Review of aeronautical fatigue investigations in the Netherlands during the period March 2005 – March 2007

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1.1 INTRODUCTION

The present review gives a summary of the work performed in the Netherlands in the field of aeronautical fatigue during the period from March 2005 to March 2007. 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
- Stork Fokker AESP BV
- Fibre Metal Laminate Centre of Competence, FMLC
- CORUS RD & T.

The names of the principal investigators and their affiliations are given in brackets at the end of each topic title.

1.2 LOADS

1.2.1 Computer aided sequencing of aircraft loads and stresses for fatigue analysis and testing, CLASS (R.P.G. Veul, NLR)

The NLR has continued to develop the CLASS computer program. The CLASS program generates a sequence of loads or stresses automatically for an arbitrary aircraft structure, and was presented in the ICAF 2001 meeting. The CLASS program reduces the effort to generate flight-by-flight load and stress sequences for fatigue analysis and testing. CLASS has been used for aeroplanes (Airbus A380, A400M), for components (Airbus and Dassault movables), landing gears (F-16) and for helicopter parts (NH90 tail).

Class is operational at Stork Fokker AESP and at Airbus Germany and is currently being considered to become an all-Airbus preferred tool for the generation of test spectra. A schematic of the CLASS program is given in figure 1.

1.2.2 Load database system for fatigue analysis (R.P.G. Veul, NLR)

In practice new questions often need to be answered concerning the fatigue life of aircraft once they have entered service. For instance, these questions could arise owing to different usage from that anticipated during the aircraft design stage, and different configurations due to modifications or repairs.

At the NLR a new procedure is being developed to allow fast generation of fatigue load and stress spectra for aircraft. Figure 2 gives a schematic of the procedure. The core of the procedure is a database in which loads data for all relevant load sources are stored for a number of configurations and flight conditions of a representative aircraft. The loads in the database must be generated using aerodynamic, elastomechanic and aeroelastic analysis methods available from the manufacturer. Once the database is available, fatigue loads for an actual aircraft can be generated by interpolation and processing of the appropriate database data for the actual configuration and usage, and further processed by CLASS (see topic **1.2.1**) into load or stress sequences for fatigue analysis. The database may also be used to generate loads data for critical load analyses as well as fatigue analyses.

1.2.3 Improvements to the ESALOAD program (R.P.G. Veul NLR)

Within the framework of the European Space Agency contract "Structural Integrity of Pressurised Structures", several improvements have been made to the ESALOAD program. The ESALOAD program is part of the ESA software package ESACRACK, and is used to derive fatigue spectra for space applications. The major modifications concern:

- Event generation and modification.
- Load curve enhancements.
- Transmissibility and load input definitions.
- Unit stresses.
- Stress spectra generation.

Some minor modifications to the ESALOAD program have also been made. Examples are the possibility to define the maximum acceleration value used in the random load curve generation process; a reply of the effective (program used) Rainflow filter size; and the possibility to select multiple items. For all the modifications, the compatibility with previous versions is preserved.

1.3 FATIGUE LOADS/USAGE MONITORING OF MILITARY AIRCRAFT

1.3.1 Transport aircraft C-130H-30 Hercules (M.J. Bos, NLR)

The NLR has been tasked to develop and sustain a loads & usage monitoring programme for the C-130H-30 Hercules fleet of the Royal Netherlands Air Force. Within the framework of this project, both aircraft of the fleet have been equipped with a data acquisition system that samples the altitude, differential cabin pressure, airspeed and vertical acceleration. Together with flight administrative data (take-off and landing weight, mission type, etc.) from various other sources, the recorded data are stored in a relational database.

In parallel, Lockheed Martin has developed a set of so-called data blocks that quantify fatigue damage key values for a number of critical locations ('control points') of the C-130 airframe, covering the entire operating range of the RNLAF C-130H-30 flight envelope and configurations. These locations have been selected by LM Aero, based on structural test and service experience. They are the same as those used on the C-130J Structural Health Monitoring System and cover the outer wing, centre wing, forward, centre and aft fuselage. NLR has constructed a fatigue life monitoring system - the so-called Data Block Method - around these data blocks, using the measured flight profiles as input. This system now serves as a high-level management tool for the RNLAF to:

- Keep track of the consumed fatigue life.
- Assess the severity of specific missions and mission types.
- Evaluate and possibly optimize the usage of the C-130 fleet.
- Assess/anticipate required structural modifications programmes or individual aircraft Fatigue Life Extended (FLE) updates.
- Provide the OEM with high-quality data in case of modification programmes.
- Rationalise decisions regarding tailnumber selection in the case of out-of-area deployment, fleet downsizing, etc.

The Data Block Method uses a data base of fatigue damage values for each possible point-in-the-sky, ground event (taxi, roll-out, take-off, etc.) and GAG-cycle. The underlying stress-spectra are based on extensive L/ESS measurements that have been performed in the past (world-wide baseline). The underlying Kt factors for each of the control points have been estimated from fatigue test data. The damage values for a complete flight are added together to obtain the total damage according to a linear fatigue damage algorithm. A 50% probability crack would be anticipated if the sum of accumulated damage is equal to 1.

In principle the Data Block Method cannot be used to optimize the prescribed ASIP inspection intervals. It employs a Miner-approach to fatigue damage, rather than using a non-linear method for crack growth prediction. The Data Block Method should therefore be regarded as a tool to determine the economical life for the individual aircraft of the fleet.

1.3.2 Helicopters (M.J. Bos, NLR)

The MoD funded technology programme *HeliDamTol* aims to improve the capability to predict in-service fatigue crack growth in helicopter airframe components, by developing an *Operational Damage Assessment Tool (ODAT)*. The *HeliDamTol* programme concerns three subjects that are considered equally important:

- The acquisition of reliable helicopter flight and loads data, and the automatic processing to stress spectra for fatigue critical locations in the airframe.
- The acquisition of reliable material data of materials relevant for the RNLAF, and insight in the effect of typical helicopter spectrum loading on fatigue crack growth behaviour (incl. variability in the near-threshold regime).
- Improvement of the NLR Strip Yield model for the prediction of fatigue crack growth.

The latter two subjects are considered in more detail in topic **1.7.5**.

1.3.3 Structural Fatigue Load and Usage Monitoring F-16 (F.C. te Winkel, NLR)

Structural load monitoring of the F-16 fleet of the RNLAF has been carried out as a routine programme by the National Aerospace Laboratory (NLR) since 1990, when the Spectrapot capable of in-flight processing of the signal of a strain gauge bridge replaced the ex-factory Mechanical Strain Recorder. In both cases the direct strain measured at the main carry-through bulkhead FS 325 is representative for the wing root bending moment.

Initially, the F-16 fleet was monitored on a sample basis. The information gathered from three to four aircraft per squadron was thought to be representative for the loads and usage experience of that squadron, i.e. it was assumed that all aircraft belonging to a specific squadron flew more or less the same mission mix. Additional operational flight administrative data such as flight duration, mission type and external store configuration were taken from a special debriefing form and since 1995 directly extracted from the Core Automated Maintenance System (CAMS) of the RNLAF. By combining the load data from the sample measuring load programme and the CAMS data from all F-16 flights, it was possible to provide the RNLAF with information on the experienced load severity per tail number.

From the sample measuring programme the severity per mission type, per squadron and per time period is available. By combining this information with the actual mission mix per individual tail number for the same period an individual damage indication can be calculated. As a damage indicator, the Crack Severity Index (CSI) is used. This CSI, developed by the NLR, is a relative figure: 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 takes into account 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 previous flights. The CSI can be used as an indicative measure for ASIP (Aircraft Structural Integrity Program) control points, owing to the fact that the CSI is a relative figure between the actually measured and reference usage and loading environment used by LM Aero for generating the current inspection schedule.

During the 1990’s a completely new fatigue monitoring system was specified by the NLR. This system was developed by RADA by extending their ACE pilot debriefing system with loads and usage monitoring functionality: FACE (Fatigue Analyser & Air Combat Evaluation system). The main features 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) that allows more extensive load monitoring of each individual aircraft.

At first glance one might consider that only an upscaling of the sample load monitoring programme took place, i.e. an increase from one to five strain gauge bridge locations and fleet wide implementation. However, the FACE system is fully integrated with other aircraft systems. This results in far more flight parameters for airframe, engine and avionics monitoring becoming available, and these can be used for purposes other than Force Management activities. Moreover, the set of measured parameters is not a fixed set, but can easily be re-configured.

Switching from a sample load monitoring programme to fleet wide individual load monitoring, combined with a flexible way of measuring a wide range of additional flight parameters, required a different approach to handling the data. This resulted in a tailor-made information system built by the NLR for managing the data storage and analyzing the collected measured flight data from the on-board FACE system together with the individual aircraft administrative operational flight data obtained from CAMS. This centralized information system enables efficient data handling for both ad hoc analyses and generating routine status reports for fleet management purposes. Fleet management information is also made available by means of an interactive interface with the use of the OLAP-tool (On-Line Analytical Processing) PowerPlay.

A similar loads and usage monitoring programme has been set up for the Belgian Air Force, with the information system being modified to suit both air forces. For both air forces an intensive measuring campaign was carried out to collect Loads and Environment Spectrum Survey data with the FACE system, in order to enable LM Aero to provide an update of the Fleet Structural Maintenance Plan.

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

Non-SLAP IAT

The fleet of 13 RNLN P-3C Orion aircraft has recently been sold to Germany (8 A/C) and Portugal (5 A/C). In the past two years the NLR has supported the Dutch government by providing updated FLE (Fleet Life Expended) figures for each aircraft, enabling the new owners to start with a known aircraft usage history.

In the past, the NLR has developed software to perform fatigue life calculations for the P-3 Orion aircraft. This software provides the operator with a validated fatigue life indicator based on the most critical wing location. The software package has been in use by the Royal Netherlands Navy RNLN (as “PLEBOI”), and is currently in use with the Spanish Air Force SAF (as “SAFORI”) and the Portuguese Air Force PoAF (as “POLICAL”). In ad-hoc applications it has also been used for usage and fatigue damage comparisons between Orion aircraft from the Royal Norwegian Air Force RNoAF, the US Navy, the Royal Australian Air Force RAAF and the Canadian Forces CF.

In relation to this, for the Spanish SAF the NLR has developed updated IAT (Individual Aircraft Tracking) procedures to monitor and control Orion fatigue accumulation on a per-aircraft basis. The benefit of this new NLR concept is that fatigue monitoring functionalities become available at the manager level, instead of only at the OEM level. It should be noted that this tool provides an *indication* for fatigue damage accumulation at one critical location, and does not give absolute lives to adapt e.g. maintenance procedures. Hence this concept has been labelled ‘non-SLAP’, to distinguish it from SLAP, which is the formal OEM proprietary concept.

Figure 3 is a schematic of the non-SLAP management tool.

PLATO

Another interesting NLR activity during the past reporting period is the development of PLATO, a P-3 Orion PLEBOI mission profile planning tool for the manager. This software tool offers interesting new features to an operator:

- An operator can design any new mission type in an interactive and graphical way and can run PLATO to calculate the associated fatigue damage, in the meantime recognizing damage contributions from different sources, such as gust and manoeuvre loading, GAG-cycle, touch & go landings and full stop landings.
- It will be possible to process operational SDRS (Structural Data Recorder System) data, to apply automated mission recognition, and from these to perform “ in-flight” fatigue damage calculations for actually flown missions.

The power of the latter feature is that a costly off-line mission analysis task by the OEM (e.g. Lockheed), is no longer required to determine the severity of operational usage. With PLATO, the operator has this feature ‘at his hands’ on a laptop. PLATO is currently being implemented within the SAF. The concept can even be applied web-based, if required.

Figure 4 illustrates the PLATO start-up screen.

P3Union

A third interesting activity (focussing on the P-3 Orion) is the development by the NLR of a dedicated web-based technical forum and website, entitled P3Union (www.p3union.com). P3Union addresses all kinds of operator-oriented issues, such as obsolescence, fleet life management, ILS, Crew Aspects etc. The forum and website is being hosted by the NLR, with the US Navy as prime sponsor. With the focus shifting to MMA over the next ten years, Orion operators should realize that in the future the US Navy may not be able to provide significant technical support to address the sustainment issues of the international P-3 community. Attention, budget, and workforce capacity will primarily be focused on MMA. Similarly, the current level of industry engineering support for the Orion may not be readily available, owing to focussing on other programmes. This may have significant impact for smaller P-3 Orion operators, who may require technical assistance in the later phases of operational service. This potential gap may now be bridged in a pro-active way by a successful and active P3Union working group. P3Union is intended to be an active platform for all Orion operators to:

- Develop tailored structural integrity plans.
- Increase awareness of available tools, analysis results, and common fleet issues.
- Share common needs and promote collaboration among operators.

As an illustration, the P3Union logo is reproduced in figure 5.

1.3.5 Lynx helicopter (A.A. ten Have, J.A.J.A. Dominicus, NLR)

The Royal Netherlands Navy RNLN will start operating the NH90 helicopter in the near future. To avoid an operational gap between phasing out the current Lynx helicopter and service entry of the NH90, it is important to closely consider the options for Lynx service extension.

Instrumental for this effort is AIDA (Aircraft Integrated Data Acquisition system), an RNLN Lynx fleetwide installation of a multi-channel structural data recorder that monitors the lifting frame, main rotor, engines and sponsons. To date, more than 30,000 RNLN Lynx flight hours have been gathered with AIDA. Information from the AIDA database is used for fleet planning and maintenance optimization tasks. In the past 2 years, the NLR has engaged in activities to support validation and certification efforts, in a dialogue with the various OEMs. Most recently, the NLR has investigated the feasibility of using pre-AIDA usage information to maximize operational service life.

