Nationaal Lucht- en Ruimtevaartlaboratorium

National Aerospace Laboratory NLR

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Review of aeronautical fatigue investigations in the Netherlands during the period March 2011 - March 2013

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Executive summary

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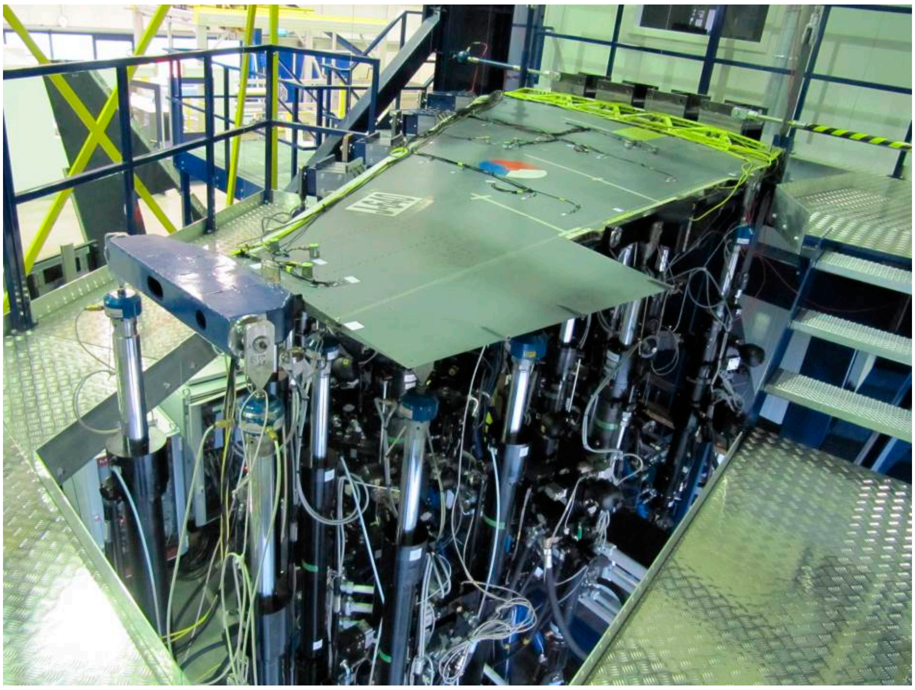

Description of work
This report is a review of the aerospace fatigue activities in the Netherlands during the period March 2011 to March 2013. The review is the Netherlands National Delegate's contribution to the 33rd Conference of the International Committee on Aeronautical Fatigue and Structural Integrity (ICAF), 3 and 4 June 2013, Jerusalem, Israel.

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Summary

This report is a review of the aeronautical fatigue activities in the Netherlands during the period March 2011 to March 2013, and is the National Delegate's contribution to the 33rd Conference of the International Committee on Aeronautical Fatigue and Structural Integrity (ICAF), 3 and 4 June 2013, Jerusalem, Israel.

Contents

Abbreviations

AE	Acoustic Emission
ANN	Artificial Neural Network
ASIP	Aircraft Structural Integrity Program
CFRP	Carbon Fibre Reinforced Plastic
CSI	Crack Severity Index
CVFDR	Cockpit Voice and Flight Data Recorder
DSTO	Defence Science and Technology Organisation (of Australia)
EBH	Equivalent Baseline Hours
ENSIP	Engine Structural Integrity Program
FACE	Fatigue Analysis & Air Combat Evaluation system
FBG	Fibre Bragg Grating
FDR	Flight Data Recorder
FML	Fibre/Metal Laminate
FH	Flight Hour
FLG	Fokker Landing Gear B.V.
FLM	Fleet Life Management
Glare®	GLAss REinforced aluminum
HOLMES	Hercules Operational Loads Monitoring and Evaluation System
HUMS	Health and Usage Monitoring System
LAM	Laser Additive Manufacturing
L/ESS	Loads and Environment Spectrum Survey
LHS	Left Hand Side
MFOQA	Military Flight Operations Quality Assurance
NAHEMA	NATO Helicopter Management Agency
NLR	National Aerospace Laboratory (of the Netherlands)
PAC	Physical Acoustics Corporation
POD	Probability Of Detection
QF	Quantitative Fractography
RAAF	Royal Australian Air Force
RHS	Right Hand Side
RNLAF	Royal Netherlands Air Force
SDE	Supportability Data Exchange
SDR	Structural Data Recorder
SHM	Structural Health Monitoring
SLM	Selective Laser Melting
TOF	Time Of Flight
TUD	Technical University of Delft

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Review of aeronautical fatigue investigations in the Netherlands during the period March 2011 – March 2013

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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 2011 to March 2013. 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 B.V. (Fokker)
- Fokker Landing Gear B.V. (FLG)

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.

2 Loads

2.1 HeliCompLoad Programme (N. Münninghoff, NLR)

Present day maintenance programmes for helicopters are often schedule based, which means that a helicopter component is replaced after a predetermined number of flight hours. Future maintenance procedures aim at maintenance 'on condition' which means that a component is replaced when it is degrading or when the actual operational life is reached. Thorough and reliable knowledge of the exerted loads on the component is essential for the development of such maintenance procedures.

Within this context a framework to calculate helicopter airframe component loads has been developed, based on a chain of relatively simple physics modelling tools that are linked to Flightlab. The results of the computations have been validated by means of comparison with strain gauge measurements on an aft pylon engine frame of a CH-47D "Chinook" helicopter during regular operational flights. The results show a good agreement for the prediction of the 3/rev vibrations of the component [1]. Trend analyses provide insight in weight, flight speed and altitude dependencies. The proposed framework is capable of calculating structural dynamic loads of an airframe component in a relatively simple and cost-effective way.

The composed framework is the first phase in a programme to develop the technology to support airframe component loads analyses. The results are very encouraging, although further validation of the method (especially to capture the 6/rev vibrations) is necessary. A second phase has started last year to perform a more thorough validation using also strain gauge measurements in the rotating system and introduce further improvements in the modelling chain.

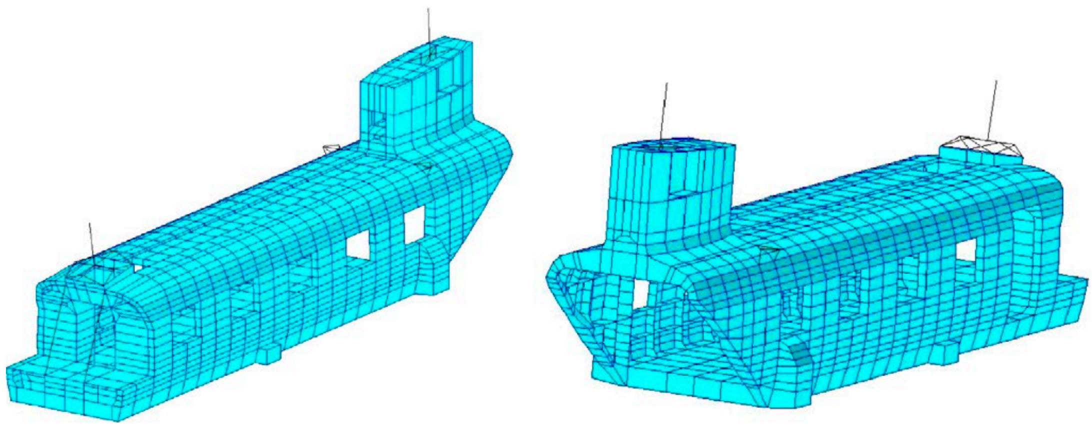


Figure 1: An impression of the stiffness model of the CH-47D "Chinook" helicopter that is used in HeliCompLoad.

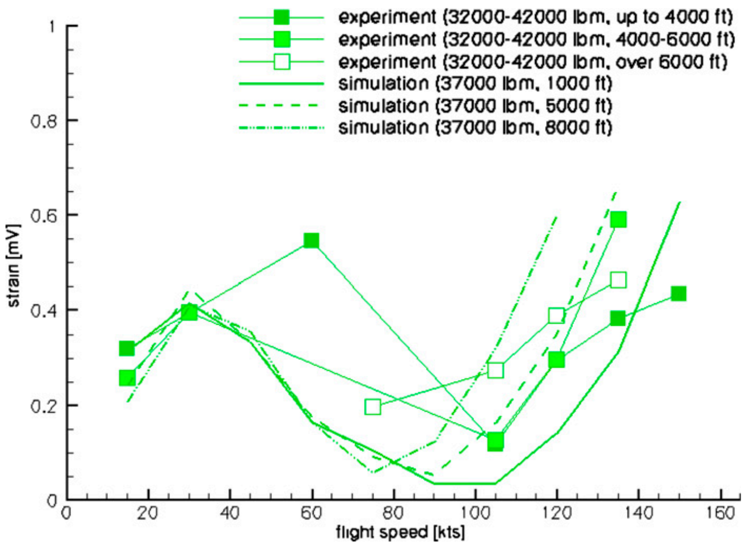


Figure 2: Comparison of measured and computed 3/rev strain amplitudes for a location in the aft fuselage of the CH-47D helicopter.

3 Structural Loads/Usage/Health Monitoring

3.1 RNLAF C-130H(-30) loads & usage monitoring (M.J. Bos and J.A.J.A. Dominicus, NLR)

The Royal Netherlands Air Force (RNLAF) operates a mixed fleet of C-130H and C-130H-30 (stretched fuselage) models. In 2001 the NLR has been tasked to monitor the usage of the fleet and to keep track of the consumed fatigue life of the major airframe components of each of the individual aircraft in the fleet. For this purpose NLR has developed an individual aircraft monitoring programme dubbed "HOLMES". This programme is based on the collection of operational data from various sources, including a flight data recorder, and the subsequent translation of these data to the accrued structural fatigue damage at twelve control points in the airframe. In addition the so-called severity factor per flight is calculated, which can be used to compute the Equivalent Baseline Hours (EBH) that form the basis for fleet life management. This programme has already been described in the previous national review [2].

Initially the limited set of flight parameters required for the computation of the EBHs was recorded with a simple data acquisition system which sampled the altitude, differential cabin pressure, airspeed and vertical acceleration. Recently, the whole fleet has been upgraded to a digital avionics system ('glass cockpit') and now features a crash-survivable flight data recorder (FDR) from Honeywell called "FDAMS" (Flight Data Acquisition & Management System) that records these four parameters plus many more. This system is easy to read out and, as such, is suitable for routine loads and usage monitoring. The additional information from the new FDR enables the development of a more advanced method of loads and usage monitoring, based on the development of virtual strain gauges that correlate the measured strains at particular points in the airframe to the flight and engine parameters that are routinely collected with from the digital databus. This can be done by means of an artificial neural network (ANN) or by a deterministic regression technique. Once such correlations have been established for all possible conditions in the flight and ground envelope of the aircraft, the gauges that have been used to measure the strains at the selected locations are no longer needed.

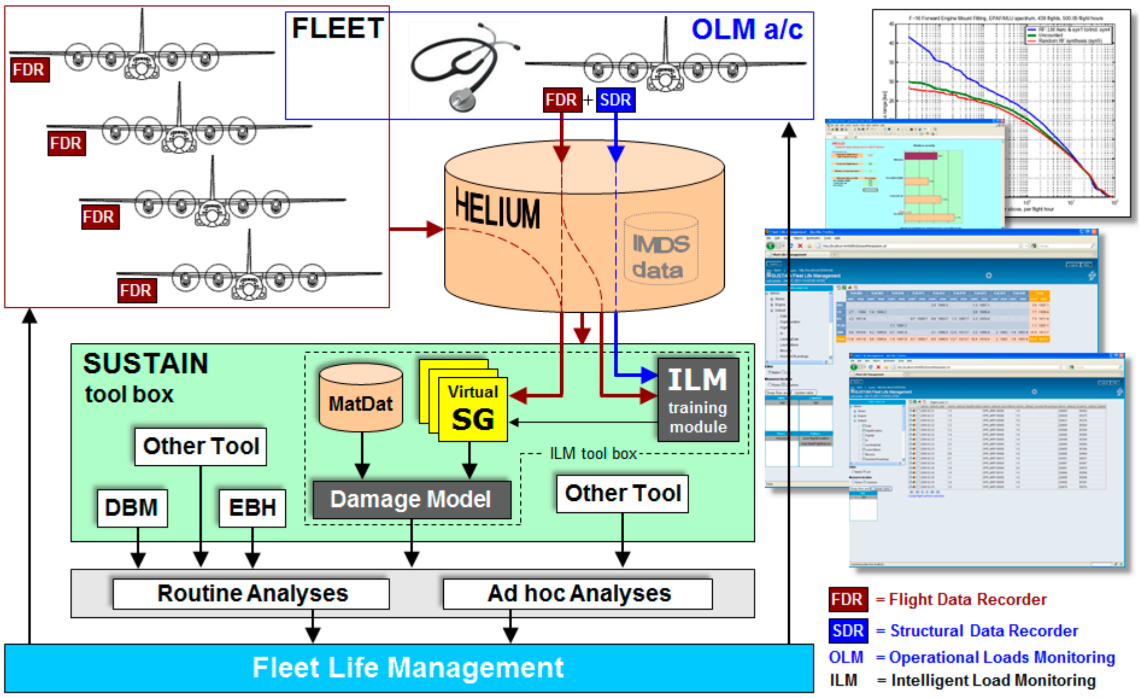


Figure 3: The Stethoscope Method for Fleet Life Management.

Based on this technique, a new fleet life management (FLM) concept has been developed called the "Stethoscope Method" [3], which to a large extent is already operational for the CH-47D helicopter fleet of the RNLAF [4]. This method is outlined in Figure 3. It involves the fleet wide collection and storage of all relevant flight, engine and control parameters that are available from the digital data bus, plus the simultaneous collection of loads data in one or more dedicated aircraft in the fleet that are equipped with a structural data recorder (SDR). The operational loads data from the SDR can be used to continually train and improve the virtual strain gauge models in a cost-effective way, since no special loads measurement flights need to be performed; the loads are simply collected during regular operations and training. By moving the strain gauges in the dedicated aircraft around on a regular basis, models will be obtained for more and more points in the airframe that are relevant from a fatigue point of view. In combination with appropriate material data and damage models this will allow the establishment of the safe life consumed so far and the assessment of the severity of in-service developed fatigue cracks for each critical point. In this respect it is essential to start collecting the digital bus data from the very first moment that an aircraft is commissioned into service. For the virtual strain gauge models this is less crucial; when developed at a later stage these models can use the historic bus data that are stored in a database to 'roll back' to day one.

For the C-130H(-30) fleet this FLM concept is being developed in collaboration with the Republic of Korea Air Force.

3.2 Loads & usage monitoring of the RNLAF helicopter fleets (A. Oldersma, NLR)

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 transport 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 (CVFDR) 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. 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. Details of the programme are provided in ref. [4].

