REVIEW OF AERONAUTICAL FATIGUE INVESTIGATIONS IN THE UNITED KINGDOM DURING THE PERIOD MAY 1999 TO APRIL 2001

compiled by

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2.1 INTRODUCTION

This review paper summarises aeronautical fatigue investigations carried out in the United Kingdom during the period May 1999 to April 2001. The format of the paper is similar to that of recent UK ICAF reviews [1]; the topics covered include loading actions, fracture mechanics, repair and NDE, as well as fatigue of metallic and composite materials, joints and structures. A list of references related to the various items is given at the end of the paper.

The author gratefully acknowledges the contributions generously provided by colleagues in the aircraft industry and universities, and at the Defence Evaluation and Research Agency. The names of the principal investigators, and their affiliations, are shown in brackets after the title of each item.

2.2 LOADING ACTIONS

2.2.1 Future Fatigue Monitoring Systems (D J Jones and Dorothy M Holford, DERA Farnborough)

On behalf of the Aircraft Structural Integrity (ASI) branch of the RAF, DERA has investigated the current situation with aircraft fatigue monitoring systems and highlighted the potential and scope for improvements, both for existing and future systems. This initial work led to proposals for a new policy that defined the scope of requirements for any new system and improvements to existing ones. A strategy was then developed that proposed the use of existing and future Operational Load Measurement (OLM) programmes to form the basis of the research work, which initially considered two main areas:

- Mathematical modelling of aircraft parameter data to synthesise strain and aircraft fatigue, together with fault and anomaly detection of the source data;
- Usage of new types of hardware for data acquisition systems.

Mathematical Modelling: Work has progressed to investigate the use of Artificial Neural Networks (ANNs) to synthesise strains at particular locations on an aircraft structure and to detect anomalies in the data. Real strain data from the current Jaguar OLM programme was used to enable initial training of the networks. This initial work showed that modelling of aircraft parameter data resulted in a reasonable level of correlation with measured strains. Further work is in progress to derive a level of accuracy for these early results.

Further work is now planned in this area, to take the modelling a stage further and synthesise fatigue consumption directly from parameters, for a whole aircraft fatigue analysis programme. This work will also investigate other types of modelling techniques in order to establish whether other methods offer improved performance in the final results and also to extend the scope of this work to engines and rotary wing aircraft, especially in relation to HUMS applications.

Data Acquisition Systems: As part of the above work ASI have purchased an ACRA 500 data acquisition system. This system has been used on a number of mini-OLM programmes in order to review the benefits and implications of utilising such systems, that potentially offer improvements in terms of cost, data sampling rates and storage capacity. This work is continuing; it should provide the hardware for the whole aircraft monitoring programme, as proposed above.

2.2.2 RAF Hawk Operational Loads Measurement Programme (S C Reed, DERA Farnborough)

The initial two phases of the Royal Air Force (RAF) Hawk Operational Loads Measurement (OLM) Programme have been completed and reported by BAE Systems. One aircraft from the Royal Air Force Aerobatics Team (RAFAT) was fully instrumented and captured over 250 sorties from the display work up and the display performance sorties. The second aircraft, from the Advance Flying Training (AFT) Unit, was instrumented and over 350 sorties were captured from the various roles within which the Hawk is flown in the Royal Air Force. The data from the OLM programme have been used to support several critical fleet life extension and fatigue management decisions. The RAF Training Aircraft Integrated Project Team is committed to continuous OLM programmes and both aircraft are undergoing data recorder upgrades this year. The RAFAT aircraft is scheduled to capture most of this season's display programme and the AFT aircraft will soon enter the fuselage replacement programme. This will ensure that the aircraft can be used to protect the structural integrity of the long-term fleet.

A paper entitled “Operational Loads Measurement of the Hawk Aircraft in RAF Service” will be presented at the 21st ICAF Symposium. This paper describes the OLM installation and objectives of the programme, gives an overview of the strain and parametric data captured, and discusses the analysis methods used. In addition, the paper explains the principal findings covering revisions to tailplane and fin monitoring methods and identification of structural features not adequately exercised by the fatigue test. Finally, follow-on activities are discussed.

2.2.3 RAF Harrier II Operational Loads Measurement Programme (SC Reed, DERA Farnborough)

The initial data capture phase for the Harrier II GR7 Operational Loads Measurement (OLM) Programme, using the Fatigue Monitoring and Computing System (FMCS) is now well underway. Over 1400 sorties have been captured from the 10 instrumented aircraft. The initial analysis of the first 200 sorties has been reported by BAE Systems. The data are currently being used to validate the fatigue meter formulae used for individual aircraft monitoring. For some locations, higher than expected fatigue damage rates have been determined; work is in progress to confirm these observations. The fleet-wide fit of FMCS to the T10 aircraft has begun and the first modified aircraft will soon return to squadron service and will commence data capture.

2.2.4 Operational Loads Measurement Programmes for Jaguar and Tornado Aircraft (N A Green and D Meadows, BAE SYSTEMS , Warton)

Three OLM programmes for Jaguar are currently underway; a general airframe OLM, an undercarriage (and associated back up structure) OLM and an engine mountings OLM. The general airframe and undercarriage programmes are almost complete. All three programmes will be used to underpin existing clearances and assist with life extensions.

The Tornado Structural Usage Monitoring System (SUMS) programme, which has been ongoing now for approximately 17 years, is being used to enable significant life extensions to both fleets of IDS and ADV aircraft to be met. SUMS data have also been used to generate new fatigue meter formulae for all RAF ADV aircraft. The system is currently being upgraded, to provide a more modern and flexible system, and the number of aircraft is being returned to the original complement of 7 IDS and 3 ADV aircraft. In addition, more locations are to be strain gauged and the rate at which the gauge responses are sampled is being increased. At the same time the ground based facility is being improved to enable more efficient analysis to be undertaken.

2.2.5 Eurofighter SHM (S R Hunt, BAE SYSTEMS, Warton)

The Eurofighter Structural Health Monitoring system is resident on every aircraft. The system in use for UK and Spain uses strain gauges to derive fatigue damage, whilst the German and Italian systems use parameters in order to calculate the strain and hence fatigue damage.

Work is underway to validate both systems based on the flying being undertaken by the development aircraft. The results from the parametric solution are being successfully compared with the results from the strain gauges. Investigations are also taking place into the use of artificial intelligence for the purpose of data integrity checking and automatic data correction.

Work is also progressing on a ground station which will be used for storage of the results of the on-board processing and provide further analysis capabilities. The facilities for installation of strain gauges on all production aircraft are in place and are being used.

2.2.6 OLM Activities on the VC10 (C Hoyle, BAE SYSTEMS, Chadderton)

a) VC10 Undercarriage Load Measurement Exercise (VULME)

In the absence of nose and main undercarriage fatigue tests, the VULME was initiated some years ago in order to produce representative fatigue spectra for the calculation of safe lives for the individual leg components [1]. The data acquisition phase was completed in February 2000. Some 131 flights have been recorded with approximately 90% containing useable data. The scope of the flying has included take off/landing/taxiing at a wide variety of both military and civil airports (to cover the undercarriages used by civil airlines before the airframes were converted to VC10 KMk2/3/4 standard).

Data from forty channels on both the main and nose undercarriages are currently being analysed. The analysis is approximately 70% complete with fatigue damages and vertical rates of descent (RoD) being derived. The maximum RoD recorded so far was 6 fps, compared to < 3 fps for the vast majority of landings at Glasgow airport.

b) VC10 Fin Star Plate Load Measurement Exercise (VFSPLME)

The fin star plates, at the top of the fin rear spar were the last of the VC10 SSIs (as part of the Fatigue Type Record) where damage tolerance (using theoretically derived loads) could not be demonstrated. As a result a fin star plate was removed from a surplus VC10 K2, initially inspected, and then strain gauged with some 20 gauges. It was then placed in a purpose built rig to undergo an off-aircraft calibration exercise. The next stage involved similar strain gauging of the fin star plates of an active VC10 CMk1(k) in the Base Hangar at RAF Brize Norton. Representative symmetrical and asymmetrical loading was then applied over a series of in-hangar tests.

The flying programme, recording at 256 samples per second, has been completed recently with the acquisition of data from 70 flights, covering all the major roles, and other known significant loading. Although the regression equations are still being refined before the data analysis can commence, the encouraging results thus far have demonstrated that the fin star plate, dependent on the loading being considered, is only receiving 25-40% of the theoretically predicted load. This reduction is likely to be due to the original conservative assumptions made for the prediction, where all the tailplane load was assumed to go down the fin spar plates. Alternative load paths appear to have reduced the fin star plate loads below those predicted on this basis.

2.2.7 Calculation of a Military Aircraft Fatigue Spectrum with Unit Load Cases to Closely Represent Operational Flying (S K Walker, S W G Feather, T Armitage, R Aaron, L Huff, M Henningsen, BAE SYSTEMS, Brough)

BAE SYSTEMS, Brough, has designed and built Hawk Lead-In Fighter aircraft for the Royal Australian Air Force. Part of this contract was to provide a Fatigue Check Stress of the critical parts of the aircraft structure. It was decided that the Fatigue Check Stress spectrum should correlate as closely as possible to the likely typical service spectrum. This would enable a best estimate of fatigue life to be made and would minimise any changes required for the creation of the loading and load case sequence of the forthcoming Full Scale Fatigue Test.

The time history of loading on the aircraft during a flight manoeuvre is complex, and although the stresses in some parts of the aircraft may be dominated by normal ‘g’ loading, this is not the case for many other parts. For example, stresses in the outboard part of the wing are affected by roll acceleration caused by aileron deflection and roll deceleration due to aerodynamic damping as well as by normal ‘g’. A vast number of load cases resulted when the required number of critical time slices in a manoeuvre were combined with other variables such as normal ‘g’ level and control surface settings. In order to simplify the analysis and yet retain a high degree of accuracy an approach has been developed based on unit load cases. Unit loading cases have been developed for each lateral and directional independent variable. The loading for these cases has been calculated using the BAE SYSTEMS Euler CFD method (FLITE3D) validated with Hawk wind tunnel model results. These loads have been mapped onto a Finite Element Model of the entire aircraft in order to obtain internal loads for each case. The comparison of predicted strains using this approach and strains measured in service is ongoing at Brough.

The manoeuvres in the spectrum have been developed from detailed analysis of in-service flying of RAF TMk1 aircraft. Typical manoeuvre types together with ‘g’ levels, control surface angles and control surface application rates have been identified from this analysis for which time histories of aerodynamic parameters have been created. The loading at any instant for any part of the structure has then been obtained by combining internal loads from the unit cases in the proportion of the aerodynamic parameters in operation at that time. Thus a time history of internal loading (or stress) may be created for any manoeuvre to be considered. The manoeuvre time histories are concatenated and rainflow counted resulting in the final spectrum for fatigue analysis using the BAE SYSTEMS local strain tracking software.

The spectrum for the RAAF Lead-In Fighter Full Scale Fatigue test will be created using the approach developed for the Fatigue Check Stress but with the manoeuvre analysis and strain gauge correlations based on RAAF in-service flying.

2.3 FRACTURE MECHANICS AND DAMAGE TOLERANCE

2.3.1 Crack Profiles and Corner Point Singularities (L P Pook, University College, London)

Crack profiles in the vicinity of a crack tip were investigated theoretically, by using standard displacement field equations, and experimentally, by using plastic foam models [2]. Particular attention was paid to crack profiles in the vicinity of a corner point where a crack front intersects a free surface. The main conclusions of the investigation are:
(a) The use of foam plastic models permits the visualisation of crack profiles in the vicinity of a corner point.
(b) Exaggerated displacement plots of crack profiles, obtained from finite element analyses, are sometimes misleading.

(c) Strictly speaking, stress intensity factors have no meaning in the vicinity of a corner point, but for practical purposes it is possible to calculate representative values.
(d) Failure to take into account warping of an initially plane free-surface, at a corner point, has led to some confusion in the finite element calculation of representative K_{III} values.

2.3.2 Finite Element Analysis of Corner Point Displacements and Stress Intensity Factors for Narrow Notches (L P Pook, University College, London)

Finite element models were used to investigate the surface displacements of narrow notches in the vicinity of a corner point [3]. Reasonably accurate two- and three-dimensional stress intensity factors were calculated by modelling a crack as a narrow notch with a semicircular tip and using stresses at the notch tip, and it was shown that the method can be used for mixed-mode situations. The extent of corner regions in which stress intensity measures, rather than stress intensity factors, dominate crack tip stresses was determined by analysis of the notch surface displacements and relevant stress intensity factors. It was shown that in nominal Mode III corner effects are local, but in nominal Mode I and nominal Mode II they are a combination of local and global effects. In addition, it was reported that it is possible to obtain worthwhile finite element stress intensity factor results from a low cost program running on a PC.

2.3.3 Stress Intensity Factor *K* and the Elastic *T*-Stress for Corner Cracks (L G Zhao, J Tong and J Byrne, Department of Mechanical and Manufacturing Engineering, University of Portsmouth)

The stress intensity factor *K* and the elastic *T*-stress for corner cracks have been determined using domain integral and interaction integral techniques. Both quarter-circular and tunnelled corner cracks have been considered. The results show that the stress intensity factor *K* maintains a minimum value at the mid-plane where the *T*-stress reaches its maximum, though negative, value in all cases. For quarter-circular corner cracks, the *K* solution agrees very well with Pickard's (1986) solution. Rapid loss of crack-front constraint near the free surfaces seems to be more evident as the crack grows deeper, although variation of the *T*-stress at the mid-plane remains small. Both *K* and *T* solutions are very sensitive to the crack front shape and crack tunnelling can substantially modify the *K* and *T* solutions. Values of the stress intensity factor *K* are raised along the crack front due to crack tunnelling, particularly for deep cracks. On the other hand, the difference in the *T*-stress near the free surfaces and at the mid-plane increases significantly with the increase of crack tunnelling. These results seem to be able to explain the well-observed experimental phenomena, such as the discrepancies of fatigue crack growth rate between CN (corner notch) and CT (compact tension) test pieces, and crack tunnelling in CN specimens under predominantly sustained load.

2.3.4 Damage Tolerance of Welded Aircraft Structures (G Bussu and P E Irving, Cranfield University)

This project [4, 5] has investigated fatigue and fatigue crack growth rates in friction stir welded 2024-T3 aluminium alloy. In particular, the effects on fatigue crack growth rates of crack location and orientation with respect to the weld line have been determined. The results have been interpreted in terms of weld induced local changes in microstructure, local softening and residual stress fields.

Samples of friction stir welded 2024-T3 containing longitudinal (stressed parallel to the weld line) and transverse (stressed perpendicular to the weld line) welds were manufactured. The samples were skimmed to remove 0.5mm from each surface. Compact tension specimens were machined from the samples so that the crack ran parallel to the weld line, at 28mm from the original plate joint line, at 6mm from the plate joint line, and on the original plate joint line (PJL). To investigate cracks running perpendicular to the weld line, surface thumbnail cracks (1mm radius) were spark machined into dogbone specimens, 400mm long and 80mm wide. The locations of the thumbnail cracks were on the plate joint line, 11mm from the plate joint line and 25mm from the plate joint line.

Fatigue crack growth tests were conducted under constant amplitude loading at an R ratio of 0.1. Crack lengths were measured using electrical potential techniques. Residual stress measurements, at similar locations as the cracks, indicated tensile stresses of up to 260MPa at 11mm from the plate joint line, up to 100MPa on the plate joint line and low or compressive stresses 25mm from the plate joint line. An example of fatigue crack growth rates at the three locations is shown in figure 1 for cracks propagating parallel to the weld line, and in figure 2 for thumbnail cracks perpendicular to the weld line. These figures show large differences in crack growth rates for the three different locations. To remove the residual stresses, samples were cold stretched to 2% plastic strain. Fatigue crack growth rates obtained after cold stretching are shown in figure 3; it can be seen that the large differences in growth rate with location were eliminated, and that all growth rates were similar to those found in the parent plate material.

It is concluded that residual stresses have the major influence on fatigue crack growth rates in welds, and that control of residual stresses is vital for damage tolerant design. This investigation will be described in detail in a paper [6] to be presented at the 21st ICAF Symposium. The work is continuing, exploring the use of MIG and VPPA welds and developing models for fatigue crack growth under variable amplitude loading, and exploring the behaviour of welded skin-stringer fabrications.

2.3.5 Experimental Studies of Crack Closure (E A Patterson, R A Tomlinson, J R Yates, University of Sheffield; M N James, University of Plymouth)

Two experimental methods are being used to investigate crack closure phenomena. At a fundamental level, transmission photoelasticity is being employed with polycarbonate compact tension specimens to gain an understanding of the mechanisms involved in crack closure. Fatigue cracks have been grown in the polycarbonate specimens [7], and the resulting photoelastic fringe patterns have been analysed using collocation techniques to obtain stress intensity factors and contact forces throughout the fatigue cycle. The results have been compared with the compliance method, which was found to be a relatively insensitive approach to quantifying crack closure. The photoelastic method, based on the multi-point over-deterministic method (MPODM), has been extended to include both the crack tip stress field and the contact stress field in the crack flanks. Solutions are achieved using a genetic algorithm and a local search technique [8]. Confirmation of the transmission data is being sought using reflection photoelasticity on aluminium compact tension specimens.

Thermoelastic stress analysis (TSA) is also being explored as a means of monitoring crack closure under service loads. TSA provides maps of the amplitude of the first stress invariant via measurement of small temperature fluctuations. The MPODM can be used to evaluate stress intensity factors [9]. TSA provides effective range of stress intensity factor directly and hence incorporates crack closure effects [10]. The practical limitations of the technique are being established.

2.3.6 Boundary Element Analysis of Flat and Curved Cracked Reinforced Aircraft Panels (P H Wen, M H Aliabadi, Department of Engineering, Queen Mary, University of London; A Young, DERA, Farnborough)

The boundary element method has become in recent years an efficient numerical tool for solving crack problems. The application of the method to thin plates and shells has however suffered in the past from the lack of available fundamental solutions and efficient formulations. Hence, the method has not previously been applied for analysis of cracks in shells. This project presents a major effort to develop a reliable computational tool based on the boundary element method for structural integrity evaluation of aircraft structures consisting of curved panels reinforced with stringers and frames. A method has been developed which is capable of analysing both flat and curved reinforced panels (with cracks), subjected to membrane and bending loads. In addition to its superior accuracy, the proposed method requires simpler modelling compared to the finite element method [11-13].

As a benchmark problem, to demonstrate the validity of the method, a simply supported stiffened flat panel subject to bending moments on the crack surfaces was studied. Two sizes of panel, $b/a = 5$ and 10, were considered to allow comparison with the known solution for an infinite stiffened panel. Normalised stress intensity factors at tips A and B were determined as functions of the relative stiffness S of the stiffener. The results for the two panel sizes were very close and were similar to the analytical solution for an infinite panel.

As an illustration of the capability of the method, a cracked cylindrical shell with longitudinal stiffeners and frames, as shown in figure 4, was analysed. The panel was subjected to uniform pressure and biaxial membrane loading as shown in figure 5. The maximum stress intensity factors on the outer surface of the panel due to pressure and membrane loads are presented in figures 6 and 7 for different curvatures of the panel k (i.e. k=0 corresponds to a flat panel). It can be seen that, for both load cases, the stress intensity factors decrease as the curvature of the panel is increased. In addition, the reduction in the normalised stress intensity factors as the crack tips approach the stiffeners is more pronounced for the more curved panels. Additional work is in progress to determine stress intensity factors for a wider range of structural configurations.

2.3.7 Boundary Element Analysis of Cracked Aircraft Panels with Mechanically Fastened or Adhesively Bonded Patches (P H Wen, M H Aliabadi, Department of Engineering, Queen Mary, University of London; A Young, P Poole, DERA Farnborough)

Mechanically fastened or adhesively bonded patches are used for the repair of damaged panels in aircraft structures, since both reduce the stress intensity factors at crack tips and thereby increase fatigue life. This project investigates application of the boundary element method to the repair of cracked aircraft panels with mechanically fastened or adhesively bonded patches. The method developed is capable of analysing both flat and curved reinforced panels [14,15]; the attachments or adhesive may deform non-linearly.

When a panel is repaired with a patch on one surface, the structural asymmetry causes the reinforced panel to bend, resulting in stress intensity factors which are much larger than those predicted by 2D modelling. To obtain correct stress intensity factors for panels patched on one side, the bending effect must be taken into account for structural design and fatigue life estimation.

As an illustration of the method, consider a centre-cracked square cylindrical shell of side 2W=180mm with a circular patch of diameter 60mm bonded to its outer surface, as shown in figure 8. The upper and lower edges of the curved shell are simply supported and are subjected to either moment load or membrane load, while the two curved edges are traction-free. The shell and patch are both of thickness 1.5mm with Young's modulus 70GPa and Poisson's ratio 0.3. The structure is discretized with 24 continuous quadratic elements on the exterior boundary, 24 discontinuous quadratic elements on the crack and 198 interior domain points, including 122 adhesion points, as illustrated in figure 8.

Numerical results for the maximum normalised stress intensity factors on the surface of shell opposite the patch for the membrane load case are plotted against curvature of the shell in figure 9 for a range of crack lengths. It can be seen that the stress intensity factors increase with increasing curvature and are much higher for curved shells than for the flat shell, demonstrating that damage tolerance calculations for patch repairs based on analyses of flat sheets may be extremely non-conservative if applied to curved structures.

2.3.8 Damage Tolerance Assessment of Multilayered Advanced Airframe Structures by Boundary Element Analysis, (P H Wen, M H Aliabadi, Department of Engineering, Queen Mary, University of London; A Young and P Poole, DERA Farnborough)

Titanium alloy structures manufactured by superplastic forming and diffusion bonding processes (SPF/DB) are increasingly being used in advanced aircraft design. These structures may contain manufacturing defects, such as localised debonds, or in-service defects, such as fatigue cracks or impact damage. In this project boundary element technology is being developed to allow for modelling of a wide range of defects in various SPF/DB structures, including four sheet X-core structures and four sheet cellular structures. Work is in progress using a boundary element multi-region plate formulation to analyse a flat panel containing cracks at various locations; promising initial results have been obtained.

2.3.9 Toughness Characterisation by Energy Dissipation Rate (J D G Sumpter, G R Sutton and P M Powell, DERA)

It is standard practice to index the toughness of aircraft materials by a K (or G) resistance curve. The basic assumption is that tearing starts at a relatively low level of applied stress intensity factor K, but remains stable as long as the resistance to tearing, K_R, increases at a faster rate with crack length, a, than the applied K. Unstable fracture occurs when:

$$\frac{dK_{applied}}{da} \geq \frac{dK_R}{da}$$

The K_R curve test is standardised in ASTM E561 *'Standard practice for R curve determination'*.

The K_R curve is held to be a material property as long as the test piece has a net section stress less than the yield strength of the material being tested. As aircraft materials become tougher, this requirement becomes very difficult to meet, even if large panels up to 2 metres wide are tested. There is thus an incentive to find a way of predicting toughness and structural defect tolerance from smaller test samples.

One small specimen found to show correlation with K_R curve panel behaviour is the Kahn tear test [16]. This test is also standardised as ASTM B871-96 '*Standard test method for tear testing of aluminium alloy products'*. It is stated in ASTM B871-96 that the test is *'not intended as an absolute measure of fracture resistance that might be used in the design of a structure'*. Nevertheless, the philosophy of the Kahn test is very close to that of the energy dissipation rate approach, which does aim to predict structural behaviour; and, in view of the Kahn test's small size (only 60mm by 40mm), and ASTM standardisation, there are strong incentives to investigate its structural relevance.

The energy dissipation rate approach to crack stability [17-19] is an elastic-plastic version of the Griffith energy balance theory. Instability is predicted to take place when an elastic-plastic energy input term, C_P, (where the suffix P denotes crack propagation at constant load) exceeds the energy dissipation rate term, D. The material toughness term D includes all dissipated energy, local and remote from the crack tip. Whilst it is believed that the energetic driving force term C_p can be linked to G as a function of plastic limit load, there is uncertainty over the extent to which D can be regarded as a material property.

Re-analysis of data from DERA panels of 2024-T3 aluminium alloy between 200 and 2000mm wide has been found to show extremely encouraging invariance of D with panel size and crack extension [20]. Typical values for D were between 1500 and 2000 kJ/m^2. Figure 10 compares values of G and D from a 2 metre wide panel. Whilst G_R rises continuously with crack extension, D remains more or less constant, suggesting that it is a fundamental characteristic of steady state crack extension.

When smaller specimens are tested, the value of D reduces and becomes a decreasing function of crack extension [21]. A theory has been formulated that D is invariant with crack growth provided crack propagation occurs in contained yield, but becomes dependent on ligament length and loading configuration after net section yield. It is postulated that scaling rules can be devised to predict the contained yield value of D from smaller test pieces under net section yielding. Data obtained to date shows some success of predicting 2 metre wide middle cracked M(T) panel behaviour from M(T) panels only 75mm wide. However, at present it is not understood how to scale D from the Kahn test to the M(T) test.