1.3.6 Chinook helicopter (A. Oldersma, NLR)

During the period 2001-2005 the Royal Netherlands Air Force (RNLAF) and the NLR initiated a CH-47D Chinook Structural Integrity Pilot Programme to monitor and control aircraft degradation, in an effort to optimise the maintenance, operational availability and flight safety. This pilot programme resulted in the development of:

- A Chinook Automatic Flight Regime Recognition (AFRR) algorithm.
- An RNLAF Structural Integrity management tool (based on a damage index) for both rotating and non-rotating components.
- A fatigue damage prediction tool (based on projected future usage).

Subsequently, the RNLAF requested the NLR to follow on from the Pilot programme with the following scope:

- Usage monitoring of the CH-47 fleet during usage in the Netherlands as well as during Out-Of-Area operations.
- Maturation of the Automated Flight Recognition software.
- Instrumentation of four Chinooks with state of the art data recorders. The recorded data include flight parameters and strain gauge information. This information will be used to correlate specific aircraft states (Flight Regimes) with damage accumulation (strain gauge output).

1.4 SERVICE LIFE ASSESSMENT AND EXTENSION PROGRAMMES

1.4.1 RNLAF Helicopter Database Management (A.A. ten Have, NLR)

The RNLAF operates a variety of helicopter types, e.g. Chinook, Apache, Cougar, Bell 412, Lynx; and in future the NH90. A large number of data are generated each and every day, stemming from flight administration sources, flight data recorders, loads recorders, crack corrosion logbooks etc. To maximize the benefit of Integrated Weapon System Management (IWSM), it will be essential to store and process all relevant data in a structured way.

For this purpose, the NLR has been contracted by the RNLAF to develop and implement a flexible helicopter database system called HELIUM (HELicopter Integrated Usage Monitoring). It is expected that the HELIUM database will be running and ready for finetuning and stabilisation by the end of 2007.

Some illustrations of HELIUM are shown in figures 6 – 8.

1.5 ENGINES

1.5.1 Single crystal material modelling (T. Tinga, NLR)

As operating temperatures in gas turbines become higher, the application of single crystal superalloy materials increases. Use of these special materials has consequences not only for materials testing, but also for modelling and life prediction. Hence the NLR has developed the capability to perform life predictions on single crystal components subjected to both creep and low cycle fatigue (LCF) [1, 2].

The modelling of single crystal material behaviour was continued by extending the capabilities of the previously developed multi-scale micro-mechanical model, see figure 9, taking into account the microstructural details of nickel-base superalloys. Detailed constitutive models were developed for the two superalloy constituent phases γ and γ'. A dislocation density-based crystal plasticity model was used for the matrix phase γ, whereas for the γ' precipitates the dislocation mechanisms of shearing and recovery climb were incorporated. The model parameters for the commercial alloy CMSX-4 were determined and the model was shown to simulate the material behaviour adequately.

1.5.2 Life assessment of gas turbine components (T. Tinga, NLR)

An integrated analysis tool for gas turbine life assessment has been developed in the past decade, combining the NLR Gas Turbine Simulation Program GSP (engine performance), computational fluid dynamics models (heat transfer, figure 10), finite element models (thermal / mechanical loads, figure 11) and lifing methods to perform life assessments for a variety of gas turbine components. Recently the analysis tool was applied to a fighter engine combustor liner. The thermal loading of the liner was obtained from a CFD combustion model and a finite element analysis was performed to calculate the creep and fatigue damage for a representative mission. The locations and directions of cracks correlated with the damage observed in practice [3].

1.5.3 Foreign object damage effects on high-cycle fatigue of γ-TiAl (R.J.H. Wanhill, NLR)

The effects of Foreign Object Damage (FOD) on high-cycle fatigue (HCF) of a commercial γ-TiAl alloy were investigated. Specimens prepared from a low-pressure turbine blade were impacted by steel balls at different velocities. After heat-tinting to distinguish FOD-induced cracks, the specimens were tested in HCF to failure. The FOD crack sizes and HCF failure stresses were used to determine the suitability of Kitagawa-Takahashi (K-T) diagrams for providing boundaries between the conditions of *crack growth* and *no crack growth* in HCF.

Figure 12 shows the final K-T diagram. Even when conservatively modified, by selecting the microstructurally short crack growth threshold, the K-T diagram had difficulty in capturing the data representing FOD + HCF failure stress ranges *versus* equivalent crack sizes. This means it will be difficult to provide secure but not excessively conservative boundaries between HCF *crack growth* and *no crack growth* for FOD-damaged γ-TiAl alloys.

N.B: This work is presented in a paper for the Poster Session of ICAF 2007.

1.6 FIBRE/METAL LAMINATES (FMLs)

1.6.1 The effect of environmental temperatures on the delamination growth rate in GLARE (R. Alderliesten, TUD)

Fatigue crack propagation in the metal layers of GLARE and the delamination growth at the interfaces with adjacent intact fibre layers has been investigated in the past. A theoretical model was developed describing the mechanisms [4], and was presented at the previous ICAF symposium [5]. This research has been extended to investigation of the effect of low and elevated temperatures on the delamination growth in GLARE, using the specimen configuration shown in figure 13. Delamination tests were performed at -55°C, -40°C, -20°C, RT, 20°C and 70°C [6, 7]. The obtained delamination growth rate versus energy release rate curves for the various temperature levels were described by

$$\frac{db}{dN} = C_d \left(\sqrt{G_{d,\max}} - k\sqrt{G_{d,\min}}\right)^{n_d}$$

where the constant C_d was found to depend on temperature (in degrees Celsius) according to

$$C_d(T) = c_1 T^{c_2} + C_{d,RT}$$

The parameter n_d remained the same for all temperature levels. The dependence of the aluminium crack growth behaviour for different temperatures and relative humidities was accounted for in the model.

The model predictions were validated by fatigue crack growth tests on GLARE laminates tested in previous research [8]. Figure 14 gives three examples comparing the measured and predicted crack growth rates for three different temperatures. The agreement was quite good at the higher temperatures, but at low temperatures the crack growth rates were overpredicted. This might be due to using the Paris crack growth rate equation, which does not account for near threshold fatigue crack growth behaviour.

1.6.2 Theoretical fatigue crack propagation model for centre reinforced aluminium (CentrAl) (R. Alderliesten, TUD)

The FML GLARE has been developed and optimised primarily as skin and stringer material for aircraft fuselages. The application of this FML configuration for wing structures requires thick lay-ups, especially at wing roots, which result in high manufacturing costs. In principle, this is also the case for thick CFRP wing structures.

An alternative hybrid wing concept has been developed by GTM-Advanced Structures [9] following a similar technology development to that for FMLs. The concept, called CentrAl, consists of FMLs bonded between aluminium skin layers. This concept has significantly better performance with respect to fatigue and residual strength than previously proposed selectively reinforced metallic concepts [10].

The theoretical methods for FMLs have been slightly modified.to understand and predict the fatigue initiation and propagation behaviour of CentrAl. In fact, the CentrAl configuration can initially be treated as a 2/1 lay-up laminate, where the fibre layer has the properties of the FML reinforcement, see prediction 1 in figure 15. Prediction 2 shows that accounting for the stiffness reduction of the central FML layer owing to aluminium crack growth results in a conservative prediction close to the observed crack growth.

The third prediction has been performed to evaluate the accuracy of the prediction model itself. The distribution of crack lengths within the FML is not known a priori and has to be calculated. The result is a very good prediction of the crack growth curve. In fact, predictions 2 and 3 prove the strength of the developed theoretical models, because the predictions are conservatively close to the actual crack growth behaviour.

N.B: A paper on CentrAl's fatigue and damage tolerance properties has been submitted to ICAF 2007 by G.H.J.J. Roebroeks, GTM Advanced Structures.

1.6.3 The effect of external stiffening elements on the fatigue crack growth in Fibre Metal Laminates (R. Rodi, University of Pisa, R. Alderliesten, TUD)

The effect of stiffening elements on crack growth in a flat plate is often accounted for by defining a geometry correction factor β that is related to the position of the stiffeners. However, it has been proven that the effect of stiffening elements can also be described by the superposition approach [11]. The effect of each stiffening element on the crack tip can be written as an additional contribution to, for instance, the K_{tip} in

$$K_{FML} = K_{Metal} - K_{Bridging}$$

The advantage of this superposition approach is that the effect of external stiffeners on the bridging stress and delamination growth within the FML can be calculated in a straightforward way, see figure 16 for the bridging stresses at several crack lengths. The delamination underneath a bonded stiffener is calculated in a similar way as the delamination at the interfaces within the FML.

Figures 17 and 18 compare predicted and experimental fatigue crack growth rates for GLARE with broken or intact central stiffeners (titanium straps). The predictions are good. Finally, figure 19 shows good predictions of the crack opening contours for GLARE with broken central stiffeners. The work illustrated by figures 17 – 19 is presented in a paper for the Poster Session of ICAF 2007.

1.6.4 Fatigue crack growth in subsurface layers of GLARE under combined tension and bending (C.E. Randell, TUD)

Owing to secondary bending, part-through cracks will occur in lap joints under tensile loading. As shown by Beumler [13] fatigue cracks in GLARE first start in a surface layer, followed later by smaller cracks in subsurface layers. Some limited delamination in the intermediate fibre layers occurs around the growing fatigue crack, but fibre-crack-bridging will considerably restrain further crack growth. This topic was studied by Alderliesten [4] who developed an analytical crack growth prediction model that included the delamination phenomenon.

In the present investigation a special specimen was designed to obtain specific ratios of tension and bending to study crack growth starting from an open hole notch. The specimen configuration is shown in figure 20. For both monolithic aluminium and GLARE the specimens were machined from thicker plates. In the test section the tensile load induces secondary bending , but the moderate tapering avoids failure at the edges of the test section.

The bending factor is defined as:

$$\boxed{k_b = \frac{\sigma_{applied} + \sigma_{bending}}{\sigma_{applied}}}$$

The bending factor can be adjusted by altering the tapered thickness dimensions and was calculated by FEM. In the present tests the k_b- values were about 2.0 or somewhat higher. The specimen width was 100 or 50 mm, and the hole diameter was 5.6 mm. The type of GLARE was GLARE 2 with 6/5 and 5/4 lay-ups and an aluminium layer thickness of 0.4 mm. Tests were also carried out under pure bending on prismatic beams of GLARE 2 with an 11/10 lay-up, again with a central hole.

The fatigue stress ratio in all tests was 0.1. Fatigue tests were carried out to different values of crack length in the surface layer. The specimens were then pulled to failure, but protruding fibers prevented crack length measurements on the fracture surfaces. The specimens were then heated at 350^{o} C for 4 hrs to separate all layers. Measurements of the crack length and the delamination boundaries could then be made. Figure 21 shows some results of the crack patterns for increasing crack sizes. Using this type of information, FEM calculations were made to derive relations for the fibre bridging stresses in the wakes of the cracks.

The calculated bridging stresses were then used as inputs for the analytical prediction model to calculate ΔK-values and corresponding crack growth rates. Agreement between these values and the observed crack growth rates of individual layers was satisfactory. The calculations do not yet include a prediction of the increasing delamination. The results are presented in a doctoral thesis [14].

1.6.5 Teardown of a GLARE window area from the Airbus MegaLiner Barrel (MLB) fatigue test article (R.J.H. Wanhill, NLR)

The NLR is carrying out teardowns and supplemental fatigue tests of GLARE (GLAss REinforced aluminium laminate) panels from several key locations of the MLB. The first location chosen for teardown was a GLARE window area, which included aluminium alloy window frames attached to the GLARE skin. The teardown began with Non-

Destructive Inspection (NDI), and was followed by low- and high-magnification fractographic investigation of the NDI-indicated cracks. The teardown had the following main objectives:

- Verification of NDI techniques and capabilities.
- Establishing the patterns of cracking in the GLARE aluminium layers.
- Determination of fatigue initiation locations and likely causes.
- Estimation of fatigue "initiation" lives and crack growth behaviour in the GLARE aluminium layers.

The NDI and low-magnification fractography demonstrated that the NLR's teardown NDI capability is excellent, and that most cracks in GLARE will be detected during teardown. The low-magnification fractography also established the patterns of cracking in the GLARE skin and the window frames. Most cracks in the GLARE skin were in the fastener hole bores rather than the countersinks. The preference for cracking in the fastener hole bores was most probably due to local secondary bending. The window frame cracks support this, since these were all corner cracks with maximum dimensions along the fastener hole bores.

Detailed fractography demonstrated the fractographic "readability" of the MLB fatigue load spectrum. The "readability" and traceability were excellent for the largest window frame crack, but less so for the largest crack in the in the GLARE aluminium layers. There are two reasons for this: (a) the crack growth rates for the aluminium layer crack were much lower than those for the window frame crack, making resolution of the crack front markings more difficult; and (b) debris and sealant obscured a considerable part of the initial growth of the aluminium layer crack.

The largest window frame crack was traced back to 0.15mm. Back-extrapolation suggested that the fatigue 'initiation" life was zero, and that there was an initial crack size of about 0.06mm. These results are best explained by the fact that the crack initiated at a fretting scar. Fretting is known to cause very early crack initiation, and the fretting scar could act as an apparent initial crack size.

