Nationaal Lucht- en Ruimtevaartlaboratorium
National Aerospace Laboratory NLR

NLR-TP-2001-245

Review of aeronautical fatigue investigation in the Netherlands during the period March 1999 - March 2001

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Summary

A review is presented of the work on aeronautical fatigue that is performed in the Netherlands in the period March 1999 till March 2001.

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1.1 INTRODUCTION

The present review gives a brief summary of the work performed in the Netherlands in the field of aeronautical fatigue, during the period from March 1999 until March 2001. The various contributions to this review come from the following sources:

- The National Aerospace Laboratory NLR
- The faculty of Aerospace Engineering, Delft University of Technology (TU Delft)
- Stork Aerospace Fokker Services
- Stork Aerospace Fokker Aerostructures

The names of the principal investigators and their affiliation are presented between brackets at the end of each topic title.

1.2 LOADS

1.2.1 Standard load sequences for aeronautical application (P.A. van Gelder, NLR)

NLR has been actively involved, especially in the period between 1975 and 1992, with the creation and validation of standardised load sequences for fatigue analysis and testing.
In order to increase the accessibility, in terms of availability and application, and also to minimise the possibility of erroneous application of the data, all sequences (known and available at NLR) have been stored on to one device (CD-ROM). An intuitive user-interface has been added to enable the generation of load/stress sequences for the appropriate stress levels from the basic data. Also an interface option for common crack-growth programs such as NASGRO (NASA) and CRAGRO (NLR) has been supplied. To complete the set of information, the relevant describing reports have been added (PDF-format) to this CD-ROM as well. The CD-ROM is expected to be completed before the ICAF 2001 meeting.
The different phases in the load analysis process are illustrated in figure 1 taken from reference 1

1.2.2 Analysis of PSD loads on aircraft in two-dimensional atmospheric turbulence (R. Houwink, NLR)

The airworthiness requirements of aircraft for gust loads are based on a one-dimensional model for the atmospheric turbulence. At NLR analyses have been carried out to determine the effects of two-dimensional vertical turbulence (Fig. 2) on aircraft loads using a Power Spectral Density approach. This more realistic modelling can be expected to yield better results for larger aircraft, as it takes into account the spanwise variation of the gust velocity. Some results are shown in figure 3 taken from reference 2. The current work is carried out using a recently updated version of a computer program developed at NLR by Noback in the early nineties (Ref. 3).

1.2.3 Aircraft ground loads (P.A. van Gelder, NLR)

It has been investigated whether aircraft touch-down speeds and aircraft ground loads (in terms of cg-accelerations) can be determined from standard on-board data recordings of Boeing 747-400 aircraft in operation at KLM (Ref. 4)

1.2.4 Fatigue load/usage monitoring of military aircraft

a. Structural fatigue load monitoring of RNLAF F-16 aircraft (D.J. Spiekhout, NLR)

The fatigue load monitoring programme of F-16 aircraft of the Royal Netherlands Air Force (RNLAF) has been continued using the FACE system.

The “Fatigue analysing and Air Combat Evaluation system (FACE)” was developed by RADA Electronics Industries Ltd in Israel and will be installed in all RNLAF aircraft. To date the installation is completed for 75% of the RNLAF F-16 fleet. The FACE system allows a more extensive load monitoring for each individual aircraft. Quantities measured to monitor structural fatigue loading include strains at five different locations: two indicative for wing root bending and outer wing bending, two at the rear fuselage dealing with horizontal and vertical tail loads and one in the fuselage center section indicative for fuselage bending. Simultaneously with these strain gauges a number of flight parameters are measured. The FACE system also monitors the engine load and usage.
During the last two years much attention has been given to the “automated” processing and analysis of the recorded FACE data and administrative data. This administrative data is taken from the maintenance information stored in the Core Automated Maintenance System (CAMS) of the RNLAF. An important feature is the “automated” coupling of data from both sources.
Under development is a tool for reporting to the RNLAF on a weekly basis the damage experience per aircraft and the next inspection time per ASIP location. For this reporting a so-called On Line Analysis Processing (OLAP) tool will be used. An example of a OLAP reporting is shown in figure 4.

b. Maritime Patrol Aircraft, P-3 Orion (A.A. ten Have, R.P.G. Veul, NLR)

NLR has developed software to perform fatigue life calculations of the P-3 Orion maritime patrol aircraft. The software provide the operator a validated fatigue life indicator based on the most critical wing location. The software package is currently in use by the Royal Netherlands Navy RNLN (as “PLEBOI”), The Spanish Airforce SAF (as “SAFORI”) and the Portuguese Airforce PoAF (as “POLICAL”). For the RNLN and the PoAF NLR has installed and for the SAF NLR soon will install a structural data recorder to support the theoretical fatigue life calculations.
In addition, NLR acts as On-site Rep for the RNLN in a multi-national full-scale fatigue testing programme of the P-3 Orion. The NLR contributions to the so-called “SLAP” programme consist of characterisation tests on more corrosion resistant material, fatigue life and crack growth calculations for different operational usage and co-development of an individual aircraft fatigue tracking program.

c. Transport aircraft (R.P.G. Veul, NLR)

The RNLAF operates two C-130H-30 Hercules aircraft that were delivered in 1994/1995. In order to optimise operational and maintenance aspects, NLR is setting up tailored C-130 structural life monitoring procedures. An in-flight data acquisition system will be installed in each aircraft, generating operational load and usage data.

1.2.5 Helicopters

a. Lynx helicopter (A.A. ten Have, NLR)

To achieve Lynx life extension and to monitor and control maintenance of the main rotor and sponson loads, the RNLN has decided for a fleetwide installation of a multi-channel structural data recording system, called AIDA. This AIDA system also replace the former engine cycle counter. As to date the fleetwide installation of the AIDA system has been finished and some 500 flighthours has been gathered. AIDA will be a cornerstone for the RNLN for continuing operational usage of the Lynx fleet until full service introduction of it’s successor, the NH-90, has been achieved.

b. Chinook and Cougar helicopter (A.A. ten Have, NLR)

NLR has supported the RNLAF with structural integrity issues concerning Chinook and Cougar. The activities concern the development of fatigue life monitoring concepts, aiming at optimising economy, safety and maintenance aspects. A Chinook pilot project for structural load and usage monitoring was started in the last quarter of 2000.

1.2.6 Prognostics and Healt Management (H.H. Ottens, NLR)

For modern fighter aircraft (like the Joint Strike Fighter) a Prognostics and Health Management, PHM, system will be incorporated. The PHM system is an extension of the already available load and usage monitoring systems. The goal of a PHM system is to predict possible failure in time so that appropriate maintenance actions can be performed. The PHM system is aimed to reduce the maintenance costs and to increase the aircraft reliability.
A Dutch PHM Consortium, DPC, consisting of Perot Nederland (specialized in adaptive and learning software development) Sun Electric System (test equipment hydraulic systems) TNO-TPD (sensor development) and NLR. DPC is developing PHM systems now focussing on JSF.

1.3 GLARE

In 2000 Airbus has decided to use GLARE material for parts of the A380 fuselage as shown in figure 5. After many years of research and development the GLARE material has now reached a maturity level.
The Fibre Metal Laminates Centre of Competence, FMLC, has been established (Fig. 6). This centre, in which TU Delft, NLR and Fokker Aerostructures participate, coordinates in the Netherlands all the research and tecnology development on application of FML's in and outside the aerospace.

