TECHNOLOGIES FOR DAMAGE TOLERANCE IN ROTORCRAFT METALLIC STRUCTURES

P E Irving[1], R Buller[1], P Wood[2], S Rigden[2], S Jenkins[2], R Cook[3], C. Palmer[3], I Austen[4]

(1) Cranfield University, Cranfield, Bedfordshire, MK 43 OAL, UK.
(2) GKN Westland Helicopters Ltd, Yeovil, Somerset, BA20 2YB, UK
(3) Structure & Materials Centre, DERA, Farnborough, Hants., GU14 6TD, UK
(4) nCode International, Sheffield, S9 3LQ, UK.

Abstract

This paper describes a collaborative programme to develop necessary technologies for damage tolerant helicopter components, as required by current civil airworthiness regulations. In contrast to fixed wing aircraft, tools for prediction of fatigue crack growth rates under high cycle, high mean stress loading spectra are required, together with the establishment of standardised loading spectra and development of stress intensity solutions for components of complex geometry subjected to multi-axial states of stress. The performance of selected models for prediction of fatigue crack growth rates has been assessed under both constant amplitude and under helicopter load spectra. It has been found that there are substantial differences in the prediction accuracy of the models. Investigation of the reasons for these differences is proceeding, and will be used to create better models for designers of damage tolerant helicopters.

1. Introduction

Federal Aviation Authority (FAA) and Joint Airworthiness Authority (JAA) regulations now require critical components and structures in helicopters to be as far as possible of damage tolerant design [1]. Despite this, all transport helicopters flying today have been fatigue substantiated using the safe life approach. Damage tolerance concepts are present in a qualitative sense in many helicopter designs [2], such as the use of multiple load paths in Thompson couplings or the multiple stringer and skin construction of the fuselage. However, achieving damage tolerance design through slow flaw growth is not yet a design reality, in spite of many published concept studies [3-6]. The work described in this paper forms part of a collaborative research project established to create the necessary data and technologies for the damage tolerance design of helicopter components and structures.

As noted by Lincoln [3], creation of damage tolerant designs by slow flaw growth requires the following tasks to be undertaken:

- Identification of critical areas,
- Development of relevant stress spectra,
- Establishment of initial flaw criteria,
- Establishment of operational limits,
- Development of a maintenance plan,
- Development of a fracture mechanics based tracking programme.

Implementation of these items in US military aircraft via the Aircraft Structural Integrity Programme (ASIP) and in civilian fixed wing aircraft has been followed for many years, and the relevant technologies of design, calculation and inspection have been developed. The major advantages of the damage tolerance approach is perceived to be increased safety arising out of allowance for the effect of manufacturing flaws and accidental damage on life to failure. Some authorities [3] claim cost savings from the damage tolerance approach over the safe life method, arising from the fact that components are only retired when visible damage has occurred rather than when the calculated safe life has expired. This would need to be offset against any increased inspection costs.

Application of damage tolerance design to helicopters must recognise a number of unique factors which must be incorporated into a damage tolerance programme. These are the following:

- Loading spectra

  There are many more cycles in a helicopter loading spectrum than in a fixed wing equivalent, and the loading is extremely irregular. For example, the fuselage structure will experience large ground-air-ground (GAG) cycles, cycles arising from manoeuvre inertial forces at perhaps 0.5 to 0.05 Hz and high frequency cycles originating from the rotor system at blade passing frequencies between 15-20Hz. Cyclic stress ratios R (R=minimum stress/maximum stress) on the lift frame resulting from these three components may vary from -0.2 to 0.8. On other components, such as a main rotor head, R ratios vary from -3 to 0.9 at various locations.

- Component design and materials selection

  Many helicopter rotor and transmission components are of complex shape and manufactured from high strength materials such as steel and titanium alloys, and are subjected to multiaxial loading. Mostly components have only one load path and hence little redundancy, which is usually the case in transmission and rotor structures.

These two aspects generate requirements for:

- Stress intensity factor solutions for complex geometry and multiaxial stressing,
- Techniques to predict fatigue crack propagation trajectories under these conditions,
- Characterisation of fatigue crack growth data in the near threshold region,
- Models to predict fatigue crack growth under highly irregular multiaxial loading,
- Techniques to reduce the number of cycles in the spectrum for damage tolerance testing of components and calculation of component lives.

In the collaborative research project, the following items are being addressed for two specific components. Firstly, the aluminium-lithium (8090 T852) main load path structure of the Agusta/Westland EH101 helicopter and secondly the titanium (Ti-10V-2Fe-3Al) bolted main rotorhead from the Westland Lynx helicopter. These are two very different applications in both material and loading. For each of these components the project seeks to:

- Develop standardised load spectra,
- Characterise fatigue crack growth behaviour for each material at low growth rates, at high R ratios and under variable amplitude loading,
- Develop models for prediction of fatigue crack growth rates and crack propagation life and to verify their capabilities against test data,
- Develop stress intensity solutions and predictions of fatigue crack propagation trajectories,
- Implement models in software for use by design engineers.

The research programme described here is currently in progress, where a helicopter manufacturer, software developers and fatigue research workers are combining to develop the above technologies for the design of damage tolerant helicopters.

