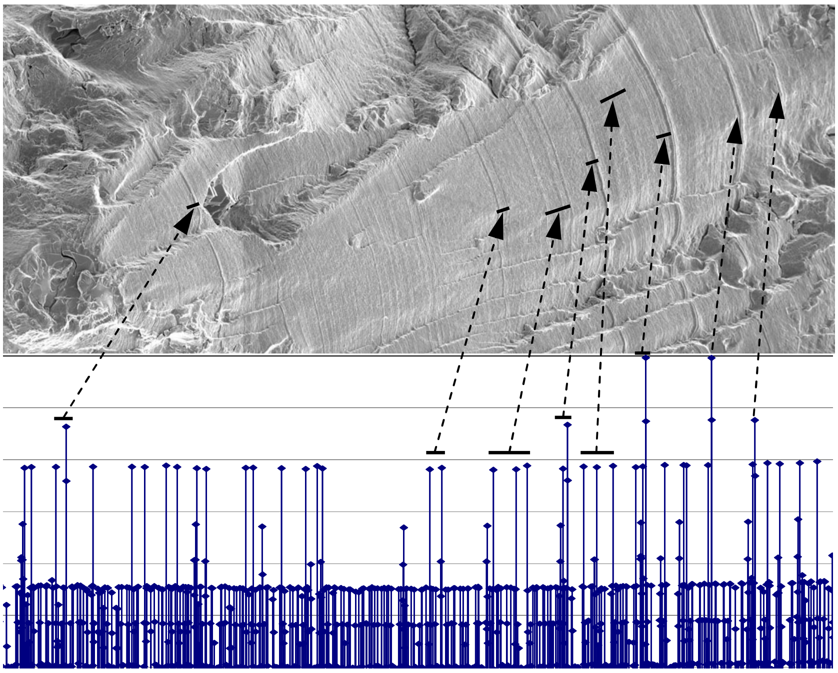
Nationaal Lucht- en Ruimtevaartlaboratorium

National Aerospace Laboratory NLR

Executive summary

Review of aeronautical fatigue investigations in the Netherlands during the period March 2009 - March 2011



Report no.
NLR-TP-2011-123

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Report classification
UNCLASSIFIED

Date
April 2011

Knowledge area(s)
Levensduurbewaking en onderhoud van vliegtuigen
Vliegtuigmaterialen
Vliegtuigmateriaal- en schadeonderzoek
Testen van vliegtuigconstructies en -materialen

Descriptor(s)
ICAF
Fatigue
Structural Integrity
Damage Tolerance

Description of work
This report is a review of the aerospace fatigue activities in the Netherlands during the period March 2009 to March 2011. The review is the Netherlands National Delegate's contribution to the 32nd Conference of the International Committee on Aeronautical Fatigue (ICAF), 30 and 31 May 2011, Montreal, Canada.

This report has been prepared in the format required for presentation at the 32nd Conference of the International Committee on Aeronautical Fatigue (ICAF), 30 and 31 May 2011, Montreal, Canada.

Review of aeronautical fatigue investigations in the Netherlands during the period March 2009 - March 2011

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This publication has been refereed by the Advisory Committee AEROSPACE VEHICLES.

Customer	National Aerospace Laboratory NLR
Contract number	-
Owner	National Aerospace Laboratory NLR
Division NLR	Aerospace Vehicles
Distribution	Unlimited
Classification of title	Unclassified
	March 2011

Approved by:

Author	Reviewer	Managing department
29/4/2011	29/4/11	9/5/'11

Summary

This report is a review of the aeronautical fatigue activities in the Netherlands during the period March 2009 to March 2011, and is the National Delegate's contribution to the 32nd Conference of the International Committee on Aeronautical Fatigue (ICAF), 30 and 31 May 2011, Montreal, Canada.

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Review of aeronautical fatigue investigations in the Netherlands during the period March 2009 – March 2011

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1.1 INTRODUCTION

The present review gives a summary of the work performed in the Netherlands in the field of aerospace fatigue during the period from March 2009 to March 2011. The contributions to this review come from the following sources:

- The National Aerospace Laboratory NLR
- The Faculty of Aerospace Engineering, Delft University of Technology, TUD
- Fokker Aerostructures

The names of the principal investigators and their affiliations are given at the end of the title for each topic. The format and arrangement of this review is similar to that of previous years.

1.2 LOADS

No work in relation to this category has been reported.

1.3 STRUCTURAL LOADS/USAGE/HEALTH MONITORING

1.3.1 CH-47D "Chinook" airframe structural integrity programme (M.J. Bos, NLR)

Prompted by severe structural maintenance issues, the Royal Netherlands Air Force (RNLAF) has tasked the National Aerospace Laboratory NLR with developing an airframe loads & usage monitoring programme for their CH-47D helicopter fleet. After an initial pilot phase during which the technical and operational possibilities were explored, a routine programme named "CHAMP" (CHinook Airframe Monitoring Programme) was started in 2007. In addition to a fleet wide installation of a Cockpit Voice & Flight Data Recorder for the collection of the relevant parameters from the digital avionics data bus, two airframes have been equipped with a state-of-the-art data acquisition system and nine strain gauges each, which are recorded at a high sample rate. The locations of these strain gauges are indicated in Figure 1. All data processing is performed off-board; no on-board data reduction is done. This has led to an extensive and ever-growing database that can be used to conduct analyses that go beyond those traditionally performed within a loads & usage monitoring programme.

The structural integrity concept that forms the basis of CHAMP has been dubbed the "stethoscope method". This method centres on the development of Artificial Neural Networks that use the recorded data bus parameters to predict internal loads at the strain gauge locations. After the creation of such "virtual strain gauges", the actual strain gauges can be relocated to monitor other key structural locations. Successive relocation of strain gauges finally results in a usage monitoring system that - in the long run - will be invaluable for structural life cycle management. More information regarding CHAMP is provided in [1].

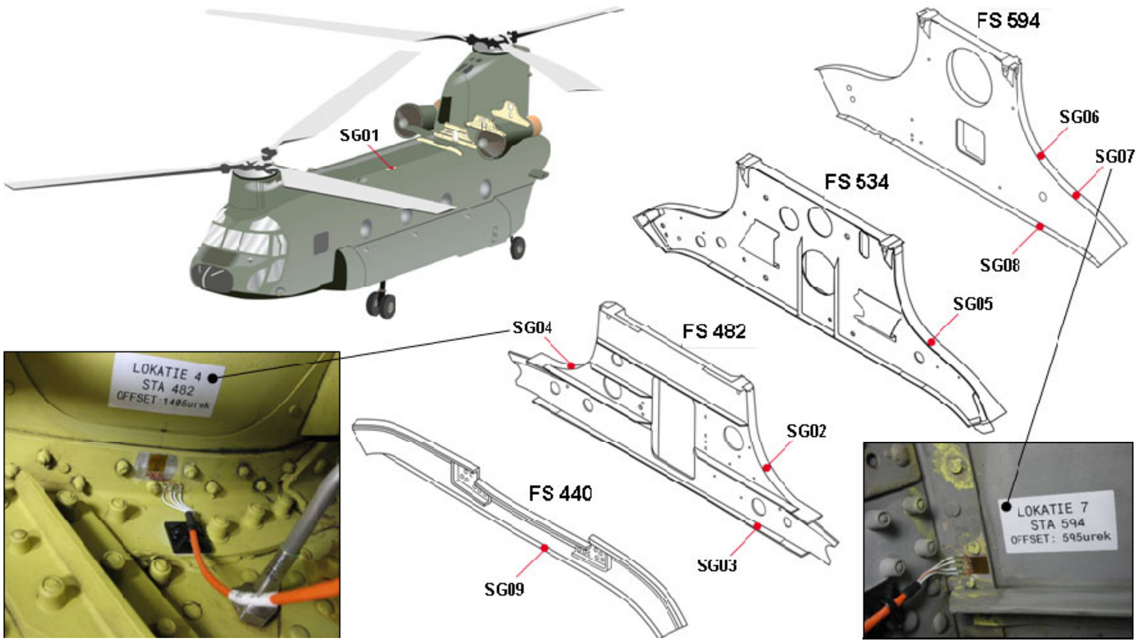

Figure 1: Location of the nine strain gauges used in CHAMP.

1.3.2 CH-47D "Chinook" referred flight parameters (A. Oldersma, NLR)

As part of the Chinook structural integrity programme, the RNLAF routinely collects flight data for its CH-47D transport helicopter fleet. The flight data are recorded by the Cockpit Voice and Flight Data Recorder, which is fleet-wide installed. These recorded flight parameters serve as the input for Flight Regime Recognition (FRR). The Flight Regime distribution, together with other usage statistics, such as weight distribution classes and speed classes, reveals differences in usage both in the Netherlands and during out-of-area operations.

However, since the actual atmospheric conditions (i.e. ambient air pressure and temperature) differ from day to day, actual performance data can seldom be compared directly. In order to be able to pool data and make comparisons, methods exist to measure the steady-state performance of turbine-engined helicopters in terms of non-dimensional parameters (also known as compound, referred or reduced parameters). In the flight test community this referred parameter technique is a widely used and accepted method, but in the field of loads and usage monitoring the referred parameter method seems still to be unrecognised. This is an important omission, since the referred parameter technique enables, for example, the use of (referred) weight classes which are independent of actual atmospheric conditions. Pooling and comparisons of data, based on performance, give added value to the current usage and FRR statistics.

Actual flight parameter recordings were used to verify the pooling of Straight and Level (S&L) flight and hovering performance data. Trends in the data clearly show the weight-related effect on performance. Unfortunately, in the case of S&L flight the operational data (especially out-of-area flights) showed large scatter, preventing the compilation of unambiguous performance curves per weight class. On the other hand, the referred hovering performance data showed less scatter: converting the flight manual hovering chart into a referred hovering chart has been shown to enable helicopter weight predictions using the recorded torque and wheel height.

1.3.3 Structural fatigue load & usage monitoring of F-16 aircraft (F.C. te Winkel, NLR)

Structural load monitoring of the RNLAF/F-16 fleet is done routinely by the NLR since 1990. During the 1990s a new fatigue monitoring system specified by the NLR was developed by RADA by extending their ACE pilot debriefing system with loads and usage monitoring functionality: FACE (Fatigue Analysis & Air Combat Evaluation system).

The main features of FACE are (a) an increase to five strain gauge locations, two indicative for wing root and "outer" wing bending, two at the rear fuselage dealing with horizontal and vertical tail loads and one in the fuselage centre section indicative for fuselage bending; (b) a flexible selection of flight, engine, and avionics parameters available via the MUX-BUS; and (c) fleet-wide implementation (since 2003) allowing more extensive load monitoring of each individual aircraft and systems.

The Crack Severity Index (CSI) is used as damage indicator. The CSI was developed by the NLR and is a relative measure of damage. For the F-16 a value of 1.0 means fatigue damage according to the reference usage and loading environment used to generate the current inspection schedule (Fleet Structural Maintenance Plan FSMP). The CSI method accounts for interaction effects between large and small load cycles (or between severe and mild flights). The fatigue damage of a flight therefore depends on the severity of the flights flown before. The CSI can therefore be used as an indicative measure for ASIP (Aircraft Structural Integrity Program) control points, since it is a relative figure between the actually measured and the LM Aero reference usage and loading environment that was used to obtain the current inspection schedule.

Switching from a sample load monitoring programme to fleet-wide individual load monitoring, combined with the flexible way of measuring a wide range of additional flight parameters, required a different approach to handling data and/or information. The NLR developed a custom-made information system for storing, managing, and analyzing the collected measured flight data with the on-board FACE system together with the administrative operational flight data from each aircraft (obtained from the RNLAF computerized maintenance/debriefing system IMDS) and a subset of data retrieved from OMISKLu (Operational Management Information System KLu). The use of this centralized information system enables efficient data handling for both ad hoc analysis and the generation of routine status reports for fleet management purposes.