1.3.1 Dutch GLARE Technology Research Program (O. Van der Jagt, GRP)

In the Netherlands a GLARE research program (GRP) is carried out which was formerly known as the GTP program (Glare Technology Program). The GRP program is co-ordinated by Structural Laminates Industries (SLI). Other participants are the Faculty of Aerospace Engineering (TU Delft), the National Aerospace Laboratory (NLR) and Stork Fokker Aerostructures (FAe). The program is sponsored by the Dutch government and is executed in close co-operation with Daimler Chrysler Aerospace Airbus (DA). The aim of the research is to achieve technology readiness for the application of GLARE laminates in very large transport aircraft. Although experiments and analysis have shown that Glare is an attractive material for application in large pressurised fuselage structures, several technical issues had to be resolved in further detail. Projects have been defined for the following working groups:

- Qualification, Properties and Design Allowables
- Calculation Methods
- Design Concepts and Certification aspects
- Manufacturing and Processing Technology
- Maintenance support

A large number of sub-projects were defined and are carried out in these working groups.

a. Qualification, Properties and Design Allowables

During that programme several mechanical properties of Glare were determined at room temperature, elevated temperature and after environmental exposure. The tests were carried on Glare laminates with different lay-ups. Among the properties determined were the fatigue crack propagation and the fatigue crack initiation properties. The results of the tests will also be used to define the final design allowables for Glare laminates.
Another important issue is the definition and the qualification of the splicing concept. With these splices it is possible to manufacture larger skin panels than with monolithic aluminium sheets. An example of a splice in a GLARE panel is given in figure 7. As a precursor to the qualification test programme for Glare splices, an experimental investigation was carried out to determine the properties of several splice details. The objective of this investigation was to demonstrate that the mechanical properties of Glare splices are generally the same or better than those of the undisturbed Glare laminates. The investigation included fatigue crack growth tests with starter cracks at the most critical locations of the splice.
Durability is also an important topic. Test results have shown excellent durability properties of GLARE, but a more extensive environmental exposure program has been defined.

b. Calculation Methods

This subject is focused on the development of calculation methods (design tools) for GLARE structures. Research aims are the prediction of basic material properties ("metal volume fraction" approach), stability of panels, residual strength and crack propagation. Both analytical and Finite Element Methods are used. A large part of the work is concerned with the validation of the calculation methods by comparing predicted test results with known test data.

c. Design Concepts and Certification aspects

Design principles have been developed to enable design details to be optimised for GLARE structures. Items such as panel arrangement, typical GLARE shell design, stringer design concepts, longitudinal and butt joints, door and window cut-outs are considered. Also certification aspects are under discussion.

d. Manufacturing and Processing Technology

This subject is focused on low cost and low risk manufacturing of GLARE fuselage skin panels. Although the material cost of GLARE is higher than for monolithic aluminium, it has been shown that a considerable cost reduction can be achieved with a clever design and manufacturing process. It implies that GLARE is a very competitive material if compared to conventional sheet materials. The laminar built-up of GLARE allows a high degree of integration of structural details in one panel, such as splices and internal doublers. For this purpose, full-size, double curved demonstrator panels have been built, with splices, internal doublers, glare stringers and door cut-outs.

e. Maintenance support

Maintenance concepts (with emphasis on cost aspects) and NDT inspection and repair methods for GLARE structures are under development, also in view of acquiring airline acceptance of GLARE.

1.3.2 Modelling of fatigue crack growth in FML's (A.U. de Koning, NLR)

New models for the description of fatigue crack growth rates in GLARE were developed. Two cases were identified: The part through the thickness initial defect and the through the thickness crack. Stress Intensity Factors for both cases were proposed and verified by comparison of the predicted crack growth rates with data obtained from tests.

1.3.3 Fatigue crack growth in Glare riveted butt joint (W. van der Hoeven, NLR)

When riveted Glare joints are loaded in fatigue, crack initiation usually is at the faying surfaces of the joint. Initially, crack growth is in the aluminium layer adjacent to the faying surface only. Once this crack has reached a certain length, crack initiation occurs in the next aluminium layer, etc. Due to this crack growth behaviour the fatigue cracks are hidden (not visible from the outside) for a large part of the total fatigue life. In order to get more information about the crack growth rates in the individual aluminium layers, butt joint specimens were fatigue tested to different fatigue lives. After the tests, the joints were first inspected with eddy current and subsequently the outermost aluminium layer and the underlying fibre layer were removed such that the crack length in the next aluminium layer could be measured. Next, the second aluminium layer was removed, and so on. The results were used to construct crack growth curves. In addition, the results yielded information about the suitability of the eddy current technique for inspecting riveted Glare joints for fatigue cracks. (Ref. 5)

1.3.4 Blunt notch and residual strength of Glare sheet material and structures (T. de Vries, A. Vlot, R. de Borst, TU Delft)

The static strength of Glare is relatively high as compared to aluminium sheet materials, but the strength can be a matter of concern if blunt or sharp notches are present. Large series of experiments were carried out to study the blunt notch strength and the residual strength if cracks are present. Various test series were carried out to collect test data on these properties, and the fracture phenomena was studied in detail. The aim was to see how this could lead to generic prediction methods of the blunt notch strength and the residual strength of crack elements built up from Glare sheet material. Five programs have been carried out defined as: (1) Blunt notch behaviour, (2) Sharp notch behaviour of through cracks, (3) Blunt and sharp notch behaviour of spliced laminates, (4) Effect of FOD in a row of rivet holes, and (5) Development of crack stoppers.
Investigation of the failure behaviour has revealed information on delamination, fiber splitting in the adhesive bond line matrix and fiber failure. Furthermore support to arrive at methods for calculating properties for different types of Glare (different lay-up) was derived from detailed strain distribution measurements employing an instrumented grid technique and finite element calculations. Plasticity effects were also considered. Some selected findings are summarised below. The investigation is documented in a PhD thesis (Ref. 6).
Filling a hole with an unloaded rivet has no influence on the blunt notch strength. The squeeze force does not affect this result.
The residual strength of cracked panels at low temperatures is larger than at RT. The residual strength is the same for a saw cut and impact damage by sharp object. A critical K-value is not applicable, but the K_R-curve incorporating stable crack growth and plasticity based on an effective crack length can be used as a material parameter. A new phenomenological model for fracture of Glare laminates with through cracks has been presented.
Spliced laminates (see Fig. 7) with blunt and sharp notches were investigated. The influence of the splices on the Glare material properties and durability was negligible. Spliced configurations were also studied incorporating doublers produced as part of the Glare panels. The aim of the doublers was to act as crack stopping elements. The effect was limited. However, much better results were obtained by locally adding UD fibre prepreg layers as crack stoppers. The local addition of two extra UD fibre prepreg layers applied to Glare 3 3/2 0.4 material creates locally a Glare 4B 3/2 0.4 material. The residual strength of this configuration in a full-scale fuselage test performed by DCAA was improved by 18%. Another full-scale test was done where four instead of two extra layers were added. This configuration was tested with a two-bay crack and the allowable hoop stress was

increased by 75%. The crack flapped in the crack stopper band region in the circumferential direction and was arrested at the next stringer, thus leading to a controlled decompression.
The residual strength of a riveted fuselage lap joint with a large through crack due to large foreign object damage was studied for a lap joint already having fatigue cracks at rivet holes. This topic is the subject of a paper at the Symposium.

1.3.5 Durability of Bonded (Fibre Metal Laminate) Repair Patches (H.J.M. Woerden, A. Vlot, TU Delft)

Fuselages of (ageing) aircraft require safe and damage tolerant repair techniques. Instead of using riveted repairs, bonded repairs can be a more efficient solution. Adhesive bonding provides a more uniform and efficient load transfer into the repair patch and reduces the risk of high stress concentrations caused by additional holes necessary for riveted repairs. Bonded repair patches of Fibre Metal Laminates (FML) combine a small mismatch in thermal expansion coefficient with the aluminium skin. Experiments have shown excellent fatigue properties, a high strength and a moderate extensional stiffness. Apparently, the FML patches are a promising candidate for bonded repairs. However, questions about the long-term durability of bonded patch repairs have yet to be answered to ensure a safe operation, even under extreme environmental conditions. This ongoing research at Delft University of Technology in close cooperation with the United States Air Force Academy started with the Ph.D. work of Fredell. Currently the focus is on the environmental durability of bonded patch repairs. The influence of environmental conditions, like temperature and moisture, on the patch effectiveness as a whole and on the different repair components (skin material, patch material, primer and adhesives) are investigated. For an exposure up to the apparent maximum moisture equilibrium level according to Fick’s laws, experiments and Finite Element modeling indicated no significant decrease in patch effectiveness, even though the degradation of the adhesive stiffness can be significant. At low temperatures the crack growth sensitivity of aluminium alloys improves in the low stress intensity factor regime as a result of the low humidity level. This has a favourable effect on the repair efficiency.