2. Standardised Load Spectra

2.1 Development of Standardised Component Loading Spectra

The assessment of damage tolerance design models requires the development of standardised load spectra for different helicopter components which contain representative selections of flight manoeuvres and mission profiles. These load spectra will be used both for analytical predictions of fatigue crack growth and realistic component testing for damage tolerance capability.

To best represent the loading spectrum of a component, by detailing the complicated mission and manoeuvre profiles and retaining definition of the small vibratory loads, the project has chosen to adapt the approach developed for the standardised loading sequences Helix and Felix [7]. Most aspects of the Helix/Felix method are retained except for the original strain data from which the loading spectra are defined.

Strain gauge data for locations on each of the bolted main rotorhead (Figure 1) and main load path structure (Figure 2) were provided by GKN Westland Helicopters Ltd. from flight tests of the helicopters for all manoeuvres defined in Ref. [7]. The twenty five manoeuvre definitions for Helix and twenty two manoeuvre definitions for Felix were used for the main lift frame and rotorhead loading spectra respectively. The strain gauge data were Rainflow cycle counted and non-dimensionalised between 0 and 100 at interval levels of 4. In accordance with the Helix/Felix method the mean loads, with alternating loads about this mean, were determined for each manoeuvre. Figure 3 gives an example of a three hour training flight sequence assembled using the Helix procedure.

2.2 Usage of Standardised Component Loading Spectra

Particular attention will be given to two aspects of damage tolerant design and testing where helicopter loading spectra can cause particular difficulties. These are:

- Small cycle omission levels in loading spectra for crack growth based component testing.

  Omission levels are important in damage tolerant testing because small load cycles, which would otherwise be omitted as non-damaging in testing under initiation based safe life, do in fact cause fatigue crack growth for cracks of realistic sizes. During component testing it is important to establish a small load cycle omission or gate level, without changing the damaging character of the spectrum, which allows realistic testing to be conducted within feasible test times. Current work has investigated the influence of increasing omission load levels on crack growth life using conventional aluminium alloy compact tension specimens subjected to a helicopter lift frame spectrum. Referring to Figure 4, the results indicate that omission load levels up to 21% of maximum or peak load can be omitted without any significant change in flights to failure. Above this level the flights to failure increase indicating that the cycles removed do cause fatigue damage. Omission up to this level can reduce test duration by up to five times.

- Load interaction effects under helicopter load spectra

  Load interaction effects in a damage tolerance regime are found to be similarly important because of the differences between fixed wing and helicopter loading spectra. Helicopter loading spectra contain significant and numerous tensile load excursions which require new models for adequate prediction of crack growth rates.

3. Characterisation of Fatigue Crack Growth Properties

The nature of the helicopter load environment necessitates investigation of fatigue crack growth behaviour at near-threshold stress intensity factor values and at high R ratios. Little or no variable amplitude loading fatigue crack growth information is currently available in open literature for the materials under investigation, although there is some constant amplitude loading data. The project group has defined a comprehensive test program to characterise their fatigue crack growth properties. The tests, which range from constant amplitude loading to complex helicopter spectra loading over a variety of specimens of increasing complexity, are described below:

3.1 Constant Amplitude Loading Data

3.1.1 Standard Specimens

Constant amplitude loading (CAL) fatigue crack growth data are required to build a materials database for the prediction models under investigation in this project. Most fatigue crack growth prediction models use CAL data in either "lookup-table" form or as curve fits to experimental data. A series of CAL tests on standard compact tension (CT) specimens (thickness t=17.5mm, width W=70.0mm) at R ratios of R=0.1, 0.4, 0.7 and 0.9 have been conducted to gather fatigue crack growth data. CAL tests are also planned using surface crack tension (SCT) specimens, the results of which will be predicted using selected models under investigation using CAL data from the CT specimen tests.

3.1.2 Complex Representative Structural Elements

To provide CAL fatigue crack growth data for model verification from complex geometries, two representative structural element (SE) test specimens were designed to approximate the EH101 main lift frame and the Lynx bolted main rotorhead.

The main lift frame SE specimen shown in Figure 5 represents the I-beam type configuration of the lift frame with a lightening hole in the centre web. The SE specimen was machined from an aluminium lithium (8090-T852) forging. Fatigue cracks were initiated from a Electro-Discharge Machined (EDM) notches. The tests were conducted under uni-axial loading at an R ratio of R=0.7.

The bolted main rotorhead SE test specimen shown in Figure 6 represents the lower corner of the control window of the mast. This is a high stress region as determined by finite element analysis of the mast. The Ti-10V-2Fe-3Al hand forged SE test specimen was designed to represent the complex stress state that occurs in the mast due to torsional, normal and bending stresses imposed during flight. NASTRAN finite element models of the SE test specimens were constructed to determine a stress intensity factor solution for a fatigue crack as described later.