A similar loads and usage monitoring programme has been implemented for the Belgian Air Force, whereby the information system was modified to facilitate both air forces. Intensive L/ESS measuring campaigns were carried out to collect Loads and Environment Spectrum Survey data with the FACE system, to enable LM Aero to provide an update of the Fleet Structural Maintenance Plan. In 2009 a contract was awarded by the Chilean Air Force (FACH) to facilitate a loads and usage programme for the FACH F-16 Block 15 aircraft (former F-16s RNLAF). Modifications were made to the information system to set up a loads and usage monitoring programme for the FACH.

Proven benefits from the FACE system and dedicated F-16 information system are:

- Valuable instrumentation package for Force Management purposes as part of the Aircraft and Engine Structural Integrity Programs ASIP and ENSIP
- Generation of test sequences based on actual measurements
- System has been a valuable source for mishap investigation
- Very flexible instrumentation package (each aircraft in theory a "unique" test aircraft)
- In recent years more ad hoc recordings are made to support several research programmes:
    - Detailed engine FAULT-code recordings
    - Flight departure margin study
    - MFOQA (Military Flight Operations Quality Assurance) trial 2007 with 18 dedicated aircraft instrumented with dedicated measurement configuration
- Information system is frequently used as a study for development of next generation decision support and simulation tools

Capabilities in progress/under development:

- Cumulative CSI for 5 strain gauge locations

- Linkage of CSIs to (clusters of) control points
- Start of follow-on MFOQA trial 2011
- F-16 dynamic effects including limit cycle oscillation (LCO) tracking capability
- Dedicated measurement campaign to evaluate/validate model Landing Gear Damage Indicator
- For general analysis, reporting, and visualization of life cycle information for the F-16, a transition will be made to SUSTAIN (SUpporting the suSTainment chAIN). SUSTAIN is a generic, weapon system independent, in-house developed secure software environment for running a variety of tools for fleet management purposes.

1.3.4 Pilatus PC-7 trainer usage monitoring (A. Oldersma, NLR)

The RNLAF has 13 Pilatus PC-7 Turbo Trainers in service. These aircraft are used by pilot trainees in the Basic Military Pilot Training programme. The PC-7 is intended to remain in service until the year 2023. According to current usage this implies the need for an airframe life extension programme from 12,000 flight hours to 15,000 flight hours. The data needed for such a programme will be collected in a dedicated loads and usage monitoring programme which has recently started. The actual loads/usage spectrum will be determined for a number of critical locations. Comparison with the design spectrum will be done to reveal potential problem areas with respect to structural life.

Figure 2: Pilatus PC-7 trainer (left) and the ACRA KAM-500 system (right).

The loads/usage monitoring involves the installation of an accelerometer and a number of strain gauges at critical locations defined by Pilatus. For data acquisition an ACRA KAM-500 system will be used. A similar system is used in the Chinook load monitoring programme. After a one year measurement campaign, the usage will be expressed as normal acceleration Nz and strain gauge exceedance curves.

1.3.5 C-130H-30 “Hercules” loads & usage monitoring (M.J. Bos, NLR)

The RNLAF C-130H-30 fleet is used in a much different way from that originally anticipated at the time of acquisition. Out-of-area operations, such as those performed under the ISAF flag in Afghanistan, severely stress the aircraft and adversely affect the airframe service life. The NLR has developed a loads and usage monitoring system called HOLMES (Hercules Operational Life Monitoring & Evaluation System) that brings together measured flight data and flight administrative data from various sources. The collected information is used to compute the expended fatigue life of critical areas of the airframe. HOLMES is operated by the NLR. Routine and *ad hoc* reports are issued on a regular basis. The information from these reports is used by the RNLAF to take informed decisions about fleet life management.

In 2010 HOLMES has been complemented with an easy-to-use graphical tool called SHERLOCK (System to provide HERcules Operational life Consumption Knowledge) that enables the RNLAF C-130 weapon system manager to evaluate the severity of flight operations in terms of accrued airframe fatigue damage. SHERLOCK can be used as a prognostic tool to assess planned missions. In addition it can be used as a post flight analysis tool, to view the flight profile and to compute the accumulated fatigue damage in a number of airframe components.

1.4 FIBRE/METAL LAMINATES & HYBRID MATERIALS/STRUCTURES

1.4.1 Fatigue crack growth in Fibre Metal Laminates under variable amplitude loading (S.U. Khan, R.C. Alderliesten, TUD)

In the previous national review, research was presented on fatigue crack propagation in Fibre Metal Laminates (FMLs) under variable amplitude loading. At that time the focus was on determining the delamination growth behaviour under variable amplitude loading conditions, with the objective of developing a model for incorporation into the mechanistic models that describe fatigue crack growth in FMLs [2]. In the period 2009-2010, the interaction between delamination progression and crack growth in the metallic layers has been assessed, and the interaction between the load interaction phenomena in the two materials has been captured [3,4]. Currently, a PhD thesis is being written and will be defended later this year (2011).

1.4.2 Residual strength of Fibre Metal Laminates containing fatigue cracks (R. Rodi, R.C. Alderliesten, TUD)

As part of an investigation into the residual strength mechanisms in FMLs, a theoretical model has been developed that describes the damage growth under quasi-static loading. This model includes the large scale plasticity effects based upon critical crack opening angles ($CTOA_c$). The damage cases considered are (i) the accidental damage scenarios earlier addressed in an empirical model (R-curve) by De Vries, and (ii) the case of a fatigue crack with delaminations as addressed, for instance, by Khan and Wilson. In the residual strength sequence, mechanisms such as quasi-static delamination and metal crack growth, fibre bridging and fibre failure are involved during failure of the laminate [5,6]. All these mechanisms and their interactions have been described by means of elastic-plastic fracture mechanic tools and implemented into an analytical prediction model. Currently, a PhD thesis is being written and will be defended later this year (2011).

1.4.3 Fatigue damage growth in arbitrary hybrid Fibre Metal Laminates (G. Wilson, R.C. Alderliesten TUD)

A theoretical mechanics-based model of crack and delamination growth under cyclic loading has been developed for FMLs of arbitrary lay-up, e.g. asymmetric, different thickness layers, different metal alloys and fibres in the same laminate; arbitrary damage, different crack lengths and delamination shapes throughout the thickness; and tension, bending, or combined tension-bending loadings. Inputs to the model are only the constituent material properties, laminate configuration, and loading conditions. The model relies on modelling of the bridging stress distribution on individual interfaces within the laminate (see Figure 3), whereas previous bridging models average the bridging distribution through the laminate. Based on the bridging stresses, the stress intensity factor seen by each cracked layer is determined and the strain energy release rates along every interface are calculated. These parameters enable determining the crack and delamination growth rates. Verification testing is underway (see the sample data in Figure 4), and papers describing each aspect of the model have been submitted to journals in the field. A PhD thesis on this topic will be defended in the fall of 2011.

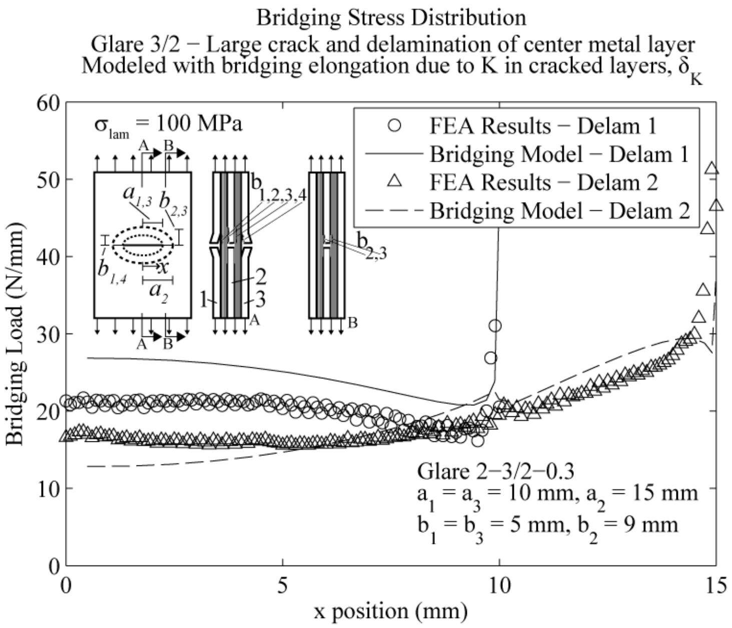

Figure 3: Comparison between bridging stresses from the arbitrary laminate bridging model and FEA for a GLARE laminate with a longer crack in the middle layer and larger delaminations in the innermost interfaces.

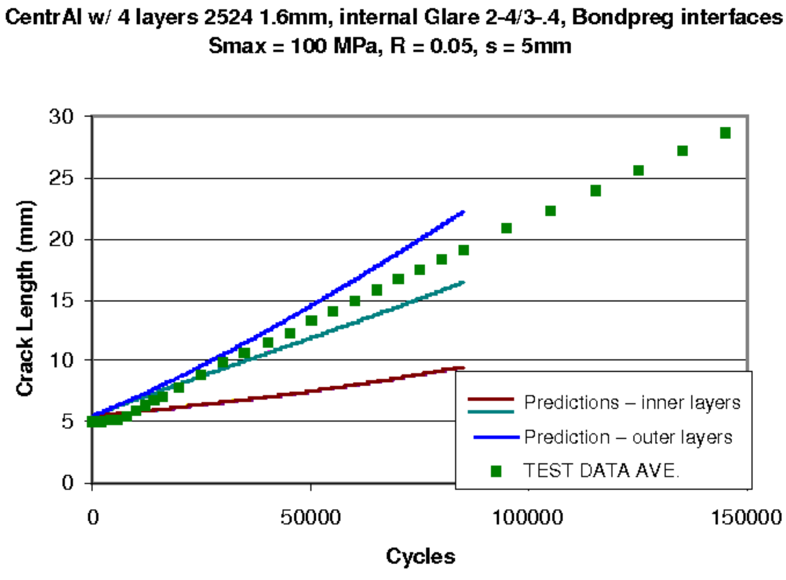

Figure 4: Sample verification test result for a CentrAl laminate. Only outer layer cracks were measured during testing: teardown is planned in order to measure internal crack lengths.

1.4.4 1441 Fibre Metal Laminate (C. Lucas, R.C. Alderliesten, TUD)

Application of the 1441 aluminium alloy in the FML concept was evaluated by testing 1441-FML coupons under static and fatigue conditions. An important result of the tensile tests was that the mechanical properties of the 1441-FML are generally better than those of the Standard GLARE. The 1441-FML stiffness was observed to be 5% higher and the yield strength even 20% higher than for GLARE. Although the required stiffness improvement can be achieved with the 1441 alloy, the implications with respect to damage tolerance had to be considered. For this reason, fatigue crack growth tests on small coupons and large flat panels were performed and compared with GLARE data and predictions.

Figure 5 gives an example of the flat panel comparisons between the 1441-FML and Standard GLARE. The case illustrated in this figure consists of cracks emanating from a large sawcut ($2a_0$=75mm). The key observation for this figure is that at lower maximum stress levels (i.e. below 100 MPa) the crack growth for both FMLs is similar. However, at higher stress levels, the crack growth resistance of the 1441-FML appeared to be lower compared to that of GLARE. From the coupon tests it was further observed that the fatigue crack growth performance of the 1441 alloy is less than that of the standard 2024 alloy over the entire growth rate regime.