1.3.6 New materials for aircraft structures (A. Vlot , C.A.J.R. Vermeeren, TU Delft)

At the occasion of the retirement of Professor Boud Vogelesang a 3-days Conference will be held in Delft (24-26 September 2001) on the subject “GLARE, the new material for aircraft”. Each day will start with a keynote lecture with the topics (i) Developments in aviation, (ii) Development of materials for aircraft design, and (iii) New materials and safety. The lectures are followed by panel discussions. In the afternoon sessions, lectures are presented by representatives of the aircraft industries, aircraft operators and members of research and development teams working on various issues associated with the application of Glare in aircraft structures. The meeting is concluded by the farewell lecture of Professor Vogelesang.

1.4 ENGINES

1.4.1 Engine usage monitoring

a. Pratt & Whitney F 100 engine (O. Kogenhop, NLR)

Since 1991, NLR performs operational engine usage monitoring of the Pratt & Whitney F100 engines installed in F-16 aircraft. For this purpose, a number of multi-channel data-acquisition systems have been installed in the RNLAF F-16 fleet registering parameters such as pressure altitude, calibrated airspeed, engine rotational speed and power lever angle. Engine damage accumulation is then calculated from the recorded engine cycles using specific algorithms. Furthermore, flight time and hot time envelopes (time spent in certain Mach number versus altitude regions) are determined to gain more insight in the RNLAF F100 mission profile. To date, more than 11000 RNLAF F-16 sorties have been collected with the multi-channel data acquisition system. On a routine basis, this operational RNLAF engine data is transferred to the engine manufacturer for evaluation purposes and could be used as a basis for tailored engine maintenance procedures, e.g. affecting inspection intervals or retirement lives.
From 1997, the Fatigue Analysing and Air Combat Evaluation system (FACE) is being introduced in RNLAF F-16 aircraft. FACE is a comprehensive maintenance management and flight debriefing system developed by RADA Electronic Industries Ltd in Israel. This system enables the recording of approximately 100 engine parameters of which a representative selection has been determined in 1999 by Pratt & Whitney. To date, more than 10 000 RNLAF F-16 sorties have been collected with the FACE data acquisition system. Ad-hoc campaign measurements will always allow other parameters to be temporarily monitored.

b. Rolls Royce Gem 42 engine (A.A. ten Have, NLR)

On a routine basis and employing the NLR developed AIDA data acquisition system, NLR supports the RNLN with Cyclic Life Control of their Lynx Gem engines. Currently the RNLN is the only Lynx operator that benefits from tailored exchange rate in Gem engine maintenance procedures.

1.4.2 Life assessment of engine components (T. Tinga, NLR)

In order to do an accurate life assessment of a gas turbine component, it is necessary to have detailed information on gas turbine loads, thermal, mechanical and aerodynamic loads, and the resulting stresses in the component.
The total loading of gas turbine components has two major contributions: thermal loading and mechanical loading. The loading of stationary (non-rotating) components like vanes is mainly thermal, caused by variations in temperature of the hot gas stream in which the components operate. Rotating components have additional mechanical loading, where the centrifugal loading due to the high rotational speed of the components is the most important contribution. At the NLR a method has been developed (Ref. 7) to determine both the thermal and centrifugal loading of any gas turbine component for any mission.
The basis of the analysis method is the Gas Turbine Simulation Program (GSP), developed at the NLR. GSP performs a transient (gas path) analysis of a mission using power settings and flight conditions as input and giving, among others, gas temperatures, pressures and mass flows as output. A computational fluid dynamics (CFD) analysis is performed to calculate the heat transfer from the hot gas to the component. The combined GSP and CFD results are then used in a finite element (FE) analysis to calculate the temperature distribution in the component (see Fig. 8). The results of this thermal analysis are combined with the rotational speeds calculated by GSP to determine the stresses and strains in the component. Finally, the obtained stress spectrum is used to perform a life assessment of the component, which completes the analysis. The method has been applied to real components like HPT and LPT turbine blades (Ref. 8) of the F100-PW-200 turbofan engine. The HPT blade is both internally and film cooled, whereas the LPT blade is solid and uncooled.

1.4.3 Modelling of thermal barrier coatings (M.F.J. Koolloos, NLR)

The Finite Element Method (FEM) was used to improve the understanding of failure of plasma sprayed Thermal Barrier Coatings (TBCs). Three types of thermal testing were simulated: isothermal furnace tests, short cycle burner rig tests and long cycle burner rig tests. The so-called global-local approach was used. First, overall temperature and stress distributions in the TBCs were obtained as functions of the type of test. These global analyses provided useful information about the locations of highly stressed areas. Next, the interesting areas were subjected to local and more refined FEM modelling. The local model provided additional information about the effects of bond coat roughness and the presence of an interface oxidation layer, especially with respect to the shear and normal stresses near and at the top coat / bond coat interface. Interface roughness resulted in small variations of the local solution shear and normal stresses around the global solution; and an oxide layer between the top coat and bond coat reversed the sign of the normal stresses and increased the shear and normal stresses considerably.
In conclusion, it was shown that the model provides information about the stress distributions that is useful for understanding the experimentally observed failure mechanisms. High normal and/or shear stresses (a) near the intersection of the top coat / bond coat interface and a free edge, or (b) in the flame area close to the interface roughness peaks, caused the top coat to delaminate. It is worth noting that a local analysis was not always required to understand failure.
An example of a local stress distribution is presented in figure 9. The results are presented in references 9 and 10

1.5 FATIGUE AND DAMAGE TOLERANCE STUDIES

1.5.1 Riveted joints

a. Stress Analysis of Mechanically Fastened Joints (J.J.M. de Rijck, S.A. Fawaz, TU Delft)

The application of the neutral line model provides a simple analytical tool to calculate the bending stresses in lap-splice or butt joints. In addition, the influence of longitudinal stiffeners can be evaluated. The validity of this method is investigated by comparing analytical to experimental results. The neutral line model uses advanced beam theory to derive analytical solutions for in-plane and out-of-plane displacements which are easily converted into tension and bending stresses. The model accounts for the eccentricities of the multi-layer joints. These eccentricities introduce a bending moment at the fasteners. For thin sheet lap-splice joints, the influence of the fastener flexibility must be considered also. Strain gage data were collected close to the most critical fastener row as a function of the remote tensile stress. A good agreement was observed between the analytical and experimental results for four types of joints. The analytical neutral line model provides a good first approximation of the stresses at the most critical fastener row thereby eliminating the need for labour intensive and time consuming finite element calculations.

b. Stress Intensity Factors Solutions for Countersunk Holes Subjected to Tension, Bending and Pin loading (J.J.M. de Rijck, S.A. Fawaz, A. Vlot, TU Delft)

For aerodynamic reasons, countersunk rivets are used in transport aircraft fuselage joints. Inspection of in-service longitudinal and circumferential joints has shown that fatigue cracks are initiated at the intersection of the faying surface and straight shank portion of the countersunk rivet hole. To study the influence of the applied stress on the crack growth rate and crack shape, an analytical study and an experimental study were conducted. In the experimental part, crack growth from a single countersunk hole subjected to tension and to combined tension and bending was investigated. By using a marker load spectrum and a scanning electron microscope (SEM), the crack shape and crack front location throughout the crack history was determined. In addition, fractographic crack length measurements were correlated to in-situ crack length measurements. The purpose of the SEM investigation is to study the crack shape development close to the bore of the hole and crack penetration behaviour at the outer surface. In the analytical part, stress intensity factors for cracks nucleating and growing from a countersunk hole subjected to tension, bending, and pin loading were calculated. The three-dimensional virtual crack closure technique was used. The experimental results are used to verify the accuracy of the newly calculated K solutions.