It is nevertheless clear from recent investigations at DERA that small tests, including the Kahn test, interpreted in terms of D, give a better ranking of the relative toughness of different airframe materials than does the conventional K_R curve test [21,22], especially when the latter is plotted in terms of the effective, as opposed to the physical, crack length. For instance, comparing 2024-T3 with an Aluminium-lithium alloy 8090-T34 showed very little difference in K_R curve behaviour measured on 400mm wide panels [22], but a factor of five increase in predicted structural critical defect length using the energy dissipation rate approach [21].

Although it has not so far been possible to understand the Kahn tear test fully in terms of structural prediction, the test is proving very useful in comparing the comparative toughness of different candidate weld systems for aircraft alloys. Because of the small size of the test piece it is possible to perform many more tests investigating effects such as crack position in the weld, and crack propagation direction compared to welding direction. Results to date show that welding always degrades toughness in a given alloy system, but that the toughness of welds from the better alloy systems can exceed that of 2024-T3 parent plate.

2.3.10 Analysis of Crack Growth from Cold Expanded Fastener Holes (Danong Dai, Airbus UK, Filton)

An analytical method based on distributed dislocations has been developed to allow a prediction of crack growth life from cold expanded fastener holes. A key part to the successful application of the method is an accurate estimation of the residual stresses induced by cold expanding fastener holes. Whilst the Finite Element Method (FEM) provides a powerful tool to simulate the 3D stress state around the hole, it is computationally expensive to use FEM to conduct parametric studies. Moreover, it is also difficult to use most commercial FEM packages to model the detailed material behaviour, which has a significant effect on the distribution of residual stresses.

Consequently, a closed-form solution for an elastic-plastic analysis of cold expanded holes has been developed. Budiansky’s method is used for both the loading and unloading analysis of cold expanded holes. A relationship between the applied expansion level and the interface pressure is given to allow an evaluation of the residual stress directly from the given applied expansion levels. Different work hardening behaviour of the material during the loading and unloading phases can be used in the solution, giving a better description of the true material behaviour. Bauschinger’s effect can also be taken into account by allowing the yield surface to either expand or contract in the stress space during unloading. More importantly, the reverse yielding criteria is modified from the uniaxial stress-strain curve according to the loading history, to ensure that the elastic unloading path always lies within the yielding surface in the stress space. The effect of reaming on the redistribution of residual stresses can also be assessed using the solution developed.

Figure 11 shows a comparison of the residual stresses obtained using the closed-form solution and FEM simulations based on the incremental theory of plasticity.

2.3.11 Initial Flaw Concept (Matthew Parry, Airbus UK, Filton)

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

The initial flaw concept is the subject of GARTEUR Action Group AG26, entitled 'The Initial Flaw Concept and Short Crack Growth', which began in January 2000 with active partners which include Airbus France, Airbus UK, DERA, FFA and NLR. The objective of the Action Group is to come to a common understanding of the implementation of the initial flaw concept, and to characterise initial flaw sizes for different hole and bolting standards, and possibly different material types. There is also a requirement to understand the effect on the initial flaw distribution of accidental 'rogue' manufacturing damage, which does not correspond to the normal build quality of the structure, but rather to an exceptionally substandard condition. This problem is being addressed through both experimental and analytical investigations.

2.3.12 Investigation of Equivalent Initial Flaw Size (EIFS) Distributions (K Brown and A Young, DERA Farnborough)

Computer models have been developed to calculate EIFS distributions quickly for simple specimen geometries under constant amplitude loading. EIFS values (typically tens of microns) have been obtained for different materials (e.g. aluminium alloys, titanium alloys) and test conditions (e.g. different maximum stress level, R ratio). The EIFS values so far obtained show a dependence on factors such as stress level - this may be a result of the crack growth behaviour assumed at short crack lengths. The models are being developed further to include EIFS calculation for cases such as crack growth from loaded interference-fit pins.

DERA is participating in the GARTEUR AG26 EIFS programme. A round robin exercise by the participants has shown that the most important factor in the calculation of EIFS values is the short crack growth behaviour assumed for the material. The GARTEUR group is considering "rogue flaws", arising from e.g. manufacturing defects, as well as the smaller initial flaws appropriate to the basic material crack growth behaviour.

2.3.13 Development of Computer Models for Prediction of Fatigue Behaviour of Mechanically Fastened Joints (K Brown and A Young, DERA Farnborough)

Research is in progress with the aim of producing computationally efficient, easy to use procedures for the prediction of the fatigue behaviour of mechanically fastened joints, with the initial emphasis on interference fit fasteners. Different techniques for calculating stress intensity factors (SIFs) for cracks at fastener holes have been investigated (e.g. Boundary Element and Weight Function Methods) which can then be used to predict fatigue crack growth rates and fatigue lives. The most promising method is a Semi-Analytical Method, which uses an approximation to the contact stresses around the hole circumference from an idealised model of a fastener in an uncracked plate. A set of formulae derived specifically for the contact stress distributions from the analytical model is then used for rapid calculation of SIFs at fastener holes.

For simple joints, predictions of fatigue crack growth rates using the SIFs from the model are in satisfactory agreement with experimentally determined growth rates; the differences may be the result of the crack growth laws assumed. The models have shown that interference fit fasteners can give R-ratios (ratio of minimum to maximum SIF during constant amplitude fatigue loading) which are much higher (0.8) than the ratio of the minimum to maximum applied load (0.1); this is very important for the prediction of crack growth rates. Future development of the method will consider different fastener and crack geometries, and address factors such as cold expansion and clamping forces. Validation of the method will be by comparison with detailed BEM analyses and experimental data.

2.3.14 Development of Robust Models for Prediction of Fatigue Crack Growth Rates in Helicopter Components (PE Irving, Cranfield University; B Perrett, DERA Farnborough)

This programme is a collaborative exercise between Westland Helicopters Ltd, nCode International Ltd, Defence Evaluation and Research Agency (Farnborough), Cranfield University and the Civil Aviation Authority. The work explores the measures required to produce robust analytical models describing crack growth behaviour in helicopter structures and dynamic components with a view to supporting damage tolerant design in rotorcraft.

Helicopter load sequences are unusual in that they contain large proportions of cycles having high R-ratio's. Basic crack growth data are difficult to generate and record in these regions. Work has been undertaken to examine the influence of the accuracy of descriptions of these data on the modelling of crack growth damage under various helicopter spectra. Constant amplitude data have been generated for a number of materials: a titanium alloy (Ti-10V-2Fe-3Al), an aluminium alloy (7010-T73651) series and an aluminium-lithium alloy (8090-T852). Tests have been carried out under the Roterix spectrum of loads representing conditions in helicopter dynamic components.

Previous work demonstrated that helicopter spectra, consisting of high frequency cycling at R ratios of 0.7-0.9, combined with underloads to zero or small compression loads, resulted in fatigue crack growth rates which were accelerated compared with calculated growth rates based on linear summation. In the present work, crack growth rates have been measured under simplified load spectra, consisting of only two or three cycle components. The ratio of number of excursions to zero load to the number of high R ratio cycles has been systematically varied, as has the number of consecutive applications of particular load cycles. Work so far has demonstrated that the acceleration effects can be reproduced in these simplified spectra, and that the critical factor appears to be the number of load transitions from low R to high R. Acceleration factors, relative to linear damage summation, are shown in figure 12 for Ti-10V-2Fe-3Al alloy.

Other work in this project has extended the results from long cracks into the short crack regime. Crack initiation and early crack growth rates have been measured in 25 samples of 7010 and 8090 aluminium alloys. The variability in crack initiation and early crack growth rates has been quantified at crack lengths up to 400 microns. Coefficients of variation in crack lengths of up to 70% were determined for 8090 at a mean crack length of 250 microns. For 7010 alloy there was excellent agreement between long and short crack growth rates at R-ratios of 0.1 and -1 when the data were correlated with ΔK. For 8090 alloy there was poor agreement between long and short crack growth rates at R-ratios of 0.1 and -1 when the data were correlated with ΔK; short crack growth rates in 8090 at R = -1 were between 10 and 50 times faster than long crack growth rates at R = 0.1. This may be explained in terms of crack closure causing anomalously low rates of crack growth for long cracks in 8090 alloy.

2.4 FATIGUE AND FRACTURE PROPERTIES OF ALUMINIUM ALLOYS

2.4.1 Optimisation of the Shot Peening Process in Terms of Fatigue Resistance (E R de los Rios, S Curtis and J Solis, University of Sheffield; A Levers, Airbus UK)

Shot peening is widely used to improve the fatigue properties of components and structures. Residual stresses, surface roughness and work hardening can be identified as the main effects induced in the surface layers of the material. The optimum values of these variables depend on the correct choice of the peening parameters. The effect of the process variables on both fatigue resistance and crack behaviour in 2024 and 7150 aluminium alloys is being studied. In order to optimise the process parameters to obtain suitable levels of residual stresses, surface roughness and work-hardened layer, leading to improved fatigue resistance, Design of Experiments (DoF) concepts are applied to the test programme [23].

In addition, tests are being performed on 2024 and 7150 alloys in gaseous and liquid environments to establish the regions of influence of different fatigue mechanisms. Models are being developed to characterise the microstructural and environmental effect on fatigue crack growth and the type of interactions between mechanical and chemical driving forces. The second stage of this work considers the effect of shot peening on corrosion fatigue.

2.4.2 A New Approach for Studying Surface Engineering Effects in Metal Fatigue (E R de los Rios, M Trull, C A Rodopoulos, University of Sheffield; A Levers, Airbus UK)

The objective of this project is to develop a methodology for the prediction of fatigue crack propagation rate and fatigue lifetime in surface engineered aerospace materials [24,25]. The model is also used to derive the conditions for crack arrest, and the results are presented in the form of a fatigue life (figure 13) and a fatigue damage map (figure 14). Two surface engineering treatments are considered, shot peening and laser peening. To evaluate the model, fatigue tests of surface engineered 2024 and 7150 aluminium alloys are being conducted, using a servo hydraulic test machine with in situ measurements under either an optical or acoustic microscope. The possibilities of healing fatigue damage by surface engineering treatments are being explored by testing previously fatigue damage specimens that subsequently have been peened or re-peened.

2.4.3 The Effect of Laser Shock Peening on Fatigue Resistance of Aircraft Materials (E R de los Rios, M Artamonov, C A Rodopoulos, University of Sheffield; P Peyre, Coopération Laser Franco Allemande (CLFA), Laboratoire pour l'Application des Lasers de Puissance (LALP), France; A. Levers, Airbus UK)

Four-point-bend specimens were laser peened using the facilities at Ecole-Polytechnique (LULI = Laboratory for the Use of Intense Lasers) under the European Improving Human Potential Programme. These facilities allowed the use of a very high intensity laser (100 J) delivering pulses shorter than those possible at LALP (2-3ns instead of the 10-20ns used at LALP). When comparing laser shock peening (LSP) and shot peening (SP), SP specimens generally display higher residual stress levels than LSP specimens, mainly in the first 100 – 200 µm. However, the stress field of LSP extends three to five times deeper than is typical for SP.

Fatigue tests results to date show that the fatigue life of shot peened specimens is longer than that of laser peened specimens at high applied stress. However, laser peened specimens have longer lives at low applied stresses.

Furthermore, the fatigue limit of laser peened specimens is higher than that for shot peened specimens. This work will be described in a paper to be presented at the 21st ICAF Symposium.

2.4.4 In-situ SEM Studies of Small Crack Growth Behaviour under Cyclic Load (P Bowen, M D Halliday and C Cooper, University of Birmingham; P Poole, DERA Farnborough)

The behaviour of small fatigue cracks in the aluminium alloys 2024-T351 and 7010-T7651 has been examined in air and aqueous sodium chloride test environments and compared with comparative long crack data. This work is part of an ongoing study of the micro-mechanics of fatigue crack growth [26,27] over a range of materials, crack sizes, environments and loading conditions.

Small "thumb-nail" cracks were naturally initiated in test samples under four point bend using a servohydraulic test machine. The cracks initiated either from intermetallic particles or from corrosion pits formed after exposure to the aqueous sodium chloride medium. Crack growth rate data were obtained from cellulose acetate surface replicas taken during testing. Detailed study of selected cracks was performed using a Hitachi S-4000 field emission gun scanning electron microscope fitted with a Raith 10kN servoelectric loading stage. This enabled the measurement of crack closure levels and crack tip opening profiles to be made by direct observation of crack opening displacement levels over a range of crack lengths. It was found that small cracks cannot be assumed to be closure free. Examples of crack closure data obtained from 2024 alloy tested in an aqueous sodium chloride environment at two R ratios can be seen in figure 15. Variations in small crack closure have been shown to contribute to the spread in fatigue crack growth rates encountered at small crack sizes. In addition, the crack path is sensitive to environment and can show extensive growth along grain boundaries in aqueous sodium chloride solution, as illustrated in figure 16.

The effect of pre-exposure to the aqueous sodium chloride environment before fatigue loading was also examined using similar testing procedures. Generally, a retardation of the onset of crack initiation was observed for 0.5 hour exposures, whilst 5 hour exposures resulted in crack initiation after similar numbers of applied cycles as encountered for tests conducted in air (figure 17). This has been attributed to the differing nature and extent of damage occurring for the different exposure times [27].

Comparative long crack data in air and aqueous sodium chloride environments were obtained from semi-elliptical surface cracks of generally similar form to those observed for the small crack testing. Again, bend test specimens were used. Cracks were initiated in air from small (0.15 to 0.25mm diameter) holes drilled at the specimen mid-position. Fine PD probe wires were subsequently attached across these precracks to enable continuous monitoring of crack growth to be carried out using the electrical potential drop method. Testing was then continued in the desired environment using a sensitive pulsed DCPD system developed as part of this programme [27] which permitted high resolution monitoring of crack growth. The resulting data were processed to determine long crack growth rates from surface cracks, examples of which can be seen in figure 18. It is planned to develop these techniques further, and preliminary data have already been successfully obtained for smaller, naturally initiated, surface cracks in aluminium alloy.

It is well known that the growth of small fatigue cracks is less amenable to predictive analysis based on LEFM than is the growth of long cracks. A statistical regression analysis of small crack growth data (obtained from this and previous related programmes of work [27]) is being carried out to develop methodologies for characterising small crack growth and predicting probable small crack growth limits from long crack data. Confidence limits have been derived to provide an envelope of expected crack growth behaviour for a range of testing conditions, and this approach was compared with extrapolations of long crack growth rates into the small crack region [27]. A comparison of short and long crack growth data for 7010 alloy tested in air over low and high R ratios showed that small crack growth rates at low R (0.05) ratio can exceed those found in long cracks at high R ratios (0.7-0.9).

The experimental procedures outlined in this work have broader applicability to fatigue and fracture studies. Both the in-situ SEM and DCPD techniques are currently being applied to titanium based materials at the University of Birmingham over a range of temperatures and often in combination with other methods such as acoustic emission monitoring of damage evolution.

2.4.5 Development of Models for the Prediction of the In-Service Performance of Metal Matrix Composites (PM Powell and R Cook, DERA Farnborough)

DERA participated in the Brite-Euram project ‘Development of models for the prediction of the in-service performance of metal matrix composites (MISPOM)’ along with Aerospatiale (France), Teksid (Italy), Erich-Schmid Institute (Austria), University Politecnica Catalonia (Spain), University of Ancona (Italy) and National University of Ireland Galway (Ireland). This three-year collaborative project, which was completed in April 2000, developed models to predict the static and fatigue behaviour of aluminium alloy based MMCs, reinforced with silicon carbide particulate.

Two materials were considered, namely wrought 2124-T4 + 17% SiC_p for aerospace applications and cast A359-T6 + 20% SiC_p for automotive applications. A wide ranging programme of experimental characterisation was carried out on the MMCs and on the unreinforced alloys [28]. This included microstructural studies, residual stress measurement, tensile properties, S/N fatigue behaviour and fatigue crack growth behaviour.

The fatigue studies showed that the introduction of reinforcement to the 2124 alloy increased its fatigue strength by up to 50%, but the extent of the increase was found to vary significantly for different test orientations, stress ratios and stress concentrations, viz, from –7% to +50% of the fatigue strength of the unreinforced alloy at room temperature. The introduction of reinforcement to the A359 alloy also increased its fatigue strength (by up to 30%) but the fatigue strengths of this MMC and the unreinforced alloy were much lower than the forged 2124 materials, with all the data for the A359 materials exhibiting fatigue strengths below 100 MPa at room temperature. This was consistent with the low tensile properties of the cast materials and was attributable to the presence of porosity. Fractographic studies confirmed that the fatigue behaviour of the MMCs was linked to the behaviour of initial defects under cyclic loading. In the case of the A359 MMC these initial defects were casting pores up to several hundred microns in size; in the 2124 MMC they were much smaller, typically being broken particles or particle clusters of the size of a few microns.

In view of the dominating effect of initial defects on fatigue performance, the fatigue crack growth studies concentrated on the near threshold behaviour of both MMCs and considered crack growth under constant amplitude loading, repeated overloads and underloads, and variable amplitude loading (ROTORIX 8 helicopter loading sequence) for the 2124 MMC.

The experimental data were used to develop and validate micromechanical and fracture mechanics based models for predicting the fatigue performance of a helicopter rotor blade sleeve and an automotive brake drum under service loading [29]. A defect tolerant approach was adopted, assuming that initial flaws are always present in the MMCs. ESI developed an analytical approach to predict the fatigue strength (at 10^7 cycles) of the MMCs, based on the use of crack growth resistance R-curves for the threshold of fatigue crack propagation. NUIG developed a finite element method to model short crack behaviour, using the J integral to characterise the crack driving force. This method was used to predict both the fatigue limit of the MMCs and the growth of fatigue cracks cycle-by-cycle. The growth of long cracks was investigated by DERA using the Führing LOSEQ crack growth model. This study demonstrated the importance of deriving representative fits to constant amplitude crack growth data for prediction purposes. The model predicted conservative lives in the case of fatigue crack growth following periodic single overloads for the 2124 MMC. There was closer agreement between prediction and experiment for periodic single underloads. The Führing model gave reasonable agreement with experiment for testing under the ROTORIX 8 helicopter rotor head sequence, as indicated in figure 19. This figure compares experimental crack growth data obtained for duplicate tests (specimen numbers XA06254 and XA06257) with corresponding data predicted by a simple linear summation model or by the Führing LOSEQ model, using best fit constant amplitude data (“visual fit”) or worst case constant amplitude data (“most damaging”).

In the final stage of the project, bench tests were carried out on a helicopter blade sleeve and on an automotive brake drum in order to assess and validate the ability of the analytical and finite element models to predict the fatigue strengths of these components in the presence of naturally-occurring and artificial defects. A successful validation of both models was achieved for the brake drum but for the blade sleeve, validation was not possible due to problems encountered in introducing adequately sharp defects. Further work is therefore planned by several of the partners to resolve this issue.

2.4.6 Structural Performance of Al-SiCp Functionally Graded and Laminated Materials (I Sinclair, Southampton University)

Considerable interest is currently being shown in mesoscopic materials architectures. Such materials offer important potential design benefits via the tailoring of a component's performance, both globally and locally, according to specific performance requirements. The present project is studying fracture mechanics issues associated with the fatigue of various layered and graded Al-SiCp MMC based materials, with particular emphasis on the influences of layer configuration, composition and interface character on structural performance.

2.4.7 Fatigue in Welded Aerospace Aluminium Alloys (I Sinclair, Southampton University)

Friction stir and plasma welding processes shows great promise for innovative metallic airframe construction, providing potentially significant savings in weight and construction time over conventional riveting. Current understanding of the fatigue behaviour of welded aerospace alloys is limited, with directed process optimisation and efficient engineering design (including lifing) for such welds clearly requiring an improved understanding of: (i) processing-structure relationships within the weld and surrounding areas, and (ii) structure-property relationships within and between the different weld regions. Superimposed on both is the development of residual stresses in and around the weld, with direct implications for fatigue performance (such as the superposition of tensile residual stresses in reducing closure levels). Work is in progress on micromechanical aspects of fatigue, including short cracks and close collaboration on the quantitative analysis of residual stresses via X-ray and neutron diffraction methods.

2.5 FATIGUE AND FRACTURE PROPERTIES OF METALS OTHER THAN ALUMINIUM

2.5.1 Damage Tolerance Design in Metal Matrix Composites (E R de los Rios, C A Rodopoulos and J R Yates, University of Sheffield)

Work has focused on the development of a technique that would map the different damage mechanisms that operate during fatigue loading. During the development period the Fatigue Damage Map (FDM), originally composed of the crack arrest and instability curves, has been extended to include (a) steady growth with maximum constrain, (b) steady growth with minimum constrain, and (c) unsteady crack growth. The FDM including the three new areas (figure 20) can now provide an accurate tool for the monitoring of fatigue damage as a function of the crack length and the applied stress [30-32]. The map can predict fail-safe and safe-life limitations.

The methodology of the Fatigue Damage Map has been used to optimise the fatigue characteristics of unidirectional Ti-based MMCs [33]. Additionally, the notch sensitivity of unidirectional MMCs is currently under investigation [34].

2.5.2 Surface Engineering in Metal Matrix Composites (E R de los Rios, C A Rodopoulos and J R Yates, University of Sheffield; S Godfrey, DERA; P Doorbar, Rolls-Royce; L Hacnell, Livermore National Research Laboratory)

This innovative work demonstrates the potential link between surface engineering and MMCs. In principle, the fatigue endurance of MMCs is significantly compromised by the thermal residual stresses developed during the consolidation process. There are two distinct damage mechanisms associated with the existence of residual stresses:

(a) Tensile residual stresses will promote the crack initiation process.

(b) The matrix material, and consequently the interface, are subjected to tensile residual stresses. Hence, the resistance of the interface to debonding is reduced significantly and crack propagation rate is increased. In this respect, the ability of the MMC to both arrest fatigue cracks and tolerate long fatigue cracks through fibre bridging are dependant on the residual stress state.

In this research programme, the compressive residual stresses induced from shot and laser peening are used to balance the thermal residual stresses in MMCs. Results from the early stages of the research have shown a remarkable increase in the fatigue life of a unidirectional SiC/Ti-6-4 MMC due to shot peening, figure 21 [35]. Further research will target the optimisation of both treatments.

2.5.3 Finite Element Simulation of Creep Crack Growth in a Nickel Base Superalloy (L G Zhao, J Tong and J Byrne, Department of Mechanical and Manufacturing Engineering, University of Portsmouth)

Creep crack growth in a standard compact-tension specimen has been simulated for an elastic/power-law creeping material, Waspaloy, at 650^{o}C using the finite element code ABAQUS. Two cases of crack growth rates, relatively fast (3.25×10^{-2}mm/hour) and slow (3.25×10^{-5}mm mm/hour), were considered. For both cases, the ratio of the load-line deflection rate due to creep ($\dot{V}_c$) to the total deflection rate ($\dot{V}$) was found to be well below unity, i.e., $\dot{V}_c/\dot{V} << 1$, which suggests creep-brittle characteristics. The development of the creep zone was determined and the creep-zone expansion rate along the crack growth direction was compared to the crack growth rate. Attempts were made to describe the crack-tip stress fields by the Hui-Riedel (HR), Hutchinson-Rice-Rosengren (HRR) and K types at different stages of crack growth.

2.5.4 The use of a Chaboche Model in ABAQUS, using Z-aba, to Predict the Behaviour of a Nickel Based Alloy under Thermo-Mechanical Fatigue (T Ward, M Henderson, DERA Farnborough; B Vermeulen, Portsmouth University)

Gas turbine engine combustor manufacturers face the challenge of achieving improved efficiency against the increasingly stringent regulations concerning NOx emissions. Revolutionary changes in design styles will be required with the increased use of high stress concentration features such as angled effusion cooling holes. To meet this challenge advanced design methods based on 3-D non-linear finite element methods will be required. Unified constitutive models such as those proposed by Chaboche and Bodner-Partom would seem to offer the best approach for the deformation modelling. No leading, commercial, finite element code offers these models as standard. However the Z-aba code provides ABAQUS users with a raft of advanced material models. Z-aba takes advantage of the ABAQUS user material subroutine but provides an elegant and simple interface. In this project [36], a Chaboche model has been developed to simulate the low cycle fatigue behaviour of the γ' strengthened, nickel-based alloy, C263. The effect of using the material parameters obtained from isothermal tests to model thermo-mechanical fatigue (TMF) cycles has been studied. The isothermal material parameters were found not to provide a good basis for simulation of the TMF cycles. Improved results can be obtained through further optimisation of the material parameters with reference to the TMF test results.

2.6 FATIGUE OF METALLIC JOINTS AND STRUCTURES

2.6.1 Flaw Tolerant Safe Life Assessment of the EH101 Helicopter Airframe (D Matthew, Westland Helicopters Ltd., Yeovil)

The aim of this programme of testing and analysis was to derive flaw tolerant safe lives for the Main Load Path (MLP) of the EH101 helicopter in support of the Civil Certification of the EH101 Civil Utility Variant. This followed on from a ‘Fail Safe Considering Flaw Growth’ analysis of the MLP that gave short inspection intervals despite Safe Lives in excess of 40000 hours by standard Miners Law fatigue analysis methods.