The largest "readable" GLARE aluminium layer crack could not be traced back to less than 0.45mm, owing to ever more closely-spaced crack markers, making them impossible to resolve. This, and the limited data, made it unfeasible to estimate a fatigue "initiation" life. Nevertheless, it was evident that this crack grew significantly slower than the largest window frame crack, see figure 22.

Overall, the teardown results demonstrated the high fatigue damage tolerance capability of the MLB GLARE skin. This conclusion is reinforced by the fact that the applied fatigue load spectrum was set at a level high enough to obtain fatigue damage during the MLB test.

This study was the opening keynote paper at the International Conference and Exhibition on Structural Integrity and Failure, Sydney, Australia, September 2006, and was also presented, in abbreviated form, at the International Council of the Aeronautical Sciences ICAS 2006 Congress, Hamburg, Germany, September 2006.

1.6.6 Miscellaneous studies of FMLs (G. van der Kevie, FMLC)

- Fatigue behaviour of HSS-GLARE, which uses a 7000 series aluminium alloy instead of a 2000 series alloy. The investigations included adhesive bonded doublers, riveted joints and (simulated) repairs.
- Optimum design and fatigue performance of GLARE stringer couplings for GLARE stringers (aluminium alloy stringer couplings are unsuitable).
- Investigation of thin GLARE laminates suitable for smaller aircraft, including the fatigue behaviour of adhesive bonded repairs of thin panels.
- Fatigue crack initiation and propagation testing of new FML variants, comprising both new alloys and new fibre types.
- Impact tests on various FMLs, and comparison with the impact behaviour of aluminium alloys and CFRPs.

1.7 FATIGUE AND DAMAGE TOLERANCE STUDIES

1.7.1 Fatigue crack initiation behaviour of friction stir welded joints in aluminium alloys (H.J.K. Lemmens, TUD)

Friction Stir Welding (FSW) is a relatively young joining technology with a high potential for the aerospace industry. FSW offers reductions of production costs, lead-times, and structural weight. Since FSW is a solid-state process, it is possible to weld high strength aluminium alloys like 2024 and 7075. The low process temperatures, with a maximum below 500°C, ensure a minimal temperature impact and a maximum preservation of strength. Moreover, FSW is a robust process without emission of dangerous gases and radiations for which protection is required.

To use FSW to manufacture an airworthy and damage tolerant structure, the fatigue and damage tolerance properties must be fully understood. These properties include fatigue crack initiation, fatigue crack growth and residual strength. Since the various (three) weld zones influence the static and dynamic behaviour of the joint differently, full understanding of the static and dynamic responses of each zone is required. Several fatigue tests on FS welded material are reported in the literature, yet most fatigue research is performed at the centre-line of the weld and not for the Thermo-Mechanically Affected Zone (TMAZ) or the Heat Affected Zone (HAZ).

The present research concerns constant amplitude + marker loading fatigue crack initiation from small-diameter holes in different locations in welds made in 2024-T3, 7075-T6 and 6013-T4. The marker loads enabled fatigue crack growth to be tracked and consequently the exact locations and times of crack initiation to be determined. Furthermore, the welds were investigated by several non-destructive and destructive methods like optical and electron microscopy, to enable explaining the fatigue initiation behaviour in terms of the material microstructures.

As a result of this research, the fatigue initiation life as function of the distance to the centres of the welds was found, thereby giving the 'weakest' locations of the welds. The relation between the fatigue initiation properties and the local material microstructure as a result of FS welding should lead to better understanding of the behaviour of FS welds applied in aircraft structures. N.B: A paper on this research has been submitted for presentation at ICAF 2007.

1.7.2 The driven rivet head dimensions as an indication of the fatigue performance of aircraft lap joints (J.J. M. de Rijck, Corus RD & T, J. Schijve, TUD)

Riveting of solid aluminium alloy rivets implies a squeezing process of the rivet with large plastic deformations to form the driven rivet head. The dimensions of the driven rivet head (diameter D and height H) depend on the applied squeeze force Fsq. High squeeze forces are beneficial for the fatigue properties of riveted joints, while low squeeze forces are associated with low fatigue lives.

In the present investigation systematic measurements were made to arrive at a correlation between the squeeze force and the rivet head dimensions. Rivets of five aluminium alloys were used covering different heat treatments. Initial rivet diameters D_0 were 4.0, 4.8 and 5.6 mm. Two sheet materials were used (2024-T3 and GLARE). The results were analyzed with the true stress / true strain relation $\sigma = K^n$, and the following relationship between the squeeze force and the driven rivet head was obtained:

$$F_{sq} = \frac{\pi}{4} \cdot D^2 \cdot K \cdot \left(2 \ln \left(\frac{D}{D_0} \right) \right)^n$$

The strength constant K and the strain hardening exponent n were derived from the test results for the five rivet materials. With this equation it is easy to check the squeeze force by measuring the diameter *D* of the driven rivet head. An example of the agreement obtained is illustrated by figure 23. For any riveting process used in the industry it is an easy task to arrive at the two constants K and n from simple calibration experiments in which different squeeze forces are applied to 20 rivets in a single riveted sheet specimen.

1.7.3 Fractography-based estimation of fatigue crack "initiation" and growth lives in aircraft components (R.J.H. Wanhill, NLR)

Guidelines and procedures were given for fractography-based methods of estimating fatigue crack "initiation" and growth lives in aerospace materials and components, both from service and full-scale tests. The guidelines and procedures represent more than 30 years experience at the NLR. Examples were given for MSD (Multiple Site fatigue Damage) in pressure cabins, cracks in a vertical stabilizer rib and main hinge fitting, cracking in GLARE (see topic **1.6.5**), and fatigue failures of superalloy gas turbine materials. This study was a paper at the International Conference and Exhibition on Structural Integrity and Failure, Sydney, Australia, September 2006.

1.7.4 Fatigue crack initiation in aerospace aluminium alloys, components and structures (R.J.H. Wanhill, NLR)

The concept of self-healing fatigue cracks in aluminium alloys has aroused much interest owing to a recent paper by Lumley *et al*. [16]. In that paper the results of S – N fatigue tests, fractography, and microstructural analysis for an experimental Al-Cu-Mg-Ag aluminium alloy in underaged (UA) and peak aged (PA) conditions were interpreted as evidence for possible self-healing of small internal fatigue cracks in the UA material.

The present work reviews fatigue studies of UA, PA and also overaged (OA) aluminium alloys, particularly the work of Lumley *et al*. [16] and Finney [17]. Then there are surveys of fatigue crack initiation in aerospace aluminium alloys, components and structures, followed by a discussion of the following topics:

- The self-healing concept.
- High-cycle fatigue of UA and PA experimental alloy specimens.
- High-cycle fatigue of UA and PA commercial aluminium alloy specimens.
- Aerospace aluminium alloy components and structures.

It is concluded that the possible self-healing of internal fatigue cracks in UA alloys is (a) limited to internal slip band cracks ~ 1 µm; (b) incapable of explaining the similar fatigue behaviour of the UA and PA experimental alloys tested by Lumley *et al*. [16] and Finney [17]; (c) not required for explaining why any UA experimental alloys can have better high-cycle fatigue properties than their PA equivalents; (d) unlikely to influence the basic fatigue properties of commercial alloys; and (e) inapplicable in practice to aerospace aluminium alloy components and structures.

This study was a contribution to the 1st International Conference on Self Healing Materials, Noordwijk, the Netherlands, April 2007.

1.7.5 Helicopter damage tolerance project *HeliDamTol* (M.J. Bos, NLR)

The helicopter damage tolerance project *HeliDamTol* has two main objectives. The first is to develop reliable methods of fatigue crack growth analysis for helicopter airframe components. The second is to incorporate these methods into an *Operational Damage Assessment Tool (ODAT)*, see topic **1.3.2** also.

To date the main thrust of the *HeliDamTol* project has been to acquire test data from standard specimens of aluminium alloys used in the airframes of the NH90 and Chinook helicopters. These data are being analysed to provide input parameters for improving the NLR Strip Yield model for predicting fatigue crack growth in safety-critical airframe components. This will lead to development of the ODAT, which is scheduled for 2008.

The required data have been obtained from the following types of tests:

- Fatigue crack growth thresholds.
- Fatigue crack growth curves.
- Cyclic stress-strain curves.
- Monotonic true stress – true strain curves.
- Simple spectrum fatigue crack growth.
- Complex spectrum fatigue crack growth.

The data analysis will be completed in 2007. So far, the analyses indicate that the tests have achieved the goal of providing the necessary input parameters for modelling fatigue crack growth. N.B: A paper on all these aspects has been submitted for presentation at ICAF 2007.

HeliDamTol's scope is being significantly extended by an international cooperative programme with the Defence Science and Technology Organisation (DSTO), Melbourne, Australia.

1.7.6 Fatigue at high stresses (R.P.G. Veul, NLR)

The computer program ESAFATIG can be used to calculate the fatigue life of metallic structural components, based on a linear damage accumulation method. The ESAFATIG program is based on S-N curves that are fitted through experimental data (mainly data from MIL-HDBK-5J with additional data for various threaded fasteners). For high stress levels (related to fatigue lives less than approximately 10^4 cycles) there are usually no experimental data, and so the calculated number of cycles is obtained via extrapolation. However, this extrapolation generally leads to unconservative results. Therefore it was decided to implement a simple more conservative low-cycle fatigue behaviour in ESAFATIG, resulting in more accurate predictions and a warning to the user when the prediction is based on less accurate points of the S-N curve.

1.7.7 New stress intensity factor solution in NASGRO 3 (R.P.G. Veul, NLR)

Two new stress intensity factors were added to the NASGRO 3 crack growth analysis program. The configurations are as follows, see figure 24.

- TC20 represents a single edge through-thickness crack, loaded in tension/compression (S_0), bending in the thickness direction (S_1), bending in the width direction (S_2), and pin loading on the crack surface (S_3).
- TC21 simulates a through-thickness crack growing through a riveted or bolted lap joint, starting from a hole and growing in one direction, loaded in tension/compression and by pin loads.

The implementations were validated by FEM using MSC.PATRAN, the NLR-C3D crack block insertion module, and MSC.NASTRAN.

1.7.8 Stress analysis of mechanically fastened joints in aircraft fuselages (J.J.M. de Rijck, Corus RD & T)

A thorough understanding of the stresses at the most critical fastener row is essential in conducting fatigue and damage tolerance analysis of mechanically fastened joints. For mechanically fastened lap-splice joints and butt joints in a fuselage structure the dominant loading condition is introduced by the Ground-Air-Ground (GAG) pressurization cycle. The hoop load is transferred from one skin panel to the next panel by fasteners in a lap joint or a single strap butt joint. The hoop load is not collinear through the joint but is offset by eccentricities in the joint. The eccentric path of the hoop load causes secondary bending. The total stress in the joint is then the membrane, secondary bending and bearing stress associated with the fastener loads on the holes. Secondary bending is highly dependent on the magnitude of the eccentricity and the flexural rigidity of the joint between the fastener rows. The theory used to derive the bending stresses is based on advanced beam theory. Schijve's simple, one-dimensional Neutral Line Model (NLM) is used to calculate the tension and bending stresses at any location in the joint. The most likely locations for fatigue crack nucleation and growth can be determined from this analysis.

The present research, which has been submitted for a paper to ICAF 2007, combines the analytical Neutral Line Model and load transmission by rivet joints between the outer rows of the joint. Load transmission by these rivet rows were ignored in the original NLM, but strain gauge measurements indicate that load transmission occurs in these rows. The analysis covers joints of both monolithic and fibre metal laminate (FML) materials. The analytical model accounts for load transfer through all fastener rows. It allows a wide range from simple to very complicated joints containing rivets or bolt type fasteners. In the case of riveted fasteners, the influence of so-called fastener flexibility can be addressed as well. Strain gauge measurements on lab specimens and in-service joints supported the analysis.

1.7.9 Structural analysis of Aircraft Structures using New Aluminium Alloys developed for the Next Generation Aircraft (Static and FD&T analysis (J.J.M. de Rijck, Corus RD & T)

To study the influence of newly developed aluminium alloys on the damage tolerant behaviour of a fuselage panel, both the material properties as well as the structural stress field in the component need to be known. A finite element model was generated that represents a typical stiffened panel in the fuselage of a commercial airplane. The model comprises the description of the skin sheet, stringers and frames, and enables the calculation of the load distribution over the considered fuselage panel, expressed in terms of nodal forces, bending moments and stresses. In this way the stress critical locations in the fuselage panel concerning damage tolerance can be found. A successive series of damage tolerant analysis tools was selected to be used for a next analysis step to investigate fatigue crack initiation, fatigue crack growth in both the longitudinal and circumferential direction, and the residual strength of the considered panel. As a result the lifetime of the fuselage panel can be assessed.

In order to also study the capabilities of newly developed alloys not only in tension dominated areas, an analytical tool has been developed to assess the strength and stability of compression-dominated fuselage panels or upper wing covers. In the analysis all the structural components can be taken into account, from flat panels to curved panels, and including integral or built-up stringers. In the analysis several standard fuselage and wing cover materials are used to serve as a baseline. From this baseline several combinations of new developments are compared, such as AlMgSc-alloys, a new generation of 2XXX/7XXX or Al-Li alloys. Analyzing a structure using these materials gives information about the strength and stability, which are then used to optimize weight, costs or production methodology.