1.5.2 Analysis of Residual Strength of Panels with Multiple Site Damage (M.F.J. Koolloos, F.P. Grooteman, H.J. ten Hoeve and A.U. de Koning, NLR)

The Aloha accident in 1988 resulted in much attention being paid to the multiple site damage (MSD) phenomenon of riveted lap joints in aircraft fuselages. MSD is a typical problem for ageing aircraft, where the large number of fuselage pressure cycles may cause fatigue cracking at multiple rivet locations. After some growth of the MSD cracks they may interact and crack link-up may occur resulting in one leading crack flanked by MSD cracks.
MSD reduces the overall structural integrity. Consequently, the residual strength of a panel with a leading crack and MSD cracks is known to be lower than that of a panel with the same leading crack without MSD. During the present investigations an advanced engineering model to predict the residual strength in flat stiffened panels with one leading crack and MSD cracks is developed. The model is based on the strip yield model with R-curve concept and the effect of stiffeners on the deformation behaviour of an MSD panel is implemented.
The developed model is verified using experimental data from the NLR and three other institutes. Several MSD crack configurations were calculated for unstiffened and stiffened 2024-T3 aluminium panels without a lap joint. The model predictions are in good agreement with the measured residual strengths (Ref. 11).

1.5.3 Effect of hole tolerances on fatigue properties (N.J. Fraterman, Fokker Aerostructures)

In order to reduce cost and assembly time it can be advantageous to have a higher tolerance and a larger nominal diameter on fastener-holes. A series of lap-shear fatigue tests are in progress to determine the effect of hole-tolerance on the fatigue properties of various fasteners and material combinations.

1.5.4 Material characterisation of 7249-T76511 extrusions (W.G.J. ‘t Hart, NLR)

To extend the operational life of the P-3C Orion aircraft, modern extrusion materials are considered to replace the (stress) corrosion susceptible 7075-T6511 alloy which is the primary Aluminium alloy in panel extrusions for the wing and horizontal stabiliser.
A test programma was carried out on 7249-T76511 material obtained from 10 extrusion panels, produced from three separate chemistry melts. The objective was to demonstrate that this material meet or exceed the mechanical and damage tolerance properties of 7075-T6511 and showed improved corrosion resistance. The material characterisation programme concerned: static strength; fatigue and residual strength; fatigue stress-life tests on coated specimens and (stress) corrosion. Figure 10 shows the effect of shot peening and the effect of protective coatings on the life time.
The main conclusion, based on the properties considered was that the 7249-T76511 properties were broadly similar or exceeded those of 7075-T6511 (Ref. 12)

1.5.5 High Cycle Fatigue Properties of Gamma Titanium Aluminide (M.F.J. Koolloos, NLR)

Gamma titanium aluminides (γ Ti-Al) containing 45-48 at% Al have received considerable attention as candidate materials for high temperature aerospace applications. These alloys possess a low density and good high temperature mechanical properties up to 750°C, which makes them attractive for components in which the stresses arise primarily from inertia. However, there is still a reticence to use γ Ti-Al due to their relatively low values of

ductility, fracture toughness, and crack growth resistance. Several investigations have shown a close relationship between the mechanical properties and the microstructure, which can show considerable variations depending on the chemical composition, and casting and heat treatment procedures.
This study was carried out in the framework of the European COST 522 programme, where a new γ Ti-Al alloy was developed. The various mechanical properties of this new alloy are characterised by all partners. The present investigation concentrates on the high cycle fatigue properties of γ Ti-Al with emphasis on the fractographic evaluation to elucidate the role of the microstructure on the fracture process and fatigue life. Moreover, some attention is paid to Foreign Object Damage (FOD) and its effect on the fatigue strength.

1.5.6 Fatigue and damage tolerance properties of Friction Stir Welded coupons, components and panels (A.U. de Koning, NLR)

A series of butt-joints were friction stir welded and tested. For two Aluminium alloys, 2024-T3 and 7075-T7 the fatigue (Kt = 1.0 and Kt = 3.1) and the crack growth behaviour (quasi static and fatigue) of welded coupons were close to the behaviour of the base material. One year of outside exposure showed no degradation of the welded material and of the heat affected zone of the 7075-T7 coupons compared to the base material. Welded 2024-T3 coupons showed some degradation of the heat affected zone.
Five welded T-stiffeners were welded to a one meter long skin sheet. The panel will be used for fatigue crack growth and residual strength tests.
Thin sheet material (thickness 0.4 mm) was welded (Jane Luc’s method) showing good strength properties (85% of the 2024-T3 material strength)
The Friction Stir Welding, FSW, process was modelled as a visco-plastic deformation process. The results show a maximum temperature at some distance of the pin.

1.5.7 Risk analysis of aircraft components (F.P. Grooteman, NLR)

For the RNLAF risk analyses on engine components have been made to determine the fleet risk. The model used consists of the following three steps.
1) Weibull analysis of failure and non-failure data
2) Monte Carlo simulation of many components
3) Fleet risk analysis

Based on the component times of known failures and current times of unfailed components, a probability function can be constructed describing the chance a component will fail given its current cycle number. For this, a so-called Weibull analysis is made resulting in a Weibull distribution function. This Weibull distribution is used in a Monte Carlo simulation, in which the crack growth and inspections of a large number of components are simulated. The Monte Carlo simulation consists of the following steps:

- Draw from the Weibull distribution the life of the component
- If the life of the component is less than the Weapon System Life simulate crack growth and inspection opportunities. The crack propagation interval is determined by a backward crack growth analysis starting from the critical crack length to the detectable crack length. The crack has to be found in this interval to prevent failure of the component. The length of this propagation interval determines the number of inspection opportunities.
- The component will fail if the crack is not found

This Monte Carlo simulation results in a probability distribution expressing the chance that a component will fail before reaching its Weapon System Life. This distribution is eventually used to determine the risk of the entire fleet based on current component times.

An example curve is depicted in the figure 11 in terms of the probability of an additional failure. The solid line is the cumulative of the dotted line, in which the discontinuties are caused by removing a component from the fleet when reaching its system life.

1.5.8 Fatigue of Structures and Materials (J. Schijve, TU Delft)

A textbook was written with the title Fatigue of Structures and Materials. The book contains 20 chapters (about 500 pages), mainly following aspects briefly outlined in figure 12. Although much information and illustrations are coming from aerospace problems, the book is supposed to apply to engineering structures of metallic materials. One chapter is added on fiber-metal laminates. The book is printed by Kluwer and is expected to be available in June 2001.

1.6 FULL SCALE FATIGUE TESTS

1.6.1 Fokker 60 fatigue and damage tolerance tests (N.J. Fraterman, Fokker Aerostructures)

The Fokker 60 is a derivative of the Fokker 50 propjet and was developed to suite the requirements of the RNLAF. With respect to the Fokker 50, the Fokker 60 has a stretched fuselage, higher design weights, an increased cruise speed and a large cargodoor (3*2 meter) in the forward fuselage. For the JAR 25 certification it was agreed by the authorities to perform fatigue tests for area with newly designed structure, for area with considerable higher fatigue loads than the Fokker 50 and/or for areas where the original Fokker 50 fatigue tests could not be extrapolated any further. Most areas have successfully been tested and have already been reported in previous ICAF reviews.
The fatigue tests on two remaining areas have recently been finalised. One the component tests represents a fatigue critical joint in the lower skin of the outer wing, subjected to flight loads due to gust / manoeuvre and ground loads due to taxi / landing. The joint is designed and certified as a multiload-path structure and will be tested for two and a half life times (=235000 fc) . The panel joint developed a crack in the coupling-plate after ~40000fc, which was repaired after monitoring the crack growth. Subsequently this repair was tested for 239000 fc until some small cracks developed. At the end of the fatigue test the panel was subjected to a residual strength test, during which the panel could carry a load of 120% ultimate load without failure.

The other remaining component test represents the upper part of the mainframe, which is part of the wing-fuselage connection. The fatigue spectrum was introduces by 8 actuators and consisted of flightloads due to gust /manoeuvre plus cabin-pressure loads and groundloads due to taxi, turning, landing impact and braking. The first part of the test consisted of a fatigue test, which was reported in the ICAF 1999 review. During the second part of the test artificial cracks of 0.05” were introduced in order the check the damage tolerance behaviour. After 164000 fc (just short of the test-goal of two lives = 180000 fc) some natural cracks were found close to the artificial cracks, however the artificial cracks did not propagate at all. During the final residual strength test, the mainframe was loaded up to 140% limit load without failure.

