produce fatigue cracks at a number of the fasteners. The fatigue loading was interrupted at intervals to allow crack length measurements using both a FlawInspecta® multi-element ultrasonic probe linked to ANDSCAN and plastic replicas, at a number of applied load levels. The ultrasonic measurements and replica crack lengths (up to 5mm) were compared to assess the difference in crack length measured using the two techniques for the different load levels. At the end of the fatigue test the fasteners were removed to compare the measured crack lengths with and without fasteners.

In addition to the experimental work, theoretical modelling of crack closure was carried out. Estimates were made of the tensile stress required to overcome the compressive stresses at the crack tip produced by the FALSTAFF loading. A semi-analytical model was used to estimate crack closure due to the compressive residual stress field produced by the cold-working (this omitted the effect of the closure at the crack tip due to the variable amplitude loading). This showed that, for a given degree of cold-working, the crack closure depended on the degree of interference of the fasteners (which could act to prop the crack open), the length of the crack (depending on its position with respect to the maximum compressive stress region of the residual stress field) and the applied tensile stress (greater stresses tending to open the crack more).

The results of the experimental and modelling work are presented in reference [27]. As examples, the modelling implied that cracks of between 0.6mm and 5mm from 1.1% interference-fit fasteners would be fully open for applied stresses of 50 MPa or more; in the absence of fasteners, cracks of 4 to 5mm would remain closed for a distance of between 1.5 and 2.6mm from the edge of the hole, depending upon the crack length and applied stress. Recommendations are made for the mark-ups required for ultrasonic measurements of crack length, which depend upon the crack length and applied stress level, and it is proposed that a specific mark-up can be applied when a minimum reliably-detectable EDM notch size has been determined.

2.6.4 VC10 teardown project (M J Duffield and D M S Taylor, QinetiQ, Farnborough)

The VC10 teardown project was conducted over a period of about six years until completion during the first half of 2006. The project was primarily intended to support the implementation of the Structural Examination Programme (SEP) derived from the damage tolerance assessment of the primary structure. The re-evaluation of the VC10 by modern damage tolerance criteria was completed during the 1990s in response to concerns regarding the fatigue clearances that had been based on design assumptions tailored to the typical usage of the VC10 by commercial airlines. A comprehensive analysis of the military use in combined air transport and air-to-air refuelling roles as flown by the Royal Air Force (RAF) was prompted by the discovery of significant fatigue cracking the that had not been anticipated as a result of the original fatigue testing undertaken for civil qualification. The results from this analysis were published in the form of a Fatigue Type Record (FTR) which itself formed the basis of the SEP.

The implementation of the SEP as originally envisaged would have resulted in an intolerable structural maintenance burden. Consequently, it was decided to take advantage of the coincident retirement of several high-time airframes from the fleet by using them as the source of samples of many of the major structural elements, joints and assemblies. These samples would be subject to detailed analysis by teardown processes designed to reveal the condition of the structure to a degree not normally possible under routine circumstances. With the process for each item carefully defined, from the controlled recovery from the donor airframe through disassembly and visual inspection to sectioning and metallographic analysis by laboratory techniques, a comprehensive assessment of the overall structural health of the aircraft was possible.

The teardown programme was a joint effort between QinetiQ at Farnborough and BAE SYSTEMS at Chadderton. QinetiQ undertook the bulk of the teardown work with technical support provided by BAE SYSTEMS, particularly with respect to the provision of engineering data which enabled the teardown examinations to be targeted and optimised. This teamwork ensured that the findings from the teardown examinations could be linked to the FTR analyses and decisions made quickly with respect to the corresponding inspections applicable to the in-service fleet.

The teardown activities undertaken by QinetiQ were organised into discrete Work Packages (WPs) in order to manage the very large number of structural samples recovered from a total of eight VC10 airframes that were retired from service for various reasons. Each WP was focussed on a specific feature or family of similar features and the teardown requirement for each sample was defined in a Statement of Work (SoW). A total of 46 WPs were identified for which 110 SoWs were prepared encompassing most of the main structural elements of the airframe; only the horizontal stabiliser was omitted from the teardown project due to the comprehensive refurbishment programme that had been previously completed.

The teardown findings resulting from the satisfaction of the objectives specified in each SoW were subject to review as they emerged. Many presentations and verbal discussions took place and various letters and reports were issued as the project progressed. This ensured that significant findings were circulated to all interested parties without delay such that appropriate action could be taken with respect to the in-service fleet. That activity continues with the on-going analysis of the teardown findings in order to further optimise and rationalise the structural inspection requirements.

The VC10 teardown project has been a major undertaking by QinetiQ and BAE SYSTEMS and supported by the UK MOD Integrated Project Team responsible for the VC10. Its success is due to the close liaison between all of the agencies involved and it has resulted in a deep understanding of the condition of the primary structure of the aircraft. This knowledge provides a basis for the assurance of structural airworthiness of the fleet as it celebrates 40 years of RAF service with an on-going commitment to the support of UK and allied forces for some years to come.

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