The Main Load Path components are machined from aluminium-lithium alloy (8090-T852) cold compressed forgings. The aim of the test programme was to generate SN curve data for the 8090-T852 material with built in damage representative of that which might occur in service. Eighty specimens of the kind shown in figure 22 were machined from a production standard roof frame and subjected to impact, corrosion or score damage. These were tested at various levels of load in single level tensile fatigue tests at R=0.1. The damage types investigated are summarised in the following table.

Damage	Application Method	Levels	Specimens Tested
Datum	No damage	-	10
Score	Machining tool with a 60° sharp point dragged at an angle of 45° across the specimen.	0.125mm deep	10
		0.25mm deep	10
Sharp Impact	Specimen impacted with a pyramidal impactor perpendicular to the specimen.	25 Joules	11
		12.5 Joules	5
Oblique Impact	Specimen impacted with the flat of a 10mm wide screwdriver at an angle of 45° to the specimen.	6 Joules	10
Corrosion	Coupon pre-drilled to represent corrosion pit. Copper electrode applied to specimen for 2 days in a salt water environment.	0.4mm deep	10
		0.2mm deep	10
		No pre-drill	3

It was observed that the level of corrosion or impact damage had only a marginal effect on coupon endurance. For example, coupons with 0.4mm corrosion pits gave very similar results to those with surface corrosion only. Therefore, all impact results were treated together and all corrosion results were treated together. As shown above, in addition to the damaged coupons, undamaged coupons were tested to generate a datum SN curve. Knock-down factors for each type of damage were calculated by dividing the datum SN curve fatigue endurance limit by that derived from the ‘damaged’ specimens. The resulting factors are summarised in the table below.

Damage Type	Damage Level	Knock-down Factor
Score	0.125mm	1.17
	0.25mm	1.32
All Sharp/Oblique Impacts	6, 12.5, 25 Joules	2.31
All Corrosion	0, 0.2, 0.4mm	2.39

Figure 23 compares the datum SN curve with the SN curve relevant to corrosion, which gave the most severe knock-down factor. Microscopic examination of the damage sites showed that there was extensive inter-granular corrosion. Figure 24 shows a micro-section through the damage site on one of the failed specimens with a fatigue crack originating from a corroded region.

Flaw tolerant enhanced safe lives calculated using SN curves with a knockdown in fatigue endurance of 2.39 are an order of magnitude less than those derived using standard SN curves. The implications of this work have yet to be agreed with the civil authorities, but the results emphasise the need for good in-service husbandry of rotorcraft structures.

2.6.2 Fatigue Testing of the Lynx Bolted Main Rotor Hub (J Nickolls, Westland Helicopters Ltd., Yeovil)

The Lynx helicopter has developed over twenty five years into a far more capable aircraft than early variants. This has brought with it a trend towards greater all up mass, with consequent restrictions in flight envelope and/or fatigue life to protect components which are costly to redesign. For further development of the Lynx, it became clear that significant main rotor head modifications would be required. The Lynx main rotor head is of semi-rigid design, with flap and lag flexibility provided by bending of a single piece forged titanium hub (monobloc) as shown in figure 25. In order to allow strengthening of the hub, it has been redesigned in Ti-10V-2Fe-3Al, which is stronger but also less stiff than the original material, Ti-6Al-4V (TA13). This has enabled larger sections to be utilised without changing the dynamic properties. In addition, the hub has been redesigned as two forgings to ease manufacture, with a rotor hub disc bolted to a separate mast (see figure 26).

This summary describes the testing performed to substantiate the redesigned hub (mast and disc). In addition to the bolted hub, the main rotor head comprises blade retention components (feathering sleeves, tie bars, and pins), control arms (pitch change horns), and lag dampers. Separate tests were conducted to substantiate the strength of these components.

Testing of the bolted hub was performed in the Westland Helicopters universal main rotor head test rig. This facility takes the form of a test pit capable of representatively loading any rotor head geometry from two to six blades. Figure 27 shows the test rig and bolted hub specimen. For testing of the bolted hub, flap, lag, and centrifugal loads were applied to all four arms, phased correctly around the azimuth of the hub. This configuration was used to test the inboard section of the hub disc, the mast, and the disc/mast joint. By installing props, outer arm sections could also be tested. Loads were applied via dummy feathering sleeves which were not substantiated by this test but provided representative load application. Loading was by a block loading programme, with factored load cases derived from typical measured flight loads.

High frequency loading blocks were alternated with blocks of low frequency loads. Each high frequency block consisted of 123025 load cycles, sub-divided into equal numbers of cycles based on vibratory flight loads from spot turn, forward flight, take off, rearwards flight, and landing conditions. Low frequency blocks consisted of 1975 cycles based on the largest ground-air-ground loads. In this manner, 2.5×10^6 cycles were applied in unpropped configuration, followed by 2.5×10^6 cycles with props installed to test the outer arm sections, followed by a further 1.4×10^6 cycles with the props removed.

Several typical test problems occurred during the test programme due to restrictions imposed by the test geometry. As resultant loads were applied to a reduced rotor radius, a more severe distribution of flap and lag bending moments was required than the flight configuration to achieve the desired test loads. In addition, factored loads were applied to account for strength scatter as required by DEF STAN 00-970. Initial testing resulted in migration of a bearing liner which caused fretting in a fillet radius and premature failure of one outboard arm test. Also the rig base plate provided a stiffer attachment for the base of the mast than would occur with an aircraft installation. Even with additional clamping, gapping of the mast/base plate joint occurred due to the factored test loads and led to a fretting condition on the mast base flange surface. It should be noted that no fretting problems similar to either of these have occurred with aircraft in service, demonstrated by a lead aircraft sampling programme where bolted heads are returned for non-destructive teardown inspection aimed at establishing overhaul periods.

The first two hub specimens produced fatigue failures which led to detail modifications to the geometry of the disc/mast joint. Four further specimens were tested to this modified standard. These resulted in fatigue failures of four masts, two inboard hub discs, and one outer arm section. In addition, one outer arm section and two disc/mast joint sections completed testing without failure. The results from the test programme were used to substantiate Safe Lives of 4000 hours for the mast and 6000 hours for the hub disc, based on Lynx Mk.3, Mk.7, and Mk.9 usage spectra. This compares to 2500 hours for the original monobloc hub.

2.6.3 Prediction of Fatigue Life for Bolted Joints (Wuxue Zhu, Airbus UK, Filton)

In order to predict the fatigue life of a bolted joint, the local stress at the edge of the fastener hole and a multiaxial fatigue model are required. However, there are many factors that may influence the fatigue life of a component in a multiaxial stress state. In this work, a multiaxial fatigue model has been developed which is based on the maximum shear stress plane and considers the influence of alternating shear stress, mean shear stress, mean normal stress, alternating normal stress and maximum principal stress (that is, the opening stress which tends to open the crack up). The various parameters in this model are determined and adjusted by reference to experimental data.

A two-dimensional analytical method has also been developed to calculate the stresses in a plate containing a hole filled with a fastener (interference fit or clearance fit) which is subject to an applied bolt load (in any direction) and a field load. This method has been validated against existing solutions and against finite element models. However, the complexity of a bolted joint means that the stresses calculated using a two-dimensional elastic model are far from the actual stress state. Consequently, modifications are required to allow for such influence factors, particularly the effect of material plasticity and thickness (three-dimensional effect). These modifications are based on the results from finite element analyses.

2.6.4 Application of Solid Phase Welding Technology to Commercial Aircraft (Gerald Shepherd, Airbus UK, Filton)

The solid phase welding process, in which a weld is produced by generating frictional heat from plunging a rotating tool into the joint line of the material section, is being considered for potential applications in aircraft wing manufacture, with a significant cost saving. Wing box ribs have been produced with the Friction Stir Welding (FSW) process, and sub-element tests undertaken to demonstrate the feasibility of the technique. The full-scale fatigue test of the A340-600 wing structure will include a FSW rib as a further proof of the process. In addition, the FSW process is being investigated for use in the manufacture of wing stringers, with qualification tests planned to facilitate introduction of the technique in 2002.

The in-service application of FSW as a repair process is also under consideration, especially with regard to the in-situ repair of leading edge wing skins up to 3mm thick, using a novel portable stirrer which is currently the subject of a patent application. Other possible in-service uses of FSW include the repair of wing spars by FSW, for which an initial feasibility study has been undertaken. The long-term corrosion resistance of stir welds has been investigated, and a test programme has been completed to demonstrate the practicability of repairs to components manufactured with the FSW process, since the addition of a mechanically-fastened repair patch may require the drilling of holes through the centre of a weld line.

2.6.5 Full-scale Panel Tests in support of the A300 Life Extension Programme (Steve Kimmins, Airbus UK, Filton)

As the Airbus A300 aircraft approach their Design Service Goal, a programme of work is underway to allow further operation up to a revised Extended Service Goal (ESG). The justification of this ESG must allow for the potential effects of Multiple Site Damage (MSD) and Multiple Element Damage (MED) on susceptible wing structures, which could develop into Widespread Fatigue Damage (WFD).

MSD/MED is a potential problem at three locations in the A300 wing structure - the chordwise joint in the top and bottom wing skins, and the tank end rib feature containing a chordwise sequence of stringer run-outs. Panel test specimens that closely simulate these locations have been constructed, and tested under a representative A300 fatigue spectrum up to a point close to panel failure. The durations of the tests far exceeded the periods required to justify the ESG. In addition, the MSD crack patterns naturally generated by the tests were not of an especially severe type, i.e. there were few adjacently cracked fastener holes.

In anticipation of these patterns, residual strength tests were performed on test panels representing the three locations. They were constructed with a lead crack one stringer bay in length and artificial damage placed at all the adjacent fastener holes in the same chordwise feature. The residual strength of each panel appears to be predictable from simple calculations that do not allow for a significant interaction between damages at adjacent holes. The full analysis of these results is still underway but the likely explanation lies with the fact that the stiffening of a wing cover is generally much higher than for a fuselage structure and the spacing of the fastener holes is of the same order as the stiffener spacing, and hence the ‘peaks and troughs’ in the crack driving force.

2.6.6 Jetstream 41 Full Scale Fatigue Test and Residual Strength Test Interpretation (M Bradley, BAE SYSTEMS, Regional Aircraft, Prestwick)

A full Jetstream 41 airframe (minus tailplane) was tested to 180,000 flights. The first 120,000 flights were with no artificial damage and from 120,000 flights artificial damage was introduced in order to compare crack growth rates with those predicted during the certification exercise. At the end of the test the specimen was subjected to sufficient residual loads to ensure it was properly exercised for all loading combinations. Following this, teardown inspections were performed on areas considered to be 'at-risk' of widespread fatigue damage.

The results of the above testing were then compared with the certification calculations to ensure that the structural inspection regime for the aircraft was satisfactory. On the wing, particularly, the beneficial effects of crack growth retardation were demonstrated and good correlation was found to a modified Willenborg method for crack lengths over 0.25". In a very few areas on the fuselage it was found necessary to reduce the inspection thresholds and periods below those predicted at certification. Otherwise, the methods used were found to be conservative.

2.6.7 Jetstream 31/32 Life Extension (M Bradley, BAE SYSTEMS, Regional Aircraft, Prestwick)

Jetstream 32 Wing Fatigue Test Completion, Residual Strength Test Completion and Interpretation. The Jetstream 32 wing was originally a safe-life design and without artificial damage testing or a residual strength test, the clearance given by the 225,000 flights of testing would only have been 45,000 flights (factor of 5). However, the crack growth from natural and artificial damage occurring during the test was used to demonstrate the damage tolerant nature of the wing. Freedom from widespread fatigue damage was also demonstrated by teardown inspections of at-risk areas. While this might lead to the supposition that a life of 112,500 flights was now possible for the aircraft, it was decided that it was uneconomic to inspect the wing-fuselage main spar boom joint and from earlier coupon testing a 'safe-life' for this part was established at 67,000 landings. This then dictated the declared life for the wing and indeed the aircraft.

Jetstream 31/32 life extension from 45,000 landings to 67,000 landings. The life extension work on the Jetstream 31 and 32 used the test results from the Jetsteam 32 Wing Fatigue Test (see above), earlier testing on a complete Jetstream 31 airframe specimen, fatigue testing on an engine nacelle specimen, quantitative read across from analysis and test results on similar Jetstream 41 structure, and fracture mechanics calculations in order to justify a structural life increase from 45,000 landings to 67,000 landings. BAE SYSTEMS believe that this meets the requirements of FAA draft Advisory Circular 91-MA on Continued Airworthiness of Small Transport and Commuter Aircraft. This has been accepted by CAA and a Supplementary Inspection Document will be issued later in the year.

2.6.8 PC-9 Fatigue Qualification Activities (J W Parish and G N King, BAE SYSTEMS, Brough)

A long-term task has been undertaken by BAE SYSTEMS with regard to fully qualifying the PC-9 aircraft for use by the Royal Saudi Air Force (RSAF). The aeroplanes were manufactured by Pilatus and supplied in two batches as part of the “Al Yamamah” program, with BAe (as it was then) being the co-ordinating Design Authority with the UK Ministry of Defence.

By assisting with the development of loading data for the RAAF PC-9/A FSFT at AMRL, Melbourne, BAE SYSTEMS has access to all the relevant airframe data to manage the RSAF fleet (flight data, spectra, strains, failure information). Testing took place from December 1995 to January 2000 at AMRL. In this time, there was a significant wing failure at 67,150 hours (to the RAAF spectrum). Cycling of the fuselage and empennage of the specimen was run on up to 100,000 hours. A Residual Strength Test on the fin and tailplane were then carried out followed by a full structural tear down.

The test damage incidents have been assessed and appropriate inspection and repair/modification actions have been put in place. Pilatus have contributed to this effort for the RSAF and world-wide fleet in their role as DA and also since they “bought into” the FSFT program. In particular, a modification package has been devised at 5,000 hours, to ensure minimum inspection and repair requirements at key areas on the wings and rear fuselage.

BAE SYSTEMS have used the data from the test, in conjunction with the Pilatus design and coupon testing data and RSAF operational data, to provide a Route to Clearance for the RSAF fleet of PC-9’s. The main activities included:

a) A check of the lives of critical areas, previously tested by Pilatus, was carried out by using appropriate flight and test data. Initial RSAF lives were calculated based on the relative damage of the RSAF symmetric spectrum (from Fatigue Meter returns) versus the Design and FSFT spectra.

b) The manoeuvre content of the RAAF Types of Flying (TOF) as tested was compared to that in the RSAF Sortie Pattern Codes (SPC) contained in the RSAF Statement of Operating Intent (SOI). This enabled relative damages to be estimated. Techniques developed for use on Hawk asymmetric spectra build-up were applied

c) Gust loading applied to the FSFT was compared to that which would be applied in Saudi via comparison of the TOFs and SPCs. Additional gust loading was added to the RSAF spectrum as a conservative measure (ESDU 69023 derived gust spectra was validated by comparison with AMRL and flight test data).

d) The overall relative damages from the asymmetric and symmetric spectra were combined to give lives applicable to the RSAF PC-9 structure.

e) The final cleared life takes into account the configuration differences between Swiss and Australian assembly – principally the jointing compound (dry on RSAF first batch, PRC on RSAF 2nd batch, JC-5 on RAAF). A literature search provided a reasonable set of Low Load Transfer data in the appropriate materials (2024-T3), whereas the Medium and High Load Transfer had relatively little data. Factors on life were then derived to assess the key LLT and MLT features.

f) Safe Life factors appropriate to DEF-STAN 00-970 Issue 1, as traditionally used on Hawk, were then considered as appropriate to the areas as tested.

All of the above gave sufficient evidence to show that primary structure is capable of withstanding 10,000 hours and associated landings to the required spectrum. In conjunction with an inspection, repair and modification program, the RSAF fleet can be successfully managed over its full life.

2.6.9 Multiple Site Damage (G R Sutton, R Cook and A Young, DERA Farnborough; S Chamberlain and I Taylor, RAF)

Work to examine strategies for dealing with MSD in ageing aircraft structures has continued at DERA in conjunction with the RAF. As reported at ICAF'99 [1], constant amplitude fatigue tests on multiple column open hole coupons, 1.6mm thick, demonstrated that significant benefits in fatigue endurance can be obtained by cold expanding holes at which fatigue cracks have formed, and MSD is present. It was reported that cold expansion was most beneficial where residual cracks were small, although crack growth was also slowed with cracks of up to 3mm present at 4.8mm diameter holes. Tests were also performed with lead cracks in addition to MSD type cracks. Lead cracks of 1, 2 or 4 fastener pitches were introduced prior to cold expansion and benefits in fatigue endurance were found in all cases due to crack retardation at successive holes as the lead crack propagated.

Recent work showed that similar benefits in fatigue endurance were obtained when fatigue tests were carried out under FALSTAFF loading, as illustrated in figure 28 for a 1 pitch main crack and MSD at each hole. In addition, fatigue tests were carried out on asymmetric butt joints containing similar damage patterns in one critical row of fastener holes. Following cold expansion of damaged holes, joints were assembled at RAF St Athan using three different types of fastener. Fatigue life varied with fastener type and clamping forces; best performance was obtained for fasteners which resulted in highest clamping forces. Further work is required to establish the significance of variations in clamping forces associated with in-situ repairs.

2.6.10 Predicting the Service Life of Adhesively Bonded Joints (A J Curley, H Hadavinia, A J Kinloch and A C Taylor, Imperial College London)

A fracture mechanics approach has been used to predict the cyclic fatigue performance of an adhesively bonded single lap joint and a typical bonded component, represented by a "top-hat" box beam joint [37]. The joints were tested under cyclic fatigue loading in either a "wet" or a "dry" environment, respectively. Several steps were required to predict the cyclic fatigue lifetime of these joints. Firstly, fracture mechanics tests were used to obtain the relationship between the rate of fatigue crack growth, da/dN, and the maximum strain energy release rate, G_{max}, applied during the fatigue cycle for the adhesive/substrate system under investigation, in both a "wet" and a "dry" test environment. Second, analytical and finite element models were developed to describe the variation in of strain energy release rate with crack length, as a function of the applied loads, for the single lap joint and the "top-hat" box beam joint. Third, the experimental results from the short term fracture mechanics tests, obtained under similar test conditions and in the same environment as were used for the single lap or bonded box beam joints, were combined with the modelling results from the theoretical studies. This enabled cyclic fatigue performance to be predicted over relatively long time periods. Agreement between the theoretical predictions and the experimentally measured fatigue behaviour for the joints was found to be very good.

2.7 OTHER ASPECTS OF FATIGUE

2.7.1 Adhesively Bonded Composite Patch Repair of Aircraft Structures (P Poole, K Brown, A Young, DERA Farnborough; R Halliburton, BAE Systems Brough)

Theoretical and experimental studies of the effects of environmental exposure and variable amplitude loading on the efficiency of bonded patch repairs to cracked aluminium alloy structures are summarised in items 2.7.2 and 2.7.3 below.

Additional research is being carried out at DERA to assess the potential of bonded patches for the repair of corrosion damage and battle damage. A three-dimensional boundary element/finite element model has been used to analyse corrosion damage repairs, where surface damage was removed by blending, blended regions were filled with an aluminium loaded epoxy and composite patches were bonded over the damaged/filled area using a 120^{o}C curing epoxy film adhesive. Initial studies assumed the presence of a small semi-circular surface crack at the deepest central part of the blended region (1.6mm deep in 3.2mm thick aluminium alloy sheet). Preliminary numerical results showed that the stress intensity factors for the cracks were reduced significantly by patching. Research on bonded patch repair of battle damage in thin sheet structures was reported in the last ICAF review [1]. More recently the use of bonded patches for the permanent repair of surface damage in thick structures has been investigated, and fatigue testing has established that patching results in marked improvements in fatigue life even when the damage extends to mid-thickness in a 12.5mm thick sections.

The feasibility and effectiveness of bonded patches for the repair of internal structure where access is limited is being investigated by BAE Systems. Up to eight artificial defects are being introduced at various locations in a Hawk tailplane, which has been mounted in a fatigue test rig. Patching and fatigue testing will commence soon.

2.7.2 Effect of Environmental Exposure on Bonded Patch Repair of Cracked Aluminium Alloy Structures (P Poole, K Brown, G Sutton and A Young, DERA Farnborough)

The effectiveness of adhesively bonded composite patches in retarding fatigue crack growth in aluminium alloy structures may be affected adversely by long-term exposure to hot-wet environments, due to moisture causing reduced strength and stiffness of the adhesive, loss of adhesion at the aluminium alloy/adhesive interface and/or corrosion under the patch. Research has been carried out at DERA [38] to investigate the extent to which such degradation may occur when cracked panels are repaired with bonded carbon/epoxy patches. In one investigation, 1.6mm thick panels containing central cracks were repaired on one side with various patch/adhesive systems and exposed at hot-wet jungle or marine sites in Australia for up to 6 years. Fatigue testing showed that such exposures had little effect in the case of precured patches bonded with a 120°C curing film adhesive, as illustrated in figure 29. In contrast, for panels repaired with wet laminated patches and exposed at the marine site for 6 years, extensive or complete patch debonding occurred; this was attributed to galvanic corrosion resulting from direct contact between the woven carbon and the aluminium alloy substrate. A three-dimensional boundary element/finite element model predicted accurately the reduction in fatigue crack growth rate observed for panels repaired with pre-cured patches and Redux 312/5 adhesive. For these repairs, the model predicted that only a small increase in crack growth rate would arise from a reduction in adhesive modulus due to water absorption during environmental exposure. Comparison of analytical and experimental results indicated that accurate prediction of fatigue crack growth rates for patched panels was possible, providing restraint of out-of-plane bending by antibuckling plates was modelled accurately.

In another study [38], constant amplitude fatigue tests showed that double-sided bonded composite patch repairs were very effective in retarding crack growth in 12.4mm thick aluminium alloy specimens. When patched specimens were exposed to and tested in 70°C/84%RH, the rate of crack growth was greater than for specimens tested in ambient laboratory conditions (see figure 30), but still much lower than for unpatched specimens. Almost identical behaviour was observed for specimens repaired with carbon/epoxy, carbon-glass/epoxy or boron/epoxy patches of equivalent stiffness. Predicted fatigue crack growth rates for patched specimens and ambient laboratory test conditions were in good agreement with experimental crack growth data. Numerical analysis showed that a reduction in adhesive shear modulus could account partly for the faster crack growth rates observed for patched specimens exposed to and tested in 70°C/84%RH. However, more pronounced debonding was observed for these specimens, which would also result in an increase in crack growth rate. Work is in progress to establish the relative contributions of increased debonding and decreased adhesive modulus.

In a third investigation [38], Boeing wedge tests were carried out to establish the extent to which environmental durability is improved if BR127 primer is applied after grit blast/silane treatment. Specimens were bonded using three different 120°C curing film adhesives and exposed to 50°C/95%RH for 2 weeks and then to salt spray for up to 8 weeks. For each of the three adhesives, application of a single coat of BR127 primer after grit blast-silane treatment resulted in significant improvements in wedge test performance when specimens were exposed to 50°C/95%RH. When two coats of primer were used, the improvements in wedge test performance were much less pronounced. For two of the three adhesives, similar trends were observed in the case of specimens exposed to 50°C/95%RH for 336 hours and then to salt spray for 1344 hours. Examination of uncracked portions of all specimens failed to reveal any evidence of ingress of corrosion in the vicinity of the bondlines, even after exposure to salt spray for 1344 hours. Thus, it appears that the absence of BR127 did not affect the susceptibility of the unloaded bondline to corrosion. Nevertheless, the use of a corrosion inhibiting adhesive bonding primer (such as BR127), with careful control of coating thickness, is recommended for critical applications where corrosive environments may be encountered. However, for other repairs, particularly those where airworthiness can be assured by a fail-safe approach, the use of a primer may not be necessary. This recommendation is supported by the good in-service performance reported for repairs involving grit blast-silane treatment without primer.

2.7.3 Effect of Variable Amplitude Loading on Bonded Composite Patch Repair of Fatigue Cracks (P Poole, K Brown, D S Lock and A Young, DERA Farnborough; M Myers, AFRL Dayton)

Research has been carried out to investigate the influence of variable amplitude loading on the effectiveness of adhesively bonded composite patches in retarding the growth of fatigue cracks, with particular reference to the role of patch debonding. Fatigue testing was carried out to establish the effects of (a) truncating ["clipping"] the variable amplitude spectrum MINITWIST, and (b) adding overloads or underloads during constant amplitude loading, on patch repair efficiency. A three-dimensional boundary element/finite element computer program was used to predict the effects of patching on stress intensity factors and fatigue crack growth rates for various load spectra, taking account of any debonding observed during fatigue testing.

Fatigue testing showed that truncation of MINITWIST (to levels III, V and VII) reduced the lives of unpatched specimens but had little effect in the case of patched specimens. Ultrasonic inspection established that debonding decreased with increasing truncation level, i.e. debonding was most pronounced for MINITWIST loading (i.e. no truncation) and least pronounced for MINITWIST VII loading (i.e. truncated to level VII). The observation that truncation level had no significant effect on the rate of fatigue crack growth of patched specimens may be explained in terms of the opposing effects of (i) reduced crack tip plasticity, and (ii) reduced debonding, with increased truncation.