1.6.2 Fatigue test on A340-500 actuator bolt (N.J. Fraterman, Fokker Aerostructures)

The actuator bolt of the centre landing retraction actuator has been designed as a break-bolt (fuse) in order to prevent rupture of the belly fuel tank in case of a landing gear tear-off. A static test was performed, which showed sufficient strength for the ultimate actuator load and the demonstrated failure load was low enough the prevent rupture of the belly fuel tank. The fatigue life of the bolt has been successfully demonstrated during a test for five lives (100000 fc) and a subsequent residual strength test. Further testing is in progress, during which the damage behaviour will be determined. During this test artificial cracks will be introduced after five lives (100000 fc) and the test will be continued for another 60000 fc (three lives) in order to determine crack growth behaviour.

1.6.3 Fatigue testing of the A340-500/600 thermoplastic fixed wing-leading edge (N.J. Fraterman, Fokker Aerostructures)

The fixed wing-leading edge is completely manufactured from thermoplastic Glass-PPS and consists of a skin panel, which is reinforced by welded on closely spaced ribs and a toppanel which is reinforced by hat-stiffeners. In order to determine the durability of the welded connection between skin and ribs, a number of coupon tests are in progress to verify the fatigue properties in shear and in peel.
The toppanel is attached to the uppergirder of the wing-frontspar and is therefore subjected to a significant shortening during each flight, which results in elastic buckling during most of the flights. The effect of repeated elastic buckling will be verified during a fatigue-test of a top-panel component loaded in compression-buckling.

1.6.4 Fatigue tests on a generic Airbus A380 fuselage panel (J.F.M. Wiggenraad, NLR)

NLR performed damage tolerance and residual strength tests on a curved fuselage panel to assess generic design concepts for the Airbus A380. The panel (1.1 m x 3.0 m, R = 2.8 m) contained five frames and five stringers and was tested in the NLR “Fuselage Panel Test Set-up”. Bi-axial loading and lateral pressure were applied simultaneously. The programme consisted of a crack propagation test, a repair, another crack propagation test and a residual strength test. A total of 77 000 flights were simulated.

1.7 FATIGUE PROBLEMS IN SERVICE

1.7.1 Prediction of fatigue loads and lifetime of a wingtip unit on a Fokker F27 aircraft (A. Hornstra, Fokker Services)

On a Fokker F27 aircraft fatigue cracks were found in the front spar brackets within the pylon of a wingtip store, mostly on the RH (Right Hand) side units. The wingtip store consists of a pylon attached to a hexagonal-shaped box with a radome underneath, with a total mass of only 30 kg.
Wind tunnel tests on a 1:15 aircraft model and in addition on a full scale wingtip store, showed oscillating forces due to flow separation and vortex generation. With the wind tunnel tests results and detailed FEM stress analyses, it was not possible to predict the experienced lifetime of about 1200 FH average for the RH unit. A guide vane, fixed at the inner side of the radome reduces the vibration levels of the wingtip unit as was demonstrated during the second wind tunnel test.

This aerodynamic modification was full scale tested with a limited Ground Vibration Test (GVT) and test flights without and with guide vane installed. The tests were performed in close co-operation with NLR. The GVT showed an asymmetrical wing torsion/bending mode of about 26 Hz. The RH unit response was 2 to 3 times larger compared to the LH unit. The first flight test (without guide vanes) confirmed the dynamic behaviour found in the GVT. The second flight test showed that the installed guide vane is quite effective in reducing the response. The lateral accelerations associated with the bending mode (of both RH and LH unit) are reduced to 35 % compared with the values for the unit without guide vane.

The fatigue life of the front spar bracket of wing tip unit, with and without guide vane installed was predicted. The test results of the first flight test showed that at cruise speed a RMS value of the lateral CG acceleration of the RH unit of 4.2 m/s^2 (RMS) at cruise speed. A Rayleigh distribution was assumed for a narrow band response amplitude. The Rayleigh distribution was used to define a proper load or acceleration spectrum. A detailed FEM stress model was used to determine the critical stresses of the front spar bracket.

The displacements and rotations at the wingtip unit CG were integrated from the acceleration. The experienced lifetime could be predicted quite well and also the effect of the guide vane on the lifetime could be predicted with a certain reliability. With the guide vane installed, the expected lifetime of the front spar bracket will be at least 120000 FH.

1.8 REFERENCE

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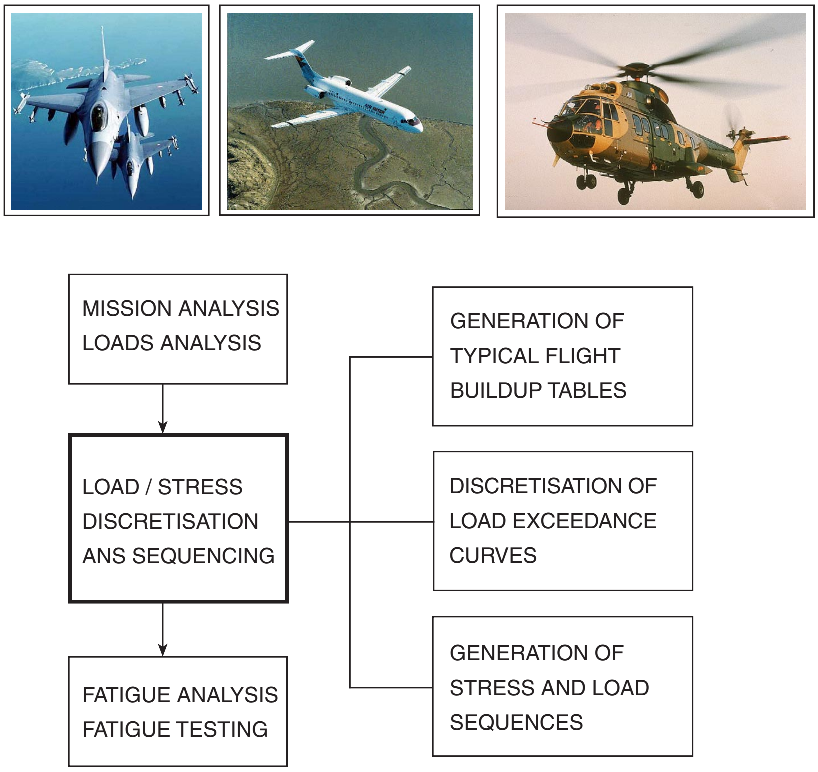