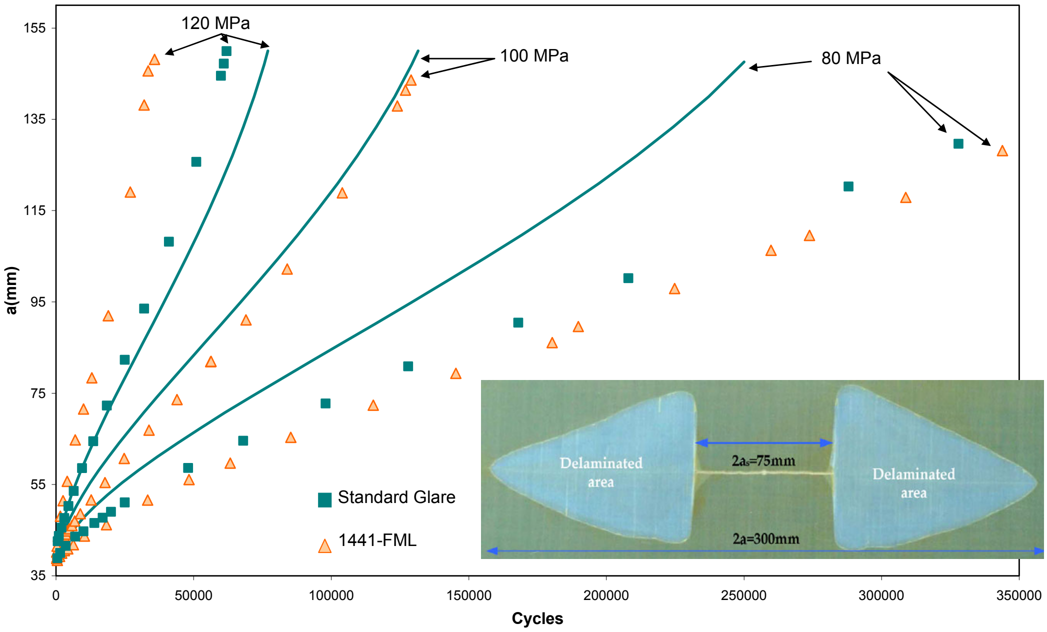

Figure 5: Comparison between constant amplitude fatigue crack growth for Standard GLARE and 1441-FML3-3/2-0.4 with R = 0.1

1.4.5 Effective bending and stress concentrations in thick Fibre Metal Laminates (A. Keizer, R.C. Alderliesten, TUD)

As part of a study on the development of FML technology for lower wing skin panels, the effect of thickness steps (ply drop-offs) and bending are being evaluated for FMLs with different aluminium layer thicknesses.

(1) Determination of the stress concentration factor (K_t) in FMLs containing thickness steps.
Digital Image Correlation (DIC) has been used to analyze the strains in FML specimens containing thickness steps. Both symmetric and asymmetric specimens were used, in order to include the influence of secondary bending. The stress concentration at a doubler run-out area is of special interest, since it influences crack initiation in the outer aluminium layer and/or delamination initiation at the bond line. Tests were performed on specimens with both adhesive interfaces as well as prepreg interfaces. The comparison between the measured strains and finite element analysis (FEA) has provided valuable insights. An illustration is given in Figure 6, where the longitudinal strains determined with the DIC are compared with strains predicted by FEA.

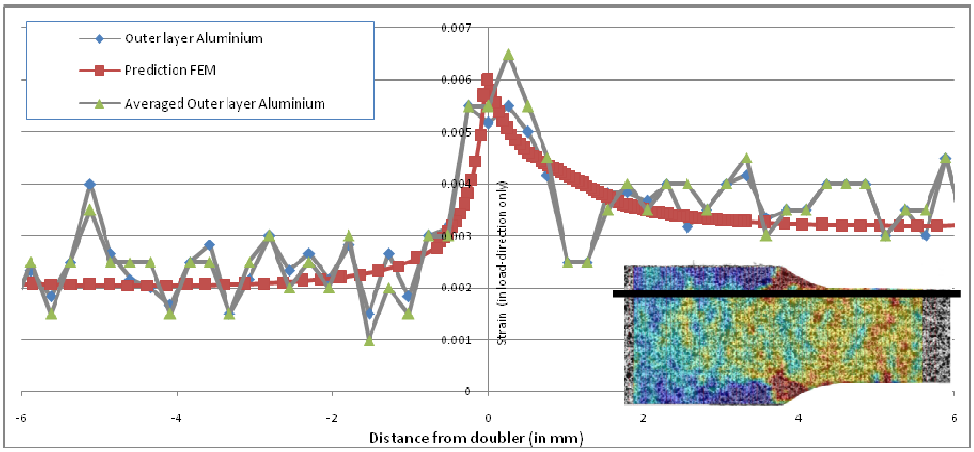

Figure 6: Comparison between DIC-determined and FEA-predicted longitudinal strains in the top metal layer (black line) underneath a thickness step.

(2) <u>Secondary bending of FML specimens</u>.
DIC has been used also to determine the bending shape to be compared with the deformation predicted by FEA. This was done by making several pictures of the specimen, such that the entire length was covered. After merging the pictures an ‘exaggerated’ view is obtained, see Figure 7. In addition, the bending factor can be obtained by comparing the strains in both outer layers. These strains can subsequently be compared with predictions made with the Neutral Line Method (NLM). Although this work is still in progress, the results appear encouraging.

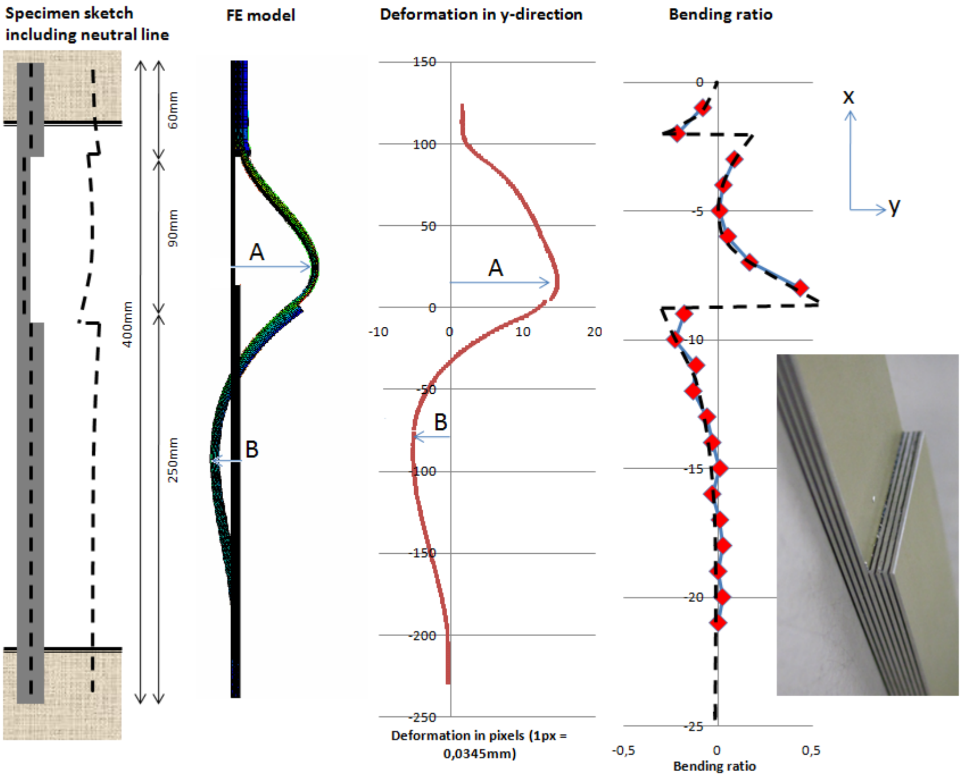

Figure 7: Illustration of the DIC evaluation of secondary bending of a GLARE doubler run-out specimen.

1.4.6 Design tools and guidelines for thick Fibre Metal Laminate wing design (C. Vinke, R.C. Alderliesten, TUD)

The focus of this research is to identify the benefits of increasing the thicknesses of individual aluminium layers. Application of thicker metallic sheets in FMLs generally has a detrimental influence on both delamination growth and crack growth performance. One way proposed for overcoming this problem is to provide additional adhesive layers at the interfaces between prepreg and metallic layers: this technique is often called Bondpreg. However, Bondpreg decreases the static performance (stiffness and strength) of the laminates. Figure 8 illustrates the correlation between the increase in crack growth resistance and the decrease in laminate stiffness.

Another way to address the problem is to increase only the thickness of the outer metal layers, while keeping the inner layers thin. The influence this approach may have on the effective stress concentration factor K_t is being investigated to assist the design of ply drop-offs and stringer run-outs on FML skin panels. This is because it has been observed that thinner aluminium layers at the outside of an FML result in higher effective stress concentrations.

Structural tests are being performed to investigate the effect of aluminium sheet thickness at stringer run-out locations. Two different base laminates will be used: FML 6/5-0.5, and FML 2/1-0.5 enclosed by two 1 mm-thick outside layers. Stringers will be bonded to these base laminates and will have different run-out angles. The geometry is illustrated in Figure 9. The results will become available in the course of 2011.

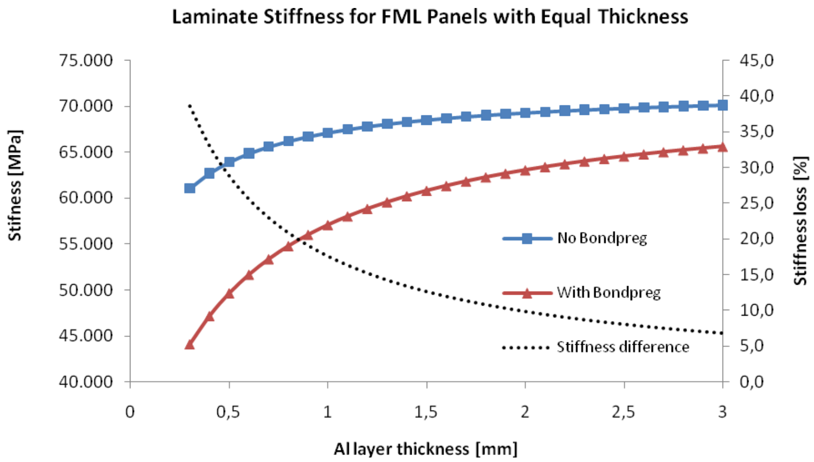

Figure 8: Influence of Bondpreg application to the stiffness of the laminate.

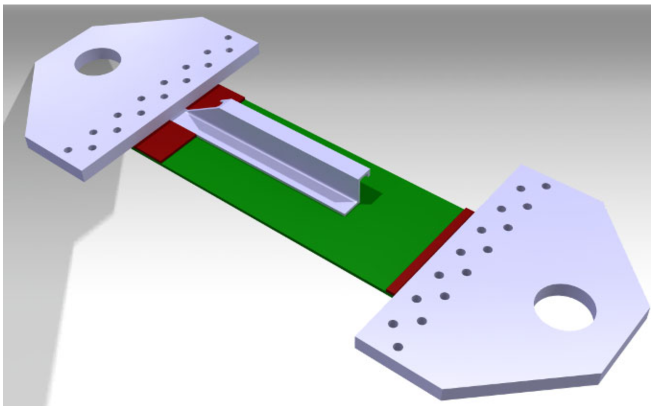

Figure 9: Illustration of the stringer run-out specimen configuration.

1.4.7 Novel Internal Tear Strap Concept for Hybrid Fibre Metal Laminate Panels (C.D. Rans, R. Rodi, R.C. Alderliesten, TUD)

A novel tear strap concept for hybrid FML skin panels has been developed and tested at the TUD. The concept mimics the effects of bonded titanium (or another material) tear straps on thin skin panels. This simulation is done by replacing the added stiffness of the bonded strap with the added stiffness from a local material substitution within the FML. For example, Figure 10 illustrates a local substitution of the 0° glass fibre layer in a tradition GLARE laminate by a 0° carbon/epoxy layer in a region analogous to a bonded strap.