This kind of analysis approach allows evaluating the potential of new aluminium alloys to improve the performance of aircraft structures in terms of weight saving and costs**.** Several generic aircraft structures have been analyzed to show significant weight saving opportunities due to new and improved aluminium alloys for the next generation of aircraft.

N.B: The present work has been submitted for a paper at ICAF 2007, lead author M. Miermeister, Corus Aluminium Rolled Products.

1.7.10 Prediction of fatigue crack growth under variable-amplitude (VA) loading (J.Schijve, TUD)

The co-operation with professor Skorupa of the University of Mining and Metallurgy in Krakow was continued. The strip yield model from the NASGRO computer software has been applied to predict fatigue crack growth in two different aircraft aluminium alloys under constant amplitude loading and programmed and random variable amplitude load histories. The computation options realized included either of the two different strip yield model implementations available in NASGRO and two types of the input material data description. The model performance has been evaluated based on comparisons between the predicted and observed results. It is concluded that the generally unsatisfactory prediction quality stems from an inadequate conception of the constraint factors incorporated in the NASGRO models. This study is reported in Ref.[18].

1.7.11 Guidelines for the use of the Strip Yield model in ESACRACK (M.J. Bos, NLR)

A report, NLR-CR-2005-295, was prepared to provide guidelines for the use of the Strip Yield model in ESACRACK. These guidelines apply to both the constant constraint-loss model of Newman (NASA) and the variable constraint-loss model of De Koning (ESA/NLR). Both models are implemented in the fatigue crack growth analysis software NASGRO that is incorporated in ESACRACK. The report addresses some of the peculiarities and caveats that are associated with the use the two models, which are deceptively similar but which require their own material input data that are not interchangeable.

The guidelines in the report have been derived from a number of reference documents; from a private conversation with the originator of the variable constraint-loss Strip Yield model, A.U. de Koning; from trial runs with ESACRACK version 4.0 (incorporating NASGRO 3.0); and from inspection of the source code of NASGRO v3.0.21. N.B: The report is not intended to replace the various user manuals, but is meant to be supplementary.

The work underlying this report has been performed within the context of the ESA project “Structural Integrity of Pressurised Structures”.

1.7.12 Strip Yield variable constraint models applied to space spectra (F.P. Grooteman, NLR)

As part of an ESA contract the NLR has examined four alternative variable constraint models to improve the current one available in ESACRACK version 3.0.21. This constraint model is part of the uniaxial Strip Yield crack growth model to simulate three dimensional effects. This is done by introducing so-called constraint factors, whose values may vary over the plastic zone.

In order to select the best performing model, the five constraint models (current and four alternatives) have been used in Strip Yield fatigue crack growth analyses for a series of space structure test cases for which experimental results were available.The 4th alternative constraint model is regarded the best of all five models. To draw a final conclusion it is recommended to update the material fit and examine the five models’ performance on aircraft spectra as well. The work is reported in NLR-CR-2006-390.

N.B: Analysis results for the current model deviated considerably from previous ones. The errors were tracked and repaired, bringing the Strip Yield model to its original state. This demonstrates the importance of checking (in an automated way) the source code on a regular basis by means of a series of test cases.

1.7.13 Ice impact protection shield (S.Bosscher, Stork Fokker AESP)

Stork Fokker AESP is responsible for the design and production of an ice impact protection shield for a new aircraft with turboprop engines. This ice shield is to prevent damage to the fuselage owing to impacts from ice shedding off the propellers. The location of the ice shield on the fuselage and the operational requirements for the aircraft lead to unusually large acoustic loads on the ice shield. Acoustic fatigue must be considered for the strength justification of the ice shield and the related loads are so severe that acoustic fatigue is in fact the sizing requirement for the ice shield.

Public domain acoustic fatigue methodologies, such as available from ESDU, are not easily applicable to acoustic fatigue due to propeller noise. This is because propeller noise spectra are too dissimilar to the acoustic loadings assumed in these methodologies. Therefore Stork Fokker AESP uses FEM frequency response calculations and comparison with products of a similar material as the means of justification. Full-scale tests are not foreseen.

1.8 FULL SCALE FATIGUE TESTS

1.8.1 Airbus A380-800 Fixed Leading Edge J-nose top panel fatigue and damage tolerance testing (J. R. Docter, Stork Fokker AESP, P.H. de Haan, NLR)

Stork Fokker AESP is responsible for the design and production of the J-nose sections as fitted to the Fixed Leading Edge (FLE) of the A380-800 wing, see figure 25. The glass thermoplastic J-Nose forms part of the aerodynamic surface of the mid and outer sections of the A380-800 fixed leading edge. It is bolted directly to the leading edge ribs and to the wing upper skin overhang via metallic lands. This J-nose concept was first used by Airbus on the A340-600 inboard FLE.

Fatigue and damage tolerance testing of the J-nose top panel was part of the J-nose certification programme. Static and fatigue testing was done on two full-scale specimens representing the J-nose top panel and manufactured by Stork Fokker AESP. This panel is between track rib 5 and hold-down rib 1 on the aircraft wing, and is a hybrid structure containing composite and metal parts.

Figure 26 gives overall diagrammatic views of the test rig and a test specimen, and figures 27 and 28 show photographs of the test rig and a specimen. The specimens were mounted on a representative J-nose structure comprising a dummy front spar (skin overhang), dummy sub-spar, ribs, spreaders and front spar lands, together with a representative section of skin and stringers.

The objectives of the test programme were:

1. Confirm the fatigue strength that provides ‘no-growth’ requirements for imparted defects and damages.
2. Confirm the ultimate static strength after the required cyclic load with imparted damages (BVID).
3. Confirm the limit strength of the structure after the required cyclic load.
4. For repairs confirm the fatigue strength after the required cyclic load and the ultimate strength.
5. Show that the A380 J-nose top panel structure will be able to withstand the repeated large non-linear buckling deformations caused by cyclic compressive loading.
6. Validate non-linear analysis by correlating strain gauge and displacement readings with finite element model predictions, see figure 29.
7. Study the progressive failure behaviour and the extent of damage incurred to the J-nose top panel.

The test programme was performed in the RT/dry condition. Environmental Knock Down Factors were applied to the test results to validate the tests for all environmental conditions.

<u>Static test</u>
Failure of the first specimen occurred at 199% Limit Load, at a front spar location, see figure 30.

<u>Fatigue and damage tolerance test</u>
Fatigue testing of the second specimen was done under flight simulation loading with already-introduced artificial damage at specified damage levels. The artificial damage locations were monitored by close visual and ultrasonic NDI, by qualified inspectors, for any possible extension of the damage.

The fatigue, Limit Load and Ultimate Load testing was completed without extension of the artificial damage. In other words, the test specimen met all requirements concerning " no crack growth" and ultimate static strength in the presence of artificial damage. The specimen was finally failed by compression loading to 190% Limit Load. Failure occurred at a front spar location, see figure 31.

A clear waviness pattern during compressive loading (Wing Up Bending load case) can be seen in figures 30 and 31, and this pattern resembles that from the FEM analysis, see figure 29.

1.8.2 Wing trailing edge movables (G. van Gool, Stork Fokker AESP, A.W. Kempes, NLR)

Several full-scale tests were carried out as part of the certification activities for wing trailing edge movables of a business jet. An aileron has undergone a full-scale fatigue and damage tolerance test at the NLR. This test finished successfully in 2006. A spoiler is in the last stage of fatigue testing, also at the NLR.

The flaps are in the last stage of fatigue and damage tolerance tests. The flaps are tested as part of a full scale test on the entire aircraft. The combined fatigue and damage tolerance test consists of 70,000 flights. The last 20,000 flights are dedicated to damage tolerance.

Artificial damages were introduced into the flaps after 50,000 flights, i.e. 20,000 flights before the end of the fatigue and damage tolerance test. A total of 5 damages were applied at several critical locations. The damage consisted mostly of sawcuts. The most severe damage was complete cutting of a stringer, including the skin attached to the stringer bottom flange. Other locations cover roller lug flanges and cracks in spar caps. After approximately 10,000 flights, and with 10,000 flights to go before the end of the test, regular NDI inspections did not reveal any unexpected crack propagation.

1.8.3 Low Cycle Fatigue (LCF) spectrum and test for the NH90 tail module (B. Vos, Stork Fokker AESP, P.H. de Haan, NLR))

The foldable composite tail module of the NH90 helicopter is attached to the rear structure by means of an aluminium hinge mechanism. The dynamic response from landing impacts, in combination with low cycle in-flight loads, represents severe fatigue loading for this hinge mechanism. The means of compliance for FAR29.571 has therefore to be by analysis and dedicated fatigue testing.

To obtain a realistic LCF spectrum, simulated landing responses are combined with a flight-by-flight spectrum for which the loads are retrieved from an extensive in-flight measurement programme. The flight-by-flight and landing spectra have been derived by Stork Fokker, using a number of severe and less severe flight types. The load sequencing within each flight type as well as the occurrence-mix of the different flight types have been derived such that the contractual helicopter spectrum, normally expressed in %-time consumption per manoeuvre type per H/C life, is correctly represented.

The flight-by-flight spectrum is subjected to an omission and truncation study determined by the aluminium structure and will be randomly combined with the landing spectrum in order to obtain an efficient test spectrum for the fatigue test on the tail module. The automatic tail folding loads will be tested separately. The tested spectrum will consist of several phases, which sequentially cover the (1) as-manufactured metal parts, (2) composite parts with BVID, (3) artificial damages in metal parts and (4) composite parts with CVID. Various residual strength tests will be performed between the fatigue phases and at the end of the total test.

Figure 32 shows the NH90 tail module fatigue test set-up in the NLR Test Hall.

1.8.4 High Cycle Fatigue (HCF) spectrum and analysis for the NH90 tail module (T. Janssen, Stork Fokker AESP, P.H. de Haan, NLR)

In-flight helicopter manoeuvres are characterized by high frequency dynamic loads superimposed on quasi-stationary fatigue cycles (LCF). Together with the NLR, Stork Fokker has developed a process to measure and evaluate the high cycle fatigue (HCF) loads using strain gauges with high sampling rates and rainflow counting methods in combination with FEM.

Damage evaluation of the obtained high frequency loads has been done using the standard Miner rule with regular material fatigue characteristics, which were reprocessed using the appropriate reduction factors on stress levels to establish safe S-N curves.

Damage tolerance for the helicopter tail structure is being substantiated via analysis of the residual fatigue strength in a multiple load path with 1 failed element, in accordance with the required repeat inspection intervals.

High frequency fatigue tests with artificially flawed specimens are being performed for items not amenable to the above analytical approach.

1.8.5 Re-design of machined flap hinge brackets of a large commercial aircraft (R.A.A. Frielink, Stork Fokker AESP)

Within the framework of continuing airworthiness, a few years ago Stork Fokker commenced the re-design of a machined hinge bracket from a trailing edge flap of a large commercial aircraft. The original load dataset appeared to be inappropriate for the re-design. Via a reverse engineering approach an additional load spectrum was predicted using cumulative damage calculations based on the standard Miner rule. A design solution has been introduced on the basis of this newly-derived spectrum.

During implementation of this design solution a second iteration loop in the design appeared to be necessary. This iteration loop put more emphasis on the representativeness of the applied fatigue spectrum. Consequently, load verification analysis has been performed based on both theory and practice.

The practical part of the verification analysis comprised an in-flight test campaign for structural loading measurements. Owing to consistency between theory and practice, a fatigue spectrum could be derived from the in-flight measurements. This fatigue spectrum has been balanced to match theoretical prediction with practical experience. Using the balanced spectrum the re-design has been certified using Miner’s rule and crack growth analysis based on Linear Elastic Fracture Mechanics.

Fractographic analysis provided for the basis of the inspection programme required to cover the mismatch between the formalisation of the final re-design and the actual in-service implementation. In addition, a static residual strength test was done to enable more flexibility in the modification implementation programme. Besides the fatigue analysis, the in-flight data contributed to definition of the applied load for the residual strength test.

The entire project has been performed in collaboration between Stork Fokker, the Aircraft Integrator and the Airworthiness Authorities.

1.8.6 A340-500/600 Aft Pressure Bulkhead (H. Kats, Stork Fokker AESP)

Fokker builds the Aft Pressure Bulkhead (APB) of the A340-500 and -600. Fokker is fully design responsible for the APB, which is located in the centre landing gear area. The APB is a fully aluminium structure consisting of vertical and horizontal panels stiffened by beams and stringers. Two Centre Landing Gear (CLG) brackets are attached to the panels, see figure 33.

Airbus conducted a full-scale fatigue and damage tolerance test for the A340-600 in Dresden. The test duration was 50,000 simulated flights with enhanced spectrum loads. The APB was inspected periodically during the test in order to detect the initiation and propagation of cracks. Special attention was paid to locations determined by analysis to be interesting. Artificial damages were also applied to follow the behaviour of cracks in the APB. An extensive teardown inspection was carried out at the end of the test.