3.2 Simple Variable Amplitude Loading

To investigate the effect of load transients on fatigue crack growth rates and to provide simple reproducible data against which the ability of different fatigue crack growth prediction models will be compared, a series of simple variable amplitude loading (SVAL) tests were defined. The test loading parameters were chosen such that they closely represent the loading conditions found in the main lift frame standardised loading spectrum. The spectrum contains typically high R ratios and have numerous load excursions which are represented by tensile overloads in the SVAL tests. Three types of tests were selected for aluminium lithium (8090-T852) CT specimens:

1. A single tensile overload of 40% is applied at fixed crack lengths during baseline CAL of R=0.7.
2. Interacting single tensile overloads spaced 3000 cycles apart. Overloads of 40% are applied at fixed crack lengths during a baseline CAL of R=0.7.
3. Single tensile overload of 40% followed by equivalent tensile underload applied at fixed crack lengths during a baseline CAL of R=0.7.

Results to date indicate that crack growth rate transients resulting from applied interacting overloads have a decreasing minimum crack growth rate da/dN with increasing CAL stress intensity factor range (ΔK) as shown in Figure 7. Alternatively the delay period following an overload will decrease as the overload stress intensity factor (Kol) increases proportionally with increasing baseline ΔK. Interestingly Figure 7 shows that crack arrest will occur (da/dN < $1x10^{-11}$ m/cycle) when overloads are applied below $\Delta K = 2.3$ MPam$^{1/2}$ (for these load conditions) which is where most crack growth under helicopter loading will occur.

Similarly a series of tests has been defined for the Ti-10V-2Fe-3Al CT specimens based on loading conditions found in the rotorhead standard loading spectrum.

3.3 Complex Variable Amplitude Loading

The ultimate aim of the project is to develop analytical models which will successfully predict fatigue crack growth under the complex variable amplitude loading (CVAL) experienced by helicopter components. Therefore, tests have been defined where the standardised loading spectra mentioned earlier will be used to subject CT, SCT and SE test specimens to helicopter load spectra. In particular the tests will investigate aspects, unique to helicopters, which effect fatigue crack growth under CVAL such as numerous tensile load excursions, high R ratios and cycle omission gating levels.

4. Modelling of Fatigue Crack Growth Rates

Several fatigue crack growth models are being investigated in this project for their capabilities to successfully predict growth in laboratory specimens under helicopter loading conditions. The models under investigation and their main features are described below. These range from semi-empirical models describing crack tip plastic yield zone sizes such as Willenborg [8] and Wheeler [9] to the analytical strip yield models describing plasticity induced crack closure such as the Newman model [10], implemented as the FASTRAN model.

4.1 Yield Zone Models

Willenborg

Willenborg, Engle and Woods [8] developed a model based on plastic zone size considerations but which ignored any compressive load excursions. The plastic zone size is calculated for each maximum applied load and if this is within the plastic zone created by a previously applied overload then a correction factor is used. A 'shut-off overload ratio' can be defined as the ratio of the overload to the load at which crack arrest occurs. The shut-off ratio could therefore be defined but it is more commonly used as a fitting parameter to give the best agreement between predictions and experimental data.

Wheeler

Wheeler [9] developed a model also based on plastic zone size considerations. The plastic zone is calculated for each maximum applied load and a retardation factor is calculated if the extent of the instantaneous plastic zone is within the plastic zone created by a previous applied load. The ratio of the instantaneous plastic zone size to the distance between the extent of the instantaneous plastic zone and the extent of the previous plastic zone is calculated. A retardation factor is equal to this ratio raised to the power m, and is used as a fitting parameter to give the best agreement between predictions and experimental data

LOSEQ

Fuhring developed a crack growth model called LOSEQ [11] which accounts for load interactions by considering the plastic zone sizes of all previous load excursions. The Dugdale model is used to represent the plastic zone and a load memory routine is used to retain all previous relevant loading. Acceleration and retardation parameters are calculated, based on plastic zone sizes and a crack closure correction formula, and applied to the calculated stress intensity factor range. There is one free parameter in the model termed the acceleration coefficient which is claimed to be a material constant but can more generally be considered as a fitting parameter to experimental data.

Kraken

The Kraken [12] model assumes that a linear log-log relationship defined by the Paris Law is the true representation of crack growth and corrections are applied to the near-threshold and near-unstable fracture regimes to maintain the linear relationship. Corrections in the near-threshold regime are based on crack closure assumptions and in the near-unstable fracture regime on an empirical fracture toughness relationship. There are no free parameters in this model though adjustments to the fitting of threshold and stress ratio data could be considered as a method of optimising the predictions. An additional 'residual K' is used to account for history effects and is based on the Willenborg model, but has been extended by Austen [12] to account for delayed retardation and crack growth rate overshoot.

4.2 Strip Yield Models

FASTRAN

The model by Newman [10] considers plasticity induced crack closure and calculates the load at which the crack becomes fully open to give an effective stress range for use in the crack tip stress intensity factor calculation. The model enables the user to define whether plane stress or plane strain conditions exist and at what point the transition between the two occurs and at what point it is complete. This is accomplished using a constraint factor α which can be varied between 1 and 3. This can also be considered as a fitting parameter to experimental data.

ESACRACK

ESACRACK [13] is constantly evolving program which currently has the NASA/FLAGRO software to perform the fracture mechanics analysis. The program is essentially a linear summation method (with no load interaction) but does have an option to activate the NLR developed CORPUS model [14]. This is an empirical crack closure model which utilises the closure relationships developed by Newman [15] and de Koning [17].