This novel concept is called ITS-Glare, and it has been evaluated for potential improvements of the damage tolerance of commercial FML GLARE. The ITS-GLARE concept was shown to result in a small improvement in crack initiation performance; up to an 80% reduction in crack growth rate (in the vicinity of the ITS); and an increase in residual strength ranging from 15-25% (depending on the location of the crack relative to the ITS and relative to an equivalent GLARE laminate. These results represent a significant improvement in damage tolerance with respect to Standard GLARE.

The ITS-GLARE concept has been proposed as an alternative to bonded titanium tear straps (commonly applied on metallic aircraft structures) and interlaminar doubler straps (currently applied on the Airbus A380). The major benefit of ITS-GLARE is that it results in improvements in damage tolerance (by means analogous to the two other alternatives) *without resulting in local thickness variations* in the finished structure. The absence of thickness variations eliminates the need to joggle stringers and match-machine the frames to which the skin panel would be joined. This represents a potentially significant improvement in manufacturing and assembly costs for stiffened aircraft structures and could help offset the higher material costs of FMLs relative to monolithic metals for both wide-body and narrow-body aircraft applications.

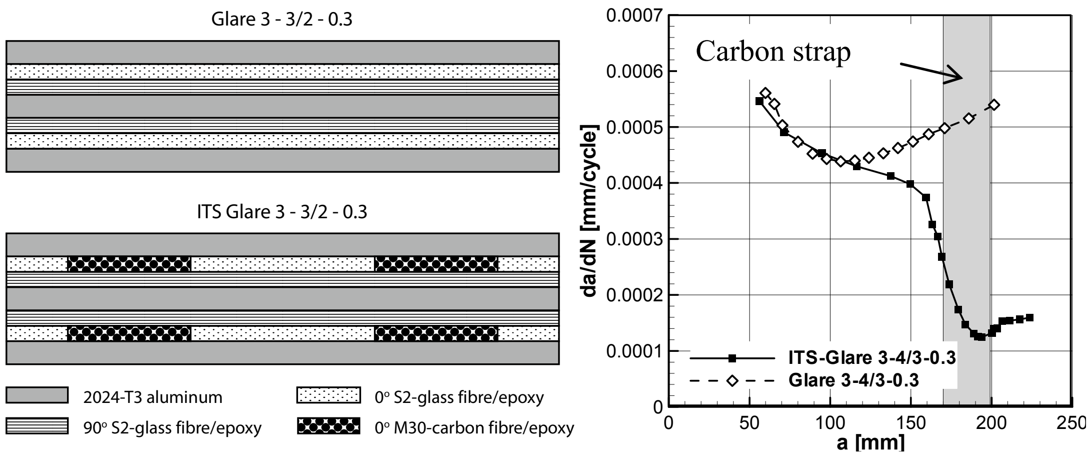

Figure 10: Illustration of the internal tear strap (ITS) GLARE concept with representative fatigue crack growth improvements.

1.5 COMPOSITES FATIGUE AND DAMAGE (NDI) STUDIES

1.5.1 Fracture of composite aerospace structures (J.H. Heida, NLR, B.H.A.H. Tijs, Fokker Aerostructures)

Composites are increasingly used in structural engineering. For the aerospace industry it is becoming more and more important to understand the damage mechanics that can affect the service life of structural composites. Several Fokker Aerostructures test programmes that include state-of-the-art composite structures were performed over the last few years at the NLR testing facilities. Examples of these test programmes are the full-scale test of the NH90 helicopter tail module, the full-scale test of several Gulfstream G650 tail components and several coupon and detailed tests. These programmes contributed much to acquiring expertise in the field of testing composite structures.

Besides experience in testing composite aircraft structures, Fokker Aerostructures and the NLR also started investigating the typical fractographic features of composites that can develop during the service life of an aircraft, either by fatigue or static fracture. Fractography of composites is not as straightforward as for isotropic materials. Post-failure mechanical damage, which obscures the original fracture morphologies, can make the fracture surfaces difficult to interpret. Figure 11 shows an example of post-failure mechanical damage that covered the fracture surface with debris. The complex build-up of composites also results in a more complex fracture surface. Figure 12 shows a Z-shaped crack through a sandwich construction in multiple directions.

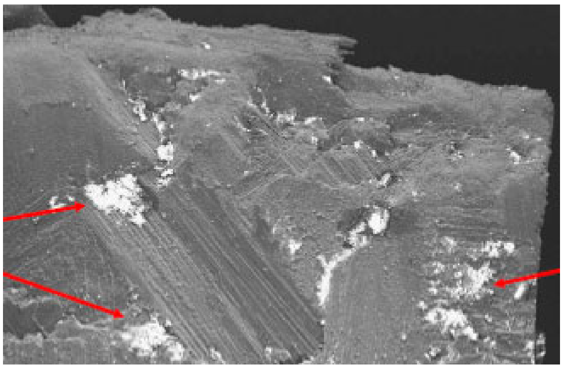

Figure 11: Post-failure mechanical damage and debris.

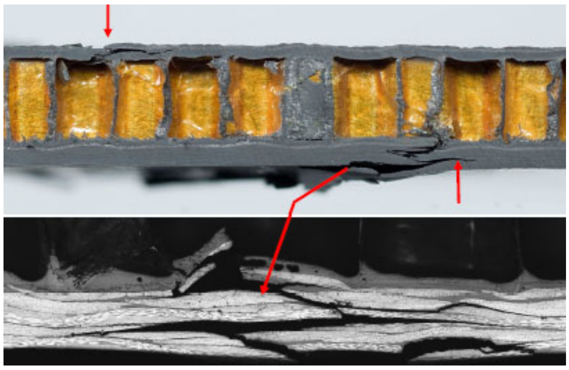

Figure 12: Z-shaped crack through sandwich construction.

During the investigation special attention was paid to identifying typical features that are caused by fatigue, such as matrix rollers, striations and the shapes of the cusps. Typical fibre failures for tensile and compressive failure, as discussed in [7], were also found on some of the fracture surfaces. The fibres that failed in compression showed typical chop marks and the fibres failing in tension showed radial patterns, see Figure 13 and Figure 14.

Non-destructive testing methods were also applied. Methods such as ultrasonic, C-scan and X-ray inspections contributed to better insight into the failures. The resulting information, for example the delamination sizes and crack shapes, was then used in combination with the fractographic results to determine the failure mechanisms.

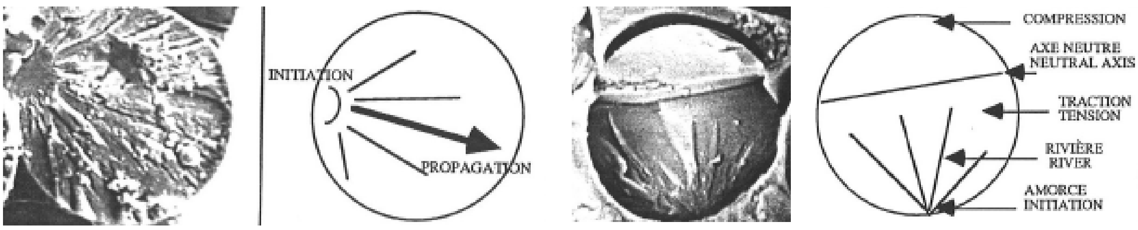

Figure 13: Typical fibre failures, from [7].

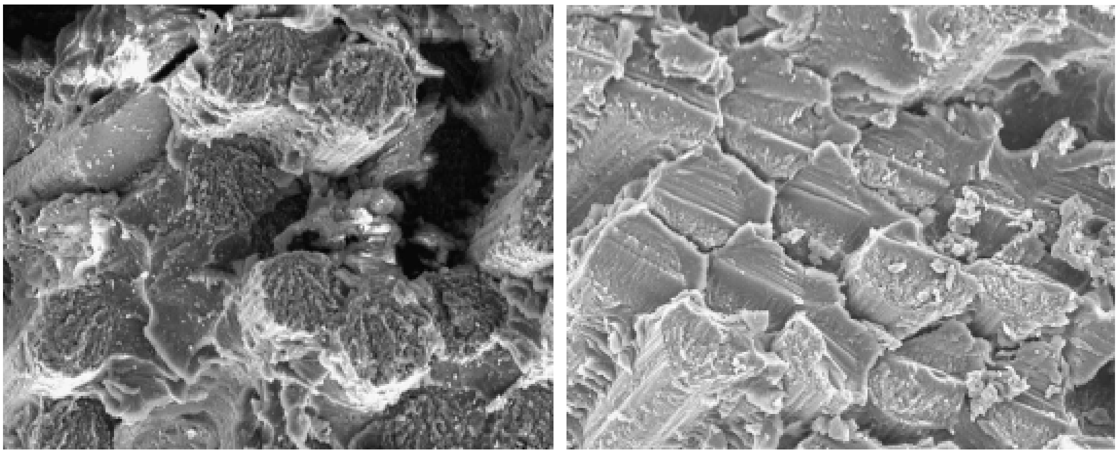

Figure 14:. Fibre failure in tension (left) and compression (right) found during the investigation

1.5.2 Non-destructive inspection of composite aerospace structures (J.H. Heida, NLR)

The NLR carried out a National Technology Programme (NTP) on “Inspection and repair techniques for composite structures”. The aim of the NTP was to develop tools and procedures that enable cost-effective inspection and repairs of composite structures for the next generation aircraft and helicopters, with spin-off to naval applications. Part of the NTP was an evaluation of promising, mobile non-destructive inspection (NDI) methods for the in-service inspection of composite aerospace structures.

The evaluation made use of carbon fibre reinforced specimens representative for primary composite structures in both current and next generation aircraft and helicopters. The specimens were solid laminates and sandwich structures with and without stiffeners, and included relevant damage types such as impact damage, interply delaminations and disbonds. A range of mobile NDI methods were evaluated including visual inspection, vibration analysis, ultrasonic inspection (including different phased array methods), shearography and thermography inspection , see Figure 15.

Figure 15: NDI methods evaluated for the inspection of composite aerospace structures.

An important aspect of the evaluation was the NDI capability for detection, sizing and depth estimation of the defects in the specimens. Further evaluation parameters were the portability of equipment, field of view, couplant requirements, speed of inspection, level of training required, and the cost of equipment. An overview of the advantages and limitations of the different NDI methods was made, and general guidelines for the in-service inspection of composite aerospace structures were given.

1.5.3 Methods for Analyzing the Lifetime Aspects of Composite Structures (W. van den Brink, G. van de Vrie, NLR)

Composite materials are increasingly used in the Royal Netherland Air Force (RNLAF) fleet of helicopters and fixed-wing aircraft. To gain more knowledge about the materials behaviour in service, the National Technology Programme (NTP) "Methods for Analyzing the Lifetime Aspects of Composite Structures" (MALACC) has been initiated. The programme consists of an experimental and a numerical part, whereby the focus is on understanding the physical behaviour of damage initiation and progression, and the fatigue performance of composite materials and structures.

In general, the simulation of structural behaviour involves a compromise between the desired level of detail and the size of the resulting model. For instance, a composite is composed of a large amount of individual fibres that could be included in the simulation. In most cases this approach is too detailed and the simulation model will grow out of bounds. Therefore the NTP project was set up with three different scales with their corresponding simulation and experimental approaches. These three scales will be developed to a level where we have confidence in the results, see Figure 16. A coupling between the different length scales is extensively described in the literature and will be dealt with in future work.