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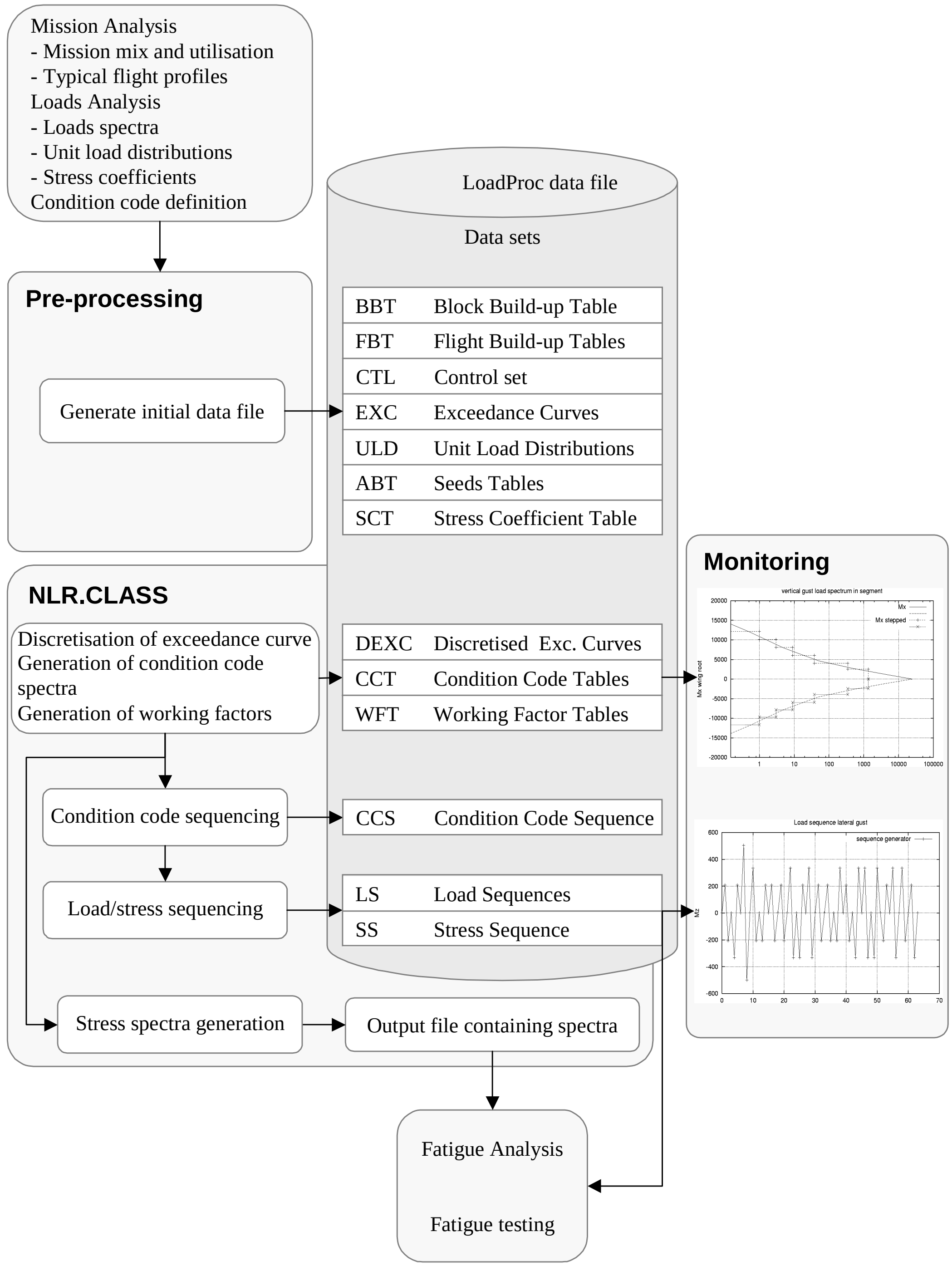


Figure 1 *Flow diagram of the CLASS programme for generation of sequences for loads or stresses for aircraft structures*

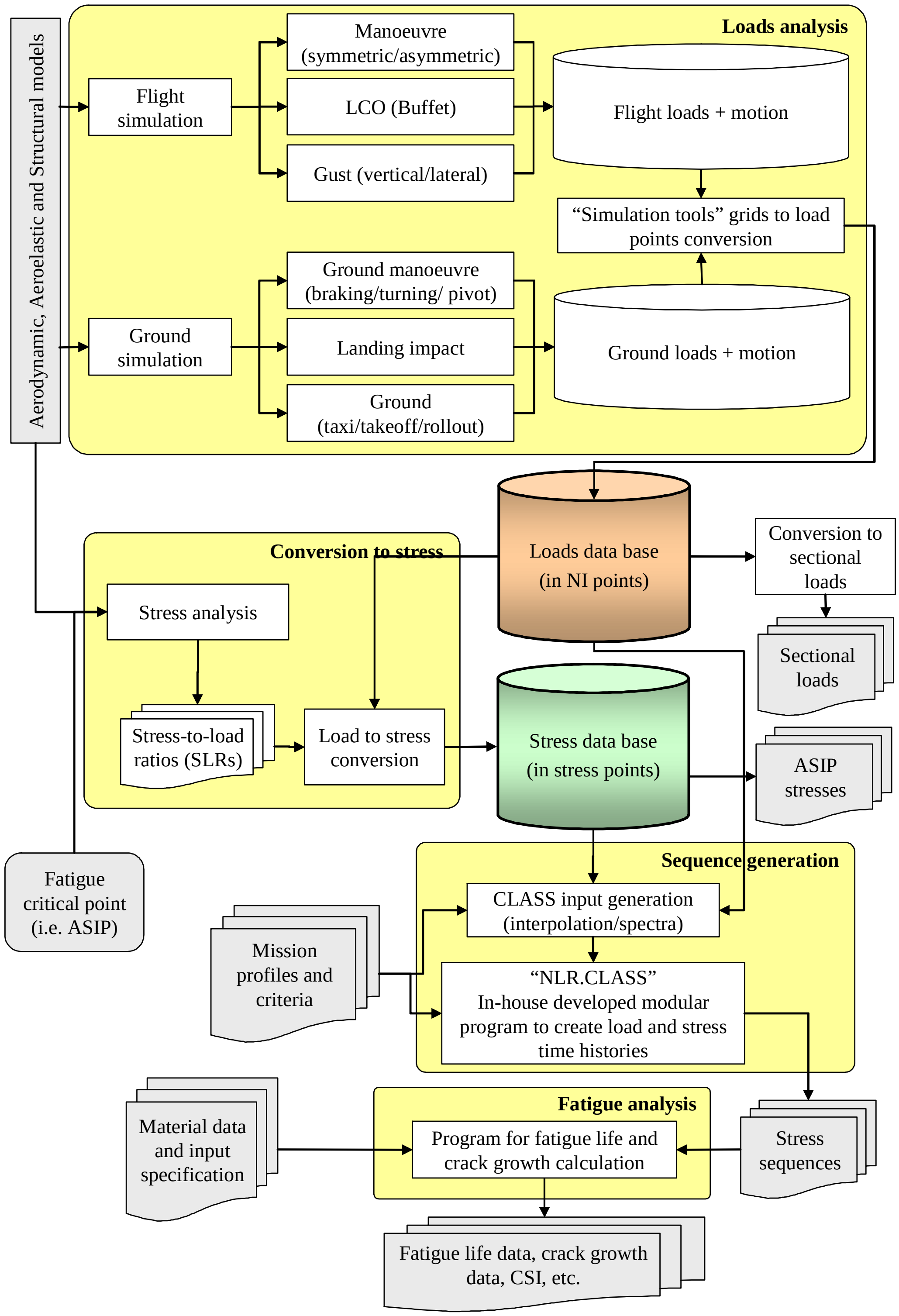


Figure 2 Schematic for a load database system for fatigue analysis

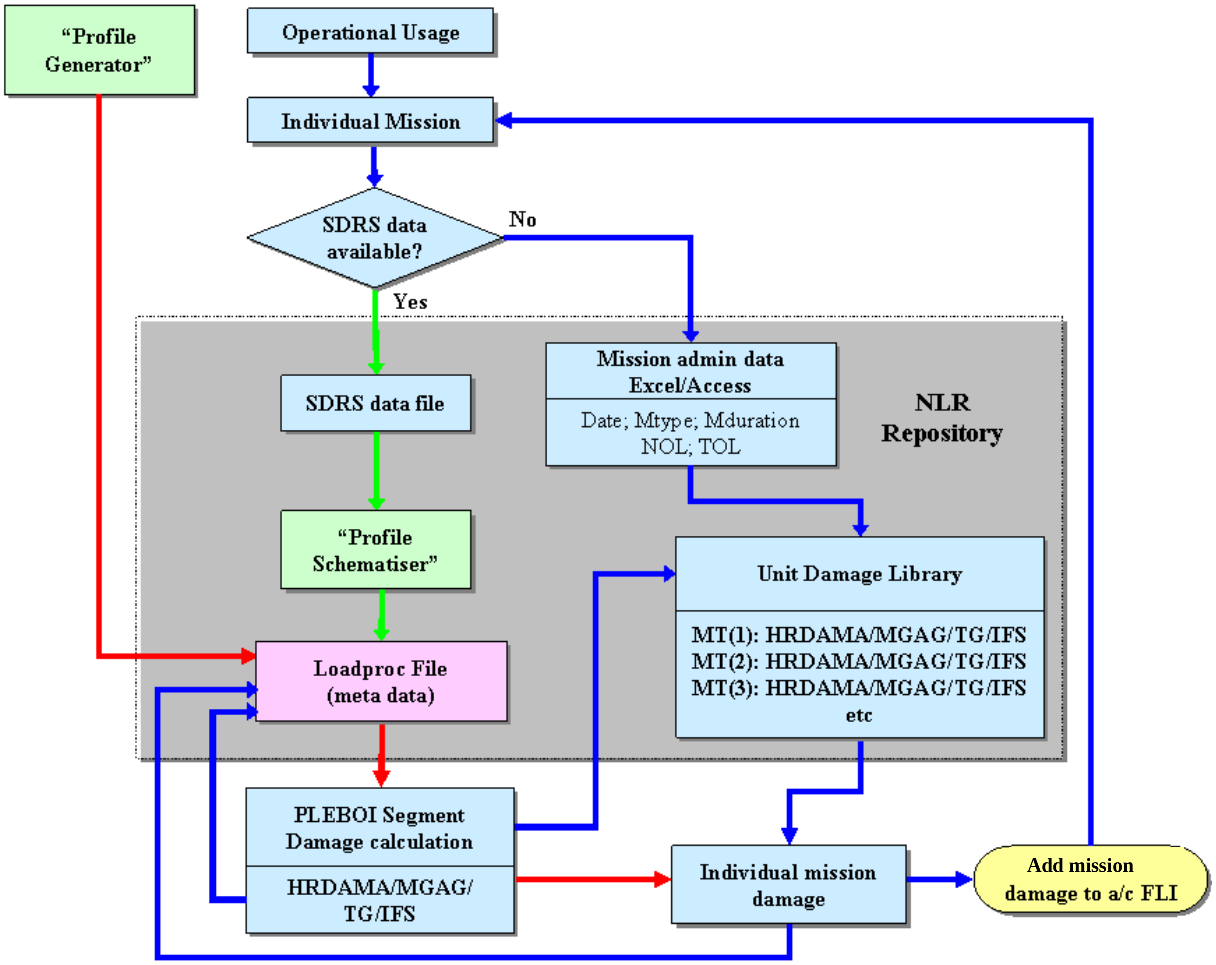


Figure 3 Schematic of P-3 Orion non-SLAP Individual Aircraft Tracking (IAT)