The three-dimensional boundary element/finite element computer program was used to calculate stress intensity factors for patched specimens subjected to constant amplitude loading, taking account of debonding observed during testing. Crack growth rates determined from the predicted stress intensity factors and experimental crack growth data (for unpatched specimens) were in good agreement with experimental measurements on patched specimens, indicating that stress intensity factors were predicted accurately by the model. The model was also used to calculate stress intensity factors for patched specimens subjected to MINITWIST III and MINITWIST VII loading. From these stress intensity factors, the loads experienced by cracks in the patched specimens were calculated and associated load spectra were derived. When unpatched specimens were tested under these simulated load spectra, the observed rates of crack growth were in reasonable overall agreement with those measured in the patched specimens. It was concluded that accurate prediction of the effect of variable amplitude loading on the efficiency of bonded patches in retarding the growth of fatigue cracks will only be possible if the extent of debonding can be predicted. Further work is required to develop a model for predicting patch debonding as function of loading variables.

2.7.4 Fretting Fatigue Under Variable Amplitude Loading (J Hooper and PE Irving, Cranfield University)

The aim of the project was to investigate effects of variable amplitude loading on fretting fatigue life and to develop damage models to predict fretting fatigue under these load situations.

The application of overload cycles at given intervals during constant amplitude loading (CAL) had dramatic effects on the fretting fatigue life [39], see figure 31. When overloads were applied at very large intervals, the lives were similar to CAL, and when overloads were applied at small intervals, the lives were similar to those predicted using Miner's law. However, applying overloads at intermediate numbers of cycles caused life to increase by a factor of two. Investigations of frictional force hysteresis loops led to the suggestion that the variations in life were due to re-distributions of stick and slip boundaries within the contact interface, and the subsequent relaxation back to the original equilibrium conditions.

Investigations into the effect of two-level and three-level block loading waveforms has shown that for repeated blocks of small numbers of cycles (with respect to the initiation life), Miner's law gives reasonable yet sometimes slightly non-conservative life predictions. The transition from overload waveforms to block loading waveforms is currently under investigation, in order to determine whether the transition is linear or if there is a critical block ratio where Miner's law becomes invalid.

A damage mechanics model for fretting fatigue has been developed, allowing fretting fatigue life prediction to be considered as a continuous process from an undamaged specimen to total failure without further assumptions about crack initiation length. This model has many advantages over other models for fretting fatigue, in that it allows the prediction of crack initiation life, propagation life, crack growth rates, and crack trajectory. The application of damage mechanics has allowed the proposal of a bi-linear damage accumulation model.

2.7.5 UK Military Airworthiness Requirements (Alison Mew, DERA Farnborough)

An update to the fatigue design requirements of Def Stan 00-970 was issued in July 2000. A significant feature of these requirements is the use of the Safe S-N curves for the analysis of major fatigue tests and in fatigue loads monitoring programmes. Other important features of the revised requirements are the need to do operational loads measurement programmes to validate any loads monitoring system, the need to do two major fatigue tests (one on a pre-production article and another, later test on a mature production standard article) and new guidance on life extension programmes.

The guidance on the Safe S-N approach given in Def Stan 00-970 has also been further developed for a number of programmes, where DERA (Airworthiness and Structural Integrity) has given specialist advice.

New requirements have been developed for an Ageing Aircraft Structural Audit, to be carried out on all UK military aircraft fleets. The key objective of the revised Audit is to build a 'snapshot' of the state of the aircraft fleet: although there may be routine structural integrity management activities, they are often considered in isolation and it is necessary to obtain a more holistic view. The first audit is to be completed no later than 15 years after entry into service, and is to be repeated at 10 year intervals thereafter, unless an exemption is justified. The Audit will comprise a thorough review of the SI-related activities for the fleet, including fatigue and static clearances, Operational Loads Measurement/Operational Data Recording (OLM/ODR) programmes, modification programmes, repair activities, and sampling and teardown programmes (more appropriate later in life).

2.7.6 Prediction of Safe Fatigue Life Using a Local Strain Method with Life-Factors (I G Hume, S K Walker, W A Lennox, BAE SYSTEMS, Brough)

Def Stan 00-970 recommends the use of an S-N approach to fatigue life estimation with scatter covered by a stress factor at high endurance, a life factor at low endurance and a combination of stress and life factors in the mid-life range. It is permissible, however, for a Design Authority to use any fatigue method it considers appropriate as long as it is shown to provide a reasonable life estimate encompassing likely scatter.

The major fatigue design tool at BAE SYSTEMS, Brough, is a computer program which tracks local stresses and strains at a notch root via Neuber's rule and, in combination with a strain-life curve, calculates crack initiation life from a linear accumulation of damage. The strain-life curves are based on constant amplitude data but with the high cycle part of the curve deliberately extrapolated below the test data points in order to account for variable amplitude loading conditions.

Attempts were made to apply Def Stan 00-970 stress and life scatter factors to strain-life curves but two major problems were encountered:

a. The stress factor was derived from constant amplitude S-N curves containing a fatigue limit and therefore the direct application of this factor to curves without a fatigue limit was considered conservative.
b. The local strain analyses use crack initiation life curves for un-notched (Kt=1) specimens. These curves are shallow when compared with S-N curves for failure of notched components and the presence of a stress factor results in excessive reduction in fatigue strength throughout the life regime.

Safe life curves have been generated using two approaches. Firstly a mean life curve was drawn through test data points on S-N axes and a safe life curve was obtained using the Def Stan 00-970 stress and life factors. Secondly a mean life curve was obtained from summing predicted lives due to crack initiation and crack growth and a safe life curve was obtained from dividing the initiation and growth lives by fixed life factors. These life factors have been recommended in earlier versions of Def Stan 00-970.

The results have shown that the results of Neuber analysis with the application of life factors alone provide a suitably safe estimate of life, as long as fretting conditions are suitably accounted for.

2.7.7 Structural Teardown (D Taylor, DERA Farnborough)

The United Kingdom forces operate several aircraft types that have reached one of the criteria laid down in policy relating to ageing aircraft audit requirements. Based on experience gained with ageing aircraft programmes, the RAF and the Royal Navy has adopted a policy, with the support of the individual aircraft type Design Authorities to carry out a programme of deep structural examination. This work commonly known as "teardown" is carried out on nominated airframes that are either fleet leader in terms of one of the lifing parameters used or have been retired from service for some other reason. The United Kingdom has found such teardowns to be one of the more valuable and cost effective elements in meeting the audit requirements.

The aim of a teardown programme is to verify not only the condition of know "hot spots" within the structure but to take the opportunity to examine areas that would not normally be open during normal maintenance procedures. The examination is carried out using detailed work instructions compiled by the Design Authority. These are supplemented by advice and experience from both the operator and DERA. The work instruction offers guidance on where and how samples should be taken. This is then transferred to the actual airframe at a dismantling or "cut line" meeting. From this the aircraft is then dismantled into large sample parts. These sample parts are then dismantled into individual detail parts to allow the work instruction inspections to be applied.

At present examinations are being carried out by DERA on Jaguar, Harrier, Hawk, VC10 and Canberra. Work consists of detailed analysis of the condition of the structure for damage from corrosion, fatigue, environmental degradation or other undetected accidental damage. Prior to dismantling Non Destructive Testing (NDT) is carried out applicable to the type of structure and material being examined. Once the structure has been dismantled into individual parts further assisted visual examination and NDT is applied. Particular attention is paid to fasteners and fastener holes. Defects found are then subject to further failure analysis.

Reports of findings are produced and forwarded to the Design Authority for their comment and review. If a defect is detected that is considered unusual or of a potential threat to airworthiness, the Design Authority is immediately alerted so that action can be taken to inspect the in service fleet. Findings from the report will be used to assist in underwriting life extension programmes and in updating inspection requirements.

2.7.8 Structural Teardown Examination of the VC10 Aircraft (M J Duffield, DERA Farnborough)

The Royal Air Force has, for several years, operated a fleet of VC10 aircraft in the air transport (AT) and air-to-air refuelling (AAR) tanker roles. The fleet comprises a combination of CMk1K mixed role aircraft and KMk2, KMk3 and KMk4 dedicated AAR aircraft.

The decision to retire the five oldest KMk2 aircraft in 2000 came at an opportune moment with respect to structural airworthiness. This was because the structural maintenance philosophy was being upgraded from the original fail-safe principle to that of damage tolerance by the adoption of the same modern criteria used to assess ageing jet transport structures in the civil world. The difficulties associated with the systematic application of such criteria to a design from the 1960’s are well known and many instances arose where the damage tolerance analyses proved incapable of providing a rational structural inspection programme. The reasons for this were manifold but were chiefly related to the absence of any reliable benchmark by which to qualify the fundamental assumptions contained within the analyses.

It was, therefore, decided to use the redundant KMk2 aircraft as a source of the necessary evidence to provide that benchmark. This was possible due to the fact that they represented a group of nominally identical aircraft with similar service histories and were all high-life aircraft relative to the rest of the RAF fleet of VC10 aircraft.

With the possible exception of the fin, none of the structural differences between Mks precluded the use of the KMk2 structures as representative of the surviving variants. The results from the teardown examination could, therefore, be read across to the other types with confidence.

Teardown can be defined as the systematic dismantling and examination of structural features recovered from redundant airframes. The initial selection of samples for this purpose on the VC10 was based on the afore-mentioned damage tolerance analysis of the aircraft and was conducted by the Design Authority (DA) [BAE SYSTEMS (Chadderton site)], the VC10 Integrated Project Team (IPT) at RAF Wyton, and the Airworthiness and Structural Integrity Group at DERA, Farnborough. Early experience with the first aircraft to be dismantled indicated that this was indeed a valuable source of information on the condition of the structure. It was, therefore, agreed that the teardown programme should be expanded to cover other aspects relating to the general concerns of ageing large jet transport structures. Work is well underway on the first and second aircraft and the overall teardown requirements for the third and fourth aircraft have been defined. The fifth aircraft is currently planned to be retained intact as a test-bed for developing and proving inspection techniques and other structural investigations.

As can be imagined, the VC10 is a large aircraft and the amount of material recovered from each airframe is considerable. This has led to a need for careful planning and control of the samples and the individual programmes of work required to dismantle and examine them. Furthermore, the recording of the findings at all stages of the work is a substantial task and much use has been made of digital imaging technology for this purpose. It is also worth emphasising that this teardown programme is very much a team effort between the various agencies involved. For this particular programme DERA have provided the resources for most of the practical aspects but the DA ultimately control the scope and extent of individual investigations and, perhaps most importantly, undertake the interpretation of the results. It is this last-mentioned activity which translates the findings from the teardown into practical measures that may need to be applied to the in-service fleet; the DA is the sole agency with the necessary breadth and depth of knowledge to perform this task.

Although the VC10 teardown programme is still in progress and full results will not be available for some time, it is possible to provide an example of the type of benefit emerging as a result of the programme. The chosen example relates to the wing lower surface skin joint at the root, or Rib O as it is commonly referred to. This joint comprises abutted inboard and outboard skin panel tongues connected by a series of inner and outer joint plates. The assembly is secured by a staggered double row of steel taper bolts either side of the joint and a typical section is shown in figure 32. In total, there are 344 bolts in each joint, or 688 per aircraft.

It can be seen that this joint configuration is efficient as the fasteners are working in double shear. However, the inner and outer joint plates render the critical section of the skin panels uninspectable except by directed NDT techniques. Furthermore, with similar stress levels across the entire chord, the joint is also a classic example of a design feature where the threat of multiple site fatigue damage must be considered. This combination of undesirable characteristics is a common attribute of aircraft designed to the now out-moded fail-safe criteria, of which the VC10 is a prime example.

The fatigue and damage tolerance analysis of the Rib O joint resulted in a calculated inspection threshold for the KMk2 of about 33400 flying hours (FH) and a repeat interval of about 1200 FH. Similar results were obtained for the other variants when taking into account the different histories and utilisation patterns. Due to the very small size of the calculated critical crack lengths, these inspections required the bolts to be removed and high frequency eddy current probes to be used to search for evidence of fatigue cracking in the skin tongues.

Clearly, to undertake such an inspection on a regular basis would be a major task with probably significant effects on maintenance costs and downtime. The extraction of the taper bolts is likely to cause damage to the hole bores leading to a requirement for reworking of the holes and the provision of an equivalent number of non-standard replacement bolts. Recovery of the joints after inspection is also likely to engender problems in fuel tank sealing; experience has shown that the joints tend to spring apart when adjacent bolts are removed and such movement may be difficult to recover in the un-jigged configuration of the aircraft during maintenance. Furthermore, the entire KMk4 fleet had exceeded the inspection threshold life whilst the KMk3s were close to doing so. Hence, the inspection requirements, if applied rigorously, threatened to compromise the availability of a significant proportion of the RAF AAR fleet.

The above-mentioned analytical results were necessarily conservative because there was no test evidence available to qualify the assumptions contained in the fatigue and damage tolerance assessments. However, all of the KMk2 aircraft had exceeded the threshold inspection life by a margin in excess of 50%. Consequently, it was argued that, if the joints on these aircraft could be demonstrated as free from any significant defects, then this margin could be interpreted as an alleviation factor on the calculated lives. As the structure is identical on all variants and the fatigue calculations for all variants used exactly the same analysis models and equivalent loads and stress data, it was also concluded that this factor could be applied with confidence to the entire fleet.

The examination of sample portions form the Rib O joints as recovered from THREE of the KMk2 aircraft became one of the first major undertakings in the VC10 teardown programme. Several chord-wise sections of each joint were selected for systematic dismantling and examination by visual, NDT techniques and, ultimately, laboratory analysis by sectioning. The initial inspection techniques applied were those that would be applied in service, consisting of assisted visual assessment of the hole bores and eddy current checks for cracks. Then, with the joints dismantled, the initial results (including nil findings) must be corroborated by further visual and NDT examinations, including the use of ultrasonic and dye penetrant techniques. Finally, any defects or defect indications resulting from these phased inspections will be subject to laboratory analysis by sectioning and microscopy techniques, both optically and using the scanning electron microscope (SEM).

It was not considered necessary to dismantle the entire joint from each wing of the three aircraft but to undertake the examination on a sampling basis both to save time and to ensure that the scope of the investigation encompassed as many aircraft as possible. The selection of the sample sections ensured that at least TWO examples of very bolt location would be examined in detail with up to SIX examples in the areas of particular interest, such as at the intermediate spar run-out locations.

This approach resulted in a programme for the detailed assessment of 1000+ bolt holes and this programme is well underway although it will take several months to complete. Initial results from the first aircraft set of samples indicate that the joints are in very good condition (bearing in mind that these structures are over 35 years old) and no defects have been found in a total of over 300 bolt holes examined to date. As there is no reason to suspect that the first aircraft is in any way atypical of the general condition of the fleet, these results indicate that the teardown of the Rib O joints is likely to be successful in providing sufficient evidence to mitigate the requirement for onerous structural inspections that might otherwise be untenable with respect to the operational effectiveness of the fleet.

Whilst the technical success of the programme is the main consideration, it should be noted that there is a substantial cost involved in this achievement. No figures are presently available to accurately quantify this cost (due to the number of agencies involved and the amount of work which remains outstanding) but there is no doubt that it is a fraction of the maintenance cost that would otherwise be incurred.

2.7.9 New lifing methodology for aeroengine components (D P Shepherd, G F Harrison, A Richards, M R Brown, A D Boyd Lee, DERA Farnborough; S Williams, T Brown, P Lord, D Bowen, Rolls Royce Derby; M Bache, P Jones, IRC, University of Wales, Swansea)

A new lifing methodology for aero engine components has been developed [40], which incorporates several novel features. The principal features of the methodology are that; firstly, fully non-linear stress analysis techniques are employed in order to accurately model the stress and strain fields experienced by the component in service; secondly, distinct crack initiation and crack propagation models are employed in order to accurately describe the different phases of life development; and thirdly, a statistical model which explicitly incorporates volume effects is employed to allow the fatigue life distribution to be described for arbitrary geometries and loadings. The method thus allows for the lifing of full scale engine components direct from plain specimen data.

The stress analysis methods employed use combined elasto-plastic-creep constitutive relations to provide an accurate description of the stress-strain evolution state under cyclic loading at high stresses and temperatures. The plastic behaviour of the material is modeled using the Mroz multi-layer hardening rule, which has been integrated with two different creep laws to give the full analysis capability. The analysis can accurately model the evolution of the hysteresis loop, even under conditions of reverse yielding, and can predict whether plasticity or creep effects will dominate. Moreover, the crack initiation phase utilises an advanced materials behaviour model to account for the effects of temperature and mean stress in the calibration data, and extensive testing work has been carried out to determine the effect of parameters such as temperature, dwell and oxidation on crack propagation rates in typical engine component materials.

The statistical aspect of the work utilises a model of the size effect to relate failure distributions in uniformly stressed material to that in test pieces with arbitrary geometry and stress field. Experience in fitting common reliability distributions to experimental results has revealed that a 3-parameter Weibull distribution is needed to achieve an acceptable fit. Consequently, a size effect model based on this distribution has been developed. When combined with the materials models and stress analysis capability, this volume model allows for fatigue life prediction in arbitrary situations.

The methodology has been applied to an extensive database for a typical aero engine disc alloy. In addition to a very substantial number of plain specimen test results, the database includes a variety of notched specimen results as well as full scale component tests. The predictions of the model have been found to provide very good agreement with both types of tests, over a range of both loads and temperatures. The fact that both notched specimen and component results are successfully predicted provides a powerful validation of the methodology, since these two situations represent opposite extremes in terms of both volume and stress level.

2.8 FATIGUE PROPERTIES OF COMPOSITES

2.8.1 Use of Through Thickness Z Pins to Improve Damage Tolerance of CFRP Laminates (P E Irving, Cranfield University)

A series of studies [41-43] have been made of the effect of through thickness pins of pultruded unidirectional carbon fibre composite on the impact and compression after impact behaviour of quasi isotropic laminates of carbon/epoxy composite. The prepreg systems used were HTA/914 and IM7/8552. The pins were 0.28mm in diameter, and were positioned in a hexagonal array with a pin areal density of 2% giving a pin spacing of about 1.8mm. The influence of pinned area, shape and extent on the impact and compression after impact performance was established. The CFRP laminates were 4mm thick.

Previous work had established that the pins have a dramatic effect on the delamination toughness in modes I and II, increasing the apparent delamination propagation energy by factors greater than 10. It was found that when impacts of between 10, 15 and 20 joules occurred on pinned areas of composite, the damage area as measured using C scan was reduced by almost 50%. Compression after impact behaviour was similarly improved, with increases in compression strength after impact up to 50% greater when impact occurred on pinned areas of composite.

2.8.2 Fatigue Life Prediction of Polymer Composite Materials (P T Curtis and A J Davies, DERA Farnborough; H Carroll, T J Matthams and D M Knowles, Cambridge University)

DERA and Cambridge University are collaborating on research to develop improved models for predicting the fatigue life of polymer composite materials. Existing fatigue life prediction models have been reviewed [44,45] and two models (proposed by Yang and Shani [46] and Hwang and Han [47]) have been selected for further study. Work has been carried out to identify the input requirements for the models, and to establish which of the model parameters have physically based links. Once these requirements had been ascertained, experimental work was undertaken to obtain data to input into the models. Previous thermography work undertaken by Curtis et al [48] on UD and ±45° specimens was broadened to encompass quasi-isotropic specimens. Various test frequencies were used and the effects on heat build up in the specimens were studied. Work is in progress to develop an analytical model, based on that proposed by Yang and Shani [46] which uses the residual and static strength values of a range of composite coupons to predict the life of the coupon.

One of the objectives of the research at Cambridge University is to extend the range of validation of the constant-life model (see item 2.8.3 below) to other lay-ups of interest, for example newly developed materials that have not yet been thoroughly researched (including both non-woven and woven reinforcements). This work is being supported by metallographic analysis of damage and failure modes with a view to exploring the feasibility of building some element of damage mechanical predictive capability into the model. Studies of damage development in plain laminate fatigue (with varying lay-up) are being undertaken to try and understand parameters in the constant-life model in terms of physical mechanisms. Current work at Cambridge includes the manufacture and testing of mixed mode bending (MMB) specimens to investigate the growth of interfacial cracking and strain energy release rates under fatigue loading conditions. Possible links between residual stiffness/strength and coupon life are being investigated. The overall aim of the research summarised in this item is to establish physically based models for predicting fatigue life and damage growth in composite structures. A realistic initial aim is to develop a model for predicting fatigue life and residual strength as function of R ratio for constant amplitude loading.

2.8.3 Development of Empirical Fatigue Life Model for Fibre Reinforced Composites (B Harris, T Adam and M H Beheshty, Bath University; P T Curtis and A J Davies, DERA Farnborough)

In research at Bath over a 20-year period, a series of programmes has been carried out on the fatigue behaviour of glass- and carbon-fibre-reinforced plastic composites. The accumulation of a substantial body of data, and detailed analysis of these data, has led to the construction of a parametric constant-life model, which provides a full description of the stress/life/R-ratio surface of a given laminate in terms of the monotonic tensile and compressive strengths of the material and three empirical parameters. With the aid of this model, predictions can be made of fatigue life for a wide range of potential service conditions, even at an early stage in the development of a new material when relatively few real fatigue data are available.

For comparison with the behaviour of a number of different CFRP laminates already studied, further constant life fatigue data have been obtained recently for a further CFRP composite and a GRP laminate of similar construction - a 16-ply $[(\pm 45,0_2)_2]_S$ lay-up. Fatigue tests have been carried out for these materials in both the virgin condition and after damage by low velocity impacts. Following analysis of these new data and a re-examination of the older database, the constant life model has been modified [49]. It now offers a prediction procedure for the fatigue response of composite materials in the virgin and impact damaged conditions which requires, in the first instance, only the tensile and compressive strengths of the composite in question. The model is equally applicable to both CFRP and GRP materials.

Validation of the constant-life model by working with fatigue data for new materials is continuing. Simultaneous attempts have been made to predict stress/life/R-ratio envelopes prior to the availability of fully comprehensive data sets in order to test the practical capabilities of the model in potential design situations; some good successes have been reported. Other areas that are still being researched are the establishment of levels of statistical validity of predictions by application of extreme-value theory and the writing of a free-standing software programme embodying the analysis and prediction of fatigue life which will be incorporated into the National Modelling Tools Program.

2.8.4 Life Prediction Methodologies for Ceramic Matrix Composites (D Shepherd, DERA Farnborough)

A statistical method for predicting the life to failure distribution for ceramic matrix composites with arbitrary fibre architecture and loading has been developed. It is based on failure models which have been developed to describe the behaviour of unidirectionally reinforced materials, under unidirectional loading. These models work by considering the fragmentation process undergone by fibres when embedded in a matrix and subjected to load. An analytical solution for the distribution of fibre fragment lengths is established, and this is used as the basis for describing the failure properties of the composite, such as UTS and pull out length. This approach has now been generalised, so that predictions can be generated for CMC components under arbitrary loading and temperature conditions.

A database for a commercial SiC/SiC CMC has been used to derive appropriate parameter values for the model. Work is continuing to extract the relevant information, so that the modeling approach can be implemented and fully validated.

2.8.5 Smart Reinforced Aluminium Alloy Laminates (P Gregson and M M Singh, Southampton University; A Rahman and P M Powell, DERA Farnborough)

Carbon fibre reinforced aluminium alloy laminates offer high specific stiffness and strength together with excellent tension-tension fatigue performance and are therefore of interest for structural airframe applications that require good damage tolerance. A programme of work [50,51] has been carried out to develop an improved understanding of the fatigue behaviour of such laminates, using embedded optical fibres carrying Bragg grating strain sensors. Laminate panels were manufactured, consisting of four plies of unidirectional carbon pre-preg (913 HTA or 920 TS) sandwiched between two outer layers of 0.5mm thick 8090 aluminium-lithium alloy. Fatigue cracks were then grown from a central, through-thickness hole under constant amplitude loading at stress ratios of R=0.1 and 0.5. A remote multiplexed grating interrogation system was constructed which allowed the strain field in the carbon fibre reinforced layer to be mapped across a 20mm x 20mm area in the vicinity of a fatigue crack in the alloy, by an array of 40 sensors. The interrogation system is based on an acousto-optic tunable filter (AOTF) which tracks each grating in turn. Software was developed to enable automatic control of the interrogation and continuous logging of the strain data.

Changes in strain amplitude have been related to the growth of delaminations between the metal and the carbon fibre layers, which was monitored by means of ultrasonic C-scans recorded in-situ at intervals during a fatigue test. Delamination between the alloy skins and the carbon fibre core was shown to be a dynamic rather than a static process, controlled by cyclic loading. Numerical simulations of the fatigue failure process have been carried out, based on ultrasonic records of the delamination profile. The results showed good agreement between the measured and predicted strain amplitudes in the vicinity of the crack tip, with the differences being attributed to asymmetric debonding in some of the tests.