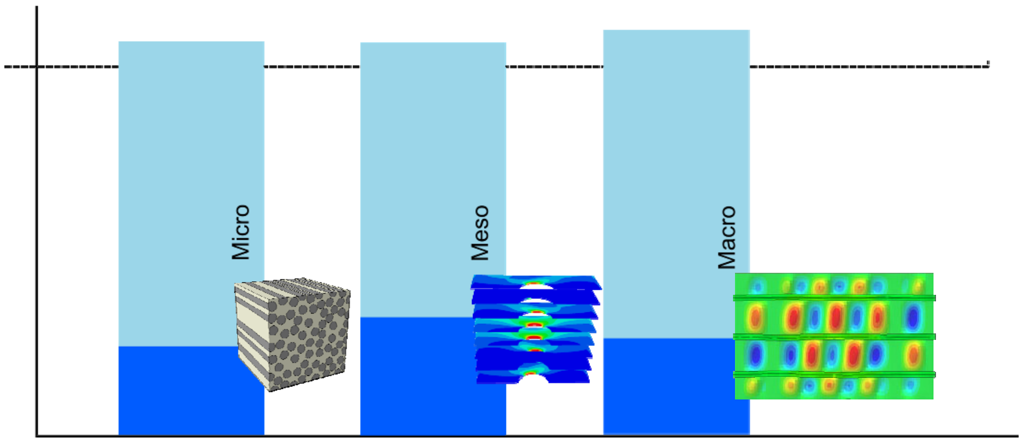

Figure 16: Division of the NTP into three scales: micro-, meso- and macro-mechanical.

<u>Micro-level</u>

On the first and smallest scale the fibres are simulated individually within a micromechanics approach. Damage behaviour is also observed in experiments at this scale by using an optical microscope. For example, several tests are performed to understand the delamination growth with a four-point bending test – see Figure 17. The general application for this approach is to understand the delamination behaviour when the interface between the fibres and matrix has to be investigated. Also, the influences of imperfections like resin-rich regions, voids or microcracks near the delamination front will be investigated. Disadvantages of this micro-level approach are the complexity of the model and the difficulty in scaling up this approach to larger structures.

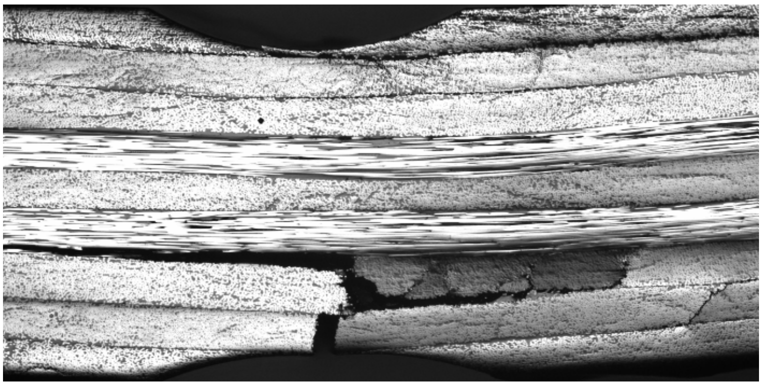

Figure 17: Through-thickness view of a composite coupon after a four-point bending test.

Meso-level

A test programme has been performed to characterize the basic properties of a fabric carbon fibre material system. Standard tensile, compression, shear, DCB and ENF tests have been done. Also, since impact damage is a major problem, a series of impact tests has been performed on the same material system, but with different layups. More impact tests are planned, including some additional tests for obtaining the (static) contact behaviour. In addition, experiments will be done with specimens subjected to cyclic loading.

The focus in meso-level simulation is on the use of finite elements, although analytic methods also have been looked at (especially for impact). The aim here is to be able to perform calculations with good accuracy but also manageable computational time and model size. In general, the distinction has been made between fracture mechanics, where individual cracks are modelled, and damage mechanics where the stiffness of damaged parts is reduced. Several (commercially) implemented damage models have been investigated, for example the Virtual Crack Closure Technique (VCCT), cohesive modelling, eXtended Finite Element Method (XFEM), Hashin in combination with material degradation, and a dedicated Abaqus fabric VUMAT with material degradation.

Macro-level

This level/scale is that of a structural component all the way to a complete aircraft. The goal for the models on this level is to derive some sort of damage indicator where the inputs are loads and acceleration of an aircraft and the output is the residual life. Work in this area will be done in the future, and will be based on analytical formulas that describe trends in damage growth behaviour as developed in the past for metallic aircraft.

1.5.4 Delamination growth in CFRP under Mode-I Fatigue Loading (R. Khan, C.D. Rans, TUD)

Saving weight is a major issue in the design of aerospace structures. This has led to an ever-increasing use of composite materials, due to their higher specific strength and stiffness. However, the possible presence of delaminations and their growth under fatigue is an obstacle to the use of composites for primary load-bearing aerospace structures.

Delamination growth in carbon fibre reinforced plastic (CFRP) composites has been researched for several decades in order to assess their reliability and performance. However, delamination behaviour is complex and there is as yet no clear understanding of the delamination mechanism(s). Owing to this lack of understanding, the development of a mechanistic model for predicting delamination growth remains a difficult task: the existing models are either empirical or semi-empirical [8].

The objective of the present project is to improve the understanding of the mechanism of delamination growth in composites in order to develop a mechanistic model for predicting delamination growth under Mode I fatigue. Two approaches are used:

(1) Identify the fatigue parameters which effect the delamination growth.

(2) Identify the mechanism by which these parameters interact in the delamination growth process.

An experimental study was conducted to investigate the effect of fatigue R-ratio on Mode I fatigue delamination growth in Double Cantilever Beam (DCB) specimens made of unidirectional carbon/epoxy. The results show that the use of a correct definition of Strain Energy Release Rate (SERR) range for the delamination growth characterization separates the effects of R-ratio and SERR range. For a constant SERR range the delamination growth rate is higher for higher R-ratios, as shown in Figure 18a. SEM investigations of the fracture surfaces under different R-ratios revealed no difference in the surface topographies. A Laser Confocal Electron microscope was used to measure the roughnesses of the fracture surfaces. The mean roughness values were similar for different R-ratios.

Using digital image correlation technique the effect of crack closure was investigated as an explanation for the R-ratio effect on delamination growth. Crack closure was observed for the lower R-ratios: however, its magnitude was too small to consider it as fully responsible for the R-ratio effect. This is demonstrated by Figure 18b, which shows the delamination growth vs. SERR after correcting for the effect of crack closure.

The project currently focusses on observing the evolution of microcracking ahead of delamination growth in DCB specimens. The objective is to identify any differences in microcracking owing to different R-ratios. A SEM is used for the observations, and initial tests have shown some evidence of the existence of microcracking.

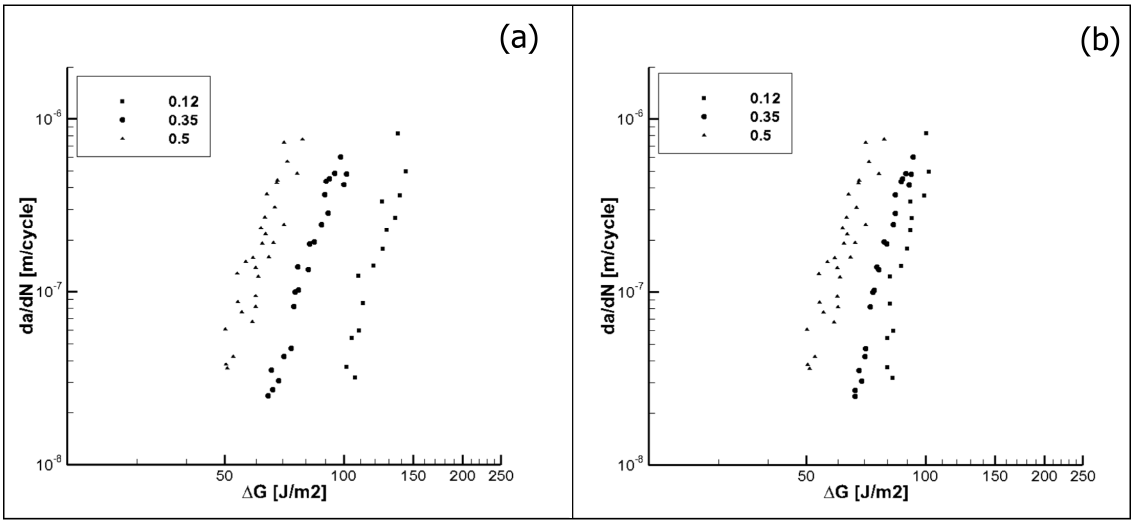

Figure 18: Delamination growth rate vs. SERR range for various R-ratios (a) without crack closure correction (b) with crack closure correction.

1.5.5 Strain energy release rate analysis of bonded composite scarf repairs (T. van Vugt, C.D. Rans, TUD)

At present, aerospace composite structures are apparently certified according to approaches similar to the traditional safe life methodology used for metallic structures, i.e. using no damage growth concepts. This approach is a significant drawback in terms of optimum design and structural weight. To fully explore the potential of composites it will be necessary to develop damage tolerance approaches similar/analogous to those for metallic structures. This development will also be driven by safety considerations. Proper safe application will not be possible without understanding the damage growth mechanisms and developing methods for predicting the damage growth.

Recently a generic concept has been proposed to predict delamination growth in FMLs and composites based on the Energy Release Rate (ERR). The first validation activities on FMLs show promising results. Currently, the purpose of this study is to show that this generic ERR-based concept is also valid for composites. A 2-dimensional representation of a CFRP wet lay-up scarf joint repair was selected as an application for this extension. The joint is between a pre-impregnated plain weave carbon fibre fabric/ 180°C curing epoxy system and a wet lay-up system of identical fabric with a room temperature curing epoxy. Both laminates have the lay-up [[±45,90]3]S and the repair plies are applied such that all plies end at the scarf interface surface.

An analytical ERR-based model was proposed for predicting the delamination behaviour of the scarf joint repair, and the model has been validated experimentally. The relation between delamination growth (da/dN) and the effective ERR (ΔG_{eff}) has been determined by experiments on mode I and mode II type specimens. For mode I the standard DCB test was used. Mode II data have been obtained using a tensile-type test in which two butt-jointed plates are connected by symmetrical application of continuous plies on both sides, with an initial delamination having an H-shape. Using the model it was found that the mixed Mode I/Mode II delamination growth behaviour of the scarf repair can be predicted from the separate Mode I and II behaviours.

1.6 METALS FATIGUE AND DAMAGE TOLERANCE STUDIES

1.6.1 Fatigue of beta processed and heat-treated titanium alloys (R.J.H. Wanhill, NLR)

A review has been made of most of the available literature on the fatigue properties of beta (β) annealed Ti-6Al-4V and titanium alloys with similar microstructures [9]. The emphasis is on beta processed and beta heat-treated titanium alloys, because beta annealed Ti-6Al-4V Extra Low Interstitial (ELI) plate has been selected for highly loaded and fatigue critical structures in advanced high-performance aircraft that are currently intended to enter service with several Air Forces around the world. The main topics in this review are the fatigue initiation mechanisms, fatigue initiation lives, and short-to-long (or small-to-large) fatigue crack growth in beta processed and beta heat-treated titanium alloys. However, some comparisons are made with alloys having different microstructures, in particular conventionally alpha + beta (α + β) processed and heat-treated Ti-6Al-4V.

In particular, the fatigue initiation mechanisms are complex. Figure 19 shows that fatigue crack can initiate at several types of site. Most of the microcracks initiated across colonies of aligned α platelets, as would be expected from previous studies. Some of these cracks extended with little or no deflection across two or more colonies and their boundaries, and a few also crossed the grain boundary α. There were also cracks along the interfaces between α platelets and remanent β, including colony boundaries; and at least one crack ran along the interface between grain boundary α and colonies of aligned α platelets (near the top right corner of Figure 19).