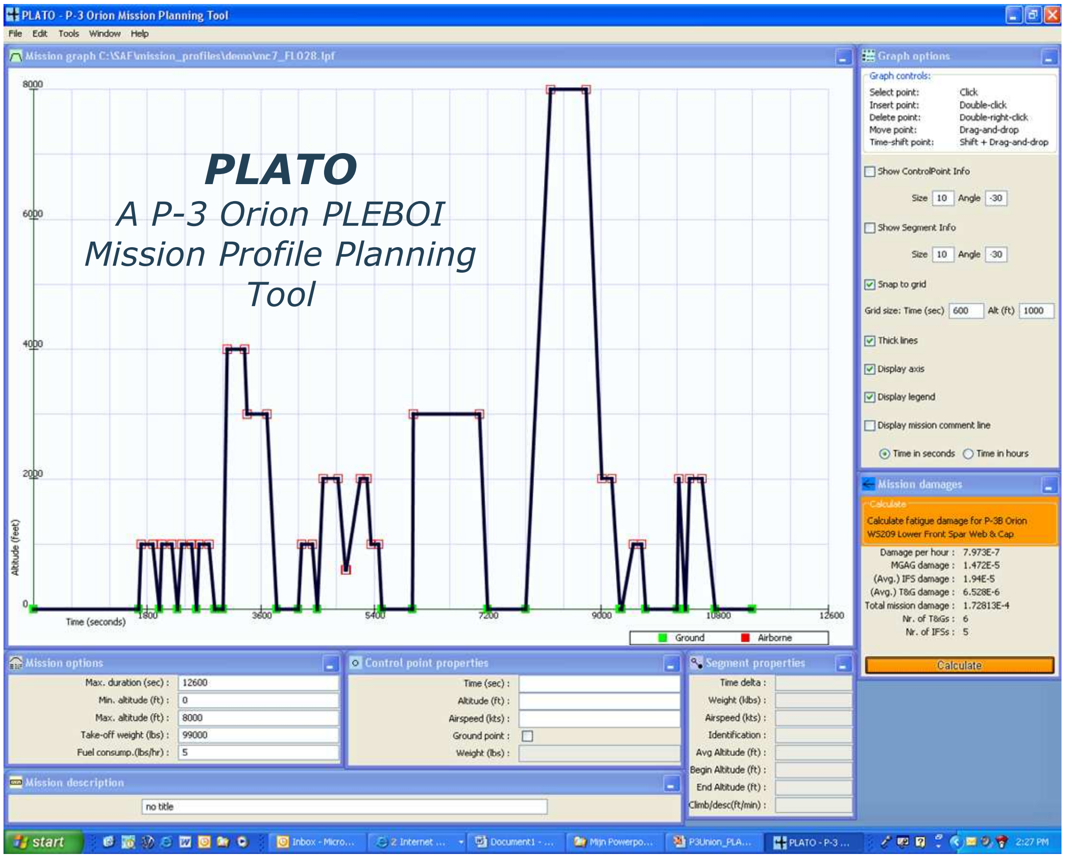


Figure 4 PLATO start-up screen

P3Union.com

The P-3A/B/C Orion Technical Information Exchange Forum

Figure 5 P3Union logo

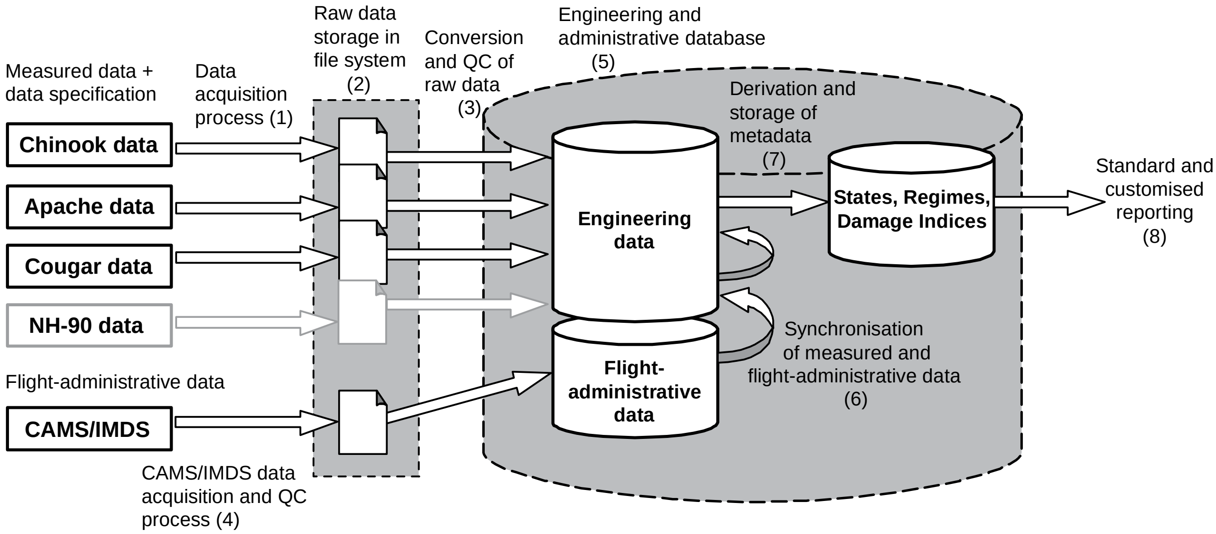


Figure 6 Schematic of HELIUM data management system

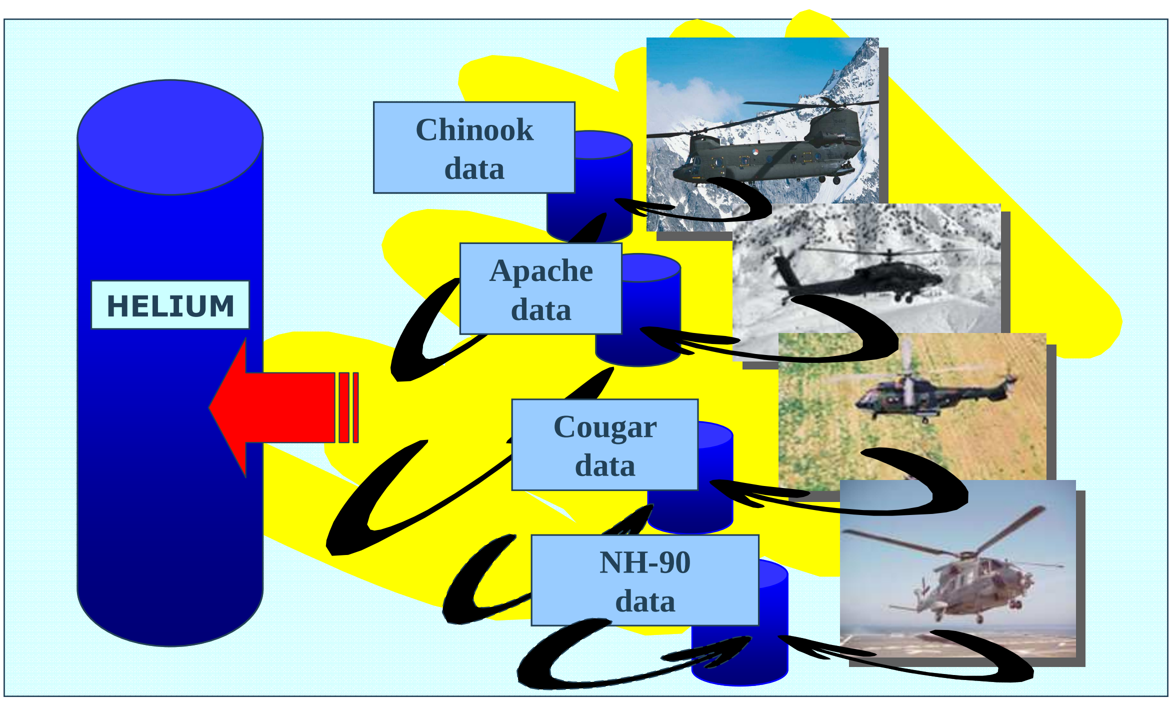


Figure 7 Schematic of RNLAF HELIUM

- ***input:*** **DMO helicopter data**

- ***processes:*** **the data will have to be**
  - loaded, acquired
  - checked
  - cleaned
  - guarded, encrypted
  - stored
  - processed, analyzed
  - correlated
  - maintained for reproducibility
  - reported

- ***output:***
  - queries
  - quick-look
  - standard reports

Figure 8 HELIUM input, processes and output

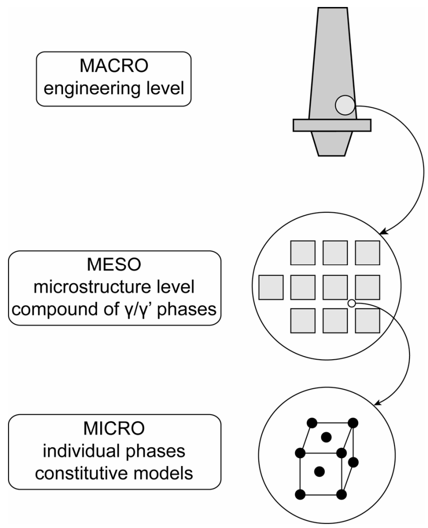
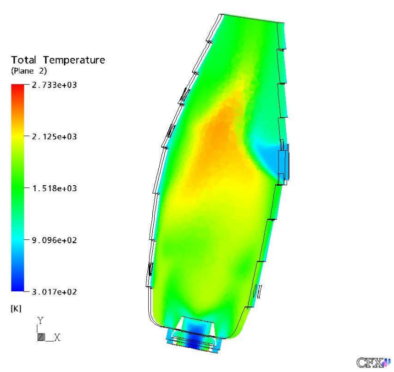


Figure 9 Overview of multiscale approach



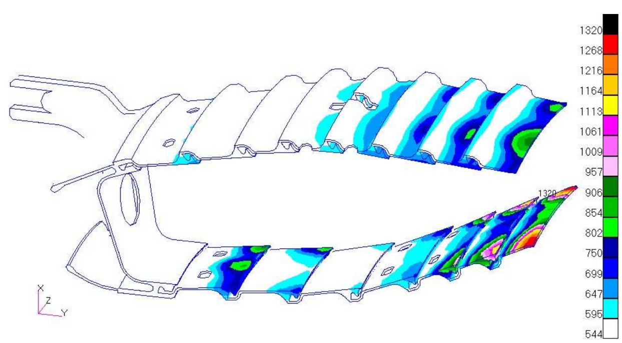


Figure 10 CFD calculated gas temperature inside the combustion chamber.

Figure 11 FE calculated temperature distribution in the combustor liner.

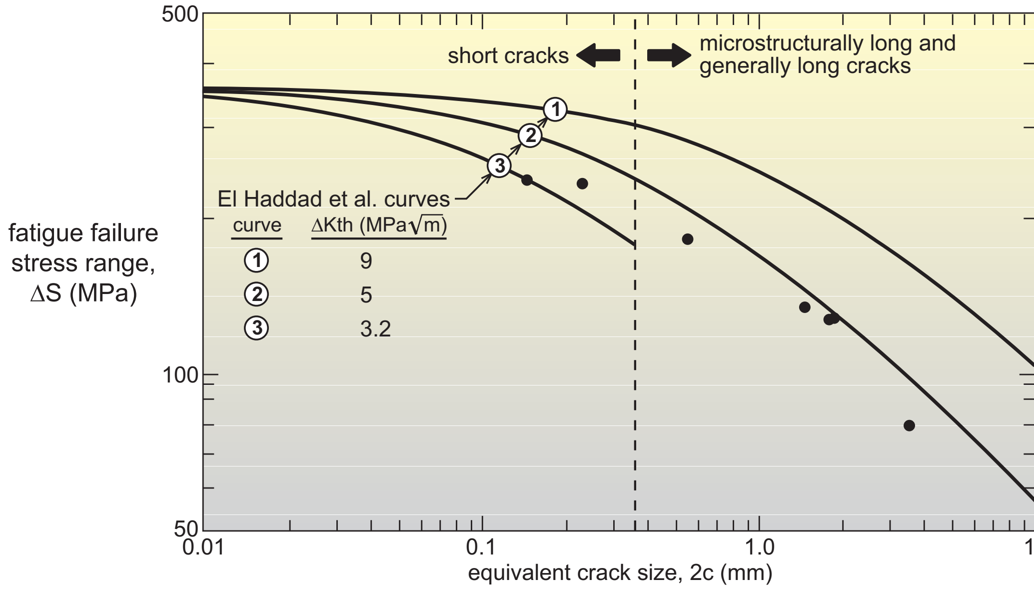


Figure 12 Modified Kitagawa-Takahashi diagram for HCF and FOD + HCF specimens of a γ-TiAl alloy

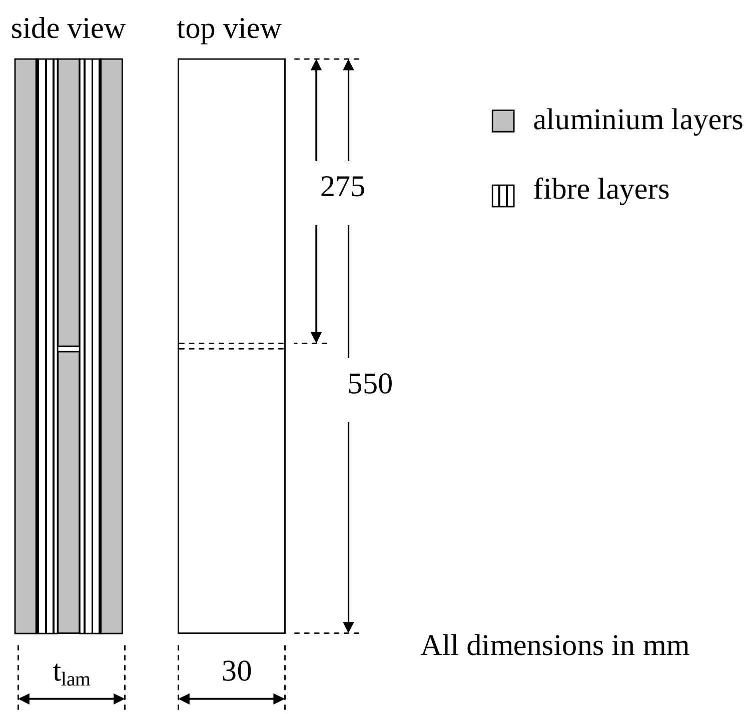


Figure 13 GLARE delamination specimen geometry schematically indicating the fibre direction