Experiments have shown that the thickness of the resin-rich layer at the interface between the metal and the carbon fibre core significantly affects the fatigue crack growth rate. An effective stress intensity factor range, which takes into account load-partitioning and crack-tip shielding, has been shown to be the most significant parameter for characterising fatigue crack growth in these materials. Further modifications to this parameter will be necessary to include the effect of adhesive properties. The experimental data measured to date will be used to develop and validate a finite element simulation of the damage accumulation process.

The direct measurement of strain using Bragg grating strain sensors has led to an enhanced understanding of the failure mechanisms in hybrid laminates. The potential of the application of fibre optic technology for structural health monitoring of fibre reinforced metal laminates has been demonstrated.

2.9 FATIGUE OF JOINTS AND COMPONENTS OF COMPOSITE MATERIALS

2.9.1 Prediction of Fatigue Damage Initiation for Impact Damaged Composite Structures (F L Matthews and A J Kinlock, Imperial College London; A J Davies and E Greenhalgh, DERA Farnborough)

A finite element based procedure has been developed at Imperial College for predicting fatigue lifetimes for composite plates containing a central hole. The method uses the stress distribution from two-dimensional FE analysis to calculate strain energy release rates which, in combination with fracture mechanics concepts, predicts fatigue lifetime via a Paris Law approach. Damage is taken to be due only to matrix cracking, and any stiffness reduction of the composite laminate is based solely on this form of damage.

Recent work has progressed from an I-beam containing holes in the web to skin/stringer panels which have sustained impact damage. The current approach involves three-dimensional finite element analysis (utilising a global/local approach) but is restricted in terms of the number of off-axis plies that can be interrogated for possible damage; the computer coding will need to be generalised to account for any number of ply orientations (any lay-up). The proposed computer model will track the development of damage in the structure (inter-fibre cracking when a threshold strain is exceeded), calculate the strain energy release rate and link the latter to number of cycles.

Complementary work has commenced where the objective is to replicate and simplify actual BVID level impact damage at coupon level. This work relies heavily on NDE techniques to ascertain the proportion of each damage mechanism typically found in composite materials namely: fibre fracture, resin fracture and delamination. Methods will be investigated to replicate this damage, these coupons will then be tested in tension-tension fatigue and the growth of the damage will be compared with actual impacted damaged specimens. This programme of work will be used to understand the role of each damage mechanism on the life of an impacted composite coupon. Another advantage of using this technique is that it should make it easier to construct predictive equations to predict the fatigue life of a specimen.

2.9.2 Adhesively Bonded Repairs to Fibre Composite Materials (M N Charalambides, A J Kinloch and F L Matthews, Imperial College, London)

The performance of carbon fibre/epoxy repair joints, bonded using an epoxy film adhesive, under static and fatigue loading has been investigated [52]. The repair joints were immersed in distilled water at 50^{o}C for periods up to 6 months and the effect of hot-wet environment on the static and fatigue strengths was evaluated. Residual strength tests, where repairs were subjected to fatigue followed by static loading, were also determined. All tests were undertaken at room temperature. It was found that there was no major effect of the conditioning on the above properties and that the repair joints had a similar static strength to that of the parent material. In contrast to the static properties, the fatigue behaviour of the repair joints was significantly inferior to that of the parent material. Finally, fatigue tests were also performed on relatively large fibre/epoxy panels with centrally placed repairs. The fatigue results obtained from the repair panels were in close agreement with the fatigue results obtained from the repair joints.

The mechanical properties of the adhesive and the CFRP were used in conjunction with finite element analysis to determine failure criteria which would predict the experimentally observed failure paths and strength of the adhesively bonded repair joints. Two material models were used for the adhesive: linear elastic and linear elastic-plastic. Two models were also used for the composite. In the first model, the composite was assumed to be a homogeneous orthotropic material with smeared properties. In the second, it was modelled as a combination of individual plies of various orthotropic/anisotropic properties, depending upon the fibre orientation angle. Three possible types of failure for the repair joints were analysed in order to predict the expected failure paths and failure loads. The general, agreement between the experimental observations, and predictions of the failure path and loads was found to be good [53].

2.9.3 Fatigue of Adhesively Bonded Joints (I A Ashcroft, Loughborough University; S J Shaw, DERA Farnborough)

Introduction

The use of structural adhesives in the construction and repair of engineering structures, e.g. aircraft, can offer substantial benefits in comparison to traditional joining techniques such as mechanical fastening. Operational benefits which can result from significant use of adhesive bonding include improved structural performance, resulting largely from the significant weight reduction adhesive bonding can provide, together with substantial reductions in both procurement and life-cycle maintenance costs. In spite of these potential advantages, a general lack of confidence in the ability of adhesive joints to withstand the full range of environmental and loading conditions experienced during aircraft operations has prevented their widespread use. In particular, limitations relating to long-term durability characteristics in warm/moist environments and an inability to accurately and quantitatively predict the lifetime and long-term performance, have been the cause of most concern to aircraft designers and airworthiness authorities. In this work, the effect of environment on the performance of three types of bonded joint has been studied. The lap-strap joint is representative of long-overlap joints, in which creep is restricted, the double lap joint is representative of short overlap joints in which global creep is not restrained, and the bonded double cantilever beam (DCB) is used to generate mode I fracture mechanics data. In all cases the adherend material was a carbon fibre reinforced polymer (CFRP) and bonding was carried out using an epoxy film adhesive.

Fatigue testing of bonded CFRP joints

The effect of test environment and pre-conditioning on the fatigue behaviour of CFRP/epoxy lap-strap joints has been investigated [54]. It was shown that the fatigue resistance of the lap-strap joints did not vary significantly until the glass transition temperature, T_g, was approached, at which point a considerable reduction in the fatigue threshold load was observed. It was also noted that absorbed moisture resulted in a significant reduction in the T_g of the adhesive. The locus of failure of the joints was seen to be highly temperature dependent, transferring from primarily in the composite adherend at low temperatures to primarily in the adhesive at elevated temperatures. It was also seen that as the crack propagated along the lap-strap joint, the forces at the crack tip tended to drive it into the strap adherend, which could result in complex mixed mode fracture surfaces.

Double lap joints were tested both quasi-statically and in fatigue across the temperature range experienced by a jet aircraft [55]. Two variants of the double lap joint sample were used, one with multidirectional (MD) CFRP adherends and the other with unidirectional (UD) CFRP adherends. Finite element analysis was used to analyse stresses in the joints. It was seen that as temperature increased both the quasi-static strength and fatigue resistance decreased. The MD joints were stronger at low temperatures and the UD joints stronger at high temperatures. It was proposed that this was because at low temperature the strength was determined by the peak stresses in the joints whereas at high temperatures, strength was controlled by creep of the joints which is determined by the minimum stresses in the joint. This argument was supported by the stress analysis.

Mode I constant displacement rate tests were conducted on DCB joints at –50°, 22° and 90°C [56]. Temperature was seen to influence the mode of fracture, which progressed from stable, brittle fracture at low temperatures to slip-stick fracture at room temperature and finally to stable ductile behaviour at elevated temperatures. This behaviour was attributed to the dependence of critical strain energy release rate on crack velocity for epoxy adhesives and a model for the fracture behaviour of viscoelastic materials was used to explain these results. The critical strain energy release rate was seen to increase with temperature and the failure locus transferred from predominantly in the composite substrate to predominantly in the adhesive. Fatigue tests were also conducted on DCB joints at –50°, 22° and 90°C and a number of techniques for determining strain energy release rate and crack propagation rate were evaluated [57]. It was seen that temperature had a significant effect on the locus of failure and fatigue crack propagation, indicating that service temperature must be taken into account when designing bonded composite joints.

Diffusion Analysis

In order to better understand the role of environment on the fatigue behaviour of bonded joints, the diffusion of moisture in carbon fibre composite bonded joints was studied experimentally and numerically, and analytical solutions for the effect of fillet shape on diffusion into joints with impermeable adherends were derived [58]. Secondly, semi-coupled finite element diffusion and stress analyses were used to model the effect of the diffusing moisture on the mechanical response of bonded composite lap-strap joints [59].

The experimental studies concentrated on moisture diffusion in adhesive films and in unidirectional and multidirectional composite substrates exposed to two different conditioning environments, namely 45°C/85%RH and 90°C/97%RH for the absorption studies and 90°C/ambient for the desorption studies. The coefficients of diffusion were determined from the water uptake plots. The analytical solutions for diffusion in joints with impermeable adherends were based on the classical theory of diffusion and were used to derive equations in two-dimensions for different adhesive fillet shapes, namely radiused fillet, triangular fillet and rectangular fillet. In the finite element analysis, the diffusion of moisture from the composite substrates into lap-strap joints was also taken into account. Both unidirectional and multidirectional composites were considered, as well as two different fillet shapes, i.e. rectangular and triangular fillet. A comparison was made between the results obtained using FEA and those obtained using the analytical solution. Fatigue test data for lap-strap joints aged and tested in different environments were analysed and a tentative link between fatigue threshold and water concentration at the site of failure initiation was made, indicating a semi-empirical method of predicting the strength of joints subjected to moisture induced degradation. Finally, semi-coupled finite element diffusion and stress analyses were used to model the effect that moisture absorption and desorption has on the stresses and strains in lap-strap joints during fatigue testing in different environments. Changes in the stresses and strains at the point of failure initiation were used to predict the effect the different environments would have on the fatigue threshold.

Lifetime Prediction Studies

Although many failure criteria have been used and proven adequate for a limited range of joint types, no globally applicable criteria have been demonstrated for adhesively bonded joints. Stress/strain based failure criteria have to contend with the problem of singularities in theoretical stress analyses and there are difficulties in applying fracture mechanics criteria to uncracked samples with poorly defined flaw distributions. In this programme a number of analysis methods and failure criteria have been assessed for the lifetime prediction of bonded joints.

The fatigue strength of adhesively bonded lap joints was analysed using finite element analysis and both strength of materials and fracture mechanics failure criteria [60]. Criteria based on the maximum principal stress provided good fatigue threshold predictions in the case of small plastic deformations, whereas the maximum principal strain, von Mises strain, shear stress and von Mises stress criteria resulted in more accurate predictions in the joints that underwent large plasticity. Elastic and elasto-plastic fracture parameters for both interfacial and cohesive cracks were also calculated. Both the J-integral and the elastic strain energy release rate were shown to correlate well with the fatigue threshold load for the different joint types.

The applicability of fracture mechanics data to the prediction of fatigue failure in uncracked lap joints was further assessed by attempting to predict fatigue thresholds in two types of lap joints at three different temperatures using DCB data and G_T as a failure criterion [57]. In most cases reasonable predictions were made, the notable exception being the over-prediction of the fatigue threshold load in double lap joints tested at 90°C. This was attributed to creep in the double lap joints, which accelerated fatigue failure. It was recommended that in order to improve current prediction techniques, efforts should be made to base predictive methods on accurate physical models of the degradation and failure processes in the joints.

The prediction of fatigue behaviour in bonded joints using Continuum Damage Mechanics (CDM) has also been investigated [61]. Two joint types were considered in this study, namely, Double Lap (DL) and Lap Strap (LS) jointsand two substrate types, namely, Uni-Directional (UD) and Multi-Directional (MD) CFRP. Damage evolution laws were derived using thermodynamics principles. The number of cycles to failure was then expressed in terms of the stresses in the adhesive layer and material constants. The stresses were calculated from non-linear finite element analyses, considering both geometrical and material non-linearities. The damage laws generated for the UD/DL joint data were then used to predict the fatigue crack initiation thresholds for the MD/DL, UD/LS and MD/LS joints. It was found that the predictions using CDM were slightly more accurate than those obtained using the fracture mechanics approach. In general, when predicting the fatigue thresholds of the LS joints using the DL joints data, or vice-versa, good agreement was obtained between the measured and predicted thresholds at ambient and low temperatures, but poor agreement was seen at the high test temperature. This was attributed to the deleterious effect of creep, which was greater in the DL joints than in the LS joints.

2.10 NON-DESTRUCTIVE EVALUATION

2.10.1 NDE for Ageing Military Platforms (E A Birt, D A Bruce, D J Harrison and R A Smith, DERA Farnborough

Because of the variety of aircraft corrosion problems a range of detection techniques have had to be developed [62]. This project has included

- development of ultrasonic high frequency imaging
- quantitative assessment and modelling of ultrasonic spectral methods
- multi-frequency eddy current optimisation
- transient eddy current methods including a rapid scanning Hall effect array probe

together with participation in two international comparative studies under the TTCP [63, 64] and Four Powers ASNR programmes. Useful information and specimens have been provided to the UK programme through these international collaboration exercises

Ultrasonic imaging and spectral techniques

Ultrasonic methods have been shown to be the most sensitive of the conventional NDE techniques for detection of corrosion in single layers of metal, allowing the earliest detection of attack. It has been found, however, that pitting corrosion can be very difficult to detect by eddy current or low-frequency ultrasonic methods. In a typical specimen, taken from a Transall aircraft, standard ultrasonic methods carried out at under 20MHz could, with some difficulty, detect the presence of some form of corrosion but could not characterise it [62].

X radiography of the affected areas was able to detect the pitting and also the presence of corrosion products, as illustrated in figure 33. Although the X ray technique can provide an indication of the severity of the pitting, it is at best semi-quantitative as the presence of variable amounts of trapped corrosion product reduces the contrast and hence the apparent depths of the pits.

High frequency ultrasonic imaging at 80MHz was demonstrated to be capable of providing detailed images of the pitted surface in excellent agreement with the X radiography, as shown in figure 34. The high frequency used, 80MHz, is well above what would normally be accepted for field inspection. It could be used, however, in conjunction with either X radiography or lower frequency ultrasonics to provide accurate measurements of corrosion morphology to facilitate optimal repair or prediction of remaining life.

The spectral method [65,66], which measures surface roughness, allows discrimination between corroded surfaces and pristine interfaces. Interpretation of the reflectivity spectrum is based on a model of the surface which is based on a probability distribution for surface displacement, i.e. material loss. In the initial work this distribution was assumed to be Gaussian, as it would be for a randomly rough surface. This technique has been shown to be able to make quantitative measurements of surface roughness down to a limit of better then 5 microns rms roughness. The use of focused rather than planar transducers has been shown to give better agreement with profilometer measurements, probably due to reducing the criticality of probe alignment.

A phase screen modelling approach has been used [62] to evaluate the limitations of the simple Gaussian surface model used to interpret the ultrasonic spectral measurements. The model was adapted to examine the effect of a non-Gaussian distribution of heights. As a possibly more realistic representation of a corroded surface, the exponential distribution was chosen. This is an asymmetric function which may be thought of as representing attack from one direction with the larger numbers corresponding to particularly deep pits. The probability density for this model is shown in figure 35a compared to the Gaussian model with the same mean and rms roughness parameters of 0.05mm. The exponential surface model has more of the corroded surface close to the original plane. The resulting values for attenuation as a function of surface roughness are shown in figure 35b.

It is clear that while the attenuation values for the exponential surface approach the Gaussian theoretical line at small values of roughness as expected, the attenuation for rougher surfaces, while increasing monotonically, is substantially less than that observed for the Gaussian model. The dotted line shows the result of an exact calculation for the exponential distribution. This confirms the expectation that for real corroded surfaces which cannot be expected to have precisely Gaussian height distributions, the simple model will be quantitatively accurate only for small values of surface roughness but will give a qualitative indication which could possibly be calibrated for known corrosion types for larger values.

Finally, the model was used to assess the effect of pitting corrosion where the surface consists of a substantial area of uncorroded material with several deep corrosion pits. These were modelled as roughened hemispheres. Initial results have been obtained for two cases having groups of pits with radii ranging from 0.3mm to 1.5mm. The spectral characteristics of this scattering are different from the above randomly rough surface models. There is little frequency dependence except at very low frequencies. At high frequencies the reflectivity can be described well by a simple geometrical interpretation, namely that the flat portions of the surface effectively scatter the appropriate fraction of the incident amplitude while the rough, hemispherical pits contribute little to the reflected wave.

Multi-frequency and transient eddy current methods

Multi-frequency eddy current methods are notoriously difficult to set up. DERA has investigated dual frequency methods, showing that they can be used to detect second layer corrosion under favourable circumstances using a model to calculate the optimal frequencies. The method has been extended for optimising multi-frequency eddy current inspection methods based on a layered structure model. By modelling the inspection process for an air-cored coil using 2D analysis and calculating the sensitivity of coil impedance to defect size it is possible to predict the optimum conditions under which the inspection should be carried out. The method has been tested on simulated inspection data [62].

In order to be able to use the above optimisation method it is necessary to have a method for measuring or predicting the magnetic field produced by the transducers used in the field, which will typically be a commercial probe of unknown geometry, probably with a ferrite core. A technique has therefore been developed for characterising the spatial frequency characteristics of eddy current probes [67]. The technique allows prediction of the probes performance from an initial calibration measurement of the impedance change as a function of frequency. In most cases, the impedance change can be fitted to a generic approximating function based on the spatial frequency spectrum for a single loop. This is then used to predict the fields produced by the coil to allow optimisation of the inspection process.

When the multi-frequency optimisation is combined with the technique for characterising eddy current probes in terms of the spatial frequency distribution this provides the basis for incorporating real-time model-based optimisation into the inspection process. This will allow scanning of structures where quantities such as skin thickness change without continually resetting the inspection instrument.

Transient eddy current methods have been shown to be capable of detecting corrosion in layered structures. Material loss detection limits have been measured for typical probes on artificial specimens [68-70]. Under favourable circumstances a capability to detect 1% material loss has been demonstrated. The variation in detection capability with defect area is significant, as shown in figure 36.

In collaboration with AMRL, algorithms have been developed to allow correction for lift-off and edge effects and to measure total material thickness. These algorithms have been shown to be necessary for inspection of real structures in the TTCP round robin exercise [64]. The transient method applied without corrections had difficulty in detecting second layer corrosion. When the correction factors were introduced there was very good agreement between the indications detected by the transient eddy current method and the areas of medium to high material loss measured by a quantitative X radiography technique by IAR (Canada).

A Hall effect array has been constructed and demonstrated, allowing the scanning speed of the transient EC methods to be greatly increased. The prototype array has 9 elements. Careful attention to probe co-calibration allows a coherent image to be collected despite variability in the individual sensor elements and the magnetic field. The sensor array is sparse in the sense that the elements have a spacing of 4 mm. This allows rapid scanning of suspect areas, the resulting scan is very coarse but the entire area is covered. Regions where there is no significant variation in the specimen are correctly represented but for regions where there are significant variations, such as in regions of corrosion, the resolution is clearly limited. It is possible, however, to continue the scan, repeatedly over-scanning the regions of interest to enhance the image resolution. The advantage of this approach is that large areas, where there is no signal activity, can be scanned very rapidly; when regions of change are encountered, over-scanning permits a correspondingly higher measurement density. This permits the local resolution to be increased up to a limit which would ultimately be determined by the size of the individual Hall elements, although a more practical limit may be around 1mm. While the Hall array was developed primarily for corrosion detection, it could also be used for rapid detection of fatigue cracks. In this case some care would have to be taken over the direction of the expected cracks, as the array elements are strongly directional owing to the linear currents produced by the exciting coil.

2.10.2 Large Area Methods for In-Service Inspection of Carbon Fibre Composite Structures (D A Bruce, R A Smith and S J Willsher, DERA Farnborough)

In a review of large area techniques prepared shortly before this project began, three inspection methods were identified for further development or investigation. These were ultrasonic arrays, lock-in thermography and phase stepped optical inspection.

Ultrasonic linear array probes have been developed [71] which can perform inspections over medium areas at realistic inspection speeds of around $4m^2hr^{-1}$. This level of performance fully met the desired outcome and will give the necessary capability for “fingerprinting” the as-delivered structure of the Eurofighter Typhoon aircraft to set up the structural integrity database. The method is adequate for occasional inspection of small military combat aircraft or for inspection of localised areas, but will not be acceptable for frequent inspection or for global inspection of large areas, particularly those liable to be encountered on future transport types.

A limitation with all of the early arrays was the use of external multiplexers necessitating the use of many cables to connect each array element in parallel. The resulting systems were heavy and bulky, unsuitable for operation in uncomfortable conditions such as scanning the underside of an aircraft structure. In this project, a further series of arrays were commissioned with the multiplexing integral with the probe. These probes can be operated with a conventional test set. Suitable modifications were incorporated in the ANDSCAN software to take account of the multiplexing and positioning of the probe. Frequencies of 5MHz and 7.5MHz were employed, the latter using new piezo-composite technology. The initial series of probes were composed of 8 effective elements, however it was possible to add a second multiplexer within the probe housing and this was taken advantage of in a second pair of 17 element arrays giving a swept width of 45mm. This is not necessarily the maximum that could be used, but was convenient given the volume of the electronic components and the data capture rates of the scanning system.

The longer arrays were used initially with the standard ANDSCAN r- positioning unit. However, it was found that a manually operated X-Y scanning system made it easier to maintain accurate linear scans which resulted in a further increase in scanning speed. The system was demonstrated at RAF St Athan using a Eurofighter fatigue test wing. The $16m^2$ structure was mapped in approximately 8.5 hours.

Models have been used to show how the lock-in thermography technique can be optimised depending on the defect depth and to derive limits on the minimum detectable defect size for a given depth [72]. They have also been used [71] to demonstrate the use of synthetic aperture focusing (SAFT) and the simpler two dimensional deconvolution image processing techniques which will be required to improve the depth penetration of the method to depths significantly below 2.5mm. The SAFT and deconvolution techniques were able to compensate for the lateral diffusion of heat energy, improving image sharpness, defect sizing capability and signal to noise ratio. It can be seen from the preliminary results that the SAFT technique in particular has considerable potential for improving the lock-in thermography method. Synthetic focusing to produce an image of the defect gives an additional method for depth estimation as well as allowing an increase in sensitivity.

The use of phase stepped optical interferometry techniques had been expected to be limited by the limitations of the electronic detection systems due to speckle de-correlation effects. It has been shown that this does not occur despite the fact that the multiple speckles imaged by a pixel are decorrelated. This means that the limitation on use of optical techniques remains the provision of an adequate stressing technique to cause a surface distortion. These methods remain less attractive than the thermal method due to this difficulty of providing a suitable method of stressing the surface.

2.10.3 Advanced Ultrasonic Spectral Methods for Adhesive Bonded Joint Assessment (D A Bruce, R A Smith, V L Weise, DERA Farnborough)

Considerable effort has gone into investigating NDE methods for bond inspection, but the capabilities of the methods which have been developed are very limited. A major difficulty in addressing the development of NDE methods is the controversy amongst adhesion specialists regarding the precise failure mechanisms. This programme seeks to investigate in detail the three major areas where it is considered that there is a fundamental lack of understanding and where it is anticipated that real progress could be made [73]. The three areas addressed are;

- Detection of contamination at the bond interfaces, particularly for composite adherends
- Detectability of "kissing" bonds or "zero-volume unbonds"
- Environmentally driven deterioration in bonded joints with metallic adherends.

In the contamination studies, it has been shown that moderate levels of common contaminants can be detected. The current work centres on improving the techniques required to apply controlled amounts of contaminant and to measure the degree of contamination to allow more control over the measurements. It is then intended to establish detectability limits and to compare these with the levels of contamination which cause an unacceptable loss in mechanical properties.

Kissing bonds have been studied [74] using a model system in which the kissing bond is artificially created by a cathodic delamination process. Normal and oblique incidence ultrasonic techniques have been used to detect these defects. Spectral measurements have been made using pulsed and swept frequency techniques. It is intended to use these to look for non-linear effects arising from the unbonded areas. As part of this work the spectral characteristics for layered systems have been modelled [75]. A basic model has been developed that can be used to predict where in the frequency domain resonances corresponding to varying joint component dimensions are located. This model can also be used to deduce which features of the spectrum should be most sensitive to interfacial changes. Experimental results also suggest that it is possible to distinguish between aluminium surface preparations used in adhesive joints using this technique.

The uptake of water in an adhesive joint can be monitored using standard normal incidence ultrasonics. Oblique incidence ultrasonics has been used to determine changes in the interface between adherends and the adhesive layer due to the water uptake.

2.10.4 Assessment of NDE Reliability Measurement for Airworthiness and Airframe Life Management (D A Bruce, DERA Farnborough)

Optimisation of an inspection strategy to provide acceptable safety at minimal cost requires a knowledge of the reliability of the inspection procedures which could be used. A methodology for assessing inspection reliability, characterising the inspection process by a 95% confidence level probability of detection (POD) curve estimated from artificial trials, has become the standard approach. This method works satisfactorily for straightforward inspection situations where the POD curve can be estimated from a large database, but application of similar methods to airframe inspection suffer from the prohibitive cost of obtaining the reliability curve from realistic trials.

Where there is limited data available to determine the reliability, the in-built conservatism of the standard method leads to wholly unrealistic estimates for the POD curve which in turn give rise to unacceptably short inspection intervals and excessive maintenance costs. It may be possible to deduce inspection reliability from in service inspection data, although the diversity of inspection situations suggests that there will still be a very limited amount of information available from which to estimate the reliability for a particular inspection task [76].