STRIPY

STRIPY is a development of ESACRACK which uses the strip yield model developed at the NLR by de Koning and ten Hoeve [16]. The strip yield model is based on the Dugdale and Barenbladt approaches and is essentially similar to Newman's FASTRAN model. The user defined α and β parameters in Newman's model however have been analysed at NLR and default values selected for typical situations. There are options on the method of describing the constant amplitude crack growth rate input data including the full Forman equation, the variables for which could be considered as fitting parameters for model optimisation.

4.3 Effect of Input Data on Model Accuracy

The calculation and representation of raw CAL fatigue data for each model is also being assessed. Input data for the models may be in tabular form or as constants for a fatigue crack growth law, so it is important to establish whether or not these variations have an affect on the accuracy of prediction results. Different methods of processing raw data have been investigated and it was found that the most accurate predictions can be made if original da/dN versus ΔK data is input in tabular form for different R ratios and subsequent values of da/dN be calculated using interpolation or extrapolation of this data. Additionally the parameters which the models are most sensitive to in terms of CAL crack growth back-prediction are being determined so that precautions are taken in the future when defining them.

5. Development of Stress Intensity Solutions

The objective of this task is to identify an efficient process of generating Stress Intensity Factors (SIF) which can be employed, for crack growth prediction, by the Stress Engineer when conducting a damage tolerance analysis.

Generally SIF's are presented in the form of geometry factors or compliance functions, as given in the following relationship:

$$K = Y\sigma\sqrt{(\pi a)} \qquad \text{... Eqn. 1}$$

where: K = Stress Intensity
Y = Compliance function
σ = Nominal stress level
a = Crack length

For many simple components this data exists, and can easily be incorporated into the crack growth model. Where this data is unavailable, numerical methods such as Finite Element and Boundary Element techniques can be employed to accomplish this task.

The project has accommodated a range of test components to investigate crack growth behaviour under simple and complex load sequences. To represent the airframe main load path structure and bolted main rotorhead mast, SE test specimens have been designed, as shown in Figures 5 and 6. It will be appreciated that there are no general SIF solutions for the SE test specimens and this information must be generated using numerical methods. Currently GKN WHL employ NASTRAN as the standard FE solver which provides two solution techniques for SIF's, notably:

- Special crack tip elements
- Strain energy release rate, G

The first method requires a fine mesh in the vicinity of the crack tip and for each crack increment the FE mesh has to be re-defined, making this solution time consuming. However, it is applicable to mixed-mode situations and simulations of the crack growth direction can be obtained. The strain energy method is very efficient, provided that the crack path is known. SIF's for mixed-mode problems cannot be separated out easily, and this method provides a single average SIF value over the crack front.

Currently the project has only utilised the Strain Energy release rate approach to determine SIF's for the SE test specimens. Constructing the FE models with all plate or all solid elements produces very similar results, as shown in Figure 8, which depicts the SIF solution for the airframe SE. The SIF solution can also be derived from test data, and this is plotted in Figure 2 for comparison. It can be seen that the SIF solution is complex, and that preliminary FE results compare well with measured data.

In addition, the project is reviewing the Dual Boundary Element Method, which offers automatic crack extension and remeshing of the surface and internal crack faces. This approach also predicts the crack path.

6. Conclusions

Two standardised load spectra for different helicopter components have been developed, using an established method. These load spectra will be used both for analytical predictions of fatigue crack growth and realistic component testing.

A comprehensive test program has been established to characterise the fatigue crack growth properties of two helicopter materials for the verification of crack growth prediction models and for the study of crack growth under typical helicopter load conditions.

Tests results indicate that omission load levels up to 21% of peak load can be omitted from a helicopter loading spectrum without any significant changes in flights to failure. Omission up to this level can reduce test duration by up to five times. Further work will determine a safe omission load level for any helicopter load spectra as omission of non-damaging cycles saves considerable time during component testing.

Selected fatigue crack growth models have been been investigated for their capabilities to successfully predict growth in laboratory specimens under helicopter loading conditions. It was found that the most accurate predictions can be made if original da/dN versus ΔK data is input in tabular form for different R ratios and subsequent values of da/dN be calculated using interpolation or extrapolation of this data.

Two representative structural element test specimens have been designed to approximate the EH101 main lift frame and the Lynx bolted main rotorhead structures. NASTRAN finite element models have been constructed to provide stress intensity factor solutions for fatigue cracks using a strain energy release rate method.