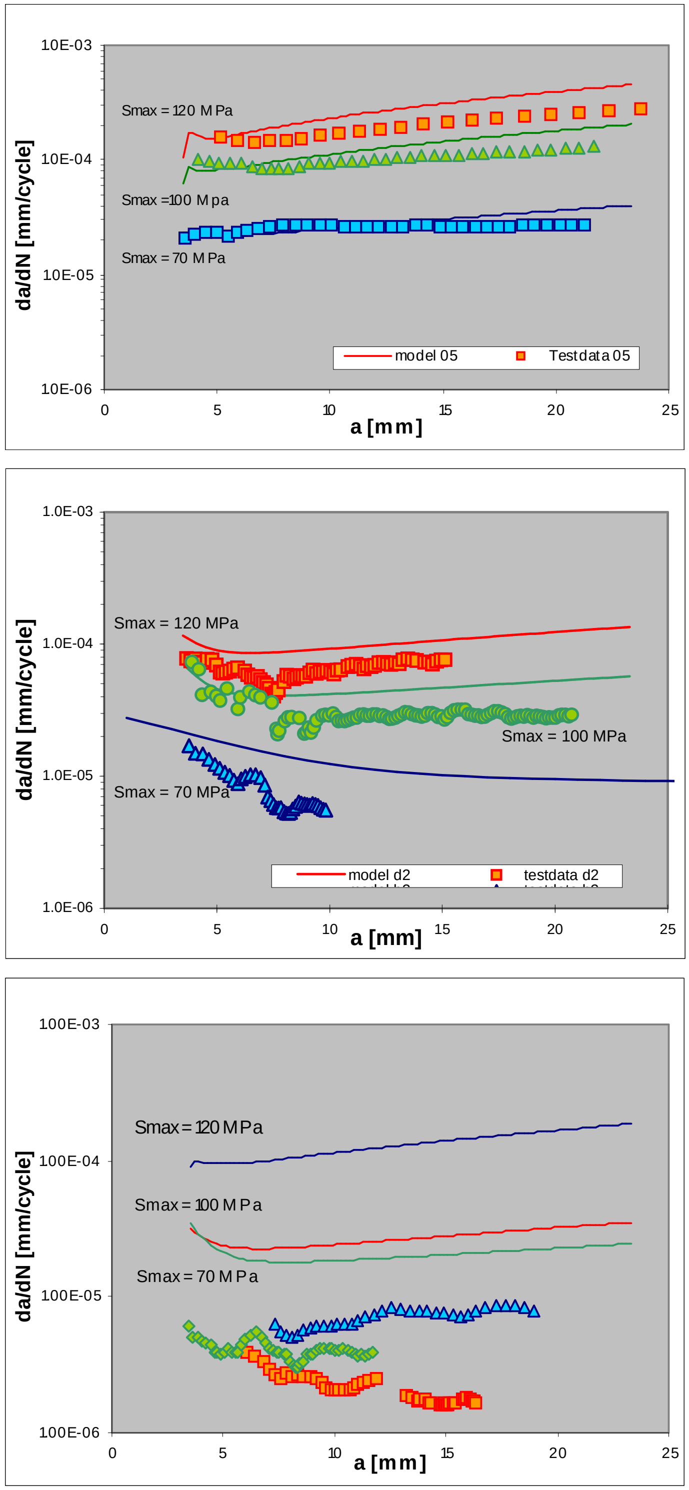


Figure 14 Comparisons between predicted and measured fatigue crack growth rates in GLARE laminates tested at three different temperatures: upper diagram 70℃; middle diagram room temperature; lower diagram -40℃

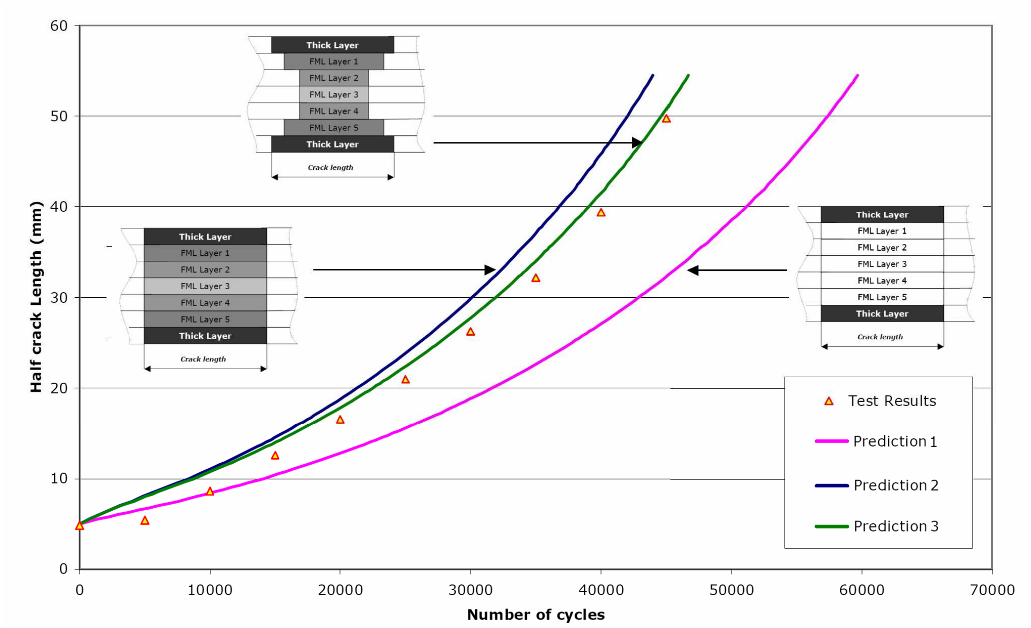


Figure 15 Illustration of the approach to modify the crack growth prediction model developed for FMLs in order to predict the crack growth behaviour of a CentrAl concept [11]

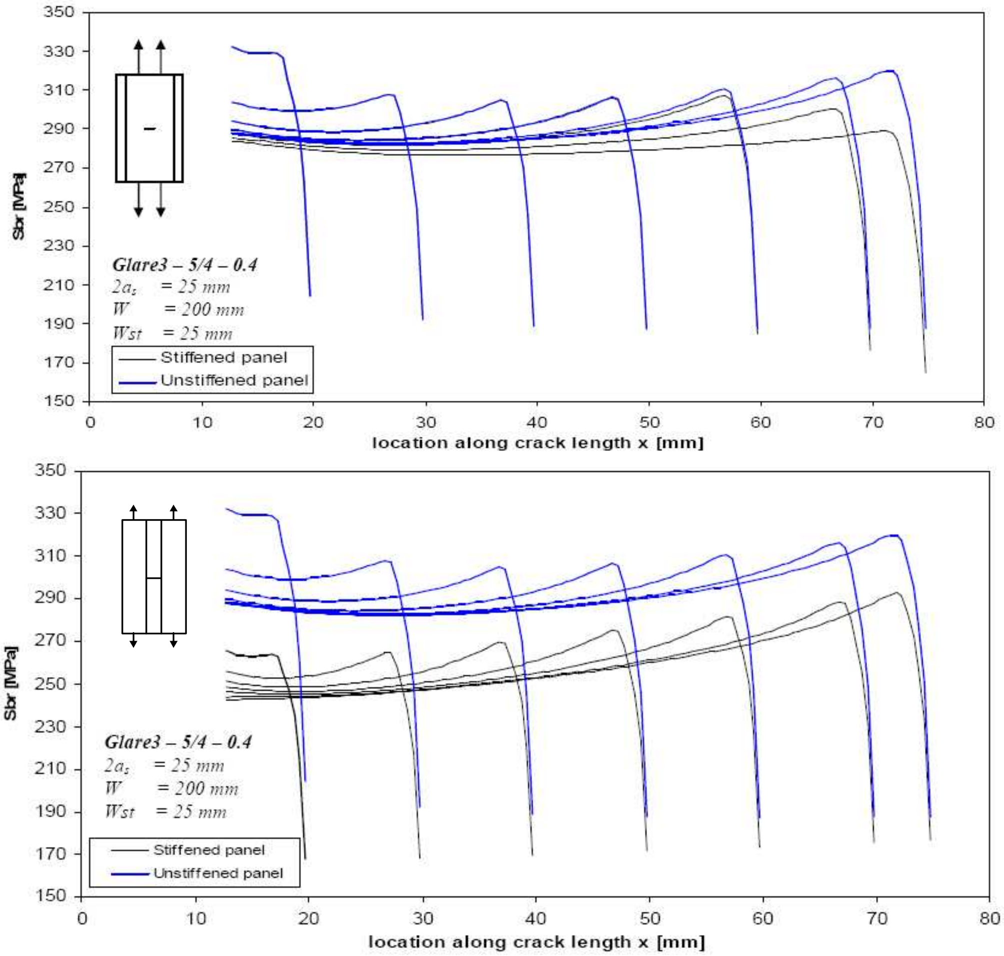


Figure 16 Illustration of calculated bridging stresses for a fatigue crack in an FML skin with and without external (double lateral or central) stiffeners. Upper: crack between stringers, lower: crack underneath intact bonded stringer [12].

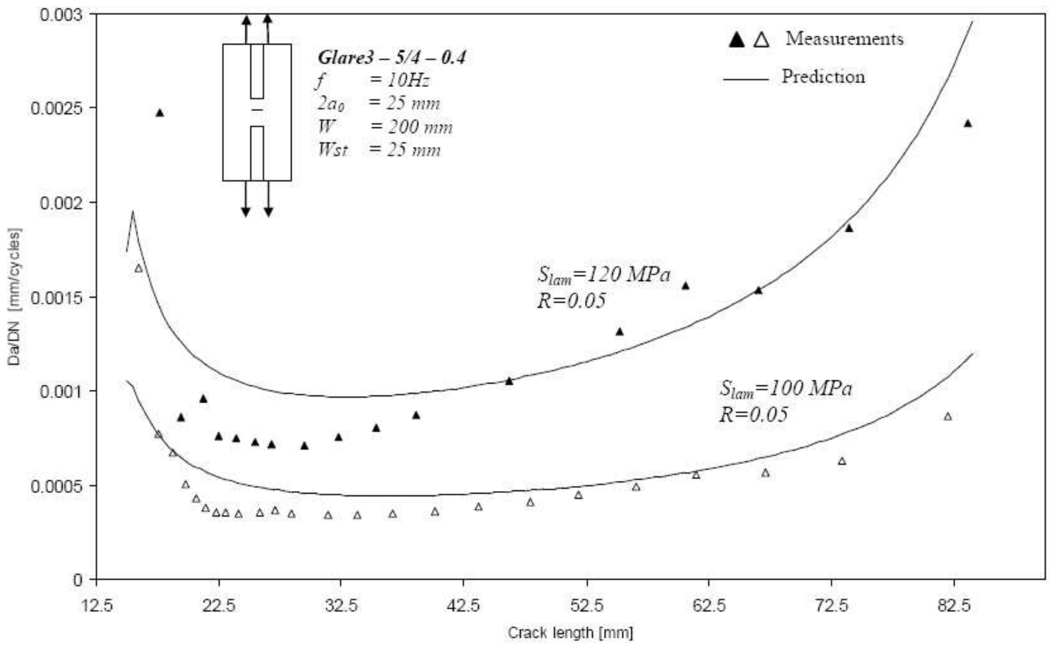


Figure 17 Comparison between predicted and measured crack growth rate of Glare3-5/4-0.4 stiffened by broken central titanium straps, W=200 mm, W_{st}=25mm, a_0=12.5 mm

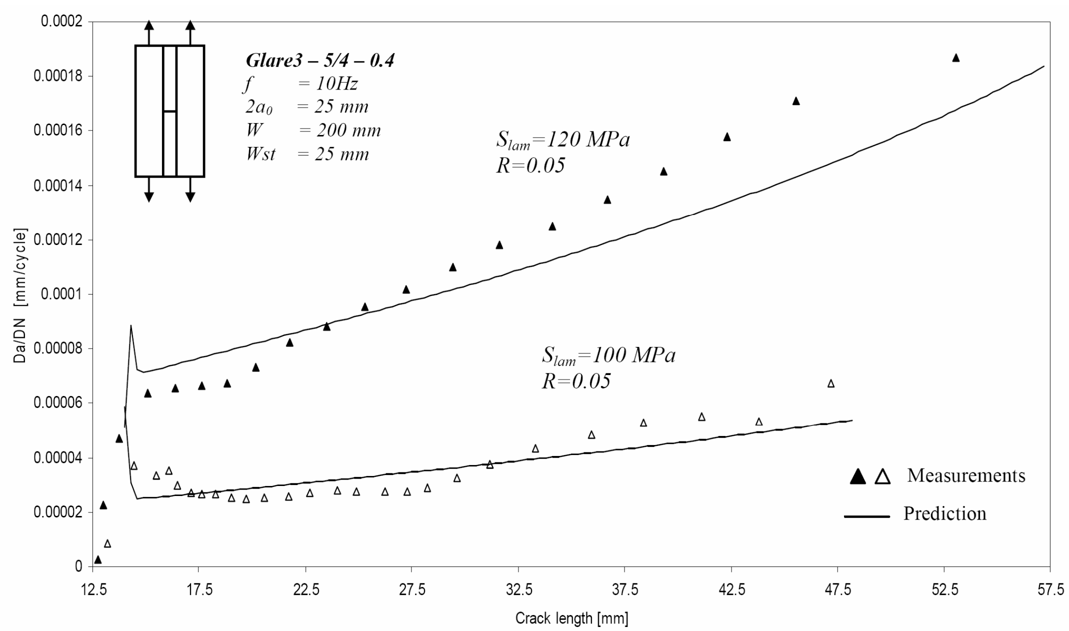


Figure 18 Comparison between predicted and measured crack growth rate of Glare3-5/4-0.4 stiffened by intact central titanium straps, W=200 mm, W_{st}=25mm, a_0=12.5 mm.