It is intended to publish this review, in slightly altered format, as a “Springer Brief” (these Springer Briefs are short monographs for general reference).

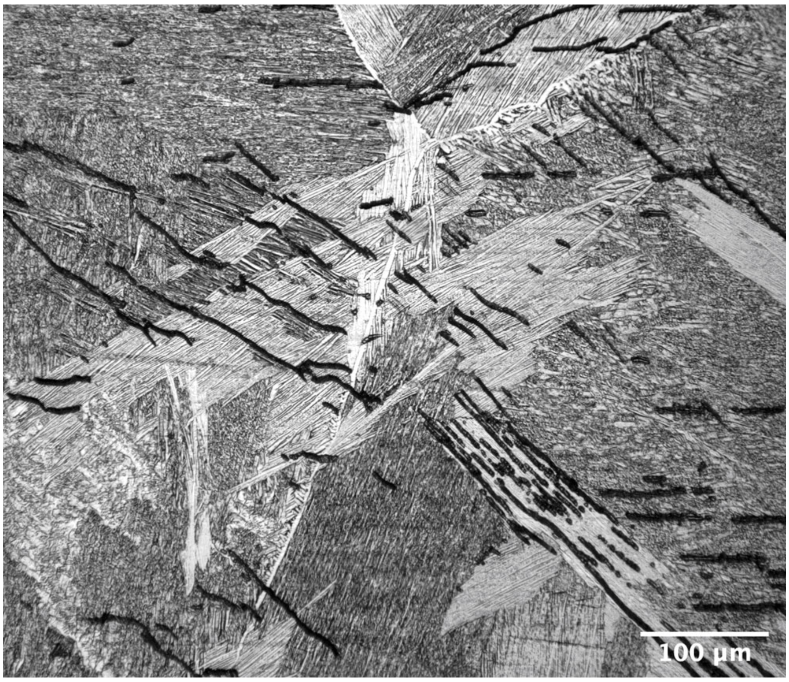

Figure 19: Microcracks on the cylindrical surface of an LCF-tested specimen taken from the β-annealed Ti-6Al-4V plate: Kroll's etch.

1.6.2 Fatigue thresholds in aluminium alloy AA7075-T7351 plate (R.J.H. Wanhill, NLR)

Reliable determinations of fatigue crack growth thresholds are important for fatigue crack growth analyses, especially for helicopter airframe components, since the analyses rely mainly on crack growth data in the near-threshold region. This region is often characterized by considerable data scatter, including scatter in the threshold values. Recognising this, the NLR and DSTO (Melbourne, Australia) have participated in a joint project on helicopter fatigue called HeliDamTol. This project had two main objectives. The first was to develop reliable methods of fatigue crack growth analysis for helicopter airframe components. The second was to incorporate these methods into an Operational Damage Assessment Tool (ODAT), intended to improve the operational readiness of a helicopter fleet.

The present work [10] is a contribution to the first main objective of HeliDamTol. This report presents experimental determinations of fatigue crack growth thresholds in aluminium alloy 7075-T7351 plate material used for the hinge beams on the NH90 helicopter carbon-epoxy composite tail boom. Interpretation of the experimental determinations was aided by fractographic observations of the thresholds and near-threshold crack growth regions.

The thresholds were determined for the positive stress ratio range R = 0.1 – 0.95. The results may be expressed as follows:

(1) For $R \geq 0.58$ the measured fatigue crack growth threshold stress intensity range ΔK_{th} is equal to the effective, or intrinsic threshold stress intensity range ΔK_o, and is given by $R \geq 0.58 : \Delta K_{th} = \Delta K_o = 1.44 \pm 0.25$ MPa√m.

(2) For $R \leq 0.58$ the measured ΔK_{th} depends on R: $R \leq 0.58 : \Delta K_{th} = [(2.35 \pm 0.25) - 1.57R]$ MPa√m.

The results include a scatterband of ± 0.25 MPa√m for all values of ΔK_{th}, and consequent variations in ΔK_{th} of about 11 – 17 %. These variations are most probably due to variable crack front topographies and profiles in the threshold region, and they represent intrinsic limitations to the accuracy of ΔK_{th} values.

This is illustrated by Figure 20, which compares the NLR's results with those of Marci [11], demonstrating very similar data and data scatter.

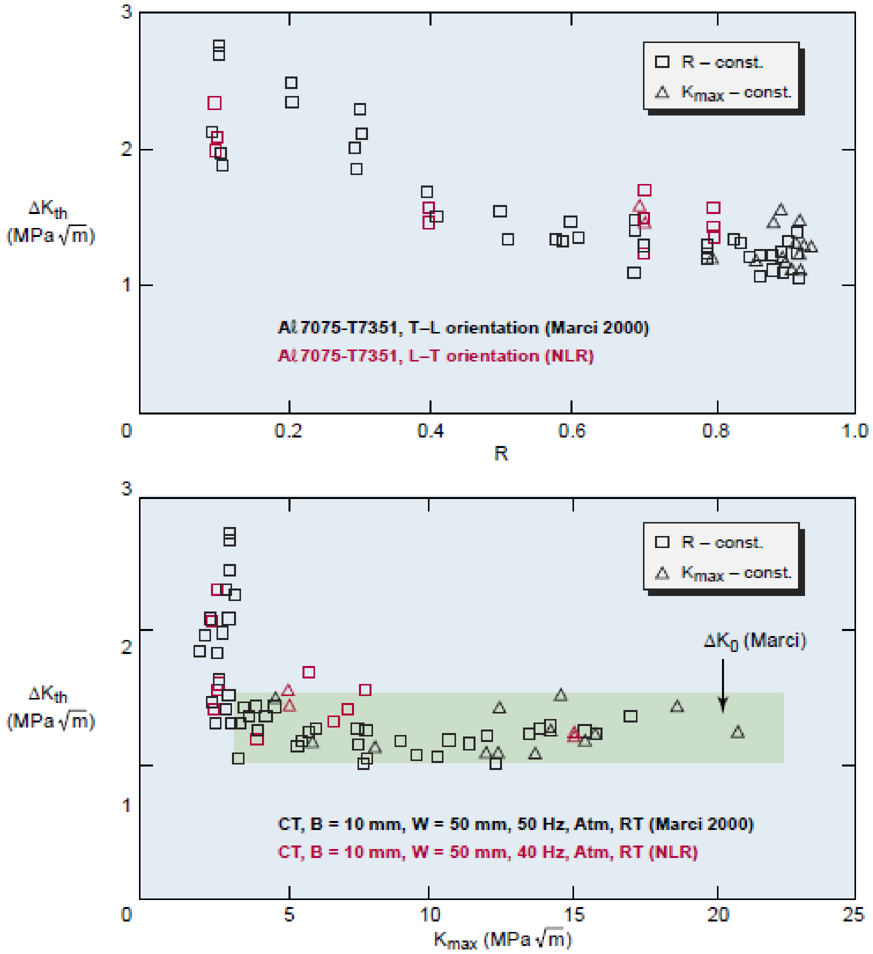

Figure 20: Comparison of the NLR's and Marci's threshold data for 7075-T7351 [10].

1.6.3 Lead crack fatigue lifing for metallic airframe components (L. Molent and S.A. Barter, DSTO, Melbourne, R.J.H. Wanhill, NLR)

A fatigue lifing framework using a lead crack concept has been developed by the DSTO for metallic primary airframe components. The framework is based on years of detailed inspection and analysis of fatigue cracks in many specimens and airframe components, and is an important additional tool for determining aircraft component fatigue lives in the Royal Australian Air Force (RAAF) fleet. Like the original Damage Tolerance (DT) concept developed by the United States Air Force (USAF), this framework assumes that fatigue cracking begins as soon as an aircraft enters service. However, there are major and fundamental differences. Instead of assuming initial crack sizes and deriving early crack growth behavior from back-extrapolation of growth data for long cracks, the DSTO framework uses data for real cracks growing from small discontinuities inherent to the material and the production of the component. Furthermore, these data, particularly for lead cracks, are characterized by exponential crack growth behaviour. Because of this common characteristic, the DSTO framework can use lead crack growth data to provide reasonable (i.e. not overly conservative) lower-bound estimates of typical crack growth lives of components, starting from small natural discontinuities and continuing up to in-service detectable crack sizes (thus encompassing short-to-long crack growth) that just meet the residual strength requirements. Scatter factors based on engineering judgement are then applied to these estimates to determine the maximum allowable service life (safe life limit).

As part of a DSTO (Melbourne, Australia) and NLR collaboration, the paper in [12] was written with the aim of introducing the framework to a wider audience.

1.6.4 Post-test Quantitative Fractography to support safe and accurate inspection intervals (B.H.A.H. Tijs, Fokker Aerostructures, T. Hattenberg, NLR)

Post-test Quantitative Fractography (QF) is of much interest when crack growth measurements during (full-scale) test programmes are insufficiently accurate, or when additional crack growth information is required in order to determine safe inspection intervals. The problem with full-scale test programmes is that the spectrum can be complex and not specifically designed to incorporate marker loads. Fokker Aerostructures, in collaboration with the NLR, therefore investigated whether QF can be successfully applied for such a complex test spectrum. Use was made of the guidelines and procedures presented by the DSTO and NLR during ICAF 2009 [13].

The investigation consisted of 2 phases:

(1) The first phase of the investigation focussed on the macroscopic features of the fracture surface. The features of main interest were the fatigue initiation areas, the fatigue crack length and the size of the crack caused by static fracture (final failure). The macroscopic features of the fracture surface revealed that fatigue cracks initiated from the stress concentration provided by an anodised hole, and that the crack caused by static fracture was rather small. Further fractographic investigation by Scanning Electron Microscopy (SEM) showed that fatigue initiation was due to local attack during the anodising process, see Figure 21.

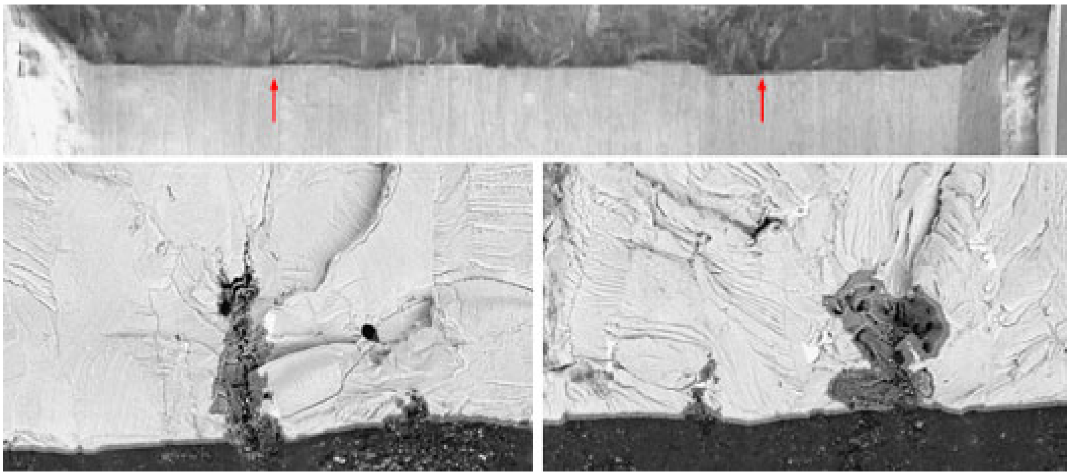

Figure 21: Fatigue initiation areas (local attack during the anodising process) at the stress concentration provided by an anodized hole.

(2) The second phase of the investigation focussed on Quantitative Fractography to support a safe and accurate crack growth interval in terms of flight cycles. The first objective of this investigation was to determine whether the fatigue crack topography could be correlated with the applied spectrum. Extensive SEM scanning of the fracture surface revealed repetitive patterns of markers, see Figure 22. Then these patterns were matched to the real-time measured strain gauge data in order to determine a safe and accurate inspection interval.