Figure 1. Phases in the aircraft loads and fatigue analysis process

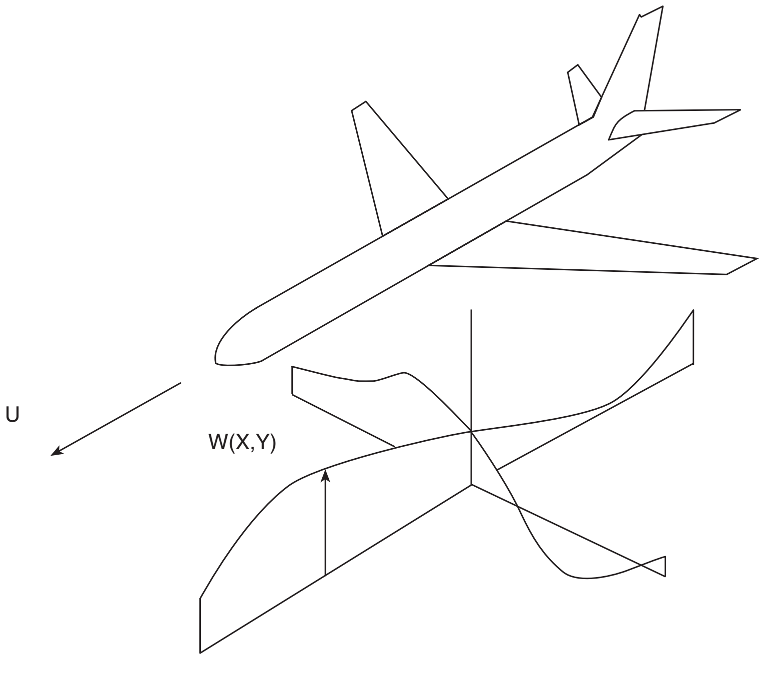
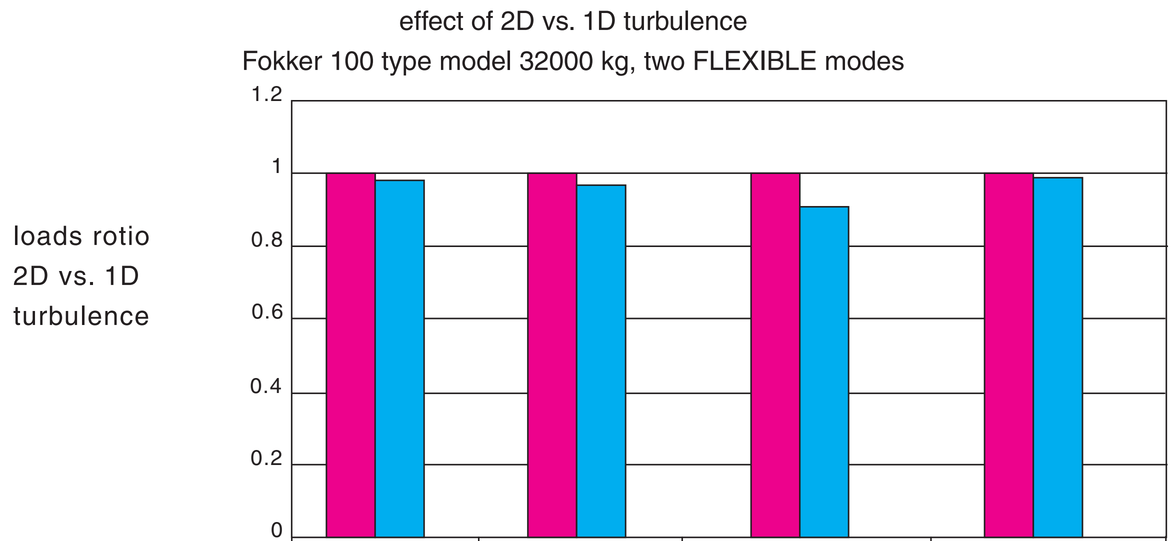


Figure 2. Sketch of aircraft flying through two-dimensional gust field



	Nz	Shear Force	Bending Moment	Torsion Moment
1D, 2 flexible modes	1	1	1	1
2D, 2 flexible modes	0.973	0.967	0.934	0.979

Figure 3. 2D to 1D turbulence ratio of gust loads for aeroelastic model of Fokker 100-type aircraft