The recorded flight parameters serve as the input for Flight Regime Recognition. The Flight Regime distribution, together with other usage statistics such as weight distribution classes and speed classes, reveals differences in usage during out-of-area operations and usage within the Netherlands. Based on the measured strain gauge data, fatigue damage rates have been developed for each operational flight regime that can be used to monitor the severity of flight operations. This has led to the introduction of a 120 KIAS operational speed limit in an attempt to reduce the maintenance burden associated with the excessive amount of fatigue cracking in secondary airframe structure – see Figure 4. The effects of this operational speed limit will be evaluated in the second half of 2013. In addition, the present strain gauges in the fleet will be relocated in order to facilitate the development of fatigue damage rates for other parts of the airframe.

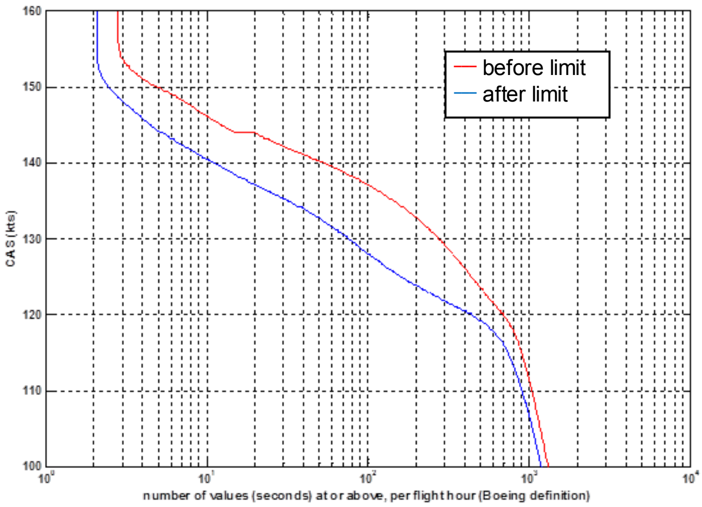


Figure 4: Exceedance plots of RNLAF CH-47D air speed before and after introduction of speed limit.

Following Honeywell Service Bulletin 55-L-714A-0002 for the Chinook T55-L-514 engine, the usage monitoring effort was extended with engine monitoring. This involves engine cycle calculations of operational flights based on Honeywell algorithms and depending on engine operating time, number of landings, engine maintenance runs and temperature exceedances. The required power turbine inlet temperature data and the flight parameters required to determine engine operating time and number of landings are obtained from the CVFDR; the engine cycles are not monitored by the digital engine controller unit (DECU).

Prompted by the success of the Chinook usage monitoring programme, the RNLAF has expressed their interest in similar programmes for the Apache AH-64D and NH90 helicopters. In 2012 the availability and suitability (sampling rate) of the AH-64D and NH90 flight parameters were investigated. Based on available flight parameters of initial sets of flights, exploratory studies were carried out to define and determine Flight Regimes for both helicopter types. The results serve as input for AH-64D and NH90 usage monitoring programmes, which will be developed within the next few years.

3.3 Multi-nation NH90 Supportability Data Exchange programme (A.A. ten Have and H. Tijink, NLR)

Under contract of NAHEMA, NLR has developed and implemented a so-called NH90 Supportability Data Exchange (SDE) service for a consortium serving nine nations that operate the NH90/MRH90 helicopter. It consists of a common shared usage database and analysis toolbox with automated web based user data input. An outline is provided in Figure 5.

The full NH90 SDE service comprises the following elements:

1. An Occurrence Reporting System functionality which provides all participating nations with a capability to store, analyze, report and share NH90 SDE occurrence data in a flexible, secure environment, allowing adequate communication of occurrence reports between military headquarters, operational fields and maintenance facilities.
2. A Reliability Assessment System functionality which provides the NH90 operator with the capability to determine experienced failure rates (MTBFs) of different helicopter components based on parts and equipment failures and maintenance task information. The SDE system contains provisions to extend reliability calculations towards Availability and Maintainability calculations, in the future.
3. An Aircraft Integrity Management functionality which offers a threefold state-of-the-art structural integrity and fleet life management tool with a standard reporting facility. Input data for SDE Level 3 AIM can be from different sources, e.g. MDS-GLIMS, Flight Data Recorder or a nation specific non-integrated retro-fit multi-channel data acquisition unit for strain gauge measurements, fed by data from the helicopter digital data buses.
4. A public and secure website (see: https://sde.nlr.nl), a moderated forum and a yearly users conference to evaluate and discuss integrity, completeness and validity of NH90 usage data, analysis results, helicopter degradation and maintenance, and to manage and direct necessary future SDE engineering support.

In 2012 the SDE Reliability Assessment System and the Occurrence Reporting System became fully functional – see Figure 6. Physically, the NH90 SDE service is running on a 24/7 basis in a secure environment, located at NLR-Flevoland.

A more detailed description of the NH90 SDE service can be found in ref. [5].

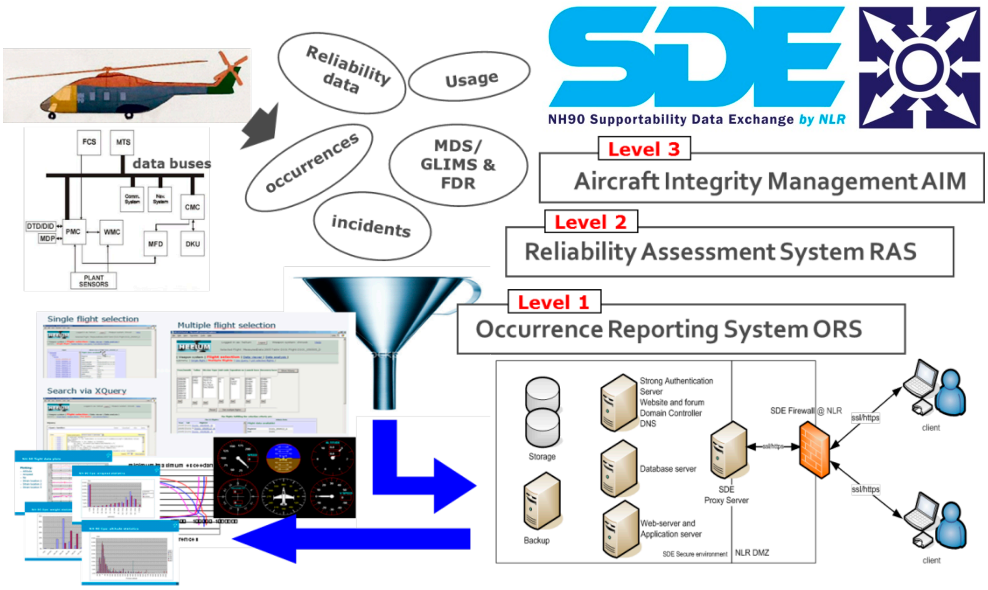
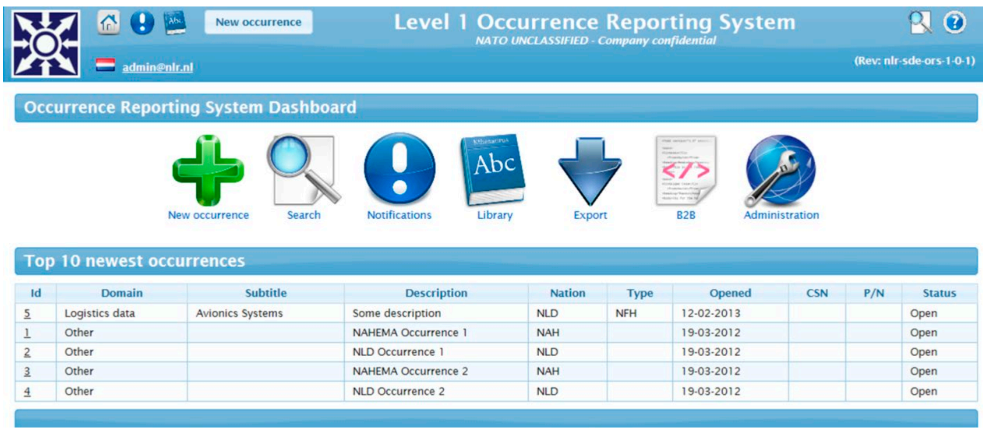


Figure 5: Outline of the NH90 SDE programme.



Id	Domain	Subtitle	Description	Nation	Type	Opened	CSN	P/N	Status
5	Logistics data	Avionics Systems	Some description	NLD	NFH	12-02-2013			Open
1	Other		NAHEMA Occurrence 1	NAH		19-03-2012			Open
2	Other		NLD Occurrence 1	NLD		19-03-2012			Open
3	Other		NAHEMA Occurrence 2	NAH		19-03-2012			Open
4	Other		NLD Occurrence 2	NLD		19-03-2012			Open

Figure 6: Screenshot of the SDE Occurrence Reporting System.

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

Structural load monitoring of the RNLAF/F-16 fleet has been 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 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 the Operational Management Information System of the RNLAF. 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 load and usage monitoring programme has been implemented for the Belgian Air Force (BAF), for which 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:

- 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, i.e. F-16 Wing Damage Enhancement Test
- 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 have been made to support several research programmes:
  - Detailed engine FAULT-code recordings
  - Flight departure margin study
  - MFOQA (Military Flight Operations Quality Assurance) trial 2007 and 2011 with 18 aircraft instrumented with a 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:

- More advanced ways of linking CSIs to (clusters of) control points
- F-16 dynamic effects including limit cycle oscillation (LCO) tracking capability
- Evaluation/validation model Landing Gear Damage Indicator with dedicated measurements
- For general analysis, reporting, and visualization of life cycle information for the F-16, a transition is in progress 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.

3.5 Damage detection with an SHM system based on optical fibres (F.P. Grooteman, NLR)

Structural Health Monitoring (SHM) systems can reduce the cost of ownership and improve the operational availability of a structural system. However, it remains a difficult task to determine the number and position of sensors to be able to detect damage. Therefore, a structural health monitoring (SHM) system design tool, schematically depicted in figure 7, is developed with which the number and position of strain sensors can be determined for a general composite or metallic structure to enable the detection of damage. The network of sensors is used to detect changes in the response of a structure from data gathered at two different states, an initial reference state and the current damaged state. The damage detection algorithm applied is based on a modal approach and is able to detect the presence and location of the damage. The SHM design tool is highly automated and allows for automatic damage insertion in the Abaqus finite element model, which is a requirement for a fast design. Damage indicator plots are automatically generated.

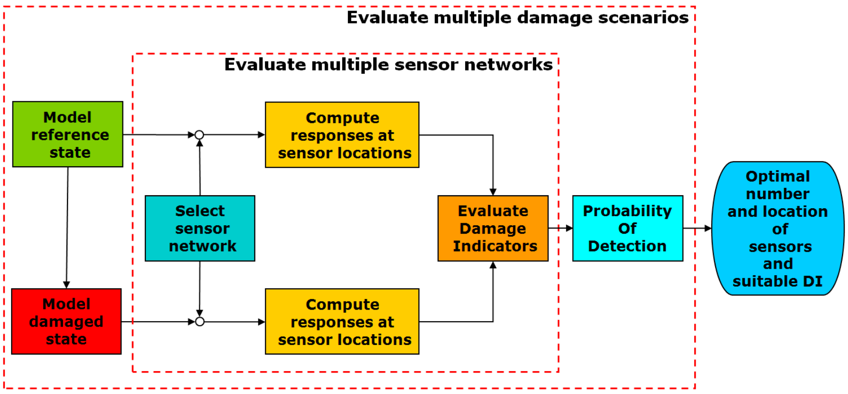


Figure 7: Flow diagram of the SHM design tool.

The SHM design tool has been applied to a flat stiffened composite aircraft-like panel, depicted in figure 8, for which several damage scenarios have been analysed, such as stringer debonding and impact damage. Especially disbonds are hard to detect with conventional non-destructive inspection methods. The numerical results showed that even small damages can be detected with a coarse sensor network. However, in a realistic application noise will be present in the measurement signal resulting in errors. A noise component can be added to the computed responses to simulate a signal-to-noise ratio (SNR) present in a realistic application. This in combination with randomly generated damages, in location and size, allows the computation of the detection capability of the SHM system, expressed in terms of the Probability of Detection (POD). The POD reflects the chance to detect a damage of a certain size within the structure, which is an important quantity for the certification of the SHM system.

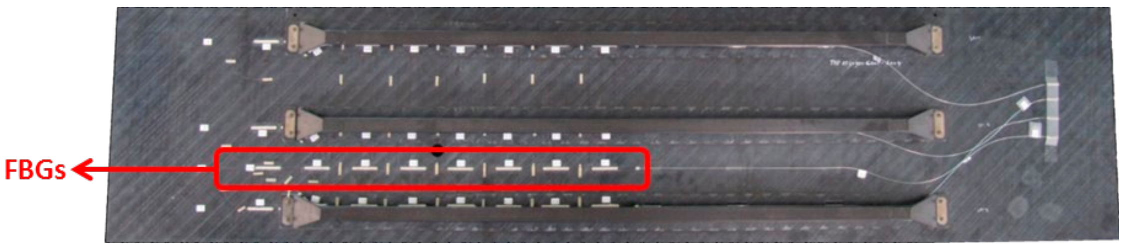


Figure 8: Three-stringer thermoplastic composite panel with installed FBGs.

The three-stiffener composite panel has recently been tested using optical strain sensors with a Fibre Bragg Grating (FBG). These have a number of appealing advantages for application in aircraft structures, such as light weight, tolerance for harsh environments, long term stability, complete passivity and no interference with other signals. The optical fibres were surface mounted, but can be embedded in the composite structure as well. Measurements were performed on the undamaged panel and two damaged configurations. Currently, the measurement results are evaluated.

3.6 Structural health monitoring (J.H. Heida and J.S. Hwang, NLR)

The NLR is conducting a full scale fatigue test on the LHS wing of a decommissioned F-16 Block 15. The wing will be tested to failure or two times the design life, whichever comes first. Information about this test programme is provided in section 7.1.

In addition to the instrumentation with conventional strain gauges (e.g. single gauges, full bridges and rosettes) also special structural health monitoring (SHM) techniques are used during the test. SHM includes acoustic emission sensors (AE) and optical fibres with fibre Bragg grating sensors (FBG). AE monitoring with a 16-channel SAMOS AE system (Physical Acoustics B.V.) and 150 kHz resonant sensors covers five critical locations on the wing: three locations on the upper wing skin and two on the lower wing skin. The monitoring of two access holes on the upper wing skin has been combined with monitoring using optical fibres with FBGs. Two FBG interrogators are used: a Deminsys system of Technobis Fibre Technologies (wave length range 830 - 870 nm) and an IFIS100 system of the Korean company Fiberpro (wave length range 1510 - 1595 nm). A total of 19 FBGs are installed on the upper wing skin, divided over three fibres. 12 FBGs are positioned alongside conventional strain gauges, 6 FBGs are placed in an array along two access holes, and one FBG is used for temperature compensation. Figure 9 gives a schematic of the placement of the SHM sensors on the upper wing skin.