In this project the effect of the in-built conservatism inherent in the standard method of POD assessment was demonstrated for simulated inspection data based on real inspection reliability trials. Alternative approaches to the prediction of NDT performance were compared to establish the minimum requirements for inspection data in order to achieve specified safety levels.

The standard methods based on linearised regression and maximum likelihood curve estimation have been examined. Both have been shown to have severe limitations when used to estimate reliability curves from small amounts of data, in particular they both systematically overestimate the 95% confidence limit on the reliability due to their incorporating large sample approximations.

A method using techniques based on Bayesian inference to provide an optimal prediction of reliability, which can be refined as further information is acquired, has been developed as has a simpler approach based on a chi-squared test.

These methods have been demonstrated on real and simulated data. The Bayesian approach has been used to calculate the necessary specimen numbers to demonstrate a safety level of one in 1000 for a given number of "misses". It has been shown that the number of specimens required is less than 70% of those required by the standard POD method for the standard situation assuming three inspections during defect growth. If the number of inspections is allowed to increase to four, the necessary number of specimens is less than 40% of those required by the standard method. The analysis methods have been used to analyse data provided by the Netherlands NRL as part of the UK contribution to a NATO working group on improving NDT reliability in the field (NATO AVT-051).

2.10.5 Collaborative Programmes on Low Cost Manufacturing (D A Bruce, DERA Farnborough)

DERA has participated in several projects to investigate the use of low cost manufacturing methods for carbon reinforced composite components. The most recent was AMCAPS II, led by Airbus UK. This project involved the investigation of RFI processing for large, thick composite structures. The requirement has been to assess existing inspection methods for thicker sections and complex structures and developing primarily ultrasonic techniques to detect novel, process-related defects. DERA's contributions involved quantitative studies of attenuation measurement in thick section material, measurement of attenuation in new generation composite materials and assessment of ultrasonic techniques for detection of novel, process related defects.

2.11 DESIGN DATA (AC Quilter, ESDU International plc, London)

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

Data Item No. 84003 Fatigue crack propagation rates and threshold stress intensity factors in high alloy and corrosion resistant (stainless) steel

Data Item No. 84003 has recently been updated and extended. The update consists of the extension of the data for alloys already featured in the Item and the addition of new data for alloys not currently featured in the Item. The opportunity has also been taken to update the text of the Items to take account of recent developments in the field of fracture mechanics. The issue of the update completes the programme of updating and extending all the ESDU crack propagation rate data.

The Item provides constant amplitude crack growth rate data for a wide range of common specifications of high alloy and stainless steels in the form of plate, sheet, bar, castings or forgings. The data are derived from results of tests, drawn from many sources in the literature, made at room temperature in laboratory air for various values of stress ratio. The chemical composition of the steels and mechanical properties of the test specimens are tabulated, and summary curves show the effects of steel composition on crack growth and of stress ratio on threshold stress intensity factor range. The effect of crack closure in creating an effective stress intensity factor is discussed. Other matters discussed are the effects of specimen size, environment, heat treatment, steel composition and stress ratio. Practical worked examples illustrate the use of the data.

Data Item No. 99002 Computerised crack resistance curves

This Data Item and the accompanying computer program will be issued in the first half of 2001. The Data Item presents a program based on the method described in Data Item No. 85031, "Crack resistance curves". The program determines either the fatigue crack length that will cause fast fracture of a component under a given applied stress or *vice versa*. The program runs as a module within the Microsoft Excel spreadsheet program and gives results in the form of both numerical and graphical data. All of the resistance curve data presented in Item No. 85031 are available within the program; the facility for users to add their own data is also provided.

Current Work

Cumulative damage program

Work on a Fortran program to perform cumulative damage calculations is in progress. The program will accompany Data Item No. 95006, "Fatigue life estimation under variable amplitude loading using cumulative damage calculations". Data Item No. 95006 presents methods of estimating the fatigue life of a component subjected to variable amplitude loading and it is one of those methods, the rainflow method, on which the program is based.

Metallic Materials Data Handbook, ESDU 00932

Work on an electronic version of the Metallic Materials Data Handbook is nearing completion. The package will be available on CD and it will be possible to run it locally on a PC or over a network. The package includes a search facility enabling the user to locate materials meeting certain combinations of user-specified criteria relating to anything from the material form through composition and physical and mechanical properties to suitability to being welded by a particular process. All the graphical data in the Handbook are included as fully interactive figures. The facility to save all or part of the data, both numerical and graphical, relating to a particular material specification is also included.

Work on delivery of the Handbook via the internet is also in progress with completion scheduled for the Autumn of 2001.

Rooke and Cartwright's Compendium of Stress Intensity Factors

Permission has been obtained by ESDU from Her Majesty's Stationery Office to include Rooke and Cartwright's compendium of stress intensity factors in ESDU Data Item No. 78036, "The compounding method of estimating stress intensity factors for cracks in complex configurations using solutions from simple configurations". Work on converting the compendium to an electronic format to enable its delivery both on CD and via the internet is in progress and completion of the project is scheduled for the end of 2001.

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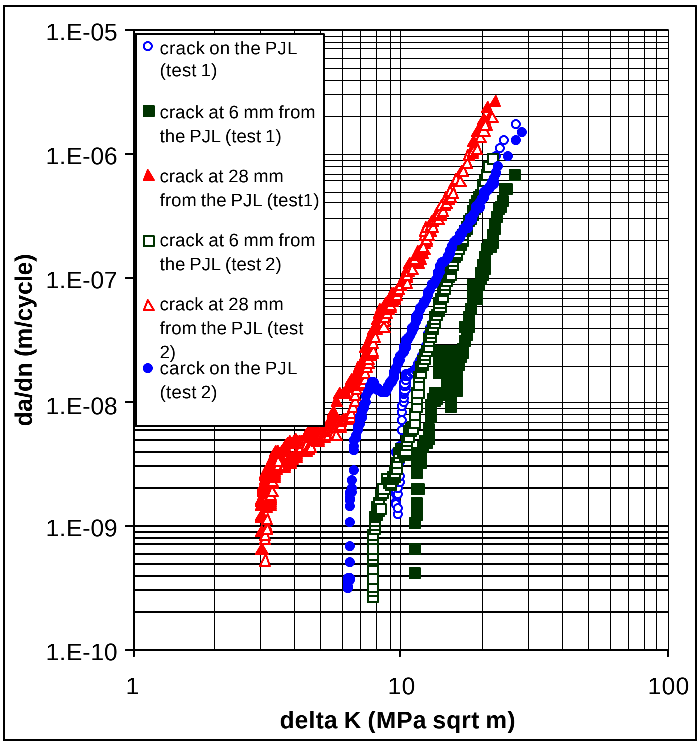

Figure 1 Fatigue crack growth rates parallel to the weld line

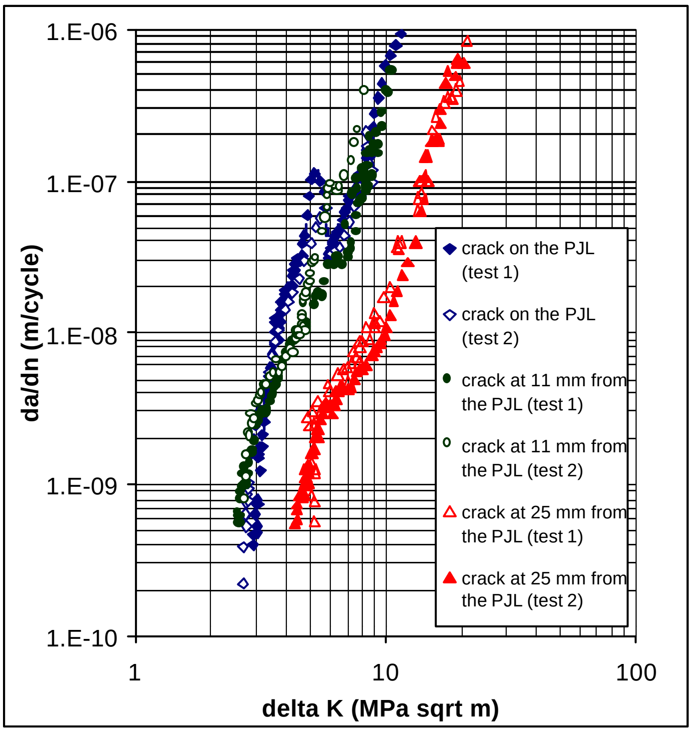

Figure 2 Fatigue crack growth rates perpendicular to the weld line

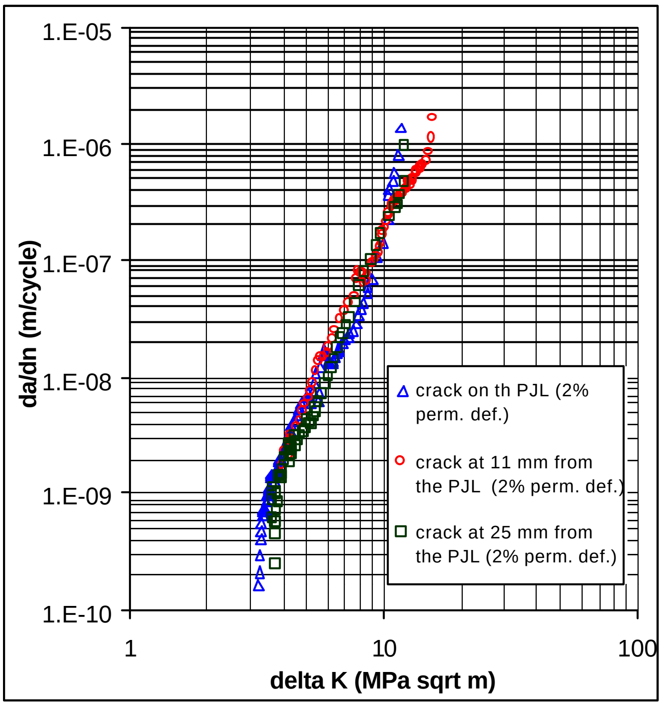

Figure 3 Fatigue crack growth rates perpendicular to the weld line after 2% plastic strain

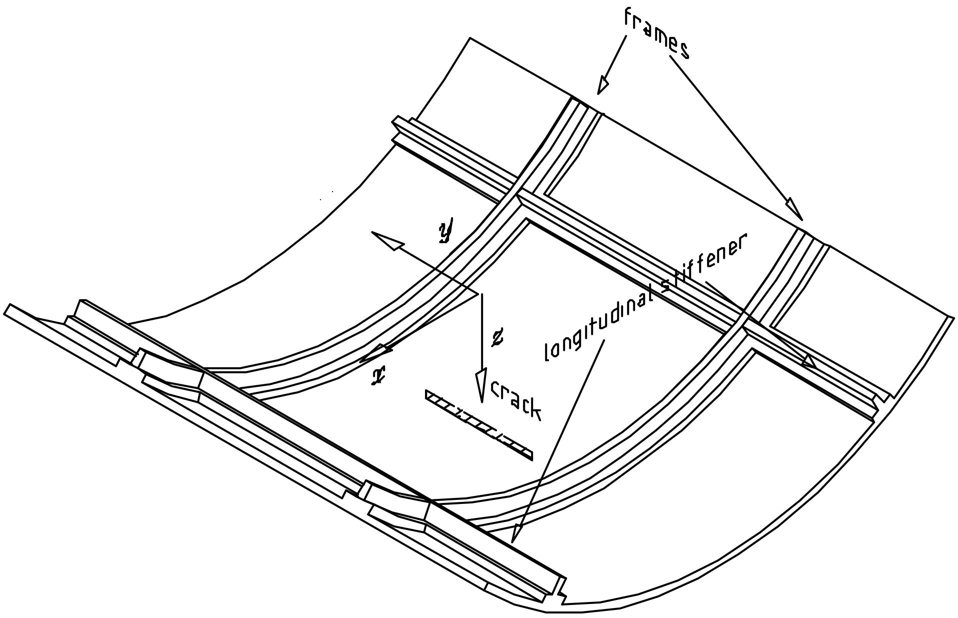

Figure 4 Cracked shell with two longitudinal stiffeners and two frames

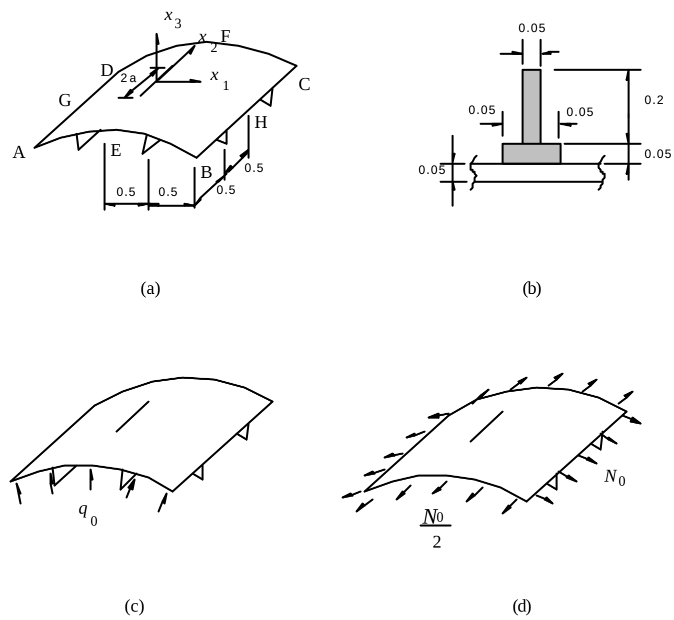

Figure 5 Geometry of shell and different types of load

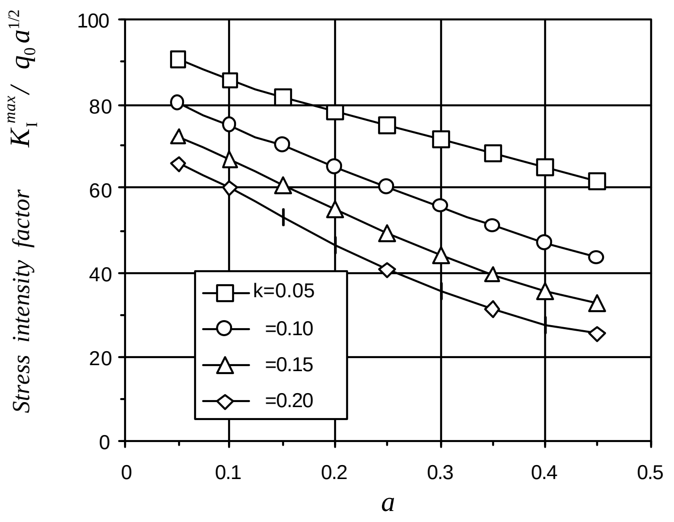

Figure 6 Normalised maximum stress intensity factors due to uniform pressure on the shell surface

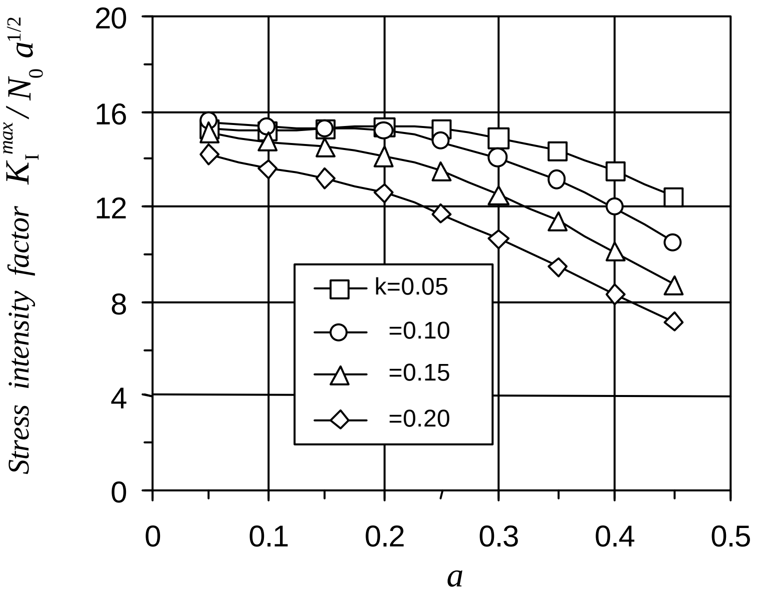

Figure 7 Normalised maximum stress intensity factors due to membrane load

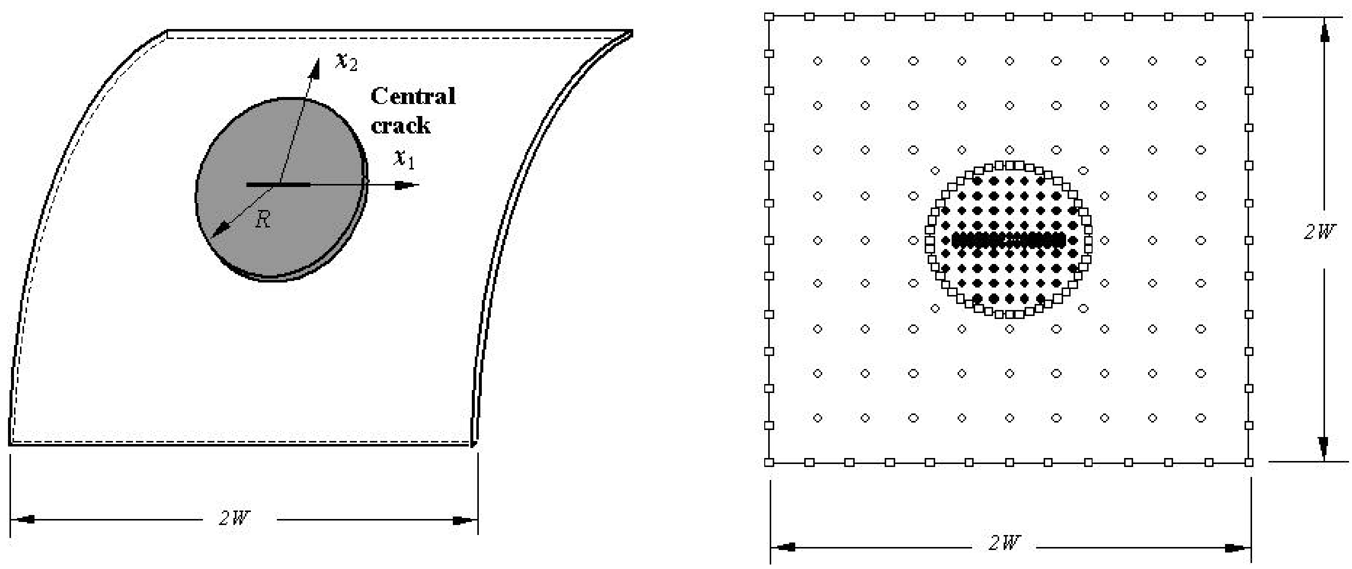

Figure 8 Cylindrical shell with a bonded circular patch

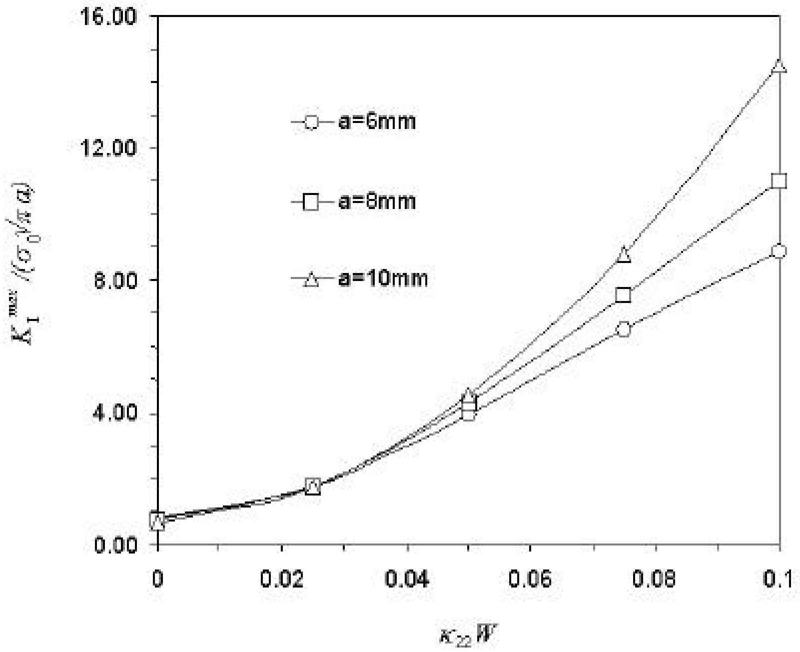

Figure 9 Normalised stress intensity factors for patched curved shell subjected to membrane loading perpendicular to the crack

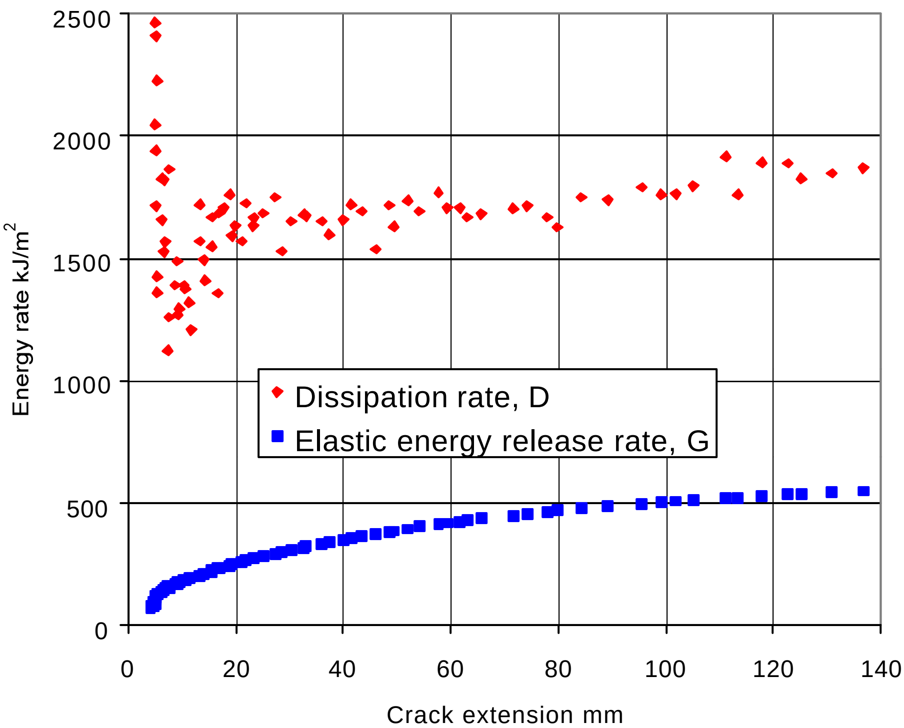

Figure 10 Comparison of G and D with crack extension on a 2 metre wide panel of 2024-T3 aluminium

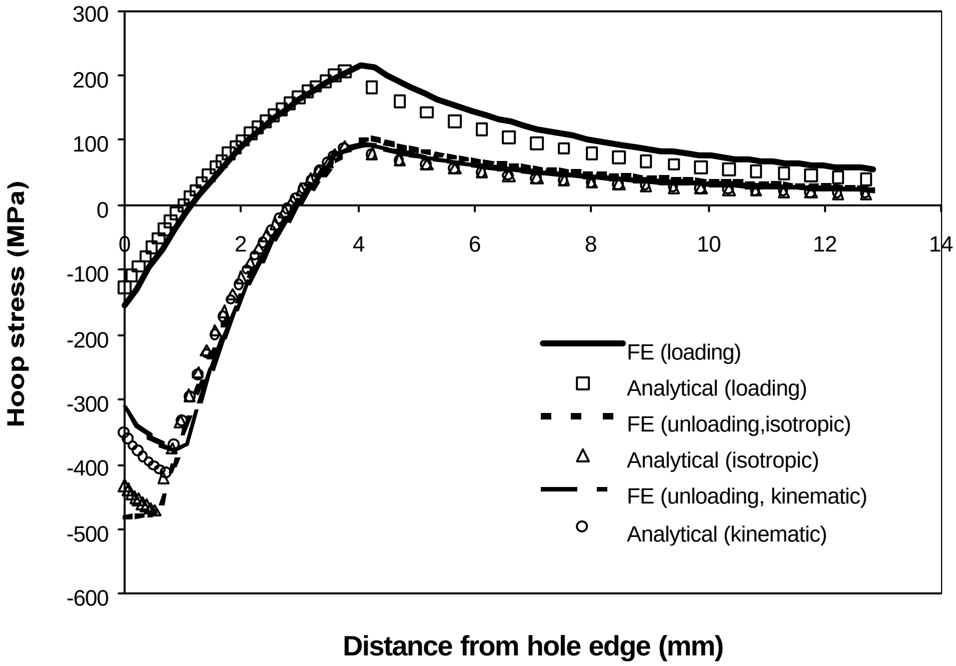

Figure 11 A comparison of closed-form solution with FE simulation

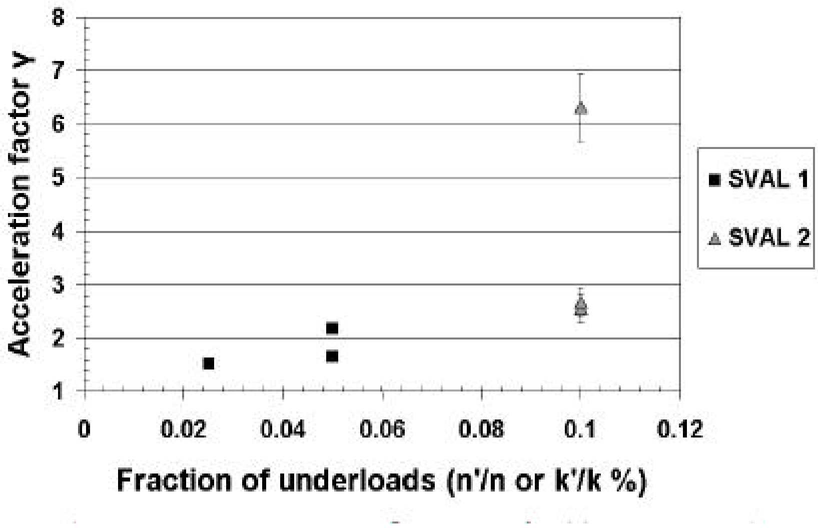

Figure 12 Acceleration factor relative to linear damage summation for Titanium 10-2-3 subjected to periodic underloads