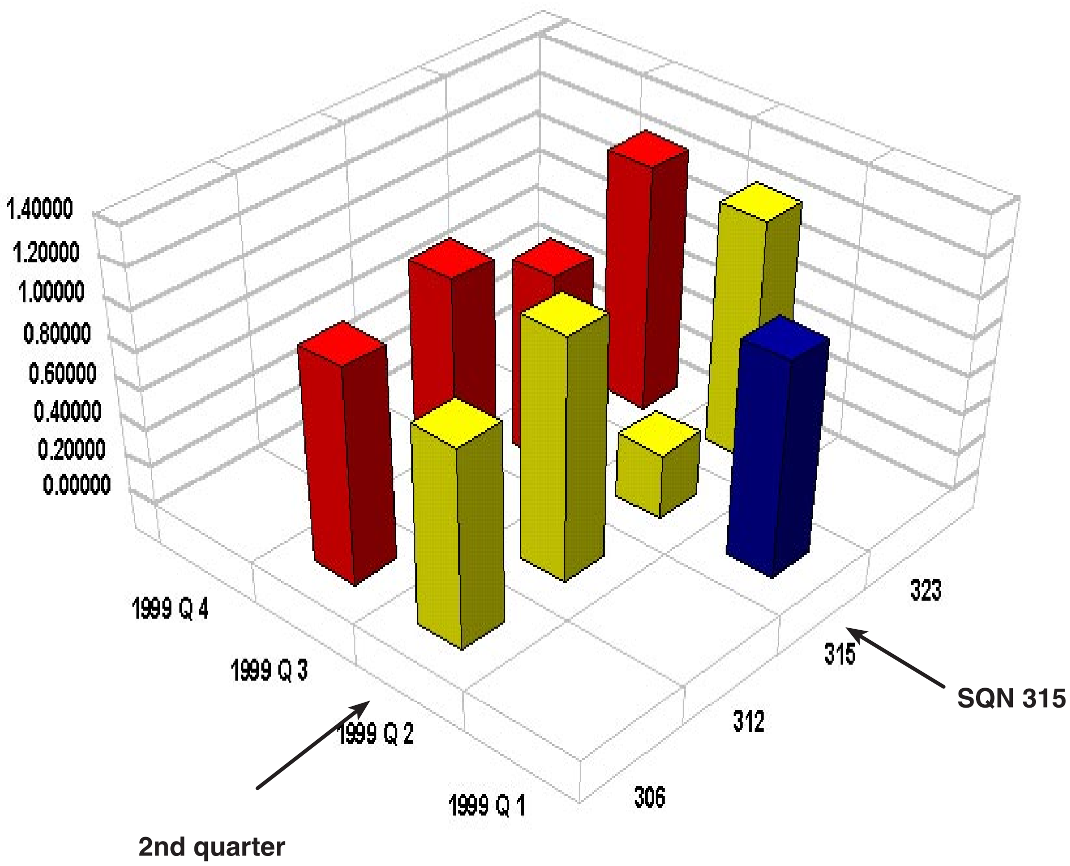


Figure 4. Example of a OLAP presentation

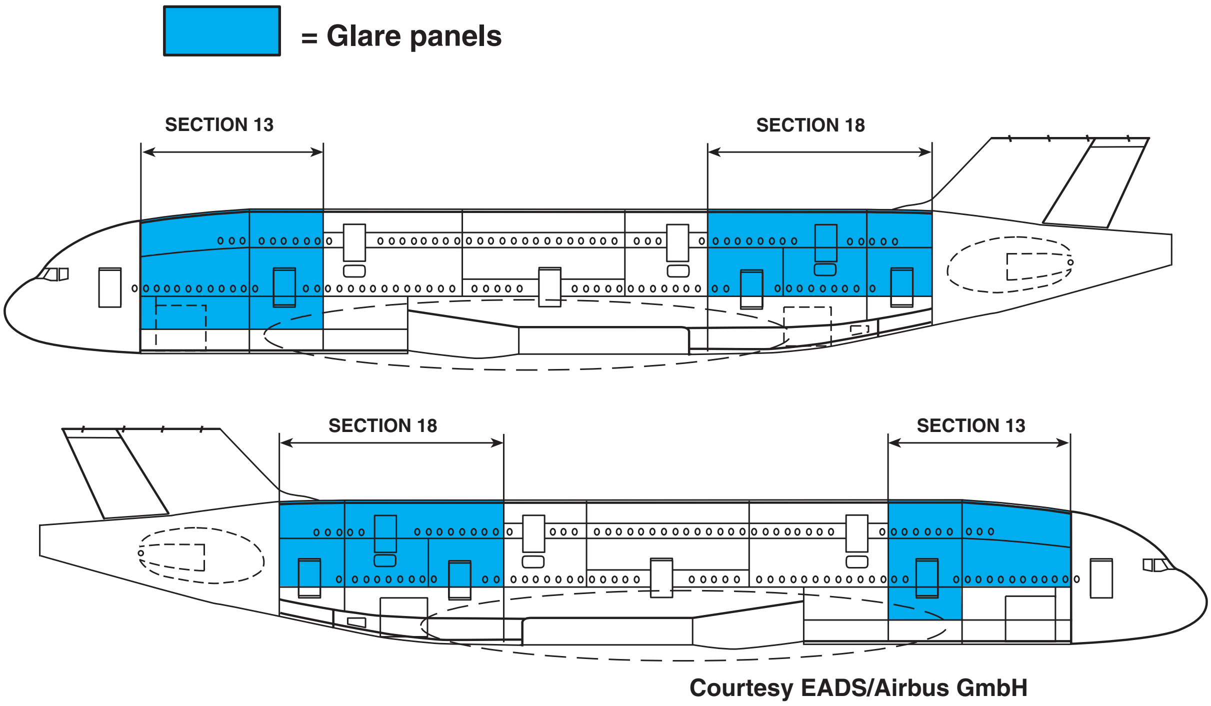


Figure 5. Application of Glare on the Airbus A380

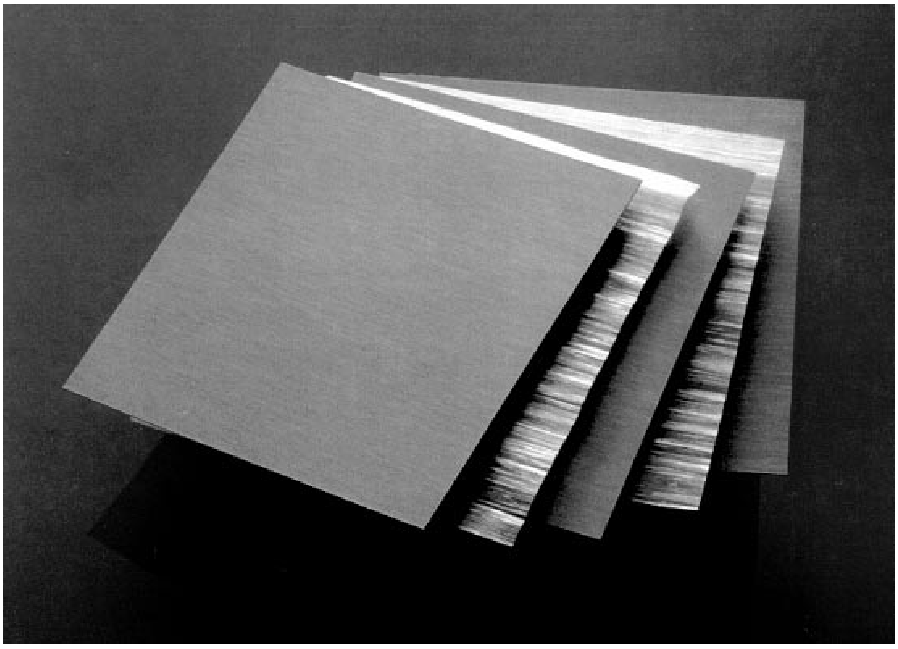


Figure 6. Fibre Metal Laminates Centre of Competence

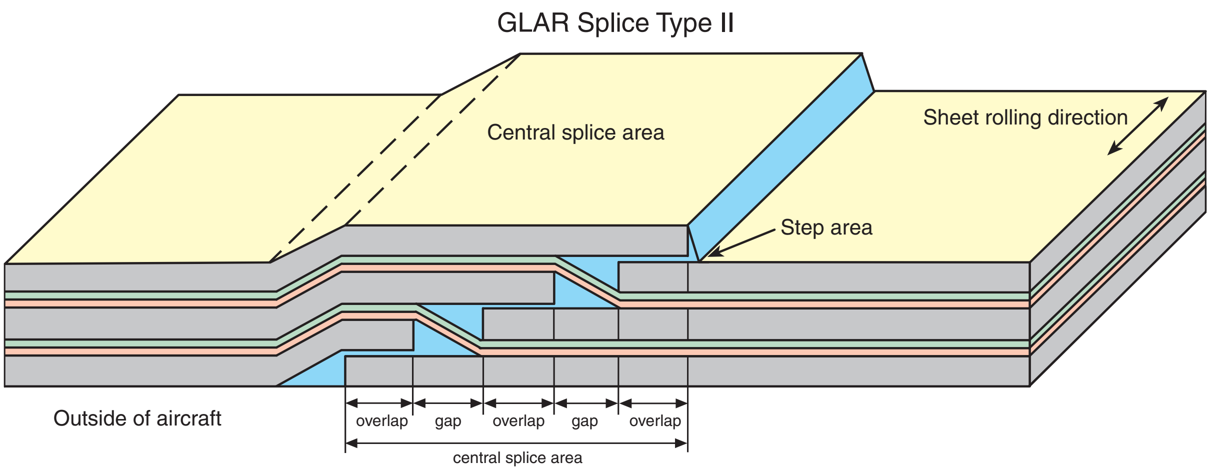


Figure 7. **CLARE splice configuration**

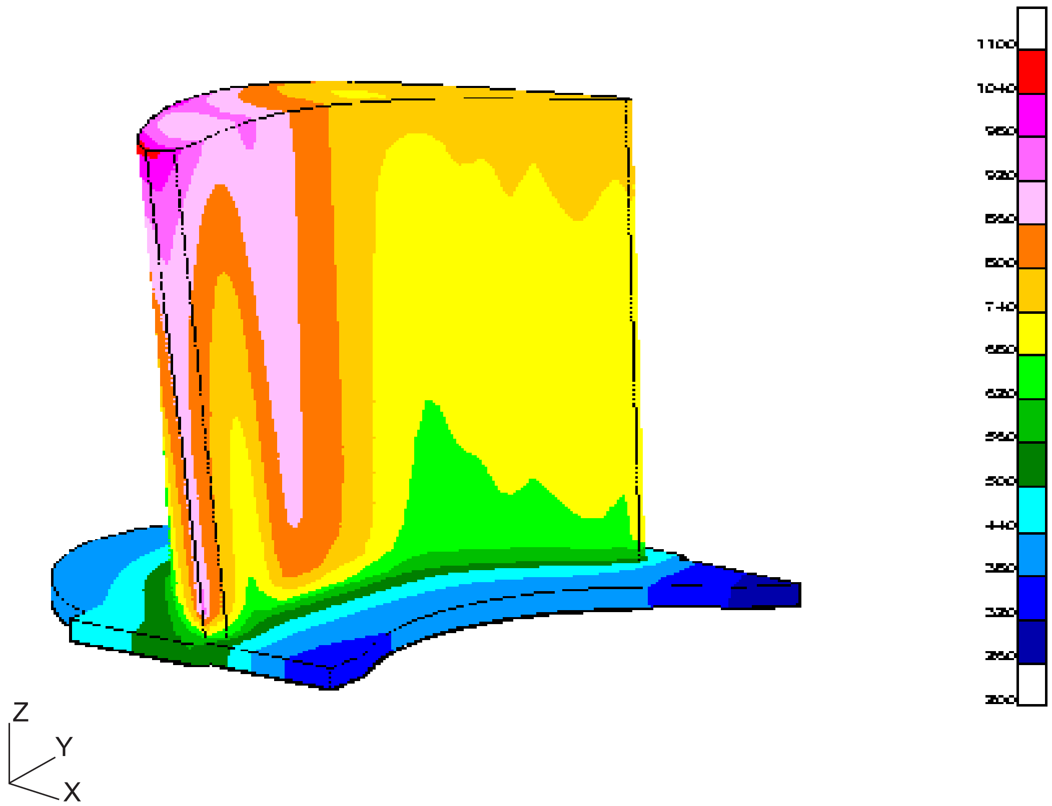


Figure 8. **Temperature (°C) in an internally and film cooled turbine blade**

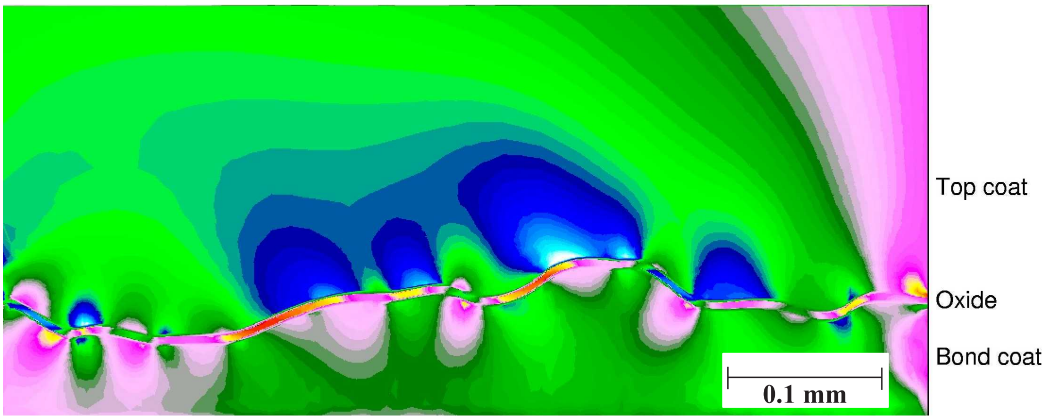


Figure 9. **Contour plot of strains in 1.0 mm top coat near free edge (located at the righthand side of the plot) during simulation.**