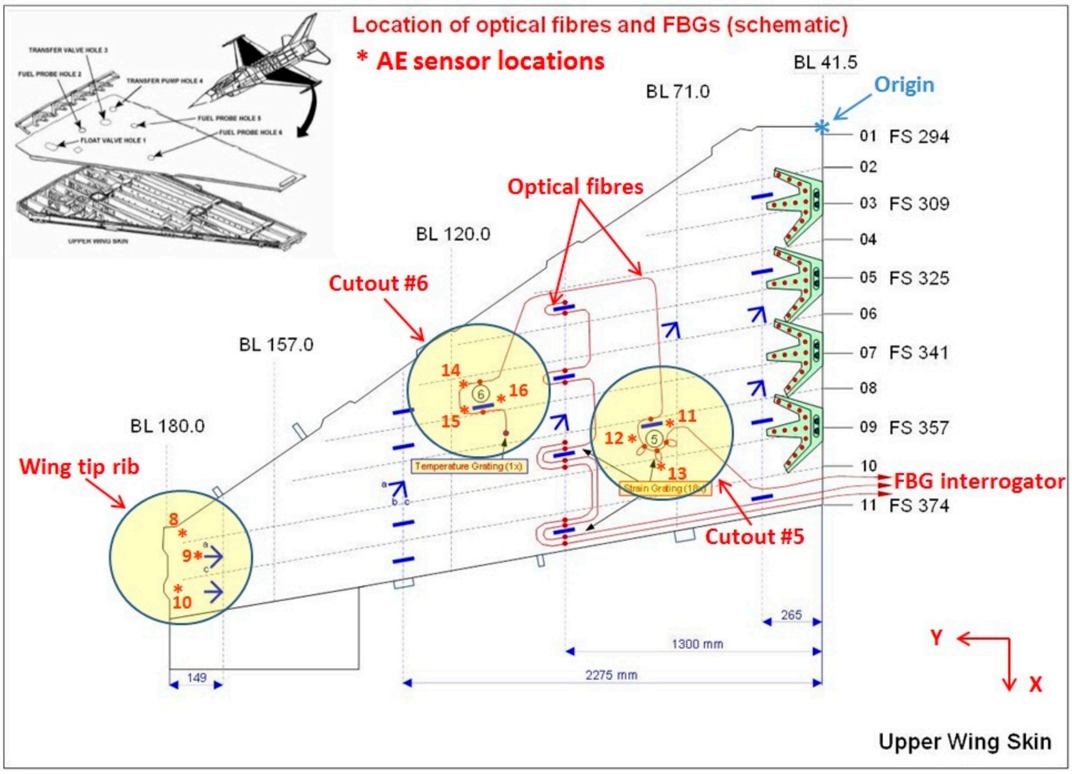


Figure 9: Monitoring of the F-16 LHS upper wing skin with AE and FBG sensors.

4 Fibre/Metal Laminates & Hybrid Materials/Structures

4.1 Fatigue crack growth in FMLs under variable amplitude loading (S.U. Khan, R.C. Alderliesten, TUD)

The topic of this study has been discussed in previous national reviews. Currently, the PhD dissertation is printed and available from the TU Delft library repository [6].

4.2 Residual strength of FMLs containing fatigue cracks (R. Rodi, R.C. Alderliesten, TUD)

As part of the investigation into the residual strength mechanisms in FMLs, a theoretical model has been developed that describes the damage growth under quasi-static loading including the large scale plasticity effects based upon critical crack opening angles (CTOAc). The damage cases considered are the accidental damage scenarios earlier addressed in an empirical model (R-curve) by De Vries, and the case of a fatigue crack with delaminations as addressed by for instance 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 the failure of the laminate. All these mechanisms and their interactions have been described by means of elastic-plastic fracture mechanic tools and implemented into an analytical prediction model.

The topic of this study has been discussed in more detail in previous national reviews. Currently, the PhD dissertation is printed and available from the TU Delft library repository [7].

4.3 Fatigue damage growth in arbitrary hybrid FMLs (G. Wilson, R.C. Alderliesten, TUD)

A theoretical, mechanistic 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 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 constituent material properties, laminate configuration, and loading conditions. This topic has been discussed in more detail in previous national reviews. Currently, the PhD defence is scheduled for June 2013, and the dissertation will then be available from the TU Delft library repository.

4.4 Directionality of damage (M. Gupta, R.C. Alderliesten, TUD)

To extend the application of Glare to aircraft structures other than the fuselage skin, in particular the lower wing skin with unidirectional Glare, it is necessary to understand the crack path propagation under off-axis loading. Crack paths in Glare under pure mode I fatigue loading (when load is in in-axis with the fibre direction) are perpendicular to the loading direction. However, a slight change in the loading angle with respect to aluminium rolling direction (and thus principal fibre directions) deflects cracks, see Figure 10. The current research project aims to develop the physical understanding and analytical models to predict these deflections under off-axis loading of Glare.

First, the fibre bridging mechanism has been identified as the parameter influencing the crack paths under fatigue loading. Experimental results obtained through Digital Image Correlation (DIC) show the strain fields at the crack tip to extend at different angles under different off-axis loading angles. Besides, it was also observed that different Glare grades have different angles of crack propagation depending on the orientation of fibres and the amount of fibres in the 2 directions – rolling direction of aluminium and perpendicular to it.

It was concluded that local crack tip parameters such as fibre bridging play a significant role in crack deflection in Glare [9,10]. Therefore, a mixed-mode model which is dependent on the effective Stress Intensity Factors is deemed more appropriate for predicting crack paths in FMLs than the widely popular T-stress theory of predicting crack path instability [11,12].

The existing analytical models only address the mode I loading calculations in FMLs. To develop the analytical model for crack path prediction in FMLs, it should also include mode II sliding in order to calculate the mode-mixity parameter that arises at the crack tip due to fibre bridging in both directions. Therefore, future actions to be performed in this researched are: (1) Develop mode II analytical model for FMLs under fatigue loading, (2) Develop the mixed-mode model for crack path prediction for fatigue loading, (3) Crack path in FMLs under biaxial fatigue loading, and (4) extend the knowledge of crack paths under fatigue loading to Large Scale Yielding conditions.

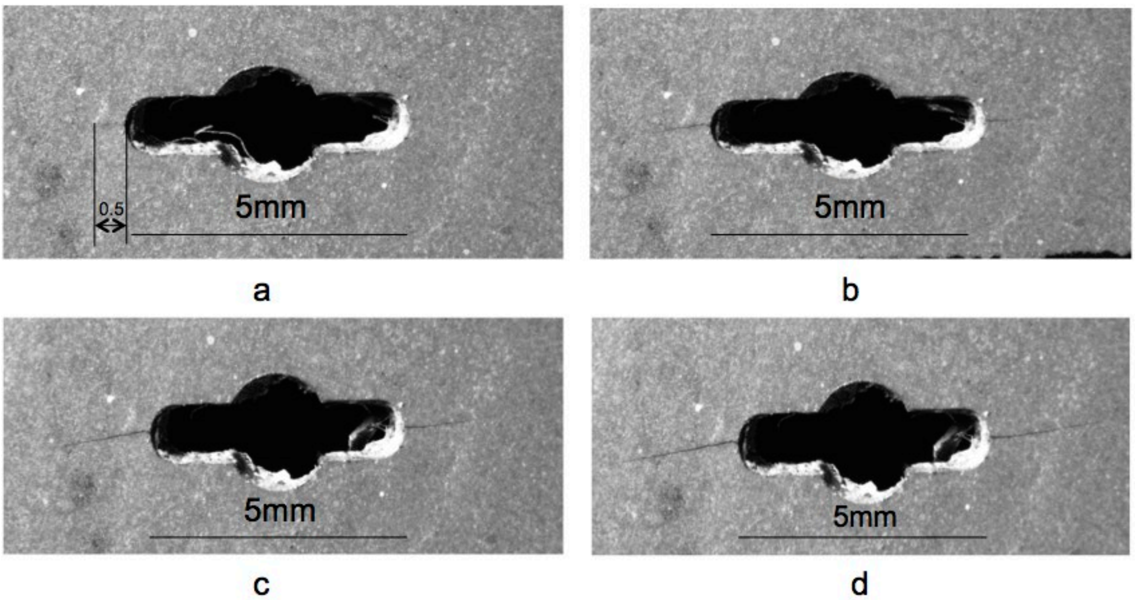


Figure 10: Crack propagation in Glare3-4/3-22.50 off-axis loading at various stages of fatigue loading - a) 19,000 cycles, b) 20,000 cycles, c) 30,000 cycles and d) 40,000 cycles.

4.5 Multiple Site Damage in FMLs (W. Wang, R.C. Alderliesten, TUD)

The state-of-the-art analytical models to describe fatigue damage growth in Fibre Metal Laminates, only address the crack propagation in the metal layers and the delamination growth at the interfaces for a single crack. In the past, engineering theories have been developed that predict the remaining residual strength of mechanically fastened joints containing Wide Spread Fatigue Damage based on a ratio of accumulated damage to intact material.

To enable the growth of multiple collinear cracks in FMLs, corrections to the single crack models have been proposed, but theoretical models describing the interaction of growth have not yet been developed. This study aims to develop a fundamental model in which crack interaction is considered in FML structures. The principles considered are basically similar to previous studies where models have been developed for single crack propagation in presence of stiffeners and straps. It is considered that the presence of multiple cracks has the opposite influence that a stiffener or strap has. It therefore is the objective to develop the theory based on the superposition principles for stress intensity factors.

5 Composite Materials & Structures

5.1 Static and fatigue behaviour of impact damaged thick-walled composites (R. Creemers, NLR)

Impact testing

Within a research programme jointly conducted by the NLR and Fokker Landing Gear BV to investigate compression design strains for thick composites, small-mass high-velocity 90 J impact tests have been executed on 20 mm thick CFRP structural elements using a one inch diameter steel impactor. None of the specimens showed much damage upon visual inspection after impact. However, microscopic photography after sectioning showed extensive damage with contact damages just below the surface and a conically shaped region of matrix shear cracks in the upper half of the specimen, from which delaminations extend outwards. Inside the conical shape of cracks the material was undamaged. In the lower half of the specimen, below the undamaged region, matrix cracks and large delaminations were identified.

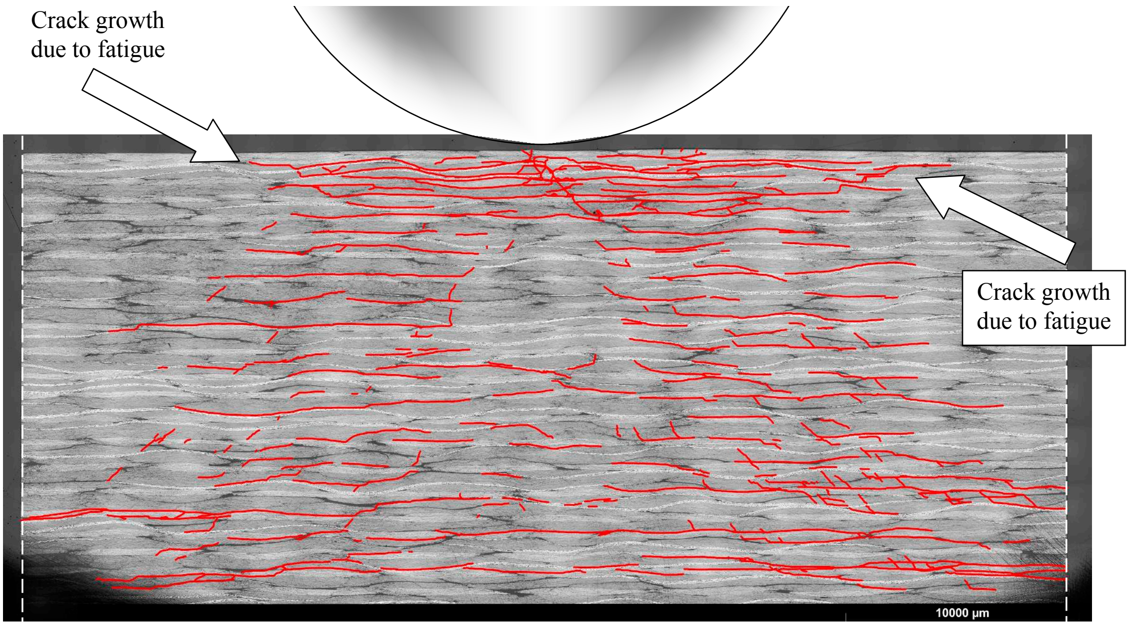


Figure 11: Example of the damage distribution in thickness direction of a 20 mm thick CFRP plate after a high-velocity 90 J impact and subsequent fatigue testing in compression.

Static and fatigue testing

Subsequent static compression testing showed that the large amount of impact damage results in a 40% strength reduction compared to undamaged specimens. This is in contradiction with data found in literature where hardly any strength reduction in thick laminates was reported for high impact energy levels. Based on the test results, it was concluded that the compression design strains of the current thick-walled composites are comparable with those for thin-walled composites.

Additionally, constant amplitude compression fatigue tests (R = 10) have been performed on the impact damaged specimens and an S-N curve was established. During these fatigue tests C-scan inspections, using Time of Flight (TOF) processing, were performed at regular intervals to detect any damage growth prior to failure. The TOF data indicated that damage growth starts at the small delaminations at the front (impact) side and that the delaminations grow in width direction of the specimen by stable delamination growth. This would not have been captured when looking at the total delamination area, because the smaller delaminations are covered by the largest one.

Figure 12: Sub-laminate buckling at the front (impact) side during compressive fatigue testing of a 20 mm thick CFRP plate after a high-velocity 90 J impact.