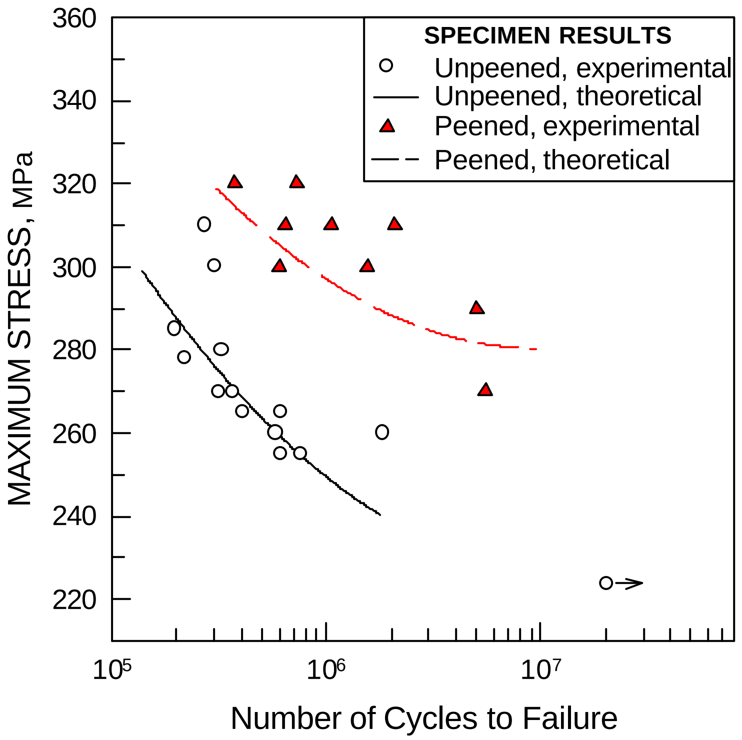

Figure 13 S-N curves, experimental data and prediction results (Four-point bending, R = 0.1)

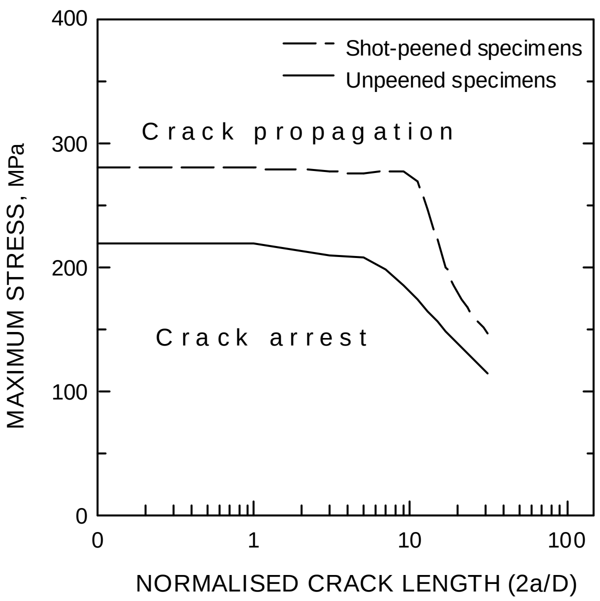

Figure 14 Fatigue damage map type diagram for unpeened and shot-peened specimens of Al 2024-T351 (four point bending, R = 0.1, D: grain diameter)

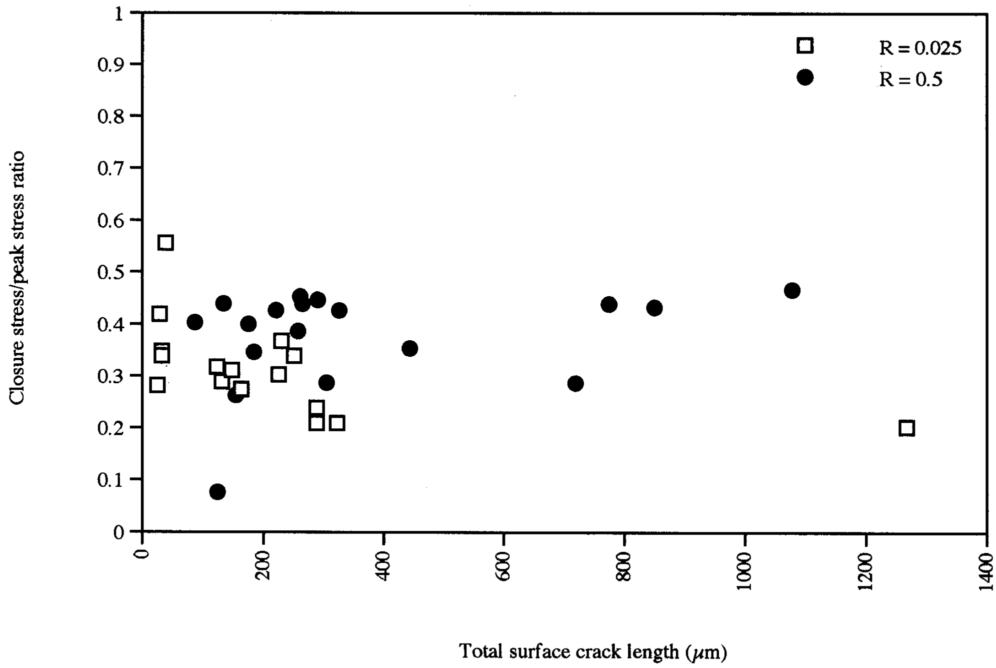

Figure 15 Closure stress/peak stress ratio versus crack length for 2024-T351 alloy tested in aqueous sodium chloride solution at R-ratios of 0.025 and 0.5

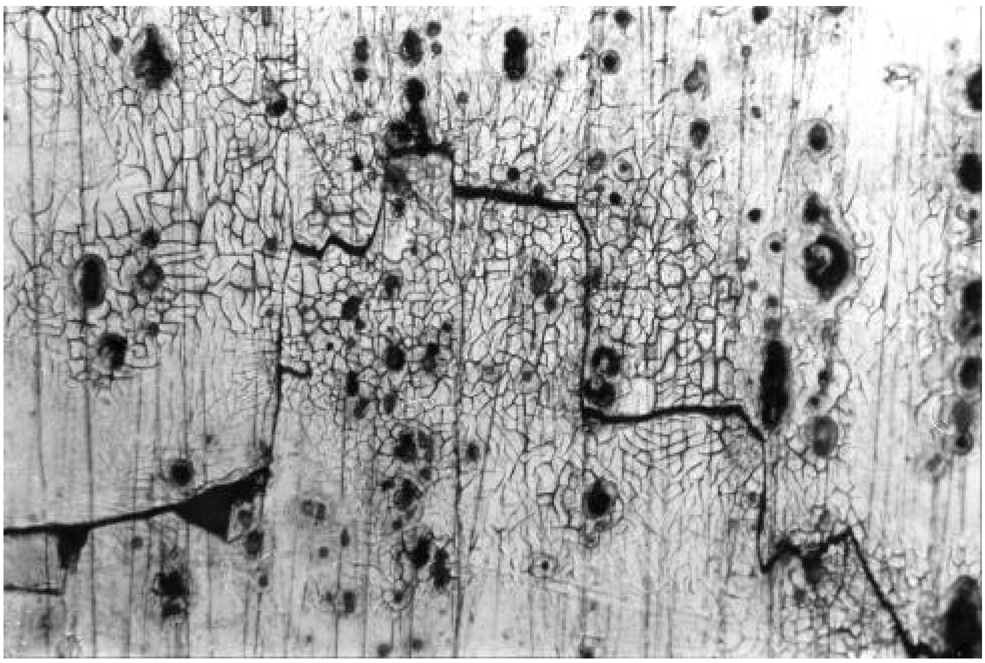

Figure 16 Surface of 2024-T351 alloy specimen fatigue tested in sodium chloride solution

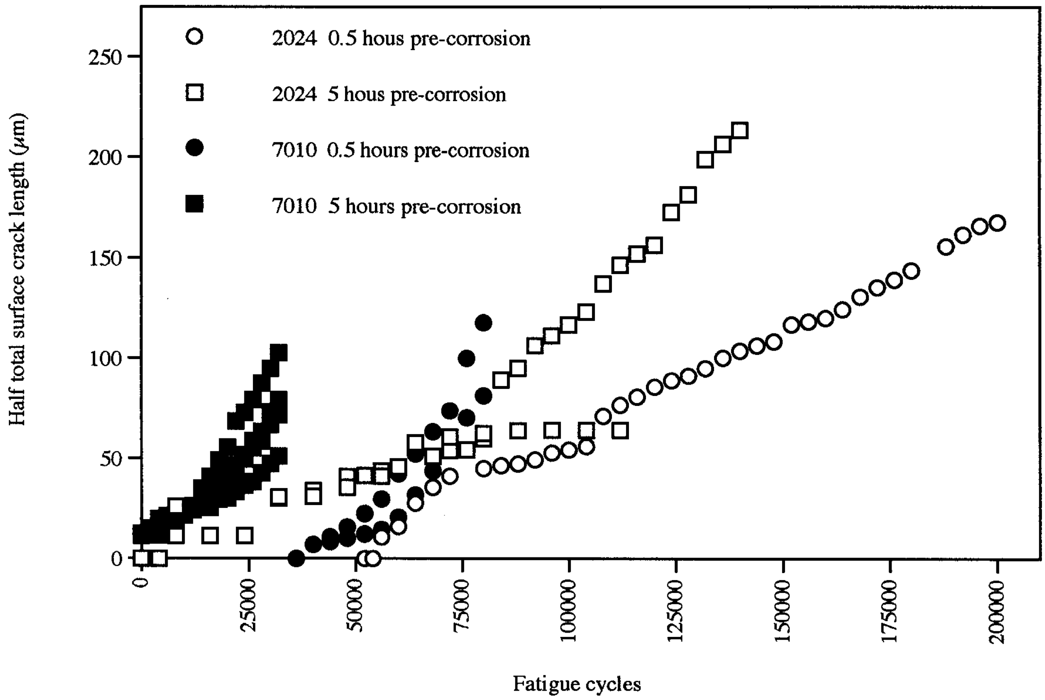

Figure 17 Growth of small cracks after pre-corrosion in aqueous sodium chloride solution

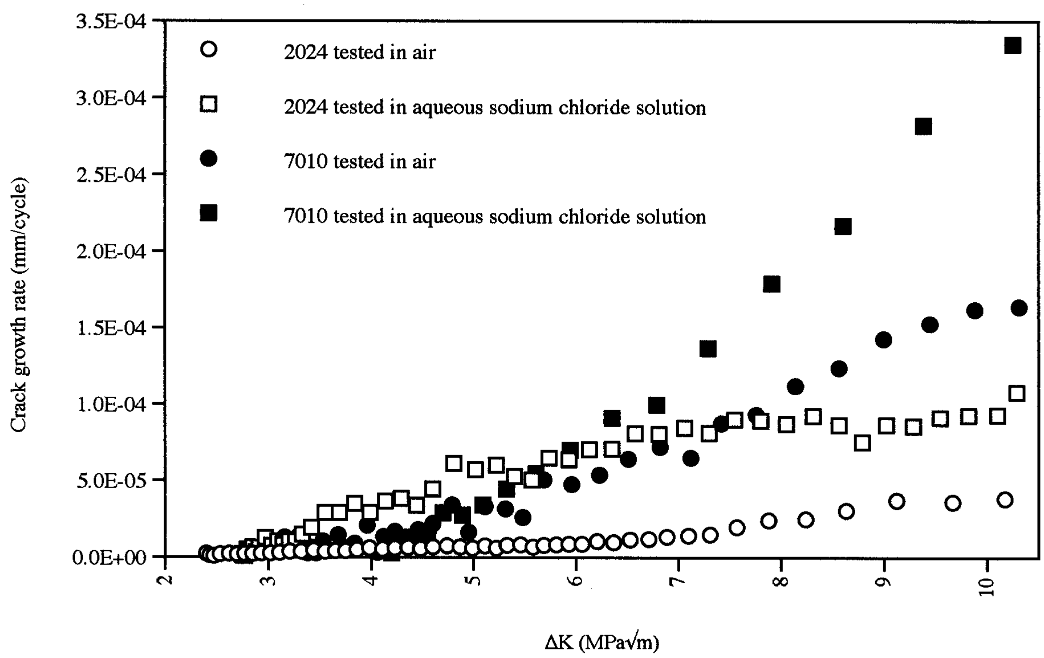

Figure 18 Crack growth rates for long surface cracks in 2024-T351 and 7010-T7651 alloys for air and aqueous sodium chloride environments

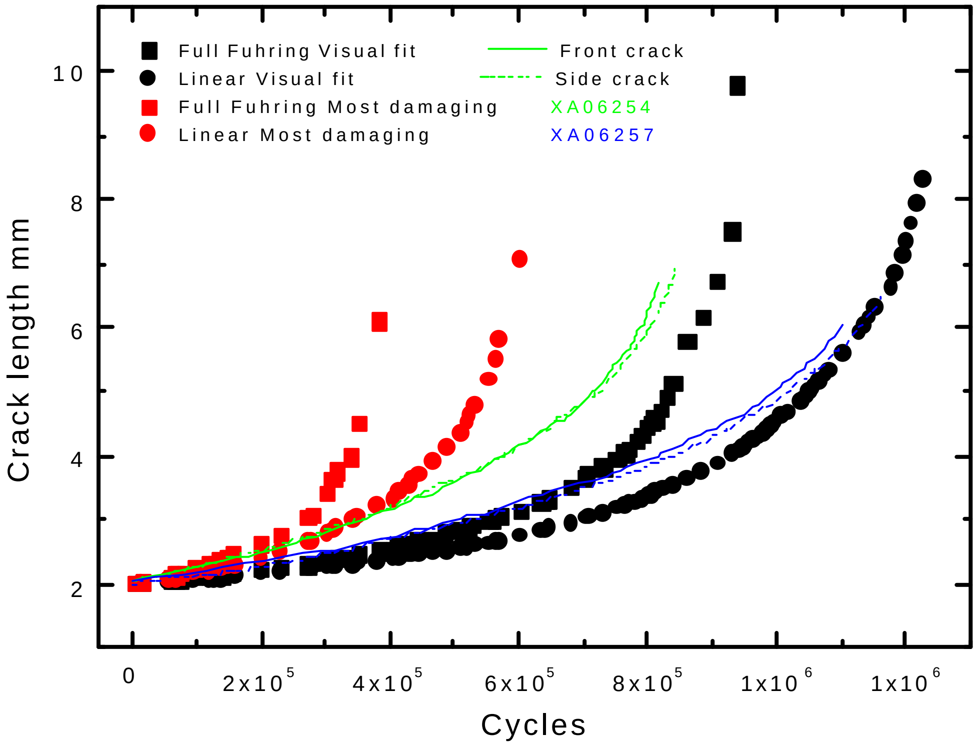

Figure 19 A comparison of predicted and experimentally determined crack growth for 2124 MMC under ROTORIX 8 load sequence, using best-fit constant amplitude data ("visual fit") and worst-case constant amplitude data ("most damaging")

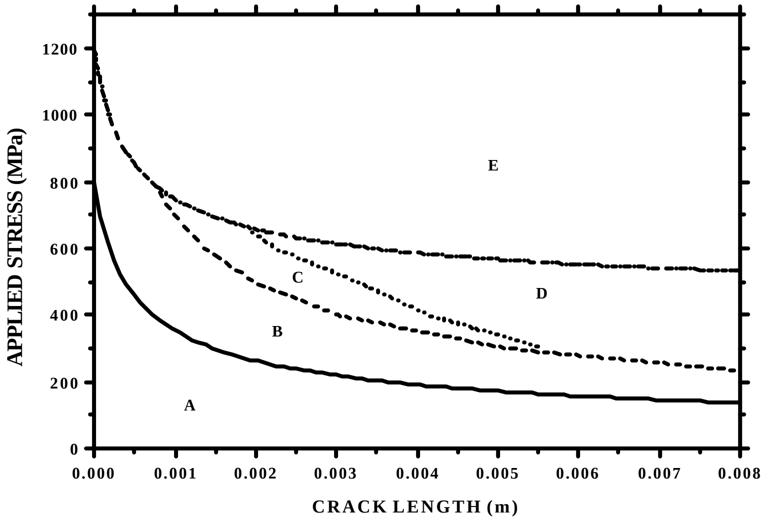

Figure 20 The FDM for an unnotched 32%, SCS-6/Ti-15-3 uMMC. (Area-A represents crack arrest, area-B is steady crack growth with maximum fibre constraint effect, area-C is steady crack growth with minimum fibre constraint effect, area-D is unsteady crack growth and area-E is crack instability)

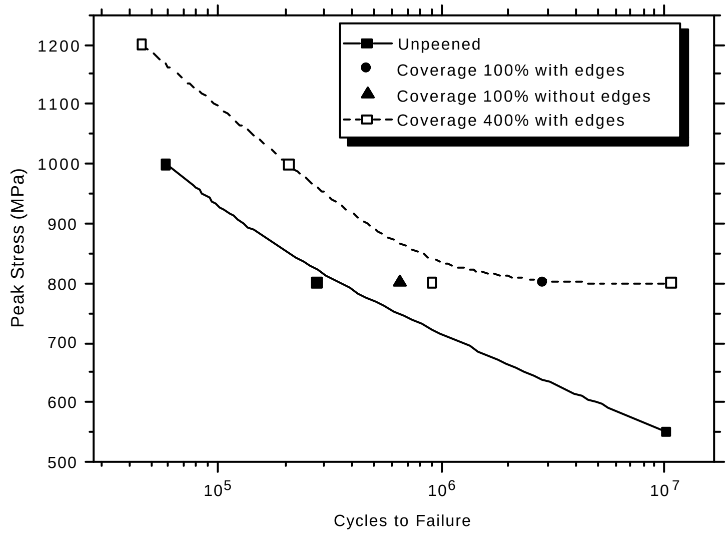

Figure 21 Comparion between S-N curves of unpeened and shot-peened experimental data. (An increase of 45% on the fatigue limit (10^7 cycles) was observed)