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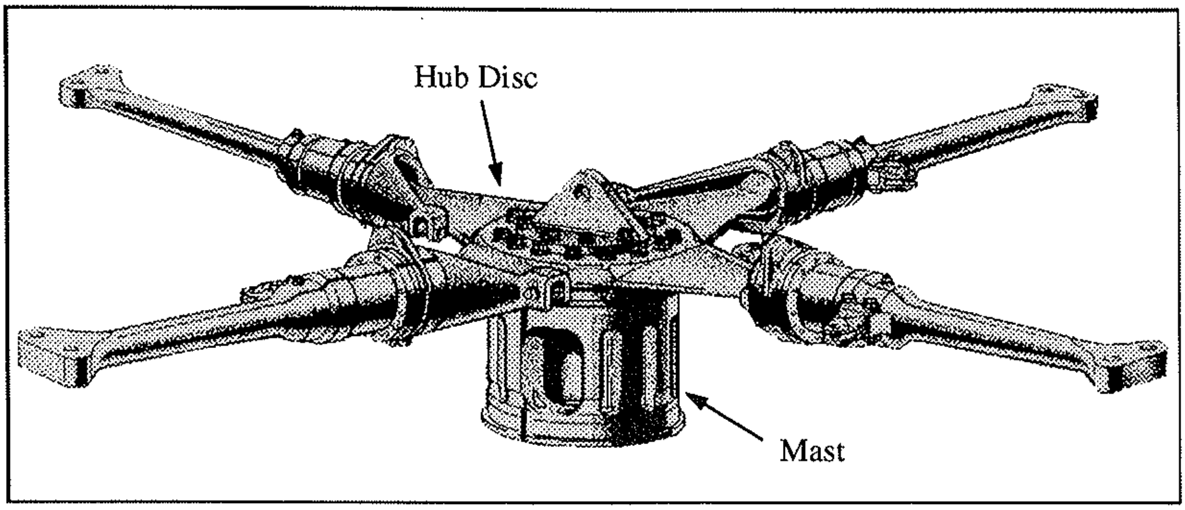

Figure 1 Bolted main rotorhead of the Westland Lynx helicopter.

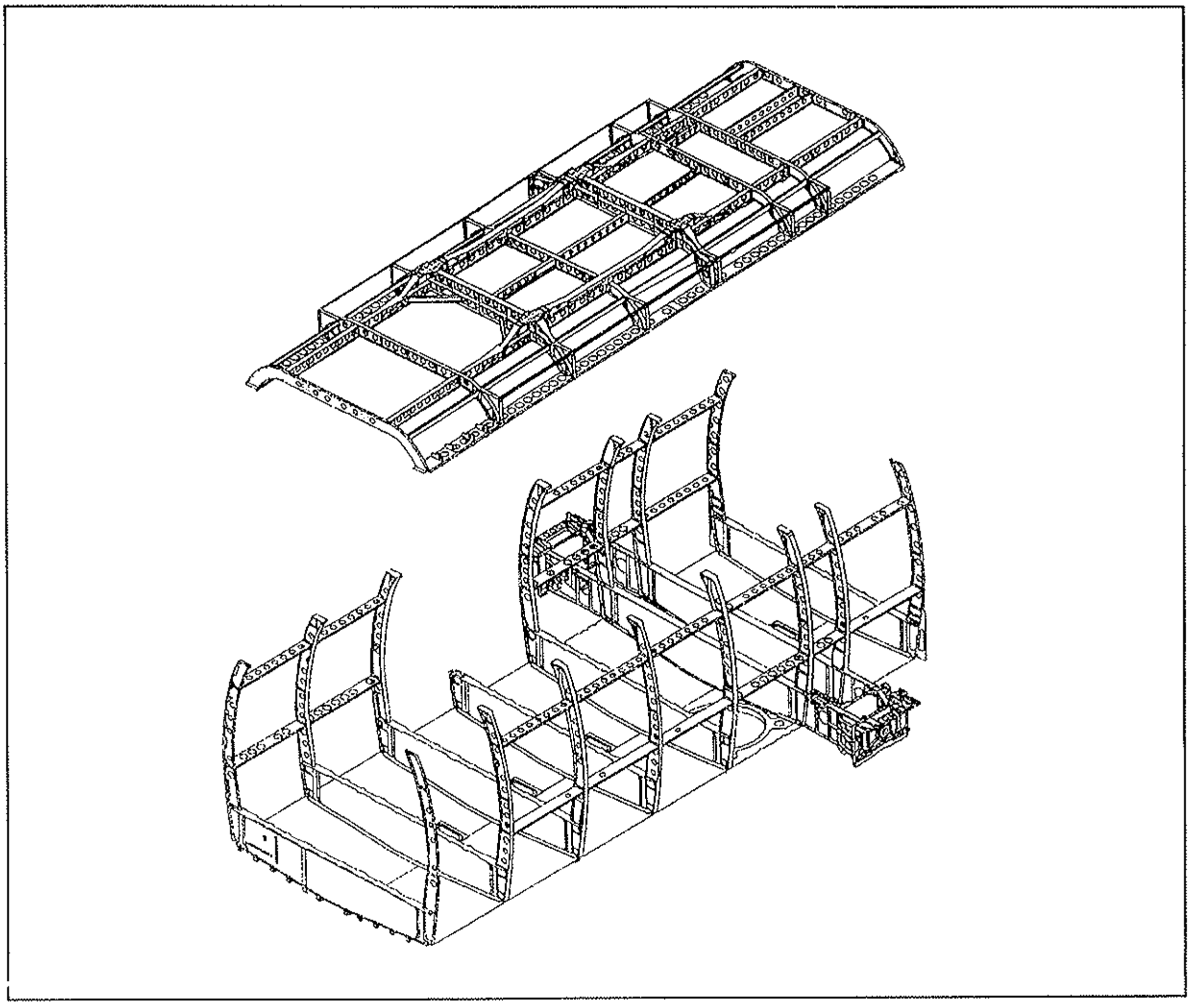

Figure 2 Main lift frame of the Agusta/Westland EH101 large transport helicopter.