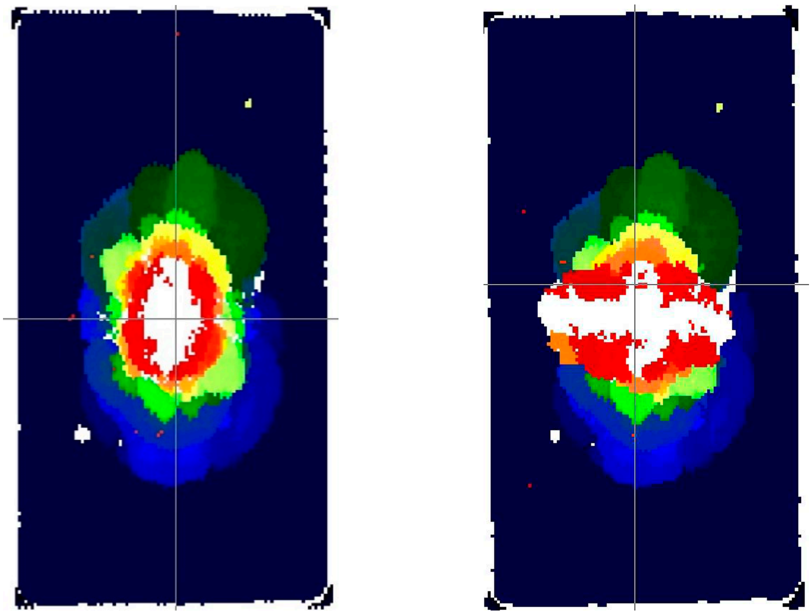


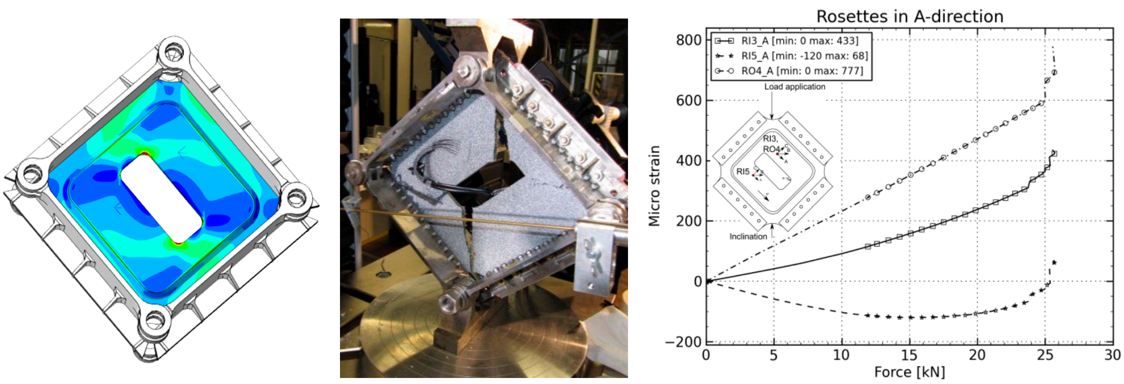
Figure 13: Damage growth starting at the front (impact) side under compressive fatigue loading in a 20 mm thick CFRP plate after a high-velocity 90 J impact.

5.2 Static and fatigue behaviour of a shear loaded fabric CFRP sandwich panel (G. van de Vrie, NLR)

Efficient operation of a military fleet of helicopters and fixed-wing aircraft critically depends on the ability of the operator to monitor the structural integrity of each individual vehicle. The NLR has supported the Royal Netherlands Air Force (RNLAF) with structural health monitoring of their helicopters and F-16 fighters for many years, with the main focus to develop and monitor damage indicators that allow an operator to quantify the amount of "life" consumed by the vehicle and schedule maintenance and repair on a per case basis. Since the primary structures of the helicopters and F-16s operated by the RNLAF almost exclusively are made of metal, development of damage indicators like the Crack Severity Index (CSI) traditionally focussed on failure behaviour of metals. However, with the recent addition of the NHI NH90 multi-role NATO helicopter to the RNLAF fleet, the need has arisen to expand the range of damage indicators to composite materials and structures.

One of the reasons that damage indicators have not yet been developed for composite aircraft structures is the common belief that composite materials do not suffer from fatigue. This is partly true as most of the composite aerospace structures are designed to a maximum allowable strain under normal operation conditions that is in the order of 3000 micro-strain. In general this strain level is below the level for which fatigue becomes an issue. However, this is only true for the in-plane properties. In cases where out-of-plane loadings are significant (e.g. ply-drop-offs, eccentrically loaded sandwich facings and local attachment points), fatigue resistance should be taken into account.

NLR has conducted an extensive research project funded by the RNLAF in order to better understand in service composite materials behaviour and to develop damage indicators that can be used for structural health monitoring of composite structures. Part of this project consisted of studying the physics behind static and fatigue failure behaviour of composite materials and capturing this behaviour in advanced simulation models. To demonstrate the performance and applicability of currently available damage models a representative demonstrator shear panel with a rectangular cut-out was designed, analyzed and tested both statically and in fatigue, under R=-1 loading - see Figure 14. No impact damages were inflicted.



a) Pre-test analysis *b) Test setup* *c) Strain result during static testing*

Figure 14: Static damage development in a shear loaded carbon fibre fabric reinforced composite panel.

Pre-test analyses of the static shear test have been performed using different damage modeling techniques, incl. progressive damage analysis and the use of the eXtended Finite Element Method (XFEM) for the prediction of damage initiation and propagation. Damage initiation in the demonstrator panel was predicted quite accurately and reasonable results were obtained for the damage progression analysis. However, due to numerical convergence and stability issues, obtaining these results was less straightforward than might be expected on the basis of the high stress concentrations in the corners of the cutout.

The fatigue test results were in line with reported results in literature on coupon scale. Based on the results it is expected that relatively simple equivalent stress methods can be used for predicting the life of CFRP components, provided that the failure mode remains similar, i.e. through crack type of failure with little or no delamination. The scatter found in the fatigue test programme was surprisingly low, as one order of magnitude of difference is often encountered for composites. A power law fit was made which covered the data well see Figure 15. Note that the applied force can be assumed to be (nearly) linearly related with stress as long as bending remains small.

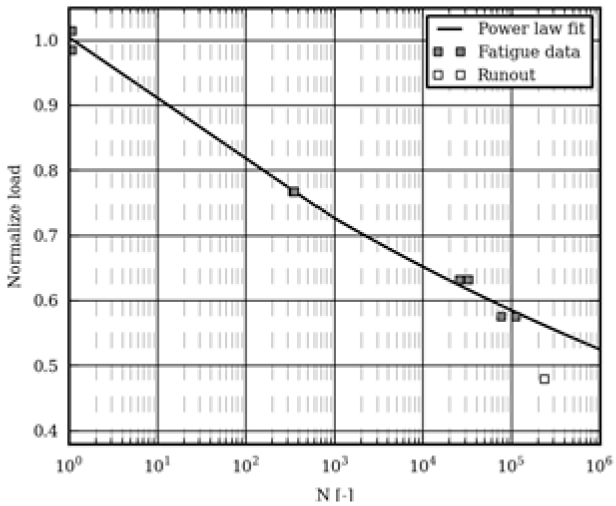


Figure 15: Fatigue life diagram of the shear panels for R=-1 loading. A power law fit has been made through the fatigue data.

Fatigue failure occurred only at relatively high load levels. It is expected, however, that with the increasing pressure on weight savings and with increasing knowledge, composite structures will be loaded higher and higher. Future work will consist of predicting the fatigue life of components. The influence of spectrum loading and more complex fatigue damage phenomena will have to be accounted for.

During the test programme novel measurement techniques were used like Comparative Vacuum Monitoring (CVM) patches to detect cracks and digital image correlation (DIC) for full field displacement and strain measurements (Figure 16). Both traditional strain gauges and CVM patches performed very well in detecting damage. Acoustic emission also gave a good indication of damage initiation and progression, although it was a more indirect way of detecting damage compared to for example CVM patches.

There was a good correlation between analysis and experiment for load and corresponding displacements and, thus, panel stiffness. The measured strains in the static tests corresponded very well with the strains obtained during the analysis, especially during the phase prior to failure and at the early stages of damage development; after that comparison became difficult as only one corner crack was modeled.

This work is published in ref. [13].

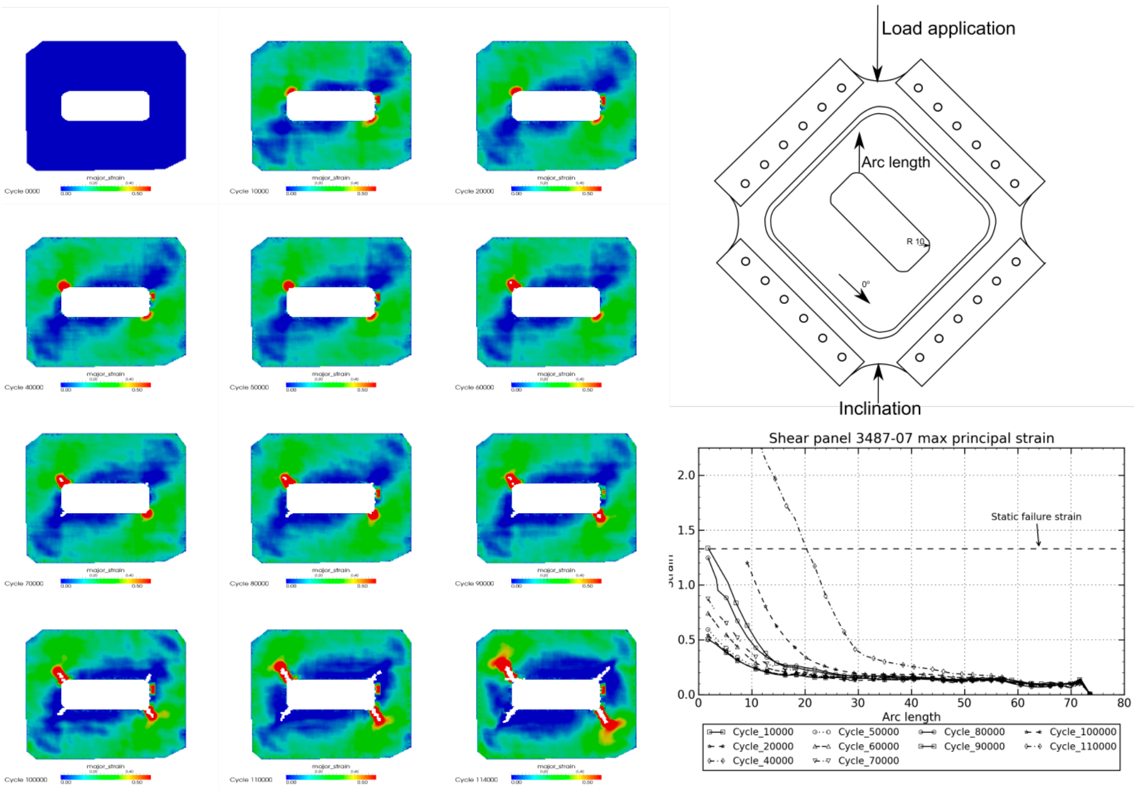


Figure 16: Fatigue damage development in a shear loaded carbon fibre fabric reinforced composite panel. The images on the left show the strain field obtained from the digital image correlation measurement at the peak load of the fatigue spectrum at different stages of the fatigue test. Initiation and development of the corner cracks can be clearly identified. The graph on the right shows how strain develops in the upper left corner of the panel.

5.3 A Damage Tolerance certification methodology for single load path, safety critical composite structures (E. Bakker, TUD / FLG)

Impact damage is of great concern for the certification of composite structures for aerospace applications. To ensure safe operation, visual inspection is used to detect impact damage and take appropriate action. However, “thick” composite structures (>10 mm) as foreseen to be used in for example landing gears typically do not show visible damage up to extremely high impact energy levels. Since such impacts are extremely improbable to occur (order of magnitude of 10^{-9} per flight), it is acceptable to ensure safety by defining an impact energy cut-off level and by using the so-called ‘enhanced safe-life’ methodology; ultimate load and “no-damage-growth” capability must be maintained throughout the entire service life of as-manufactured structure, impacted with cut-off level impact energies at critical locations. The enhanced safe-life methodology essentially is a combination of ‘safety-by-retirement’ and ‘safety-by-improbability’ (of occurrence). This research explores the opportunities to expand the level of safety by additionally ensuring ‘safety-by-inspection’ for thick composite structures, thereby adopting the damage tolerance methodology. Furthermore, the advantages such as expected weight savings are explored by deriving the minimum qualification requirements through a systematic approach.

The key to adopt the damage tolerance methodology is to introduce a Non-Destructive Inspection (NDI) technique that allows earlier detection of damage. The extent to which damage can be detected depends on the threshold of detectability of the NDI technique. Appropriate inspection intervals need then to be established to define an approach that meets safety requirements.

In line with the above, a systematic top-down damage tolerance methodology is proposed using a requirements-based engineering philosophy to capture all relevant requirements. In this methodology the inspection method, its probability of detection, the inspection interval, the probability of load occurrence, the probability of damage occurrence and the residual strength of structure after impact are related to a single safety parameter: probability of failure. By defining an acceptable level of safety, a minimum residual strength after impact can be obtained. The required strength is directly related to possible weight savings compared to the enhanced safe-life methodology. This approach only illustrates the validation steps during requirements capture. For certification, the entire validation and verification cycle must be completed, including validation of the qualified structure.

The proposed methodology creates insight in the effect a design choice has on the level of safety. At the same time, the level of safety is quantified and can be used for comparison to other systems and for in-service validation of structures. Additionally, significant weight reductions can be expected compared to the current enhanced safe-life methodology.

5.4 Delamination growth analysis on a Butt-Joint T-stiffened panel (P. Lantermans, Fokker)

A damage tolerance analysis has been performed on a 3-stringer test panel, which is tested as part of a technology development programme. The skin and stringers are made from thermoplastic composite material (Carbon/PEKK) and connected with a butt-joint between stringer web and skin (Figure 17). There are no continuous fibers from one plate to the other. The joint is made entirely by the filler (PEKK with 20% short Carbon fibers), which transfers all loads from stiffener web to stiffener cap and skin.

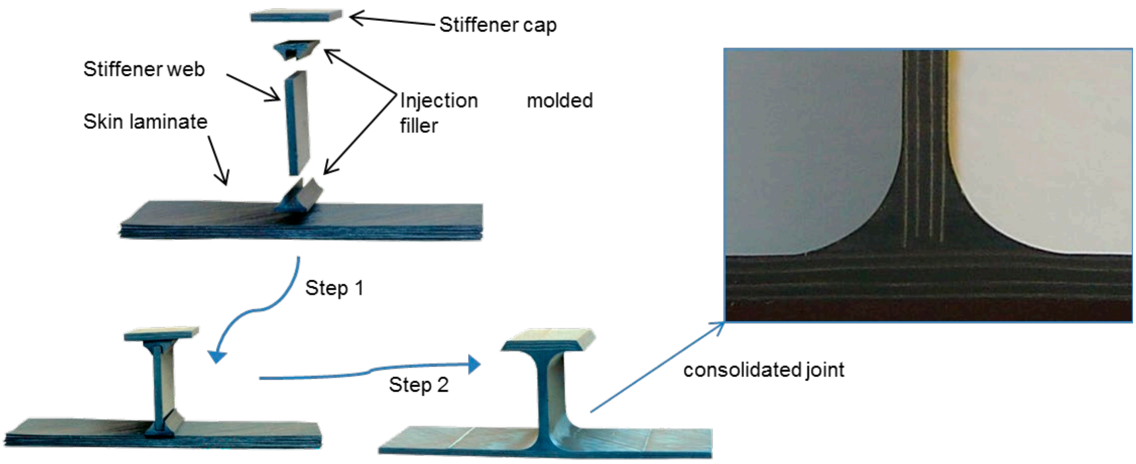


Figure 17: Butt- joint parts and assembly.