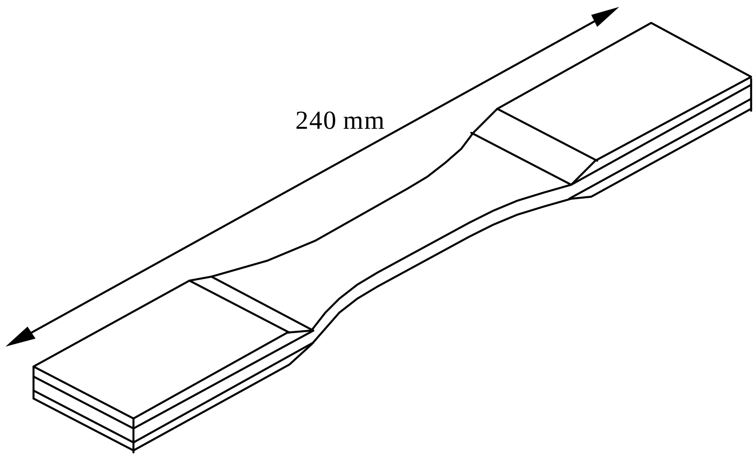

Figure 22 Specimen geometry

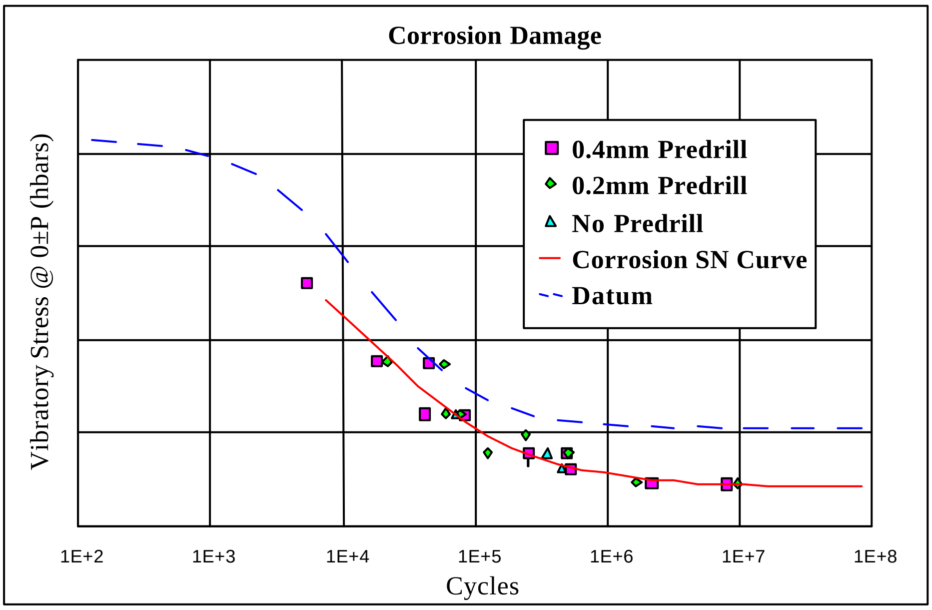

Figure 23 Comparison of corrosion data with datum S-N curve

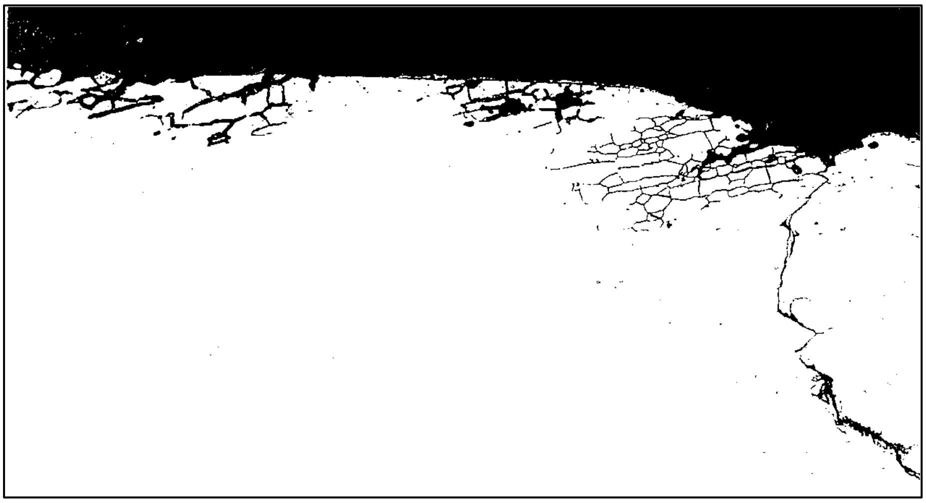

Figure 24 Corrosion damage microsection

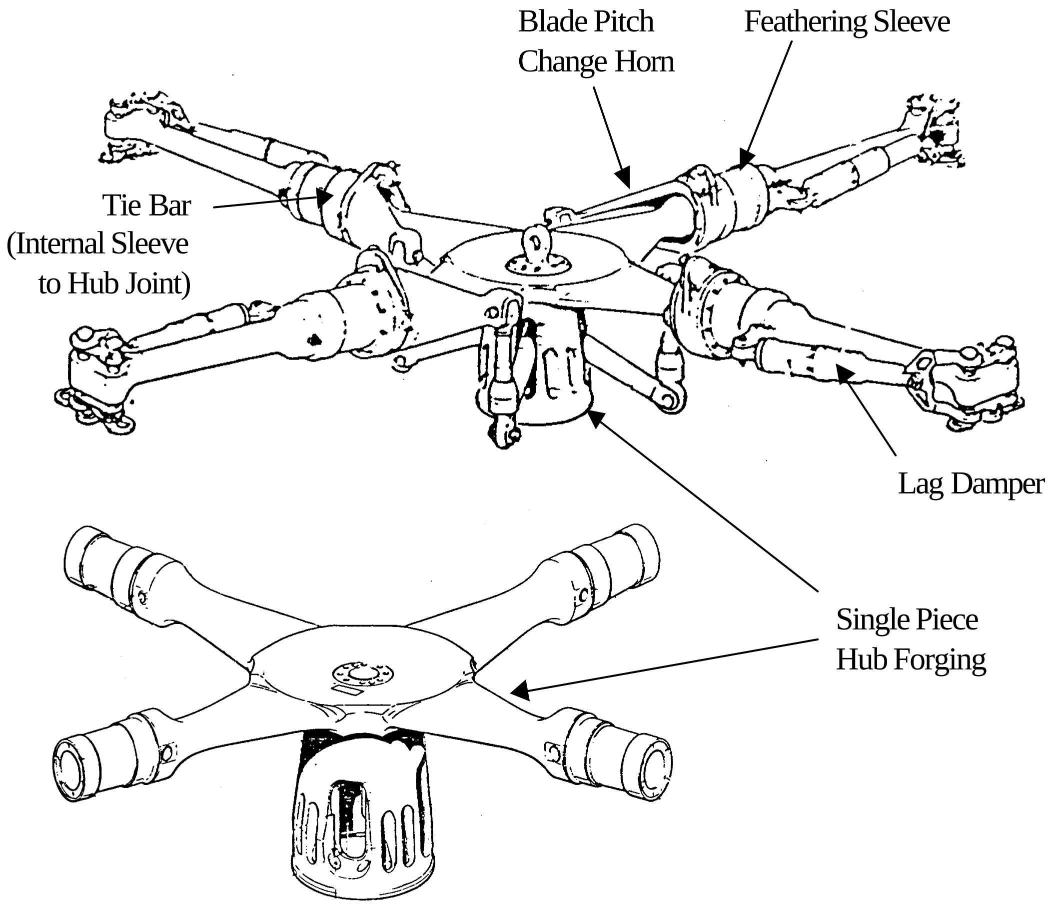

Figure 25 Existing Lynx main rotor head

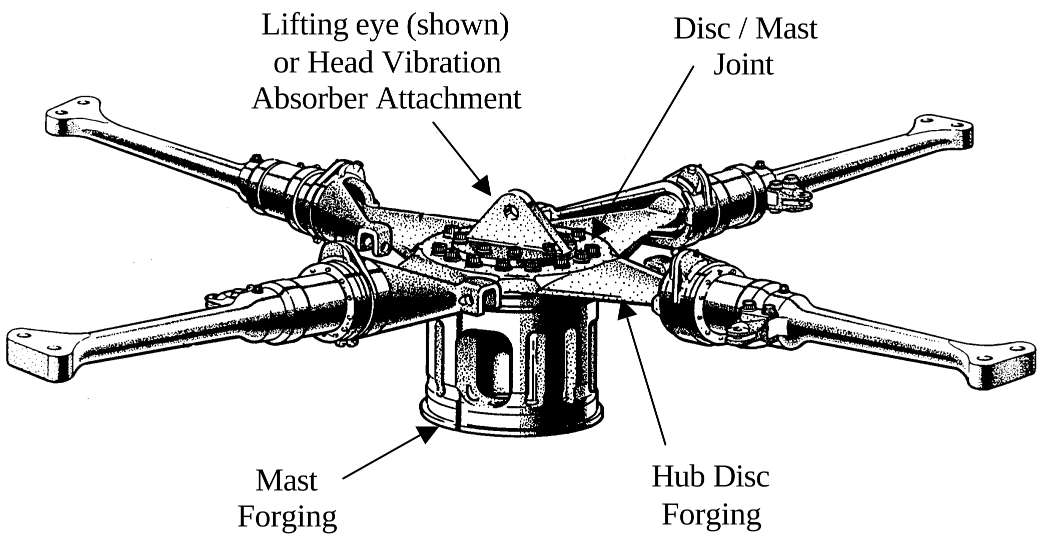

Figure 26 Lynx bolted main rotor head

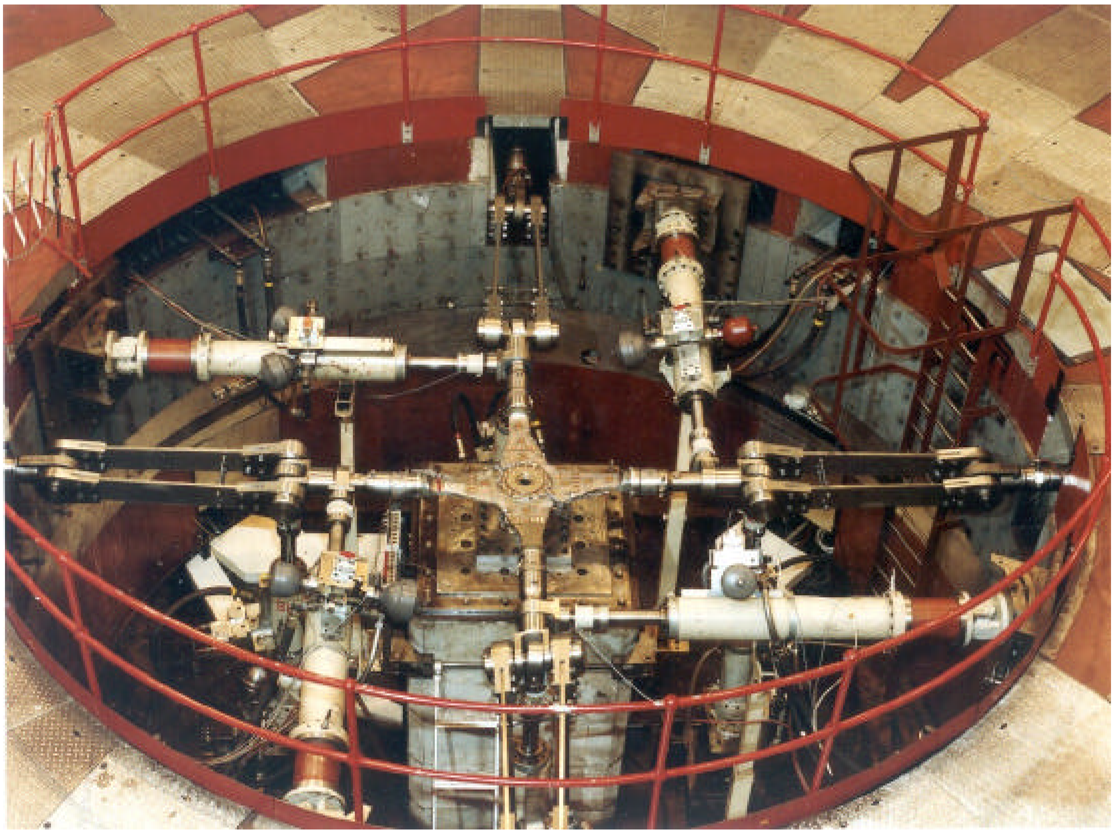

Figure 27 Westland Helicopters universal main rotor head test rig

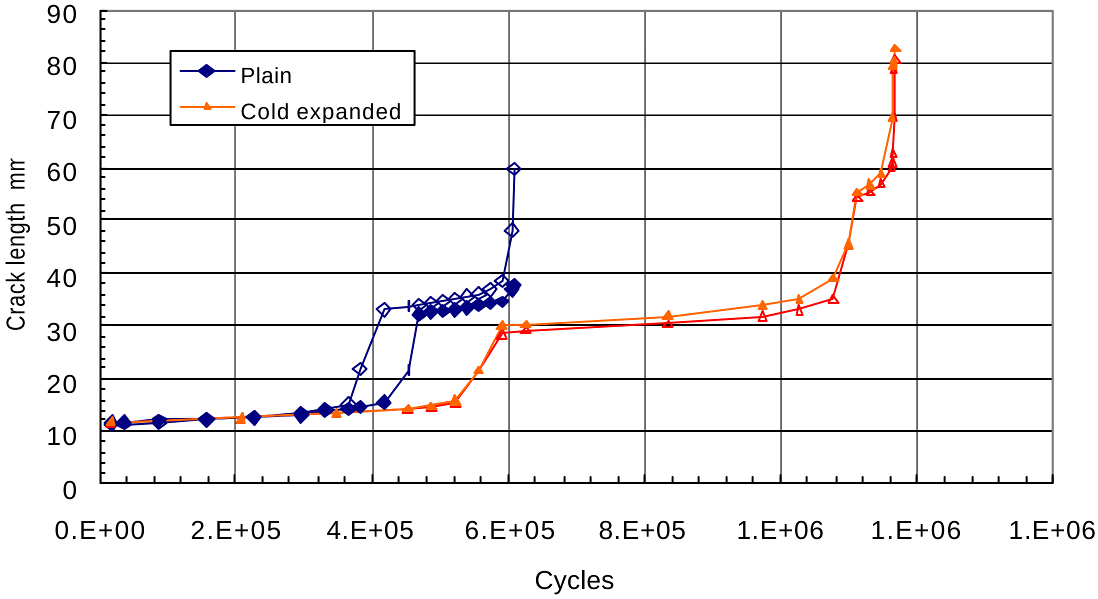

Figure 28 Effects of cold expansion on crack growth of a 1 pitch main crack (truncated FALSTAFF loading)

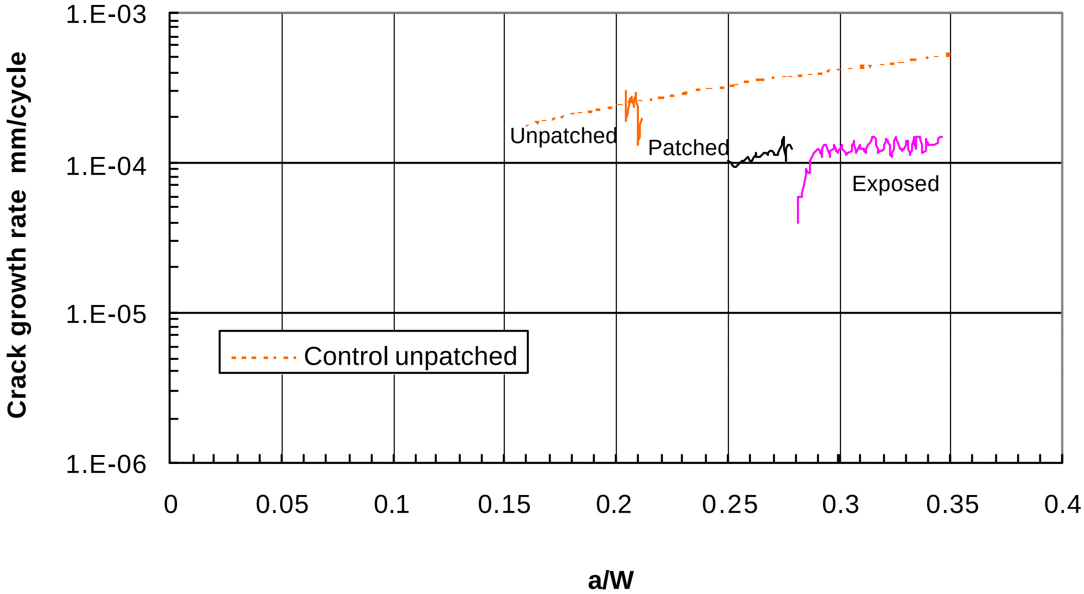

Figure 29 Effect of patching and exposure on the fatigue crack growth rate of a panel with a precured patch bonded with Redux 312/5 adhesive (jungle site, 6 years exposure)

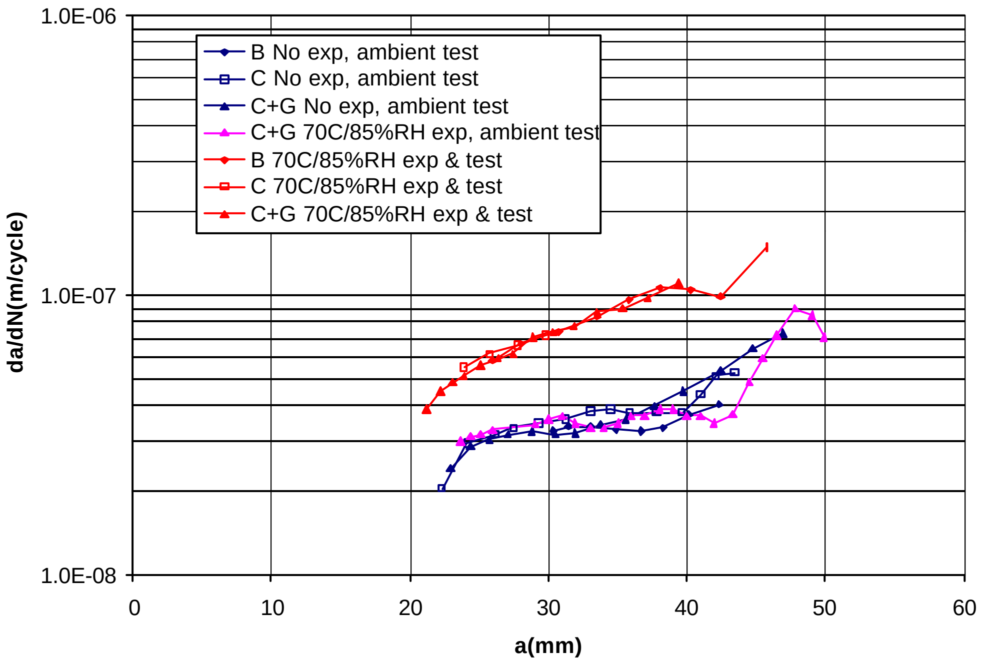

Figure 30 Effect of exposure and test environment on crack growth rate

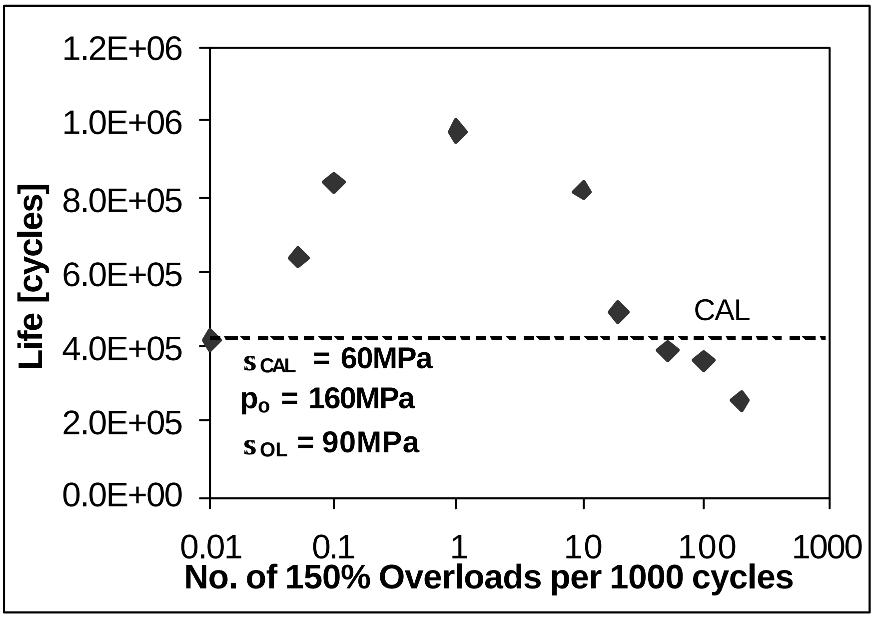

Figure 31 Effect of periodic single overloads on fretting fatigue life in aluminium alloy 2618-T6

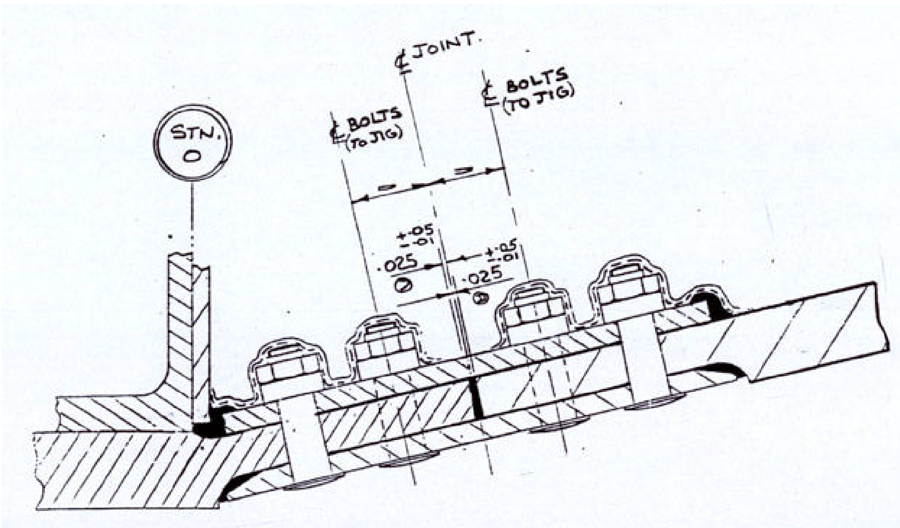

Figure 32 Lower wing skin joint at Rib O – typical section

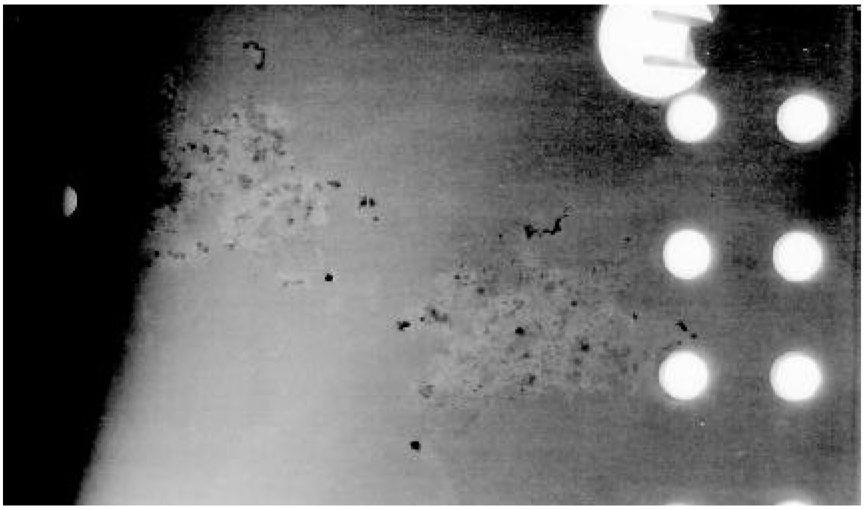

Figure 33 Pitting corrosion in a Transall 160 specimen imaged by X Radiography showing individual pits surrounded by a halo of displaced corrosion product

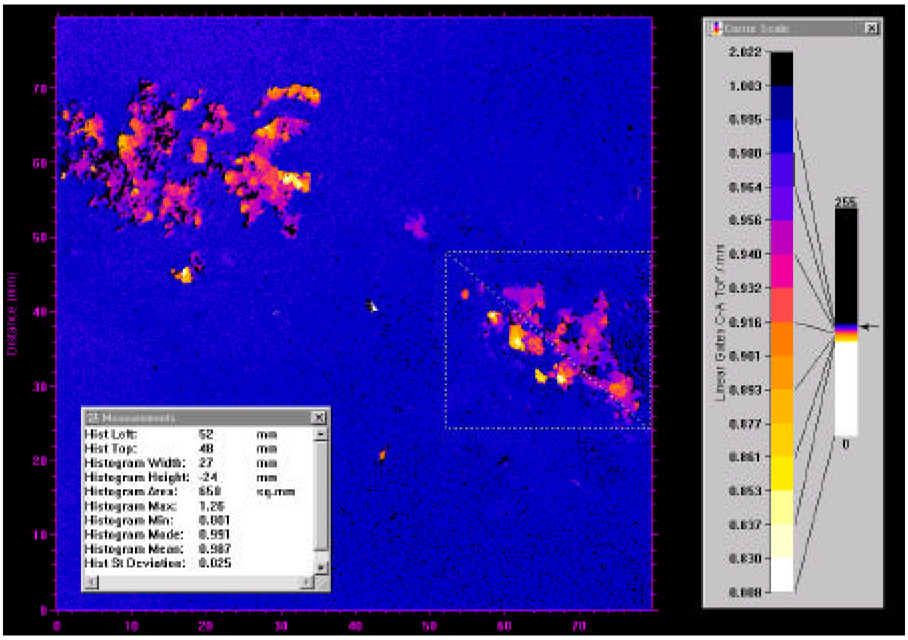

Figure 34 High frequency, 80 MHz, Ultrasonic depth (time-of-flight) scan of pitting corrosion. Individual pit depths can be measured to around 0.02mm

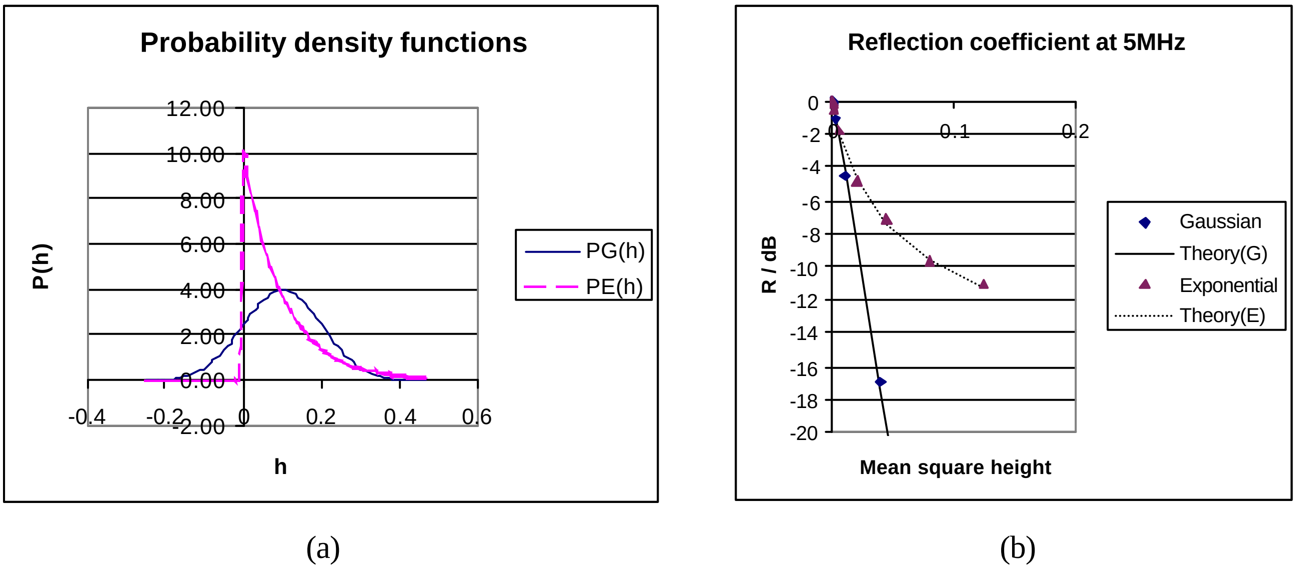

Figure 35 Comparison of (a) distribution and (b) reflectivity from exponential and Gaussian random surface models

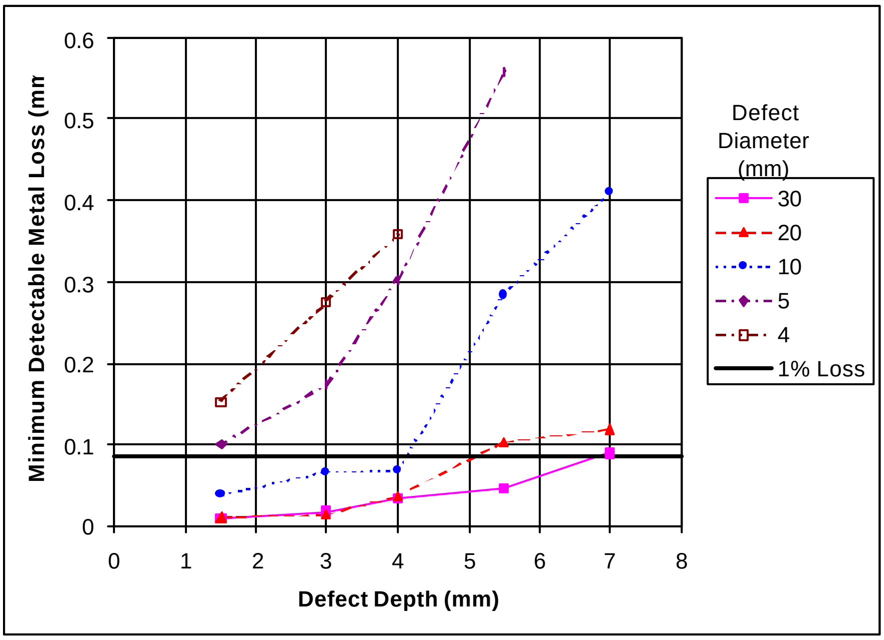

Figure 36 Minimum detectable thinning detected by transient eddy currents as function of depth of defect for multiple layers with 8.5mm total thickness. The solidline indicates a metal loss of 1%