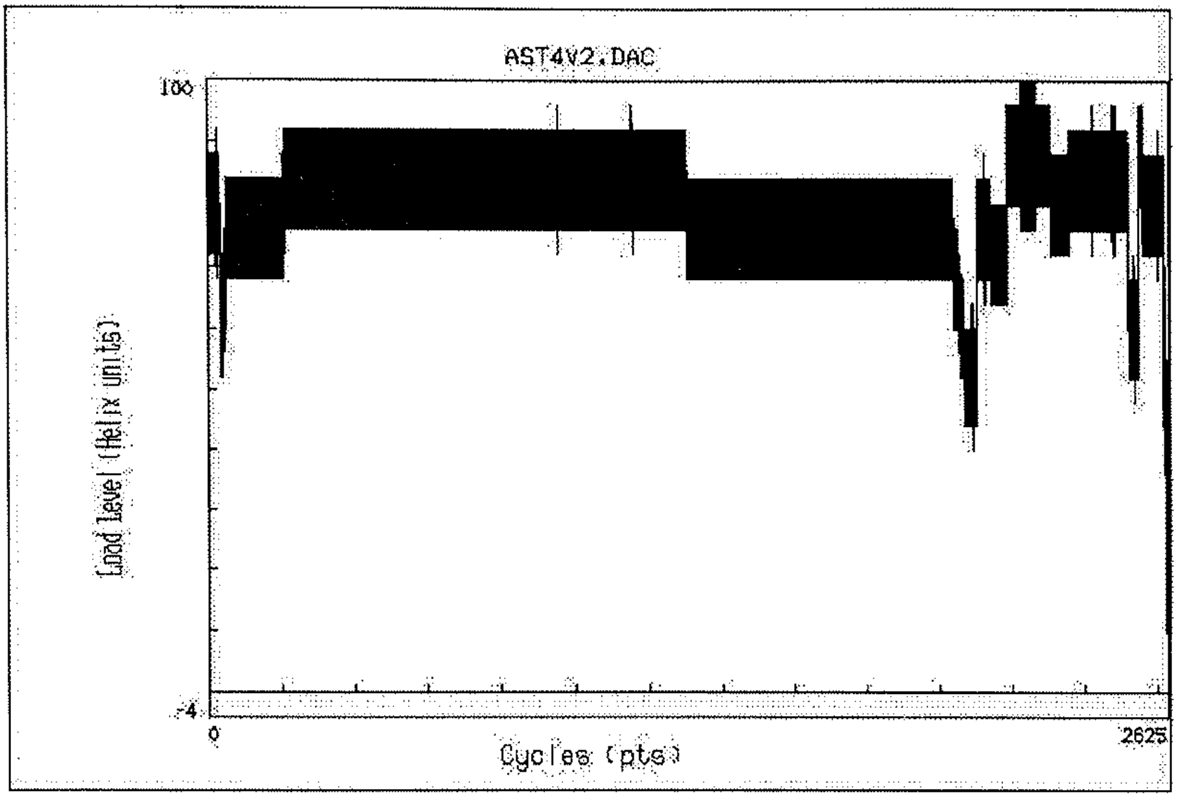

Figure 3 Three hour training flight loading spectrum assembled using the Helix procedure.

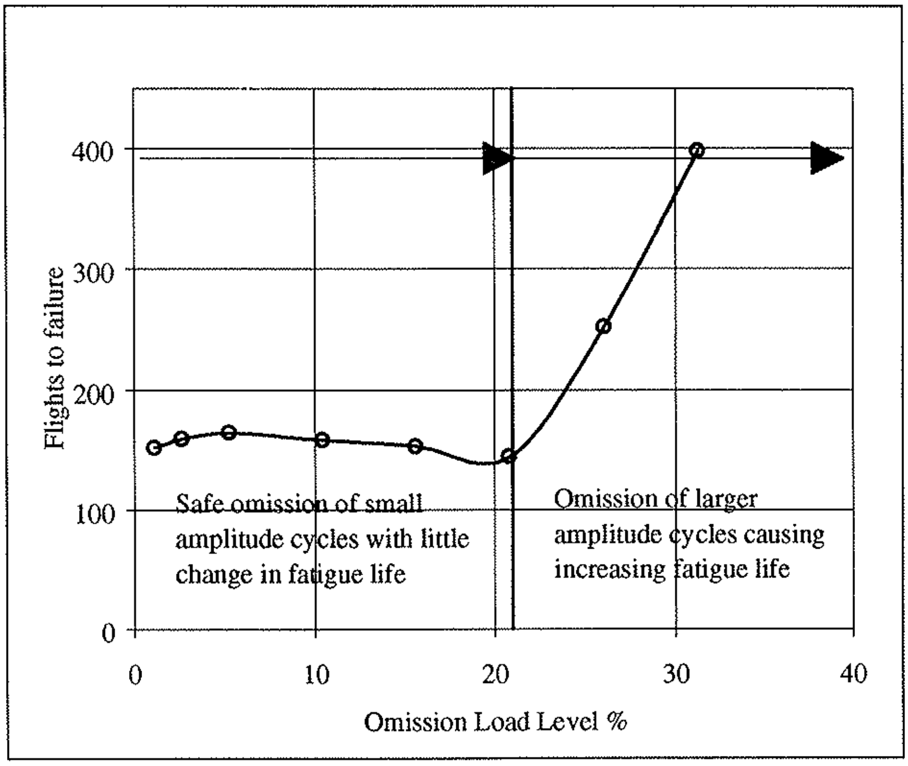

Figure 4 Results of omission load level tests on aluminium alloy 7010 T73651.