A partial delamination between the stringer web and the surface of the skin at the run-out on one end of the central stringer was selected as the baseline damage scenario for test and analysis, as shown in Figure 18 and Figure 19. This initial delamination location was intended to be “conservative” based on the idea that other delamination interfaces and/or impact damage states would likely be more complex and thus require more energy to grow. The initial delamination length was selected to be 65 mm based on the post-impact damage state in a full-scale test article as shown in Figure 18. During fabrication of the test panel a foil insert will be used to create an artificial delamination representative of an impact damage at this location to facilitate analysis correlation.

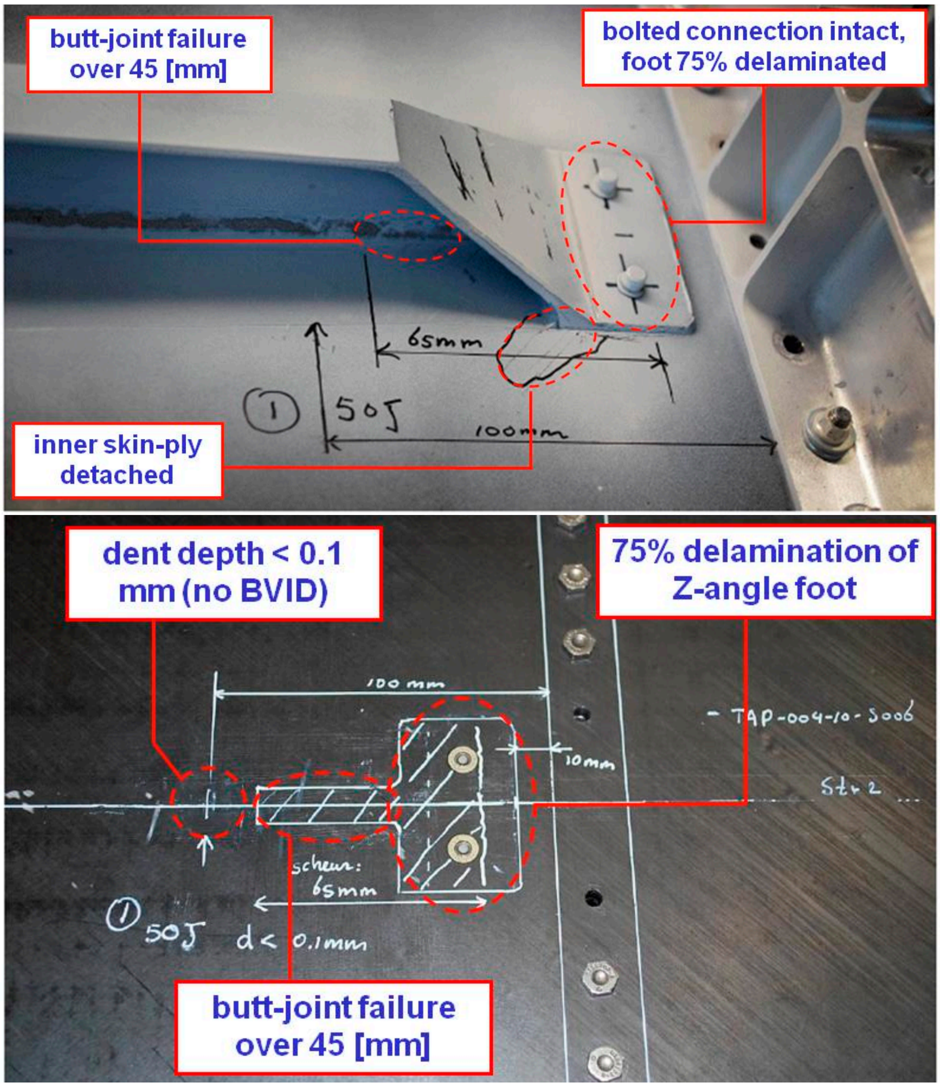


Figure 18: Post-impact damage state at stringer run-out.

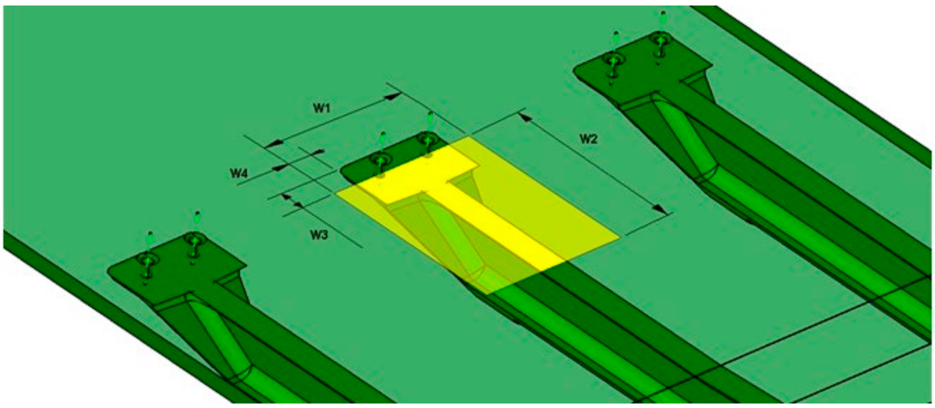


Figure 19: Modeled delamination.

Fatigue and damage tolerance analysis has been performed for the damage / delamination scenario using a fracture mechanics approach. This approach used the results from a detailed finite element model together with fracture toughness coupon data to predict delamination growth under constant amplitude cyclic loading. The objectives of the analysis were as follows:

- to predict the threshold constant amplitude cyclic load below which delamination growth will not occur (growth threshold);
- to determine at what cyclic load level significant delamination growth is expected under constant amplitude cyclic mechanical loading. ("short growth" analysis, i.e. 1.2 mm damage increment);
- to predict the direction of delamination growth;
- to predict if delamination growth is stable or unstable (stability indicates possible arrestment).

The analysis methods used to predict delamination growth are still evolving and the accuracy of the predictions relative to the test results depends on many factors, including:

- how well the idealized damage/delamination state represents the actual post-impact state;
- the availability of fatigue fracture toughness coupon data (i.e., da/dN vs. Gmax) for the delamination interface and loading of interest and how well the coupon data represented the "in-situ" toughness at the sub-component scale;
- the amount of scatter in composite fatigue test data (typically more than for metals) as well as the sensitivity of the predicted number of cycles to small changes in cyclic load.

Despite these limitations, conservative interpretation of the analysis results can generally be used to determine whether the threshold of delamination growth will be exceeded at a given cyclic load level. Also, the analysis results can be used qualitatively to determine the relative criticality of different damage states, damage locations, structural configurations, and loadings.

Several unanticipated behaviours were predicted:

- threshold cyclic load levels (~ -400 kN) well above Ultimate Load;
- no delamination growth for realistic cyclic test loads; and,
- nonlinear response beyond load levels of -400 kN due to panel beam-column effects resulting in decreasing strain energy release rates (G's) at the crack front with increasing load.

Analyses were also performed for other initial delamination lengths (100 mm and 125 mm) to evaluate the predicted stability of the delamination growth.

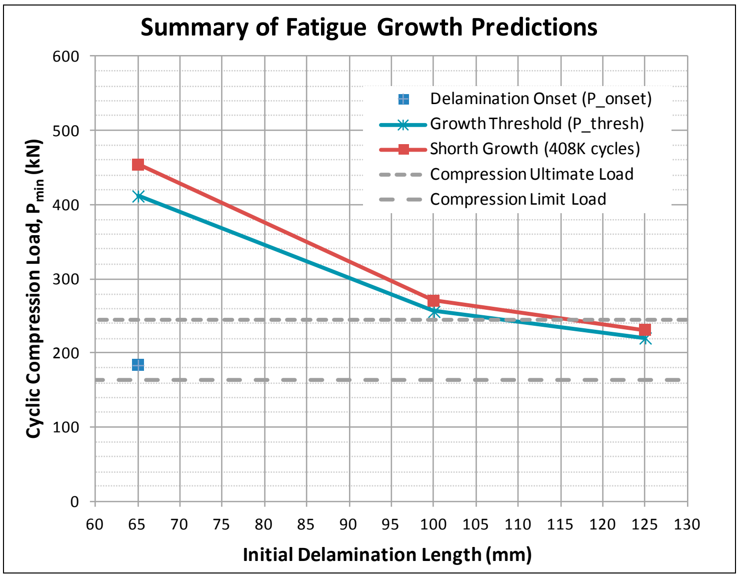


Figure 20: Fatigue Growth Predictions.

As can be seen in Figure 20, the predicted cyclic loads decrease with increasing delamination length, indicating unstable growth behaviour. The structural response for these initial delamination lengths is highly nonlinear, which precluded additional characterization of fatigue delamination growth at higher cyclic loads.

In lieu of the fatigue delamination growth characterization at higher cyclic loads, additional "short growth" fatigue delamination analyses were performed at the two additional initial delamination lengths (100 mm and 125 mm). These results were combined with the results from the threshold analysis (P_{thresh} values plotted as "run-out" at 1,000,000 cycles), to produce the "P vs. N" curve of Figure 21.

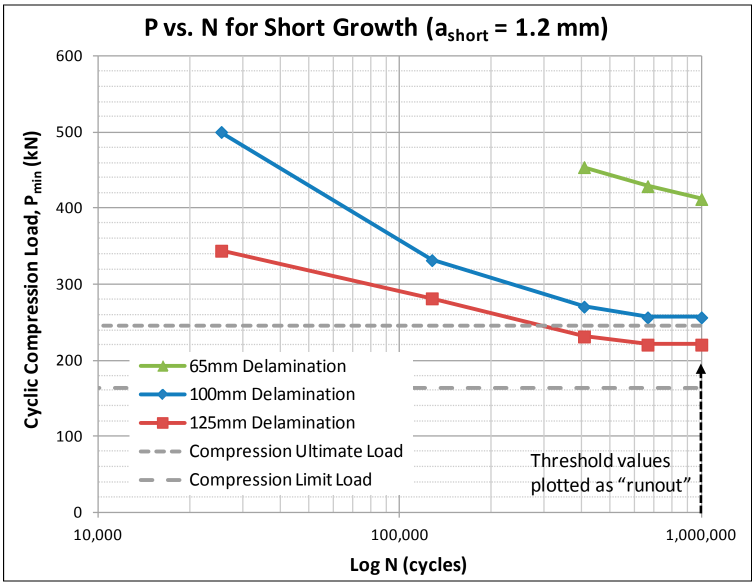


Figure 21: Short growth analysis.

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

The topic of mode I delamination growth in CFRP has been discussed in more detail in the previous national review [2]. The work has almost been completed. The effect of the stress ratio on the delamination resistance has been studied in detail. In particular the contribution of fibre bridging and crack closure has been assessed. The conclusion was that both may have an influence on the delamination resistance, but that the influence is marginal and does not explain the stress ratio effect.

As a consequence, the delamination resistance is evaluated with a two-parameter model approach, in which both the cyclic and monotonic load component of the load cycle are considered. It is concluded that the stress ratio effect appears to be part of a surface in a 3D representation of the delamination resistance, see Figure 22.

Fractographic evaluation of the fracture surfaces revealed that the two-parameter models proposed in the literature, which multiply the cyclic and monotonic component, may not well represent the physical process of delamination growth. Evaluation of fractographic features such as striations in the fibre imprint and hackle formation in between the fibres, seems to indicate that superposition of both load component contributions seems more appropriate.

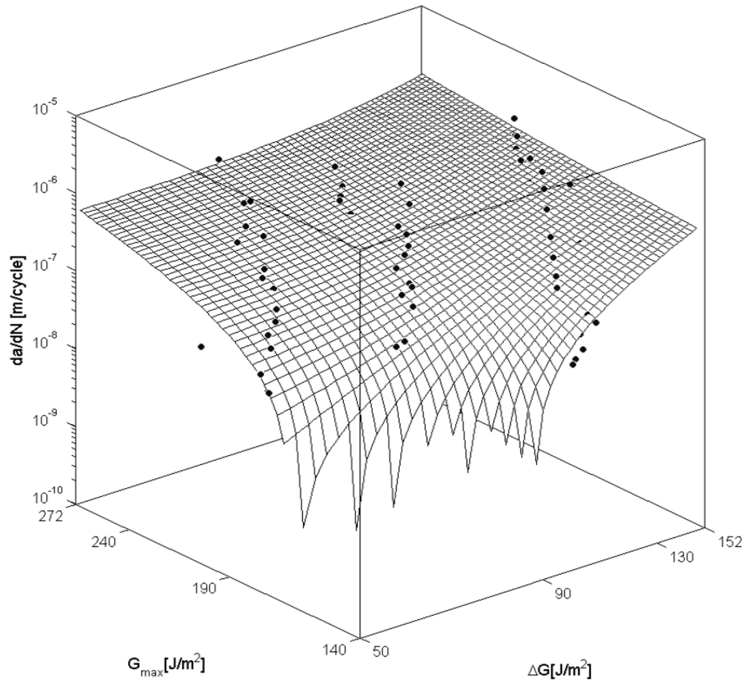
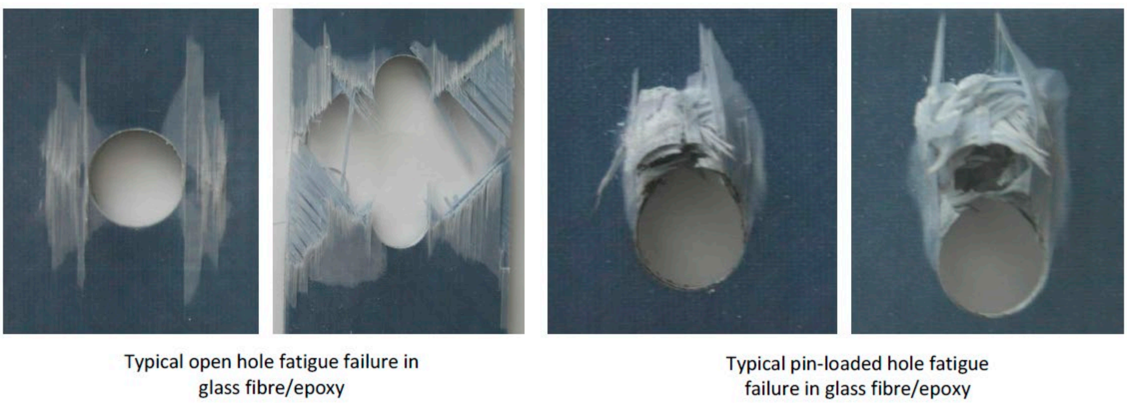


Figure 22: Illustration of the surface describing the mode I delamination resistance based on Gmax and ΔG obtained with performing DCB tests at three different stress ratios.