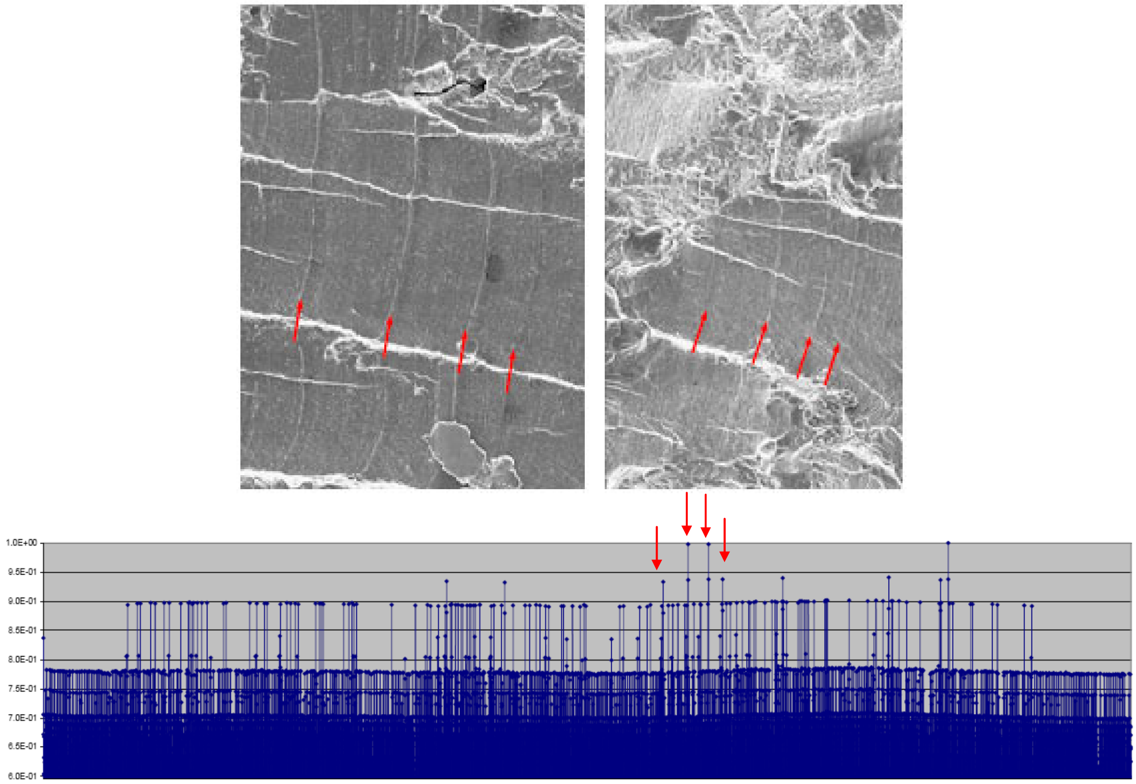

Figure 22: Scanning Electron Microscope (SEM) fractographs of well-defined markers and the normalized (filtered) test spectrum.

1.6.5 Constraint formulation for the ESACRACK model (R. A. Huls, NLR)

The ESACRACK Strip Yield model contains a constraint factor to accommodate three-dimensional effects. This factor was originally derived by comparing predicted and measured crack growth curves and thus was used as a fit parameter. However, the physical background of the constraint factor enables a second way of deriving it: determining the factor using finite element calculations. A three-dimensional parametric finite element model of a middle tension (MT) specimen with elastic-perfectly plastic material behaviour was therefore created. With the model a routine for obtaining the constraint factor as function of location in the specimen was made. Results were generated for different load sequences and a new constraint formulation was developed. This new formulation was added to ESACRACK and evaluated against a range of experimental crack growth curves. The new formulation performs slightly better than the previous derivation, and has the important advantage of a physical basis.

1.6.6 Scatter in CA and VA fatigue crack growth lives (F.P. Grooteman, NLR)

Much scatter can be observed in the fatigue crack growth lives of metallic components. A significant part of the scatter is caused by the variations in material properties. This means that many similar tests, i.e. tests using the same specimen geometry and type of loading, have to be done to sufficiently characterize the scatter. However, only a few experimental datasets are known where a sufficiently substantial dataset has been obtained.

At the NLR a series of crack growth tests has been performed on 160 mm wide centre-cracked (MT) specimens with an initial crack length of 3 mm, i.e. in the so-called “long crack” regime. The specimens were manufactured from two different batches of aluminium alloy AA7075-T7351. To date (February

2011) 43 constant amplitude (CA) tests (Smax = 80 MPa, R=0.1) have been performed and 31 variable amplitude (VA) tests using the reference gust spectrum mini-TWIST.

A preliminary survey and analysis of these tests indicate significant scatter and a considerable difference between the scatters for the CA and VA tests, such that the results cannot be pooled. A larger amount of scatter was observed for CA loading than for VA loading. Owing to these results, the current test sets will be extended in the near future. The extended datasets will subsequently be used as input for probabilistic analyses.

1.6.7 Fatigue and Damage Tolerance of Friction Stir Welded Joints for aerospace applications (H.J.K. Lemmen, R.C. Alderliesten, TUD)

The research on fatigue in Friction Stir Welding (FSW) has been reported before in previous National Reviews and ICAF symposia. The main objective of the research was to investigate the effect of fatigue initiation and crack propagation in aerospace aluminium sheets welded by FSW. The result of the research is the development of prediction models that can be used to evaluate the damage tolerance concept for metallic FSW-welded structures. Preliminary design rules have been formulated and recommendations given for further research and application of FSW in aerospace primary structures. All the research work has been reported in a PhD dissertation successfully defended on January 14th, 2011 [14-18].

1.6.8 Application of marker loads for fatigue research using fractography (M. Krkoska, R.C. Alderliesten, TUD)

The aim of the project is to increase the knowledge about crack propagation mechanisms involved in variable amplitude (VA) loading. Research on the effect of positive peak loads (overloads) is more commonly reported in the open literature. This is probably in recognition of the crack closure concept and an expectation that no crack will in fact propagate at loads below the crack closure level. However, compressive peak loads (underloads) often occur in service and it is recognized that these loads can have a detrimental effect on crack growth behaviour. Therefore, an experimental programme on the effect of underloads on crack growth was initiated, focussing on the 2024-T3 and 7075-T6 aluminium alloys [19]. These are industry standard aluminium alloys and are widely used in aerospace applications. Quantitative and qualitative fractography, as well as stereography, have been employed in this study.

The incorporation of underload cycles in the laboratory test sequences was shown to be a very effective method in marker production. Easily visible markers were obtained right down to the early stages of crack growth. However, introduction of underload cycles in the testing sequence can significantly reduce the fatigue crack growth life of the test component. The amount of crack life reduction will depend on the material (7075-T6 is less sensitive) and the amplitudes of applied underload cycles. It was shown that an increase in underload amplitude was directly linked to the reduction in crack growth life. At the same time, no significant load interaction between applied load cycles was observed.

Sensitivity of crack propagation to individual grain orientation was also observed, especially for the 7075-T6 alloy. However, only minimum sensitivity of crack propagation to grain orientation was observed for 2024-T3. Local variations in fracture surface appearance and crack growth directions occur on a grain-to-grain scale, and these variations suggest that different failure mechanisms are occurring. These observations have a significant implication: markers could be easily identified in some grains while no markers were observed in neighbouring grains. There are also irregular and incoherent crack fronts. Figure 23 shows examples of these variations.

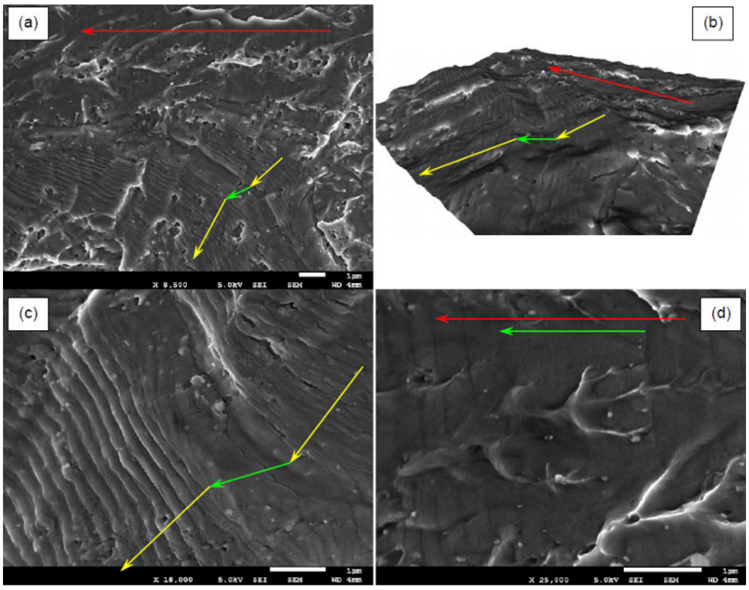

Figure 23: SEM micrographs of fracture surfaces for 7075-T6 alloy. The examples depict differences in striations formation as observed in relation to local crack plane orientation. (a) overview of fracture surface with two distinctively different crack topographies; (b) 3D model of the overview; (c and d) different appearance of striations and markers (more ductile and tearing-like features, respectively) when formed on differently-tilted local crack planes.

1.7 FULL-SCALE AND DETAIL FATIGUE TESTING

1.7.1 Certification test activity for a medium size jet empennage (G. van Gool, J. Meuzelaar, J.E.A. Waleson, Fokker Aerostructures)

Certification approach

Following the Building Block approach from AC20-107B, the certification approach for this programme can best be described by the figure below. This approach was also taken into consideration for F&DT certification.

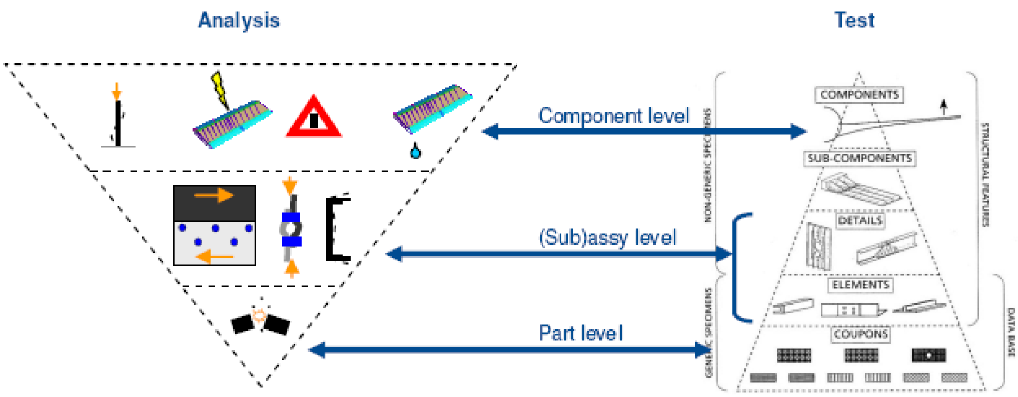

Figure 24: Certification approach.

Coupon/element/subcomponent testing

In the framework of the development and certification of thermoplastic induction-welded control surfaces, a large number of screening, qualification and certification tests have been conducted. The main structural composite material for these multi-rib constructions is carbon fibre reinforced thermoplastic. Certification tests at the coupon/detail level were performed on notched laminates, bolted and welded joints (peel and shear) at RTD and critical environmental conditions. Joint tests were performed under static and dynamic loading. B-values and S-N curves were constructed from these tests. A noteworthy detail was the fatigue behaviour of the welded joints: these showed relatively small scatter and fatigue-insensitive behaviour, even with artificial defects present in the specimens.