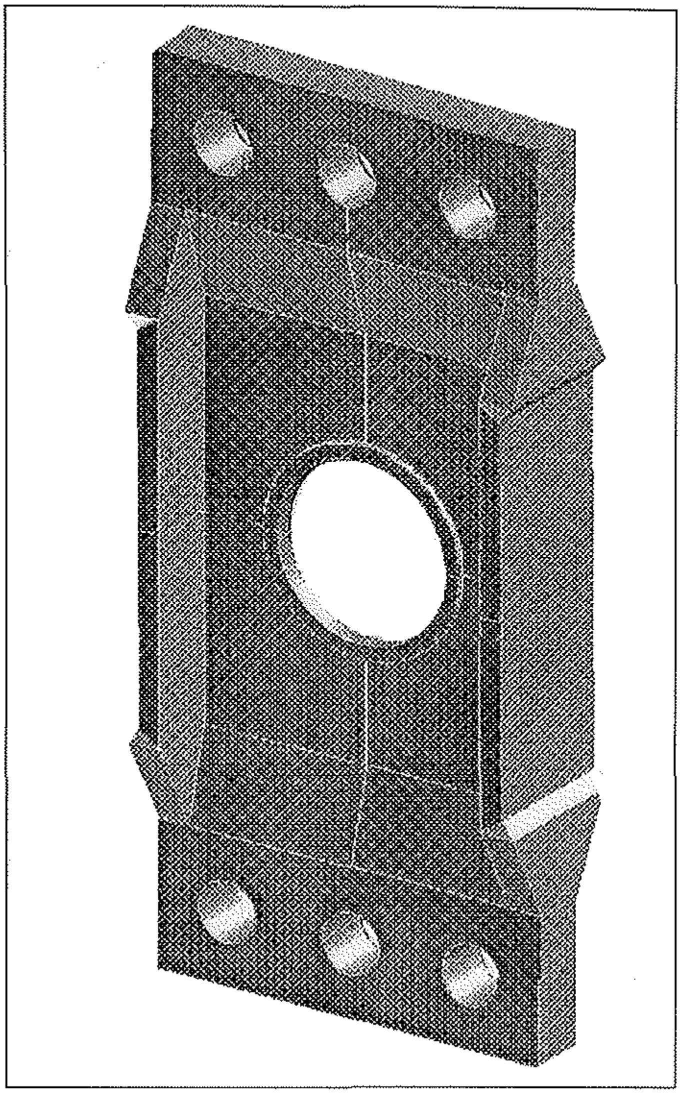

Figure 5 EH101 main lift frame structural test element

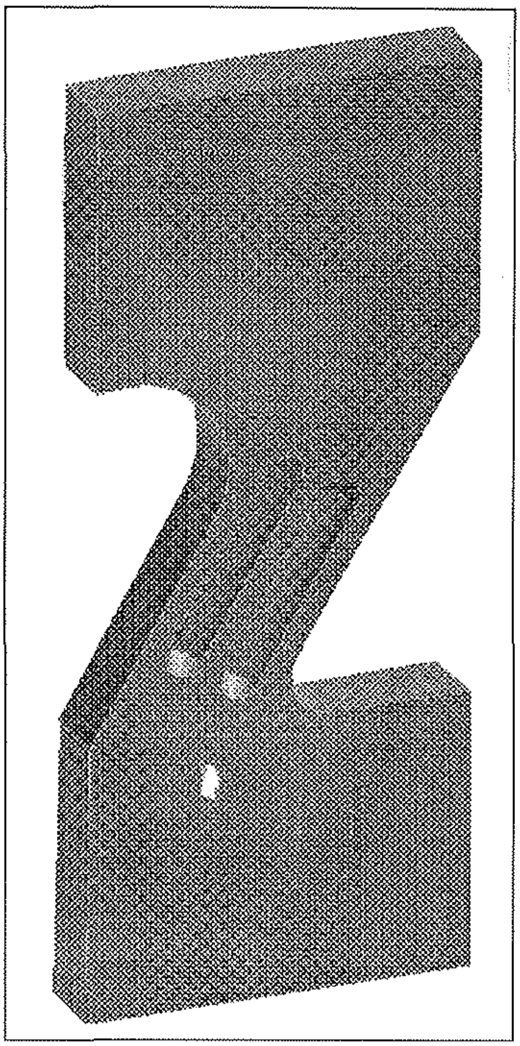

Figure 6 Lynx bolted rotorhead structural test element.