5.6 Fatigue damage evolution in GFRP under fatigue loading (B.C.W. van der Vossen, R.C. Alderliesten, TUD)

A small study has been performed in which different unnotched and notched GFRP specimen have been tested under fatigue loading to monitor the damage progression over time. The objective of the study was to assess the validity of predicting fatigue lives using S-N data obtained for different notch geometries. Although theoretically the stress concentration factor can be calculated for the various geometries, the fatigue life cannot be related using this factor. It is observed that the damage progression leading to failure is substantially different for the three investigated cases, see Figure 23. It is concluded that predicting the fatigue life based on laminate stresses can only be done once S-N data is used for similar damage growth patterns [14].



Typical fatigue failure in un-notched
glass fibre/epoxy

Figure 23: Comparison between failure patterns of unnotched, open hole and pin-loaded hole configurations in composites.

5.7 Enhanced material qualification of composite materials by probabilistic analyses (F.P. Grooteman, NLR)

According to the certification standards, in the design of structures use has to be made of restricted values for the material properties, the so-called allowables. These are lower bound values taking into account the scatter in these properties to arrive at safe structures, see figure 24. In general, this scatter is more pronounced in composites than metals.

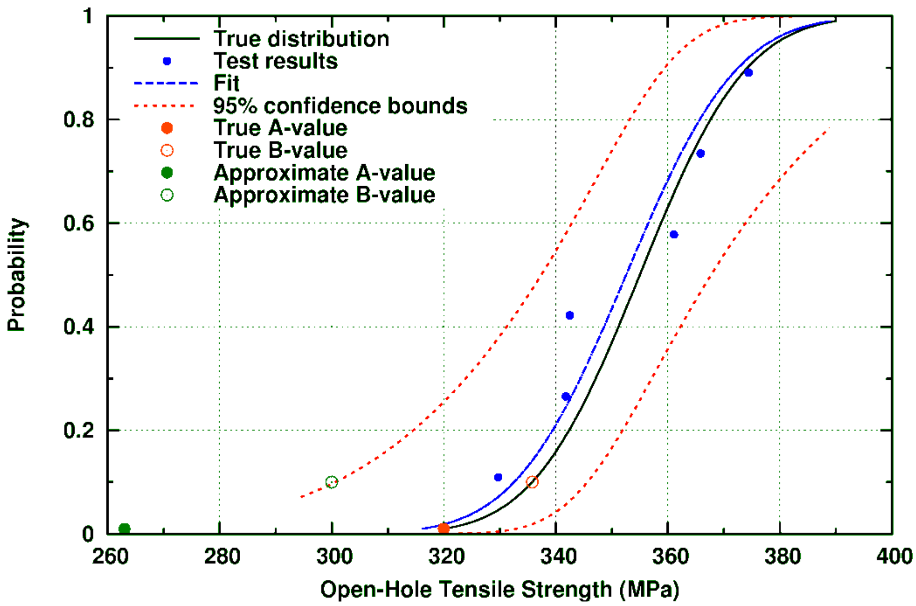


Figure 24: Allowable strength determination for a sample size of 6 data points.

A probabilistic framework is developed, as part of the investigation and development of a certification methodology for single load path, primary structural elements, employing large thicknesses, complex shapes, manufactured by Liquid Resin Transfer Moulding (RTM) processes and using large tow sizes.

To capture the sensitivity of the structural response of thick composite components to variations in the materials and the manufacturing process requires an enormous amount of tests if standard methods are applied. The costs of these activities (financial and time) are high and can form a barrier towards the commercial introduction of composites in landing gear of civil aircraft. It is therefore important to develop new methods in order to substantially reduce the number of tests that are required in the generic building block levels.

By means of a probabilistic approach, the number of tests can be substantially reduced and mainly restricted to the determination of the distribution functions for the basic lamina properties together with a number of validation tests on some laminates. Hence, allowables for other laminates (stacking sequences) can for the larger part be determined by analyses and only require a limited number of additional tests to validate the results. Such a probabilistic approach requires an accurate deterministic model with which the response of the material can be predicted and consists of a finite element model simulating the tests incorporating a sophisticated material model developed by Fokker Landing Gear.

Within the project a large number of coupon tests are performed to generate the stochastic datasets required in the probabilistic framework.

6 Metallic Materials & Structures

6.1 NASGRO stress intensity factor solution for a center through crack in a finite width plate subjected to a symmetric remote displacement field (R.A. Huls, F.P. Grooteman and R.P.G. Veul, NLR)

As part of the ESA contract Structural Integrity of Pressurised Structures (SIPS), NLR determined a new stress intensity factor (SIF) solution for a center through crack in a finite width plate subjected to a symmetric remote displacement field, figure 25. This new stress intensity factor solution will be implemented within the NASGRO fracture and fatigue analysis suite as crack configuration TC24, extending the limited set of displacement controlled configurations. It is a variation of the existing NASGRO crack configuration TC14 implemented by Southwest Research Institute (SwRI): an edge through crack in a finite width plate subjected to a symmetric remote displacement field.

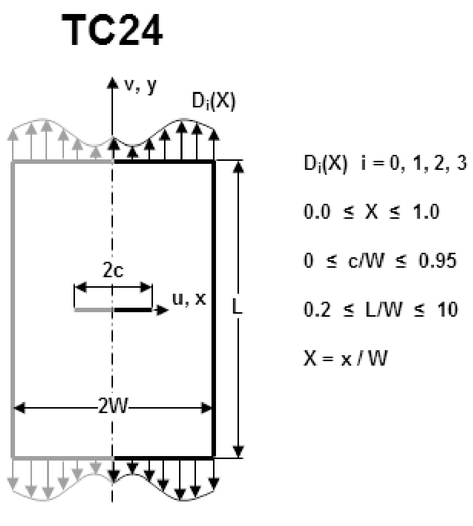


Figure 25: Center through crack in a finite width plate subjected to a symmetric remote displacement field.

A large number of linear elastic stress intensity factor calculations were performed in Abaqus 6.11 using crack tip elements to represent the stress field at the crack tip. The models are half models around the symmetry plane of the plate. The displacement loading is also fully defined on these half models. Calculations were performed for various L/W ratios (length over half plate width) and a/W ratios (half crack length over half plate width) for four displacement fields defined by Chebyshev polynomials. Both plane stress and plane strain states where taken into account. Furthermore, all SIF calculations were performed for two boundary conditions: 1) the remote ends are free to contract or expand perpendicular to the loading direction; 2) the remote ends are restricted in the perpendicular direction from expansion or contraction (fixed grip loading).

The accuracy of the results was verified by comparison against solutions obtained from literature. The finite element results of the normalized stress intensity factors (SIF) are implemented in a NASGRO fracture mechanics module. Together with a number of test cases and a proposed graphical user interface (GUI), this forms the bases for the implementation of TC24 in NASGRO.

6.2 Fatigue crack growth in highly loaded launcher components (F.P. Grooteman, R.A. Huls and R.P.G. Veul, NLR)

Launcher structures often contain components that are subjected to high load levels causing global plastic deformation. The current design approach for such highly loaded components is based on a classic damage tolerance approach consisting of a linear elastic fracture mechanics (LEFM) crack growth analysis. Load interaction effects are generally not taken into account. It is unclear whether the LEFM methodology can still be applied at these high load levels. Life prediction could therefore be improved by evaluating whether LEFM is still applicable and whether a retardation model can be applied. Further gain may be obtained by taking into account the scatter in loads and material properties by means of a probabilistic approach. To examine these issues, ESA started the technology research programme (TRP) “Fracture control/Damage tolerance methods for highly loaded launcher components”. NLR is the main contractor with Snecma and Cenaero as sub-contractors.

The main objective of this ESA TRP project is to examine the benefits of more advanced damage tolerance approaches for the prediction of highly loaded structural launcher components than the classic damage tolerance approach. The study examines application of load interaction models, probabilistic methods and other new methodologies for these types of structures.

The first part of the project consisted of a number of surveys for representative examples, state-of-the-art load interaction models, state-of-the-art probabilistic models and new methodologies like XFEM.

In the second part of the project a verification of the load interaction models, probabilistic models and new methodologies is performed. The main questions considered in the load interaction verification are whether linear elastic fracture mechanics (LEFM) can still be applied for these highly loaded structures and which, if any, retardation model should be used for these structures. The main questions in the probabilistic method verification are how probabilistic methods can best be incorporated in fracture mechanics and the applicability of such an approach. The new methodologies verification focuses on the applicability of XFEM and the gain that can be obtained over the traditional crack tip elements.

In support of the verification part, an experimental campaign was formulated and executed on IN718 specimens, a typical launcher component material, to create a data set against which the most promising models are evaluated. Static tests were performed to obtain static material properties and to verify material quality. Constant amplitude fatigue crack growth tests were performed on middle tension specimens (MT) under moderate loads with and without an additional plastic deformation before creating the starter notch, to evaluate the effect of plastic deformation on the fatigue crack growth rates. Constant amplitude displacement controlled tests were performed on single edge crack tension (SET) specimens to evaluate the behaviour at loads above the yield limit. Finally, variable amplitude (VA) displacement controlled fatigue crack growth tests for a representative launcher load spectrum were performed on SET specimens at comparable high load levels.

Slanted crack growth (single or double shear) occurred for the large scale plasticity SET specimens loaded by the CA spectra, while the specimens loaded by the VA spectra showed square crack growth, as depicted in figure 26. Plastic pre-deformation did not affect crack growth behaviour. Fatigue crack growth showed Paris like growth behaviour for all load ranges considered. Significant retardation was observed under variable amplitude loading, but this was not captured by the generalized Willenborg retardation model.

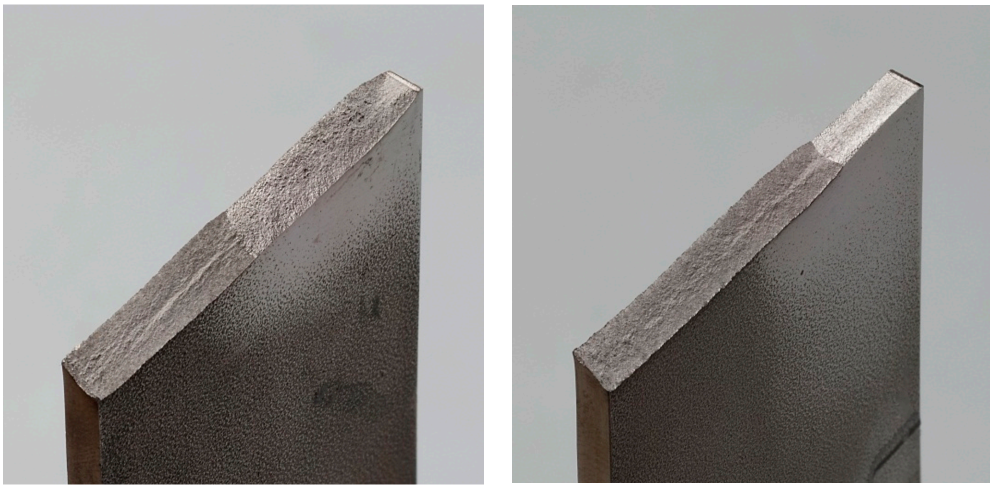


Figure 26: Fracture surface of CA (left) and VA (right) loaded SET specimen, from right to left showing the notch, the pre-crack, the (slanted) crack and the final rupture surface.

6.3 Fractographic observations on the mechanisms of fatigue crack growth in aluminium alloys (M. Krkoska, R.C. Alderliesten, J. Schijve, TUD)

Fatigue crack growth experiments are carried out on sheet specimens of aluminium alloys to obtain information about the behaviour of the moving crack tip during the increasing and decreasing part of a load cycle, associated with the crack opening and closure of the crack tip. For this purpose special load histories have been developed. Modern SEM techniques are applied for observations of the morphology of the fractures surfaces of a fatigue crack, see for example Figure 27, and to obtain information about the cross section profiles of the striations. Corresponding locations of the upper and the lower fracture surfaces are explored in view of the crack extension mechanism. Most experiments are carried out on sheet specimens of aluminium alloys 2024-T3, see Figure 28, but 7050-T7451 specimens are also tested in view of a different ductility of the two alloys.

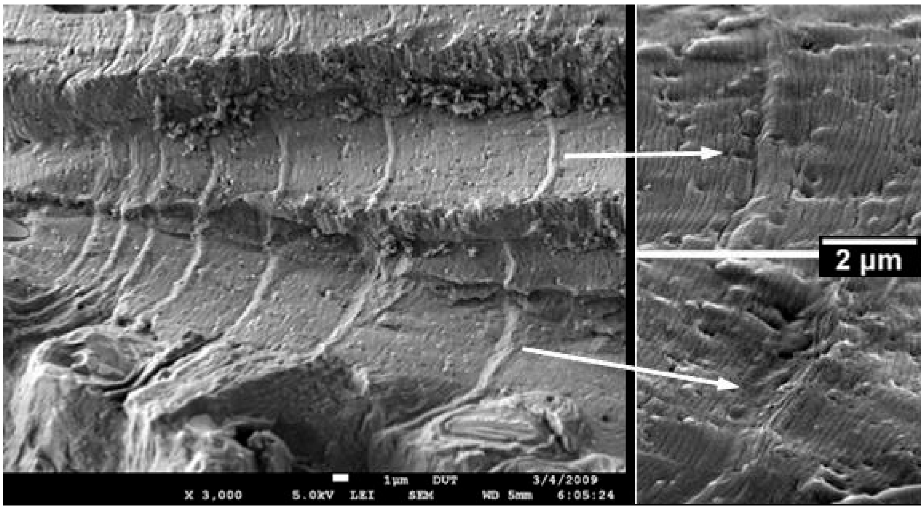


Figure 27: Fracture surface highlighting the influence of crack plane orientation on the ridge shapes (Kmax = 14 MPa √m). Crack propagates from left to the right.

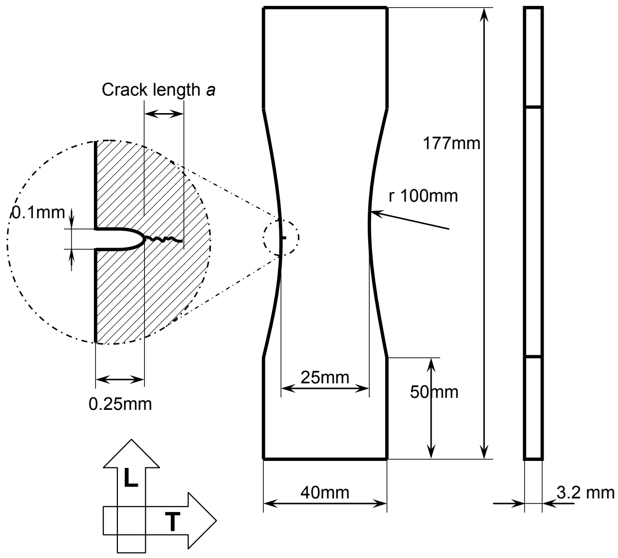


Figure 28: Illustration of the specimen geometry and dimensions.