At the element level, so-called shaker tests were done to establish the sonic fatigue resistance of the induction-welded joints. A very flat S-N curve was found, confirming again the low fatigue sensitivity of the welded joints.

At the sub-component level a complete section of an elevator/rudder torsion box was tested for direct effects of lightning strike. The test article featured fully representative structural details (lay-up, welded joints, fasteners), surface treatments (primer, paint) and electrical bonding details (protected and unprotected skins). Four Zone 1A and 2A strikes in the high-voltage laboratory demonstrated sufficient protection against lightning strike (local damages only).

<u>Overview of full scale/component test program</u>
Several full-scale component tests were carried out as part of the certification activities for the empennage. The vertical fin is part of the full-scale airframe test. The horizontal stabilizer, elevator and rudder are evaluated in dedicated component tests. The horizontal stabilizer and elevator were tested for static strength, fatigue and damage tolerance at the National Aerospace Laboratory (NLR). The rudder test was carried out by VZLU (CZ). A schematic of the test principle is shown in Figure 25.

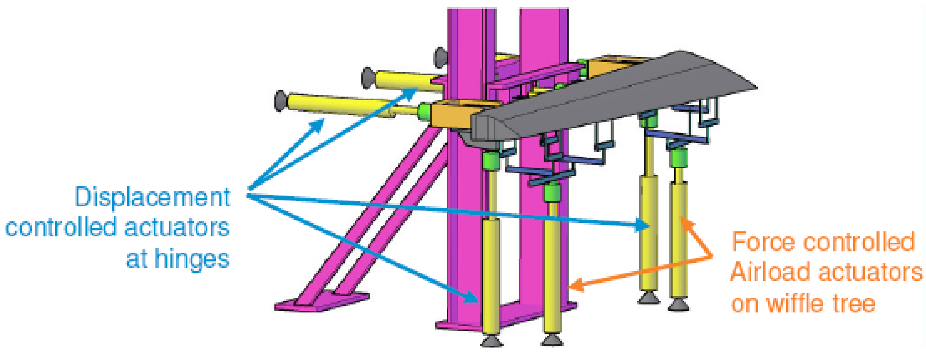

Figure 25: Test principle.

All tests consisted of different phases. The test activities started with a shakedown, followed by the fatigue design life (using a scatter factor of 1.5 and LEF of 1.15), including Level 1 damages. After the fatigue campaign, static LL and UL tests were carried out. The subsequent Damage Tolerance phase was tested for two inspection intervals. At each interval different Level 2 damages were applied. Level 2 damages mainly consisted of impacts in the composite material. Finally all three test articles had Level 3 damages applied and were tested successfully with a daily load. The most severe Level 3 damage was a cut through the front beam and upper skin of the elevator.

The horizontal stabilizer test was a pure force-controlled test. The elevator and rudder tests used a combination of force and displacement control.

All the component tests finished successfully. NDI did not reveal any unexpected crack or delamination propagation.

Thermal loads in fatigue spectrum

Thermal loads in the tailplane structure with a composite skin and aluminium ribs and an aluminium spar were determined for the fatigue and damage tolerance analysis. The temperature in the structure was derived from the static temperature and thermal heating. There was a good correlation of stress in the FEM model and a thermal test for the aluminium substructure.

Omission level of test spectrum for composite structure

The test spectra for the composite tailplane structure and thermoplastic control surfaces were derived without truncation and with a commonly used omission level. The omission level was confirmed by a fatigue test of the critical joint of each component at one representative stress ratio. Other omission levels were determined for cycles with other stress ratios based upon a Goodman diagram for notched coupons.

Initiation of crack growth from artificial damage

An artificial damage was inflicted in a steel lug in a full-scale test by means of spark erosion. Coupon tests were used to guarantee that crack growth would occur during testing and that failure of the lug would not jeopardize the full-scale test. A similar damage was inflicted in compact tension (CT) specimens. A spectrum of stress intensity factors equal to that of the test was applied to the CT-specimens up to the start of crack growth. This enabled establishing the required size of the damage in the full-scale test and the number of test flights up to initiation. The rest of the crack growth in the CT-specimens was used to determine the da/dN curve and to establish the shut-off ratio used in the Modified generalized Willenborg crack growth model. It should be noted that the spectrum load sequence was accounted for by using a peak-retention rainflow counting method.

1.7.2 Fatigue testing of the NH90 tail module (E. Mantel, Fokker Aerostructures)

The NH90 helicopter tail is a hybrid structure, consisting of carbon panels and aluminium parts. The fatigue loads are both in the regular and high-cycle fatigue domain. The regular fatigue spectrum originates from the mission types: the main contributors are flight loads from the horizontal stabilizer, vertical fin and tail rotor, and the ground-air-ground cycles. The high cycle fatigue loads come from the vibrations imposed by the dynamic system of the helicopter.

Safety aspects require compliance with FAR29.571. A mixture of different analyses has been performed to cover the high and low cycle fatigue loads. The fatigue and damage tolerance substantiations are supported by full-scale and detail testing. The full-scale test is used to support the fatigue and damage tolerance substantiations of the metal and composite parts of the NH90 tail module. Since only one test article is available for the full-scale test, the certification requirements for both the metal and composite parts had to be met with respect to the test conditions and schedule.

For the composite parts it is necessary to use a Load Enhancement Factor and an Environmental Knockdown factor. The Load Enhancement factor (LEF) depends on the number of lives to be tested. Fokker has chosen an LEF such that the number of lives are equal to those obtained from the required scatter factor for the metal parts.

The Environmental Knockdown factor (EKDF) is a load increasing factor, which compensates the RT-conducted test for the potential decrease of fatigue properties of the composite parts due to life-cycle moisture and temperature effects. The EKDF-fatigue to be used for the spectrum loads is different to the EKDF-static to be used for the residual strength checks (UL or LL) during the test programme.

Both factors (LEF and EKDF) are necessary for the composite parts, but not for the metal parts. In practice this means that the fatigue damage accumulation of the metal parts is accelerated.

For the test sequence Fokker has chosen the following approach:

(1) Fatigue test cycling (with LEF and EKDF-fatigue)

(2) Damage tolerance testing for metal parts only (no factors)
(3) Limit load test for metal parts (no factors)
(4) Ultimate load test for composite parts (with EKDF-static)
(5) Damage tolerance testing for composite parts only (with LEF and EKDF-fatigue)
(6) Limit load test for composite parts (with EKDF-static)

The fatigue phase of the test covers as-manufactured metal part damage levels barely visible impact damages (BVIDs) and minimum production quality (voids/delaminations/inclusions) for composite parts. The aim of the ultimate load test is to show "no detrimental growth" of these damages.

The aims of the damage tolerance tests are to establish or confirm the analytically determined safe inspection intervals. These tests include higher damage levels: artificial damages in metal parts and clearly visible impact damages (CVIDs) in the composite parts. The purpose of the limit load tests is to show adequate residual strength.

Figure 26 provides an overview of the full-scale fatigue test setup.

In the test spectrum the flight loads are randomly distributed to reflect and respect the occurrences and flight sequence. This randomizing process is done by means of the load-sequencing program CLASS [20].

Figure 26: Overview of the full-scale fatigue test setup of the NH90 tail module.

During the test different monitoring techniques are being used:

- Continuous logging of actuator displacements, also used for test safeguarding purposes
- Continous logging of strain gauges, also used for test safeguarding purposes
- Crack wires
- Comparative Vacuum Monitoring patches (CVM)

In addition, so-called commissioning loads have been applied to the test article during several test phases. The results of these commissioning loads have been used to verify the proper functioning of the measuring system and also to monitor any stiffness changes of the test article.

1.8 SPECIAL CATEGORY

1.8.1 Fatigue of Structures & Materials (J. Schijve, TUD)

As reported in the previous review, a second edition of the book “Fatigue of Structures and Materials” was published in 2009. In 2010 the book was selected for a 2010 Textbook Excellence Award of the Academic Author Association in the USA. Currently the book is being translated into Chinese for publication in 2012.

1.8.2 A short course on failure analysis of metallic materials (R.J.H. Wanhill, NLR, S.A. Barter and S.P. Lynch, DSTO, Melbourne)

As part of a DSTO (Melbourne, Australia) and NLR collaboration a short (1-day) course on failure analysis of metallic materials, especially aerospace alloys, has been prepared in PowerPoint format [21]. The course is in two main parts. Part I covers the general approach to failure analysis, causes and types of failures, failure diagnosis via fracture modes, and summarises the required specialist knowledge and techniques. Part II consists of nine case histories taken from the DSTO and NLR archives. The case histories were selected to illustrate a wide variety of problems. Figure 27 gives an overview of the course contents.

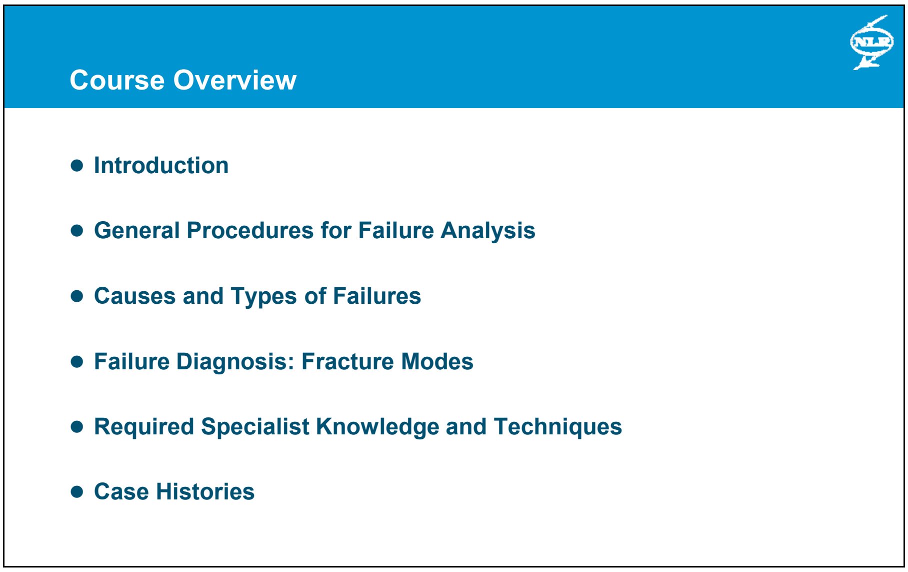

Figure 27: Overview of a DSTO (Melbourne, Australia) and NLR short (1-day) course on failure analysis of metallic materials, especially aerospace alloys.

1.8.3 A course on FML technology for primary aircraft structures (R.C. Alderliesten, C.D. Rans, J. Sinke, R. Benedictus, TUD)

TU Delft has developed a multi-day course that presents the relevant aspects related to FML technology for primary aircraft structures. The course has been developed to introduce OEMs and first tier suppliers into the hybrid technologies for primary aircraft structures, and to instruct on best practices for the development process of FML components and structures.

The course consists of multiple parts that cover a general overview of the FML technology development, the determination of static, fatigue and damage tolerance allowables and evaluation methods, the design approaches for mechanical joining, the manufacturing and production aspects, the inspection, testing and certification aspects, the environmental aspects, and general best practices.

The course has been given successfully to a large OEM.

1.8.4 Discussing fatigue crack growth models in FMLs (R.C. Alderliesten, TUD)

Much research has been performed on the FML technology within the Netherlands and particularly by TUD. In recent years, other industry and academia have initiated efforts to explore the FML technology for structural applications. A paper recently published by Po-Yu Chang and Jenn-Ming Yang (USA) has been discussed based upon the knowledge generated over the past years at TUD. The prediction model proposed by these authors appears to have deficiencies, which are being discussed in a letter to the editor of International Journal of Fatigue [22].

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