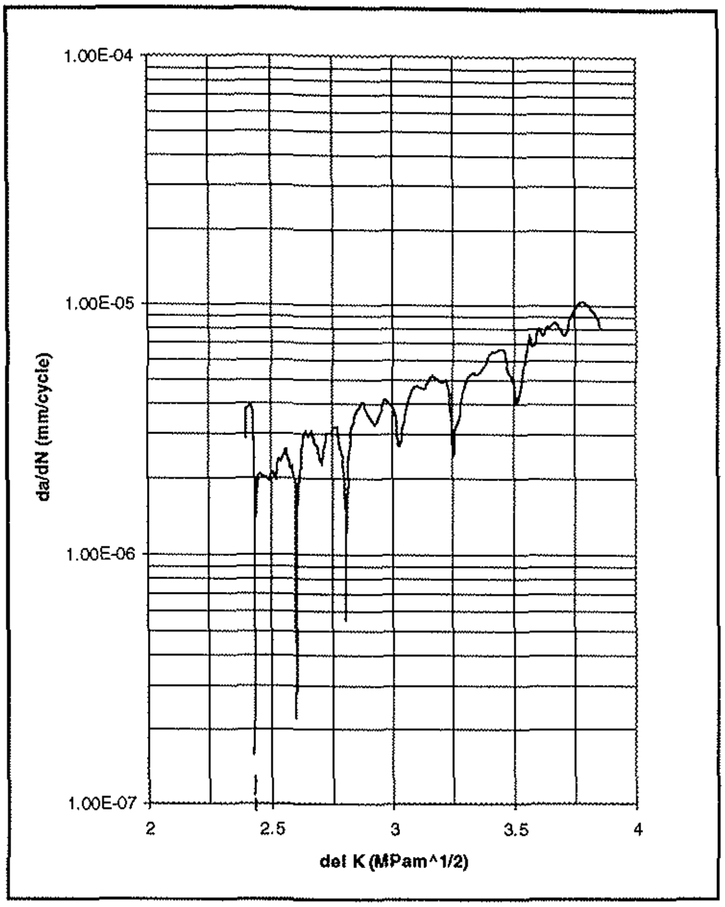

Figure 7 Effect on the fatigue crack growth rate of 40% tensile load excursions on 8090 T852 with respect to increasing baseline stress intensity factor range (ΔK).

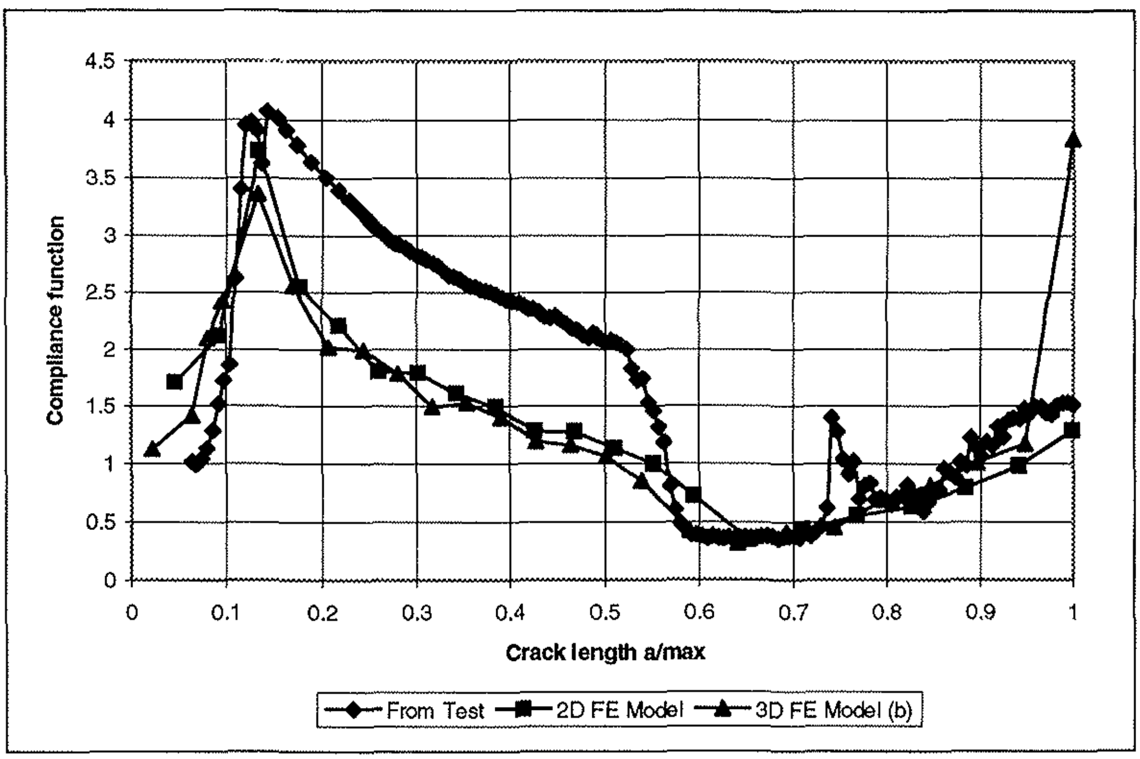

Figure 8 K solution using 3D and 2D finite elements in NASTRAN and comparison with K solution derived from test data.