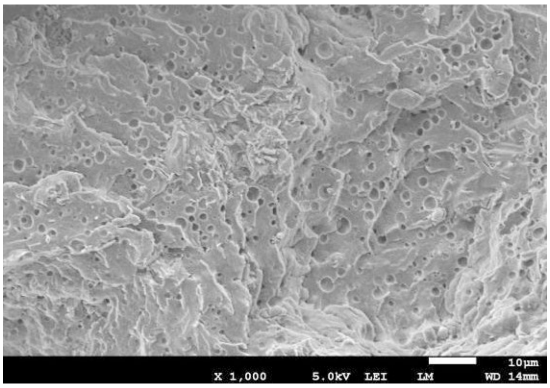
6.4 Mixed-mode fatigue disbond of metal-to-metal interfaces (D. Bürger[1,2], C. Rans[1,3], R. Benedictus[1])

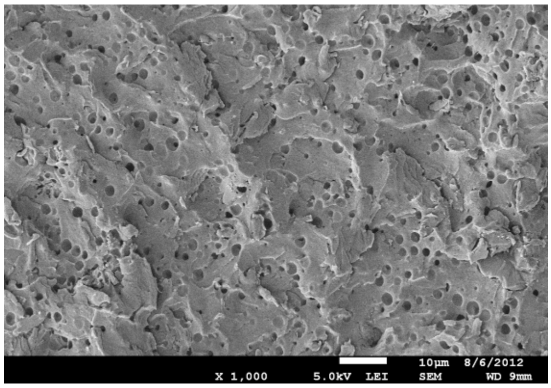
[1]*Delft University of Technology, Delft, Netherlands*
[2]*Aeronautical and Space Institute, São José dos Campos, Brazil*
[3]*Carleton University, Ottawa, Canada*

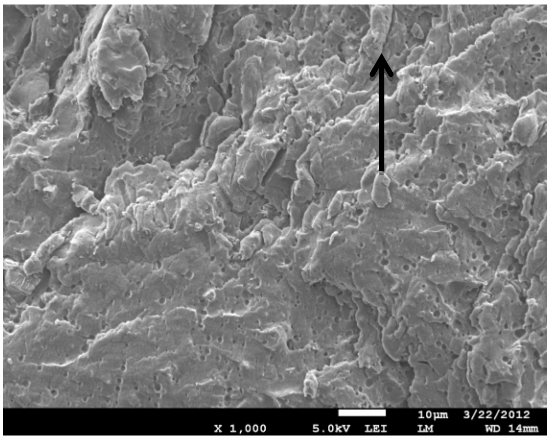
Currently, most metallic aircraft joints are made with rivets. The main disadvantage of a riveted joint is the high local stresses induced by the rivet, which lead to a shorter fatigue life. Adhesive bonding provides a promising alternative to riveted connections, providing a more efficient mode of load transfer and eliminating local stress concentrations associated with the fastener load transfer and fastener holes themselves. As a result, bonded joints exhibit better fatigue properties than their riveted counterparts. Despite this advantage, the widespread application of bonding for safety-critical structures is currently limited by the lack of robust fatigue and degradation prediction methods, and a lack of reliable damage inspection techniques.

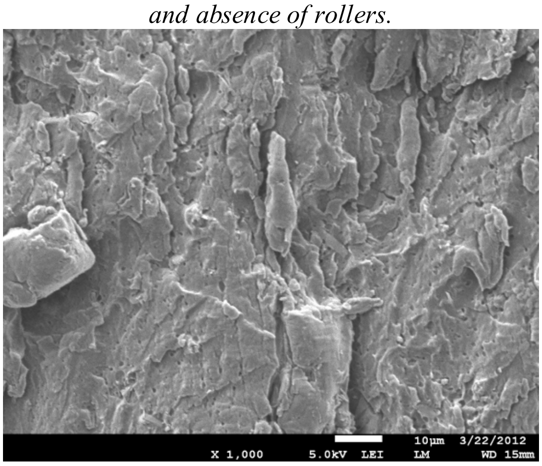
Failure in a bonded interface is classified according to the type of loading: peeling (Mode I) and sliding shear (Mode II). Most joint designs experience a combination of both loading modes called Mixed Mode. Empirical disbond models for pure Mode I and pure Mode II are well-known in the literature. However, Mode I and Mode II effects cannot yet be combined satisfactorily in a Mixed-mode theoretical model. As a result, the most accurate Mixed-Mode models are based on curve fitting experimental data obtained at different Mixed-Mode ratios. This approach results in an excessive experiments number, which is both costly and time consuming.

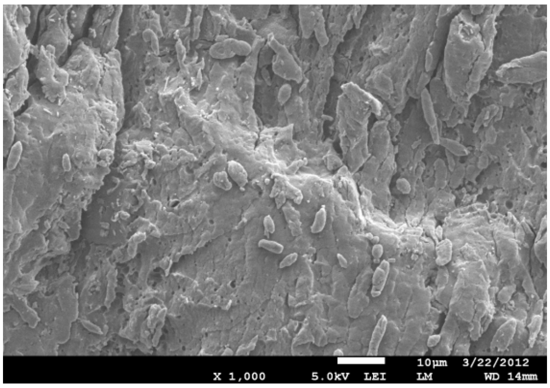
In the last two years, a series of fatigue experiments at fixed mode ratios was performed to identify the effects of pure Mode I and II, and Mixed-Mode loading. Fractography results indicate that no visible interaction exists in the Mixed-Mode loading, see Figure 29. In addition the fractured surfaces indicate three different regions. Mode I dominant (at high Mode I), Mode II dominant (at high Mode II) and a Mixed Mode region, between 30% and 70% of Mode II (G_{II}/G_{total}).
Additionally, the research is evaluating the effect of a supporting system (carrier cloth) on the Mixed-Mode fatigue disbond. Preliminary results, supported by fractography, indicate the carrier cloth bears a part of the Mode II loading, resulting in a slower crack growth rate for Mode II and Mixed-Mode loading. As for Mode I the carrier cloth seems to have a detrimental effect over the threshold value.


Pure Mode I (0%) – presence of scarps.

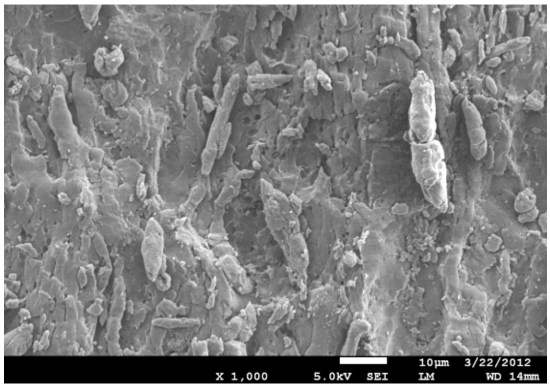

High Mode I region (30%) – presence of scarps and absence of rollers.


Mixed-Mode region (50%) – small amount of non-detached rollers (arrow) are detected.


Mixed-Mode region (70%) – both rollers and scarps are present in the surface.



High Mode II region (80%) – presence of rollers and absence of scarps.



Pure Mode II (100%) – presence of rollers.

Figure 29: Fractured surface evolution – the disbond grows from left to right. From pure Mode I, with the presence of scarps, to pure Mode II, with the presence of rollers.

The next research steps are to incorporate the above findings in a Mixed-Mode fatigue disbond model and to validate this model in a structure where the mode ratio is a function of disbond length.

6.5 Disbonding of Bonded Repairs (J.A. Pascoe[1], R.C. Alderliesten[1], C.D. Rans[1,2], R. Benedictus[1])

[1]Delft University of Technology, Delft, Netherlands [2]Carleton University, Ottawa, Canada

A model was developed for disbond growth in bonded repair patches under constant amplitude fatigue loading with a constant R-ratio. The model uses the strain energy release rate (SERR) as a similarity parameter, linking it to the disbond growth rate through a Paris relation. Finite element analysis, employing the virtual crack closure technique, is used to determine the strain energy release rate (SERR) as a function of disbond length. Fatigue testing of material coupons was used to determine the Paris parameters. Figure 30 depicts the model in a flow chart.

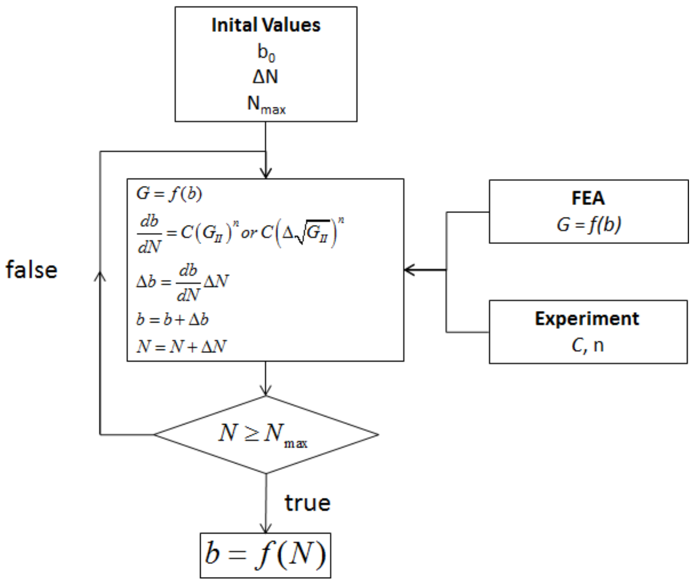


Figure 30: Flow chart of the developed disbond growth model.

Key results of the research were that the SERR can be used as a similarity parameter for the prediction of disbond growth, that a reduction in temperature retards the disbond growth rate (at least for the FM73 adhesive used in this research), and that interaction effects may exist if there are multiple disbonds with different lengths present in the specimens.

A full report is publicly available from the TU Delft repository [8]

6.6 Fatigue of welded joints (J. Schijve, TUD)

Fatigue life prediction of welded joints is a well-known problem. Recommendations are covered in international codes. About a decade ago a new concept was proposed to characterize the fatigue severity of welded joints in a structure by replacing the root of the welds by a rounded notch with a specific root radius. It is labelled as the "effective notch stress concept". With a finite element analysis of a welded structure stress concentrations of the various welds can then be analysed. Although this might seem to be helpful for design purposes, there are elementary shortcomings due to unrealistic elements of the original concept. This is discussed in ref. [15] which also includes an improved effective stress concept based on a ratio of the root radius and a dimension of the weld bead. Furthermore, it is emphasized that an effective notch stress concept can be informative for fatigue limits, but not for S-N curves as proposed in the original concept. Recommendations for future research are included.

6.7 Specimens for fatigue tests and fracture toughness tests (J. Schijve, TUD)

Two well-known specimens are the compact tension specimen (Figure 31), C(T), and the middle cracked tension specimen, M(T). The advantage of the C(T) specimen is that the size is small. The disadvantage is that the stress intensity factor increases rapidly during crack extension. In terms of application for designing an aircraft structure it should be recognized that the boundary conditions in a structure are essentially different from the boundary conditions of the compact tension specimen. Boundary conditions of the symmetric middle cracked specimen represent the conditions in a real structure much better. This was recently emphasized in [16].

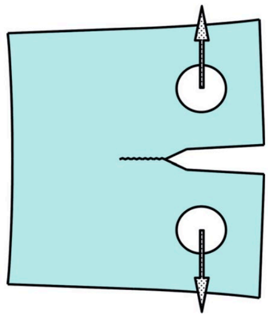


Figure 31: A load applied to a C(T) specimen is causing displacements of the specimen.

6.8 Fatigue crack growth characterisation of β-annealed Ti-6Al-4V ELI (E. Amsterdam, NLR)

Beta (β) annealed Ti-6Al-4V Extra Low Interstitial (ELI) alloy has a chemical composition and manufacturing process intended to optimise its fatigue and fracture properties, notably in the thick sections required for large primary aircraft structures. This alloy has been selected for the main wing carry-through bulkhead and other fatigue critical structures, including the vertical tail stubs, of an advanced military aircraft intended to enter service with the RAAF and the RNLAF. Since adequate fatigue data are currently unavailable for β-annealed Ti-6Al-4V ELI, a joint DSTO/NLR project was initiated to generate these data using a batch of material held by DSTO.

The workload was split between NLR and DSTO. Samples were cut from a 102 mm thick section by means of electro discharge machining. The following types of tests have been performed at NLR:

- Cyclic stress-strain tests
- Strain – life tests under constant amplitude (CA) loading
- Fatigue crack growth tests (CA)
- Fatigue crack growth threshold tests (CA)

The cyclic stress-strain curve obtained at NLR was similar to the one obtained at DSTO. The test results indicated an on-going shake-down for each series. The strain-life results were generally consistent with data from DSTO and (limited) open literature data. There was no difference between the L- and T-oriented samples. There was no clear difference in the fatigue crack growth rate between the LT- and TL-orientation either. There was a clear dependence of the crack growth rate and the threshold on the stress ratio, however, even though there was a large spread in the threshold (see Figure 32). The threshold values lie between:

7.7 and 9.4 MPa√m for R=0.1

5.8 and 6.6 MPa√m for R=0.4

4.8 and 5.2 MPa√m for R=0.7

In order to build up experience on quantitative fractography (QF) for short cracks, some tests were performed on AA7050-T7451 first before considering the β-annealed Ti-6Al-4V alloy. Constant amplitude tests with high and low stress ratios and low delta stress intensity factors were performed. The blocks with high and low stress ratios could easily be observed on the crack surfaces (see Figure 33) and the fatigue crack growth rates were similar to those reported in literature and by DSTO. The measurements also indicated that crack growth occurred at delta stress intensity factors below the threshold value as established from the CA tests.

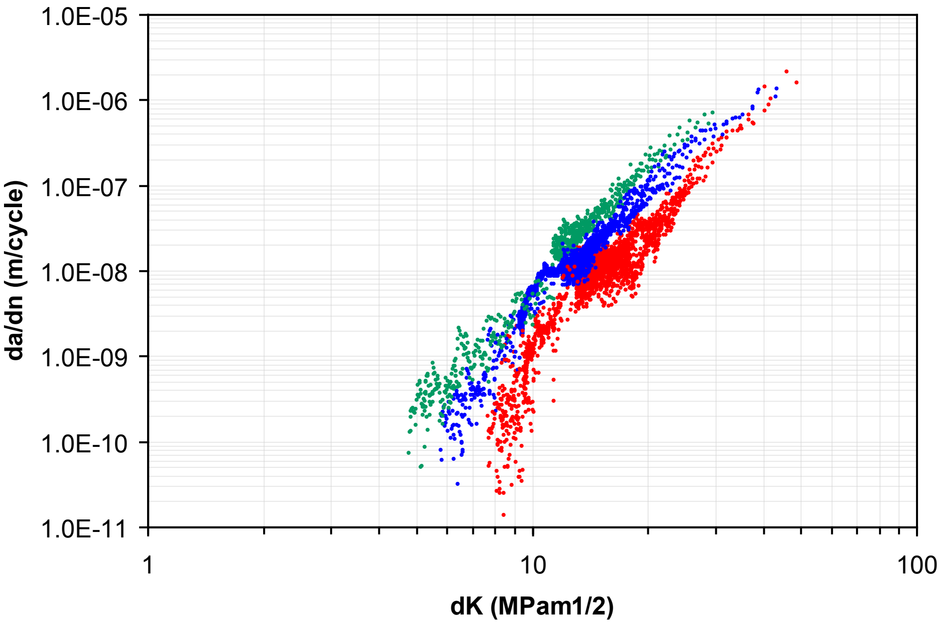


Figure 32: Crack growth rate in beta-annealed Ti-6Al-4V as a function of ΔK for all long crack growth and threshold tests performed at NLR: two L samples, two T samples, three stress ratios, viz. 0.1 (red), 0.4 (blue) and 0.7 (green).