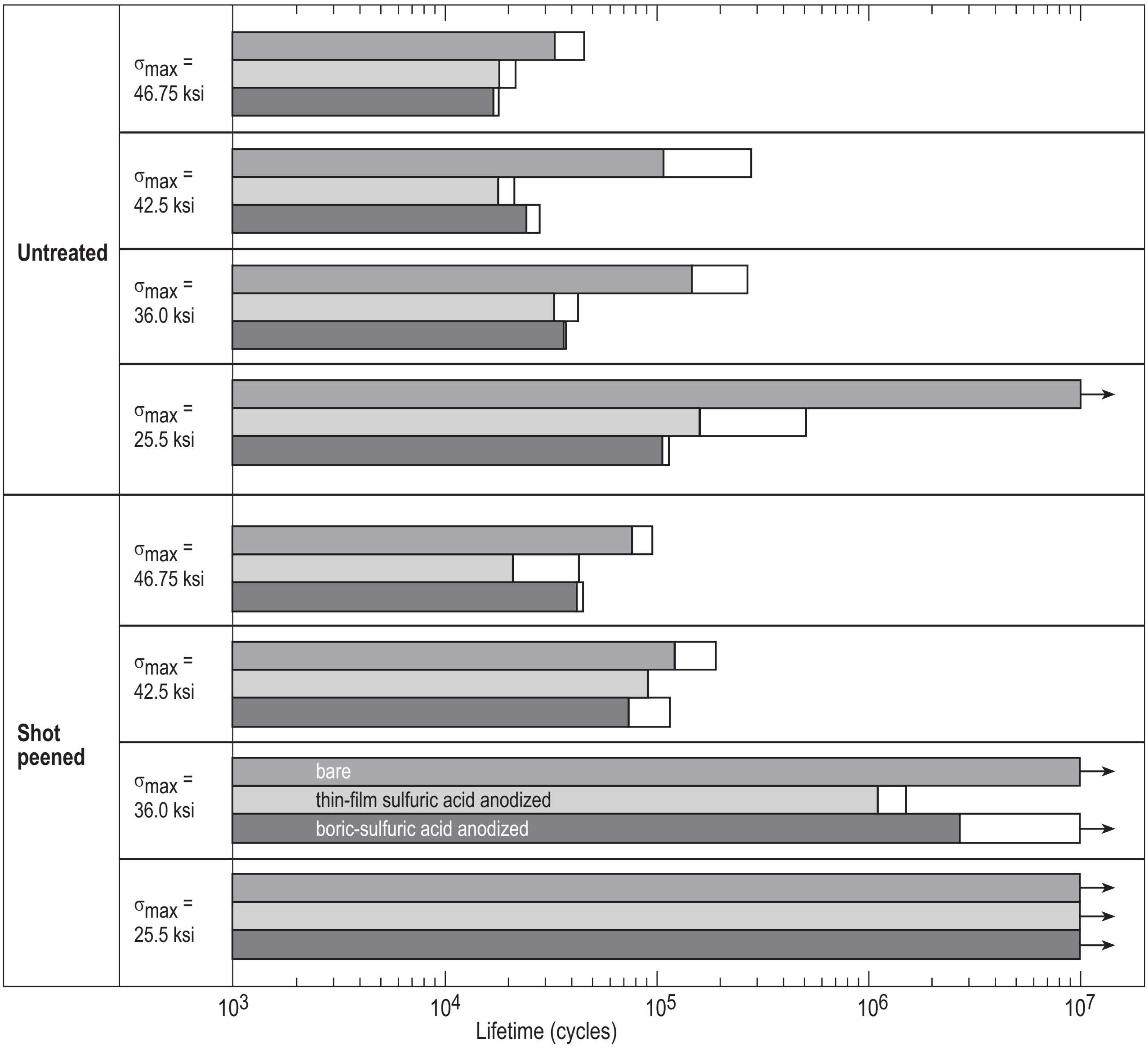


Figure 10. **Results of coating fatigue tests, constant amplitude loading, R = 0,1.**

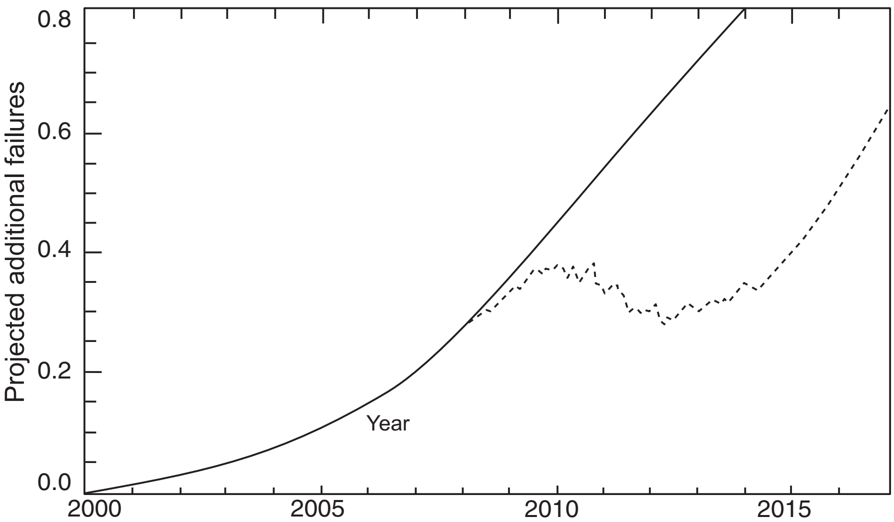


Figure 11. Projected additional failure of components based on a risk analysis

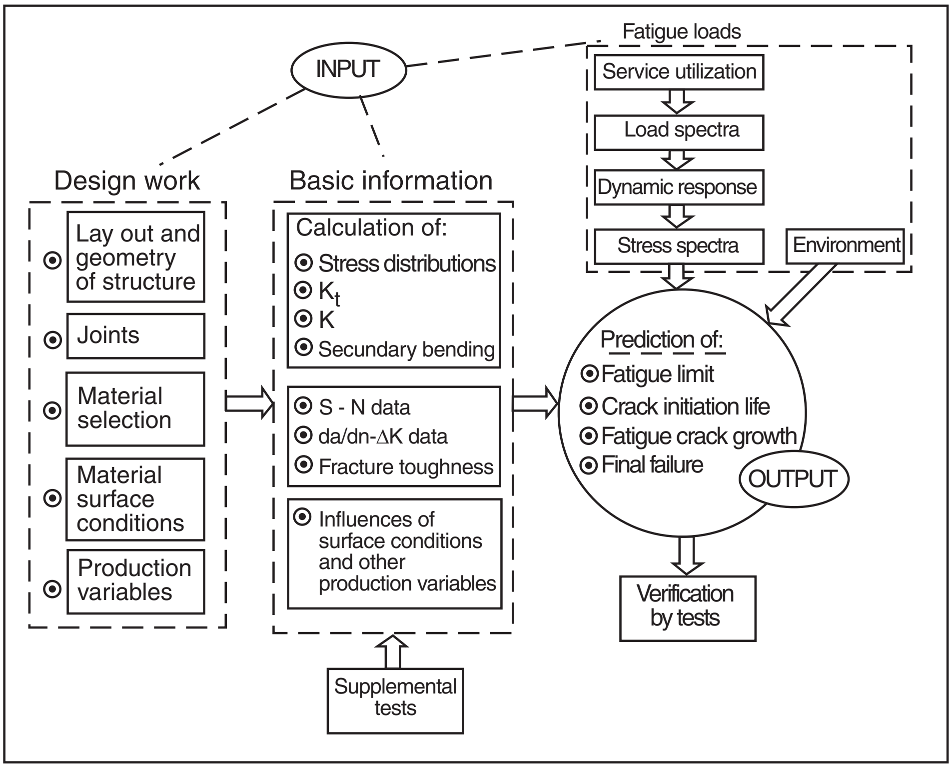


Figure 12. Survey of the various aspects of fatigue of structures.