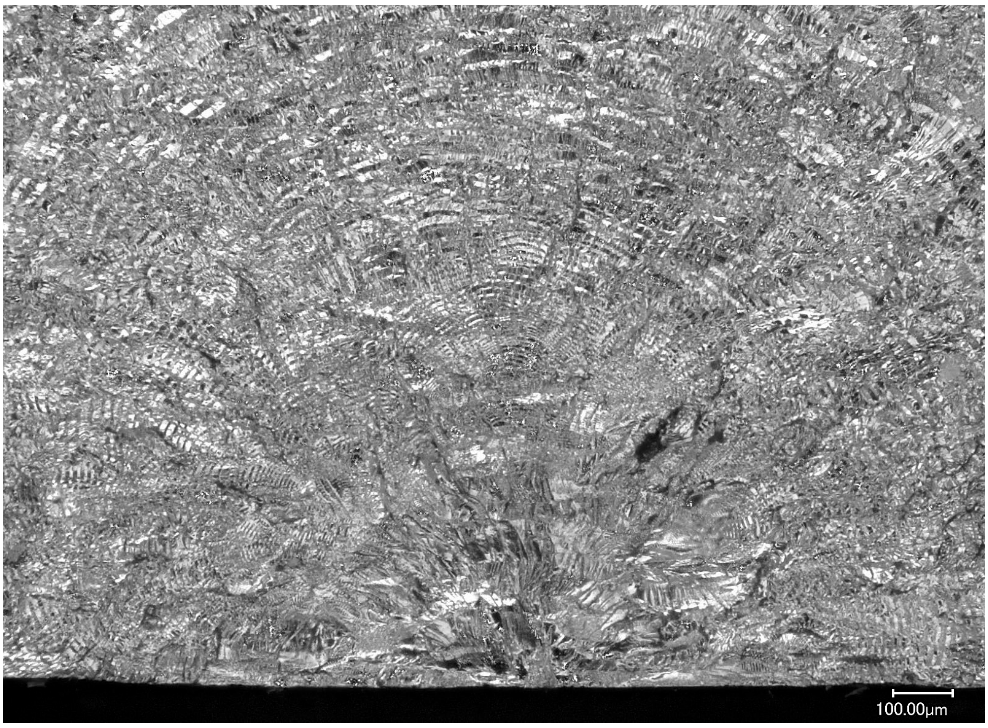


Figure 33: Fracture surface of AA7050-T7451 used for QF.

6.9 Tensile and fatigue tests on Inconel 718 fabricated by additive layer manufacturing via the laser powder deposition and powder bed route (E. Amsterdam, NLR)

The objectives of this project were:

- to assess the current ability of the laser additive manufacturing (LAM) technology to deposit a large specimen of defect free Inconel 718 and to investigate the microstructure, tensile strength and fatigue properties of LAM Inconel 718 coupons in comparison to those of coupons from regular Inconel 718 plate material.
- to investigate the influence of the type of production process, LAM vs. selective laser melting (SLM), on the grain size and the associated mechanical material properties.

Both SLM and LAM specimens of Inconel 718 were fabricated. Cross-sections and tensile- and fatigue tests were performed on coupons from these specimens before and after a hot isostatic pressing (HIP) treatment (for SLM samples only).

All tensile test coupons from one LAM specimen failed in the gripping area due to the presence of a region of high porosity and oxides. The coupons from the other LAM specimens had lower yield and ultimate tensile strengths than the samples made from plate material; this was attributed to the difference in grain size. The SLM technique yielded smaller and more uniform grain sizes in comparison with the LAM technique. The HIP treatment on the SLM samples led to an increase of the grain size, however, but resulted in less defects and therefore better fatigue properties compared to the as-manufactured SLM samples. The fatigue properties of the as-manufactured SLM samples were better than those of the as-manufactured LAM-samples.

Figure 34: SLM specimen (left) and fatigue coupons (right) of Inconel 718.

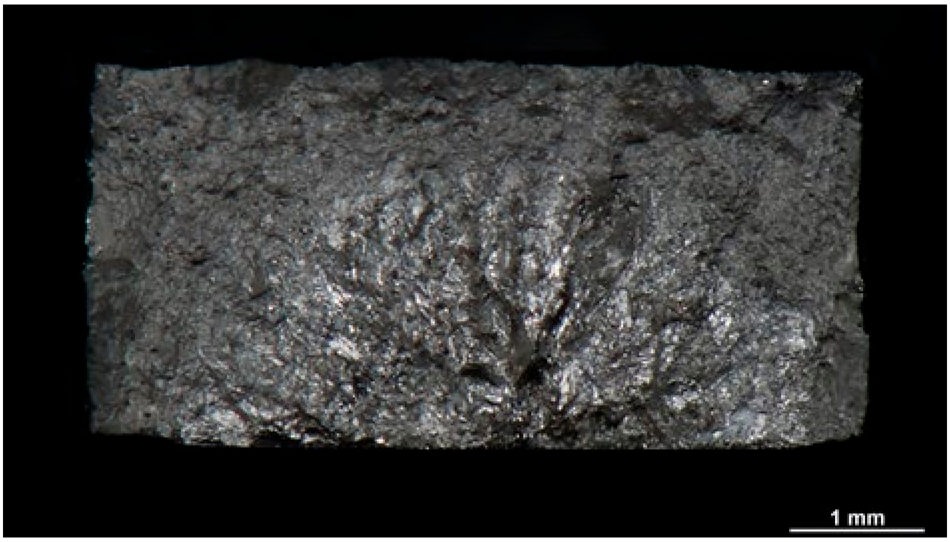


Figure 35: Fatigue fracture surface of a LAM Inconel 718 coupon.

7 Full-scale Testing

7.1 F-16 Block 15 wing damage enhancement test (M.J. Bos, NLR)

The NLR has been contracted by the Royal Netherlands Air Force (RNLAF) to conduct a so-called damage enhancement test on the LHS wing of a decommissioned F-16 Block 15 of the Royal Netherlands Air Force (RNLAF). The wing has a known history and has accumulated 4,200 flight hours. The results of the test programme will be relevant for both the RNLAF and the Chilean Air Force (FACH), who will share the programme results with the RNLAF on a government to government basis.

The damage enhancement test essentially is a durability test that aims to grow in-service cracks of sub-detectable size to a size where they may readily be detected. This significantly enhances the teardown inspection programme that is conducted in parallel on the RHS wing of the same aircraft – see section 7.2. The main objective of the test is to determine if the ex-service wing contains damage not accounted for in the early durability test programme that was performed in the late 1970s or in the current durability and damage tolerance analysis (DADTA) of Lockheed Martin (LM). Other objectives are to generate data (i.e. crack growth curves for the locations that are most critical) that can be used for an assessment of the current ASIP inspection programme and to establish the most likely fail scenario, incl. an estimate of the associated technical end of life time.

In order not to miss any potentially critical cracks because of insufficient loading conditions, the test setup is fairly complex and involves the use of 23 load actuators and the application of a representative fatigue load spectrum that is based on on an extensive loads survey that was taken of the RNLAF fleet in 2004. The same spectrum has been used by LM in the development of the current structural maintenance plan. The added advantage is that the observed crack growth rates in the tested wing will be representative of those in service. This will enable a translation of the test results to service conditions and provide an experimental validation of the theoretical crack growth curves provided by LM in the DADTA. It must be realized that the results are not meant for certification purposes, however. The available budget and time only allows a relatively simple approach which, for instance, precludes the presence of the leading edge flap and flaperon (the calculated interface loads will be applied, however) and the pressurization of the fuel tank. It is noted in this respect that the durability test involves a simplification by considering the wing only configuration; the design of a representative wing root support has been dealt with using calculations with the F-16 Block 15 coarse grid Finite Element Model (cgFEM) of LM that is available at the NLR. The support structure has been equipped with calibrated load sensors to monitor the distribution of the reaction forces over the four wing attachment fitting stations and the shear ties. Additionally the wing has been instrumented with approximately 80 strain gauges that are continuously monitored, a suite of acoustic emission sensors and a number of CVM sensors on the lower skin in the wing root area.

No artificial damages have been applied to the test article. Only naturally existing damages, caused by in-service fatigue loading, corrosion, etch pits, tool marks, etc. are considered. The durability test will cover two design lifetimes or less (in case of untimely failure). No residual strength test will be conducted at the conclusion of the test.

The durability test has started in March 2013 and is expected to be completed by the end of June 2013, after which an extensive teardown and inspection will be conducted.

An impression of the test setup is provided in the figures below.

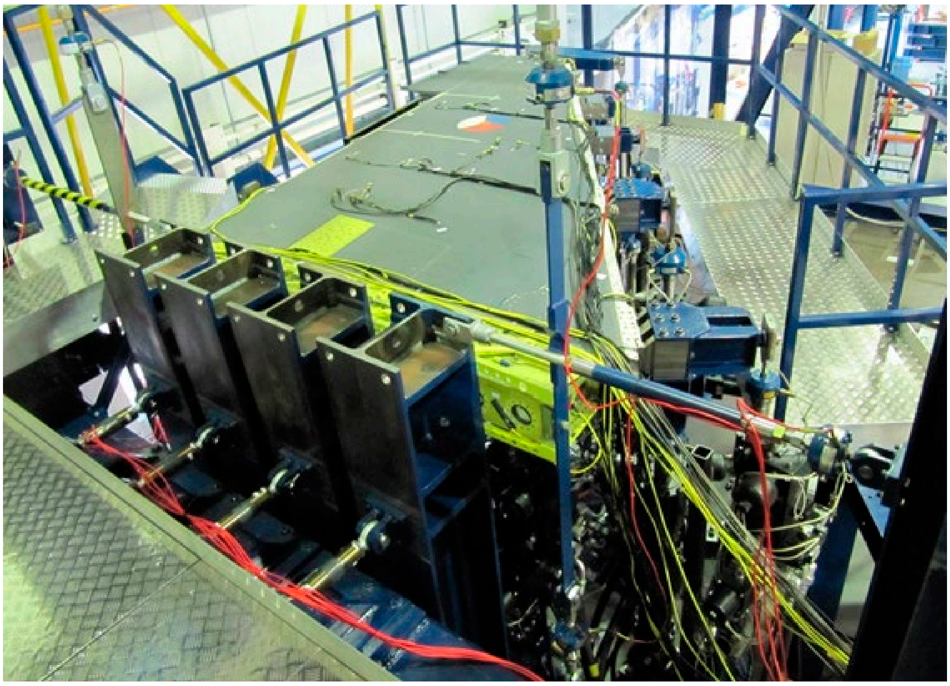


Figure 36: Impression of the F-16 wing test setup. The fully instrumented backup structure is clearly visible in the foreground.

Figure 37: Impression of the F-16 wing test setup. Some of the acoustic emission sensors are visible in the foreground, at the wing tip.

7.2 F-16 Block 15 wing inspection and teardown (A. Oldersma, NLR)

The NLR has been contracted by the RNLAF to disassemble and inspect the RHS wing of a decommissioned F-16 Block 15 with 4,200 service hours. Part of the disassembly work is done by RNLAF maintenance personnel. This teardown programme is conducted in parallel to the F-16 Block 15 wing damage enhancement test programme that is described in section 7.1 and the wing originates from the same aircraft. The objective is to provide an assessment of the current wing status of the F-16 Block 15 fleet in terms of fatigue cracking and other damages such as corrosion and mechanical damage. The inspections include the mandated inspections according to the F-16 Aircraft Structural Integrity Program (ASIP) as well as inspection of other high loaded wing areas in the wing skin, spars and attach fitting areas. The work has started in 2012 and is expected to be completed in July 2013.

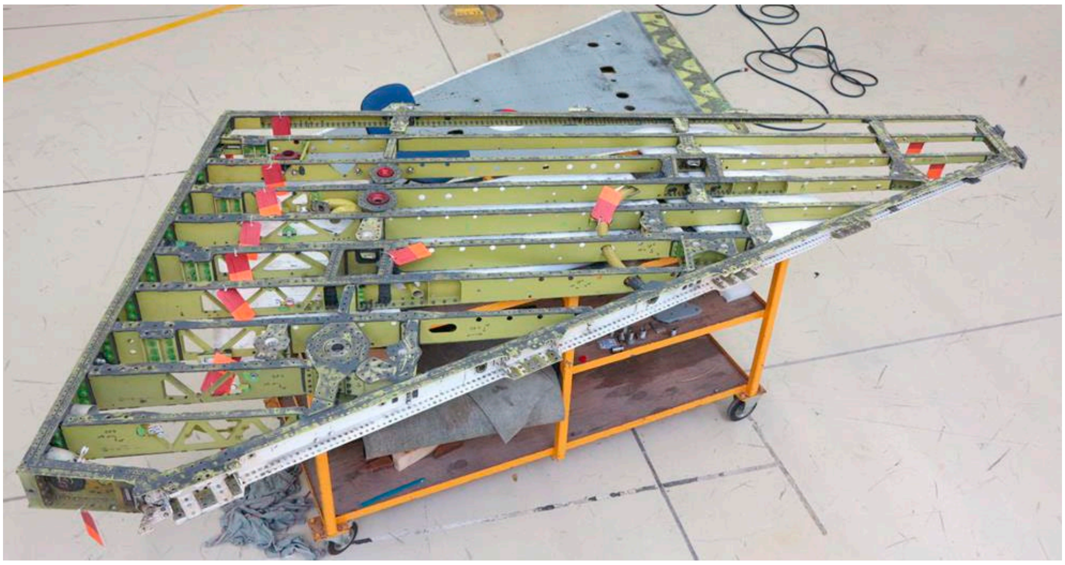


Figure 38: Teardown of RNLAF F-16 Block 15 RHS wing with 4,200 service hours.

8 Special Category

The NLR (E. Amsterdam) has co-organized the Fifth International Conference on Engineering Failure Analysis in The Hague, July 2012.

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