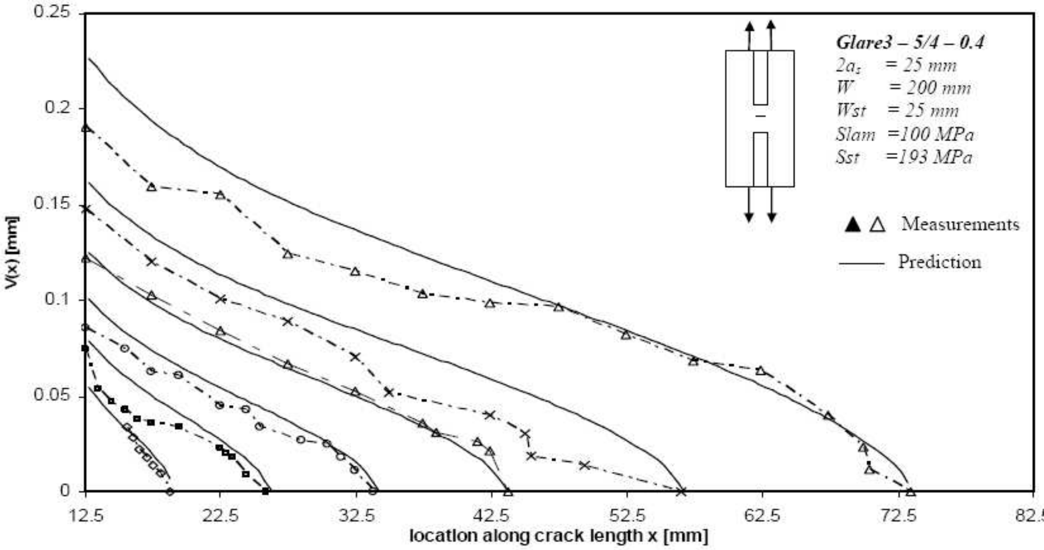
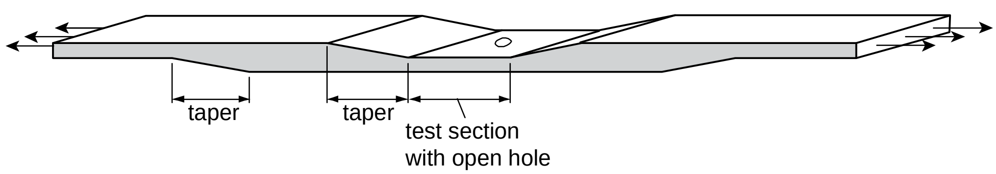
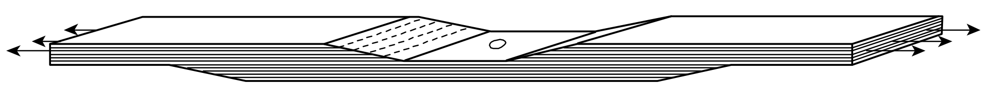


Figure 19 Comparison between predicted and measured crack opening contour of Glare3-5/4-0.4 stiffened by broken central titanium straps. W=200 mm, W_{st}=25mm, a_0=12.5 mm, S_{lam}=100 MPa and the fatigue stress ratio R=0.05



(a) Monolithic aluminium plate machined to tension-bending specimen



(b) GLARE plate machined to tension-bending specimen

Figure 20 Machined open hole tension-bending specimen configurations

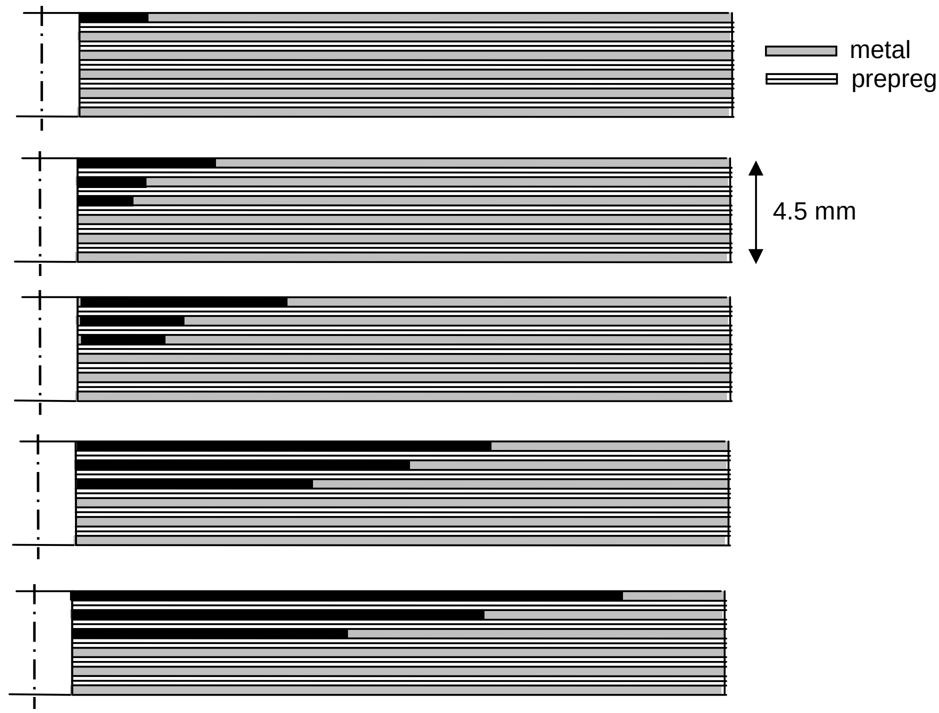


Figure 21: Typical fatigue crack patterns on one side of an open hole during successive stages of crack growth for GLARE 2A-6/5-0.4 tested at Smax = 100 MPa: specimen width 100 mm.

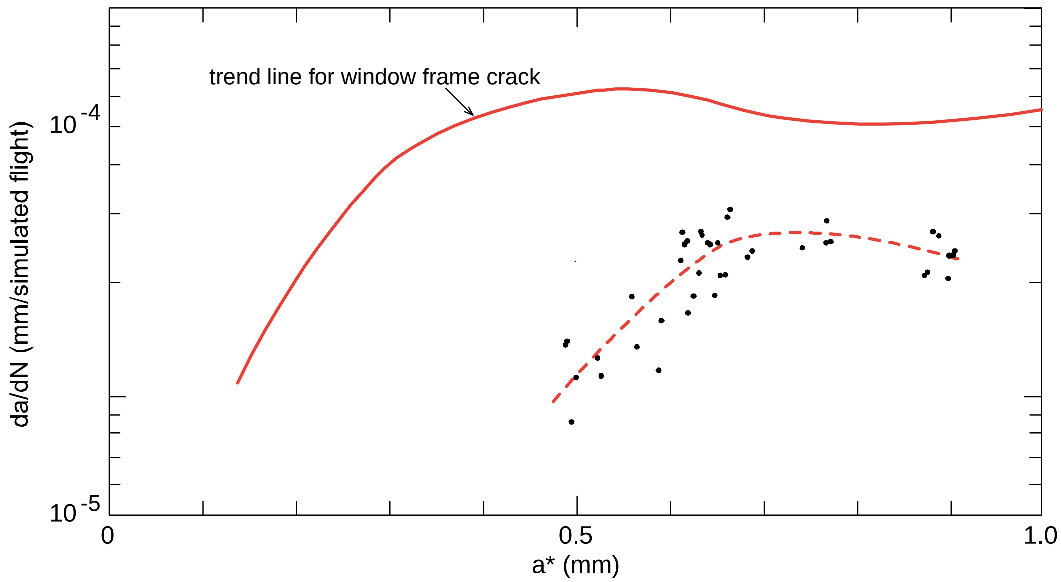


*Figure 22 MegaLiner Barrel (MLB) full-scale fatigue test: fatigue crack growth rate curves for the largest window frame fastener hole crack, in monolithic 7175-T73, and the largest 'readable' fastener hole crack in an aluminium layer (0.3mm 2024-T3) of the GLARE skin of the window: crack growth rates versus mean crack lengths, a**

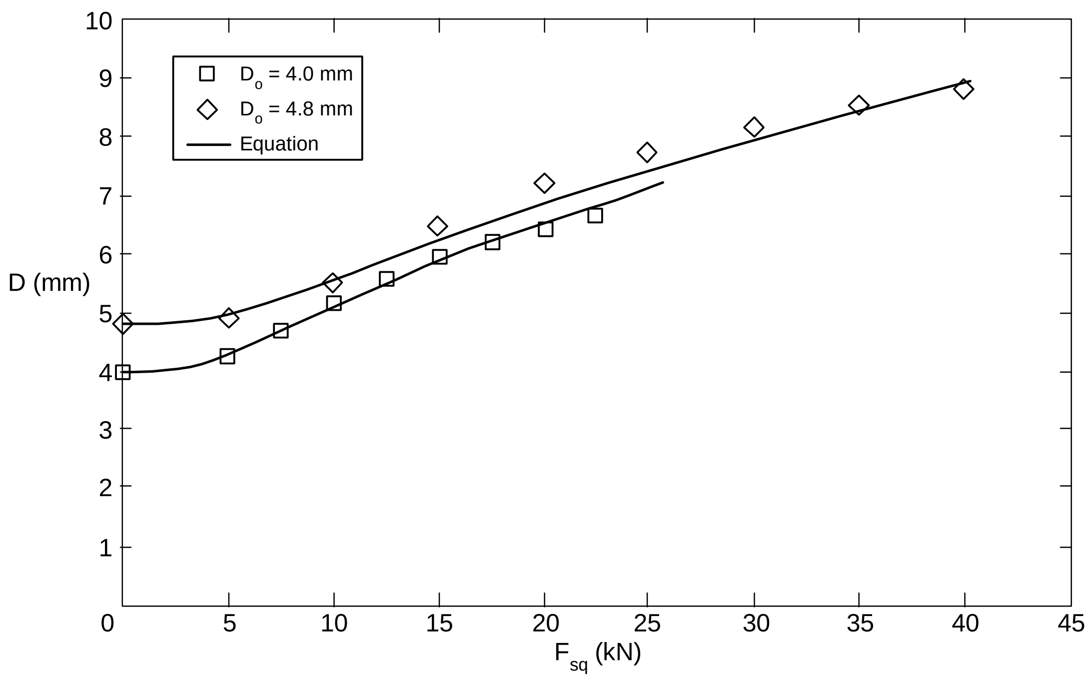


Figure 23 Illustration of the agreement obtained between experiments and predictions of the Diameters, D, of driven rivet heads as functions of the applied squeeze load, F_{sq}: 2024-T3 sheet, AD rivets (2117-T4)

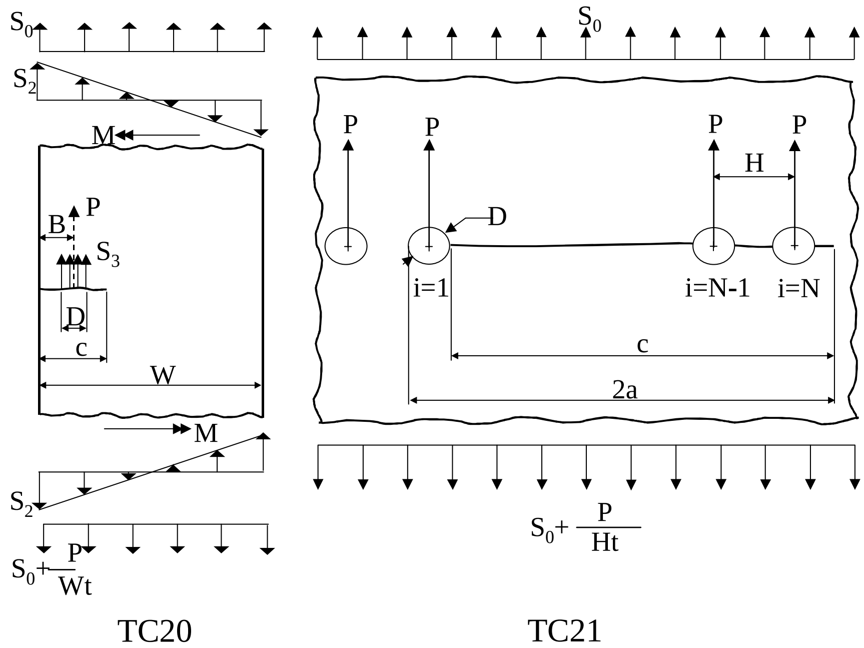


Figure 24 Two new stress intensity factor configurations added to NASGRO 3

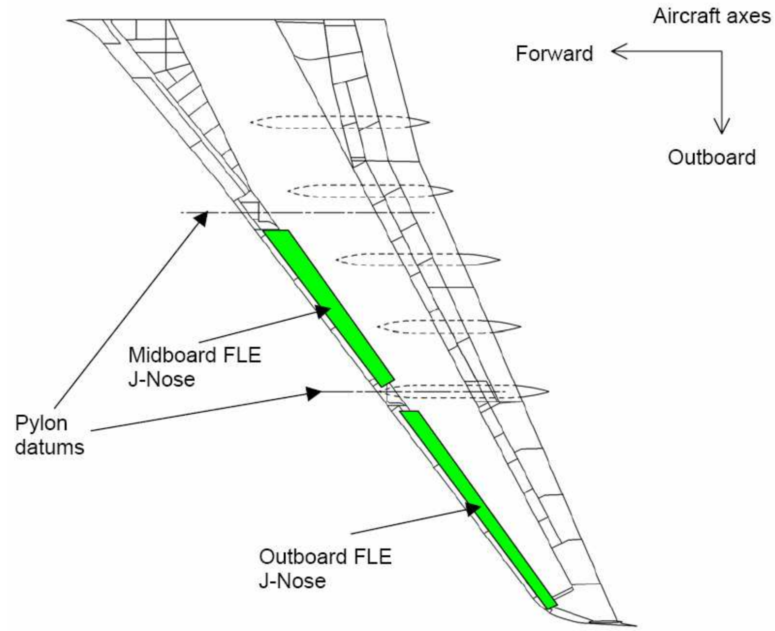


Figure 25 Top view of the A380 wing with the location of the J-Nose highlighted. The slats and engine pods are not shown

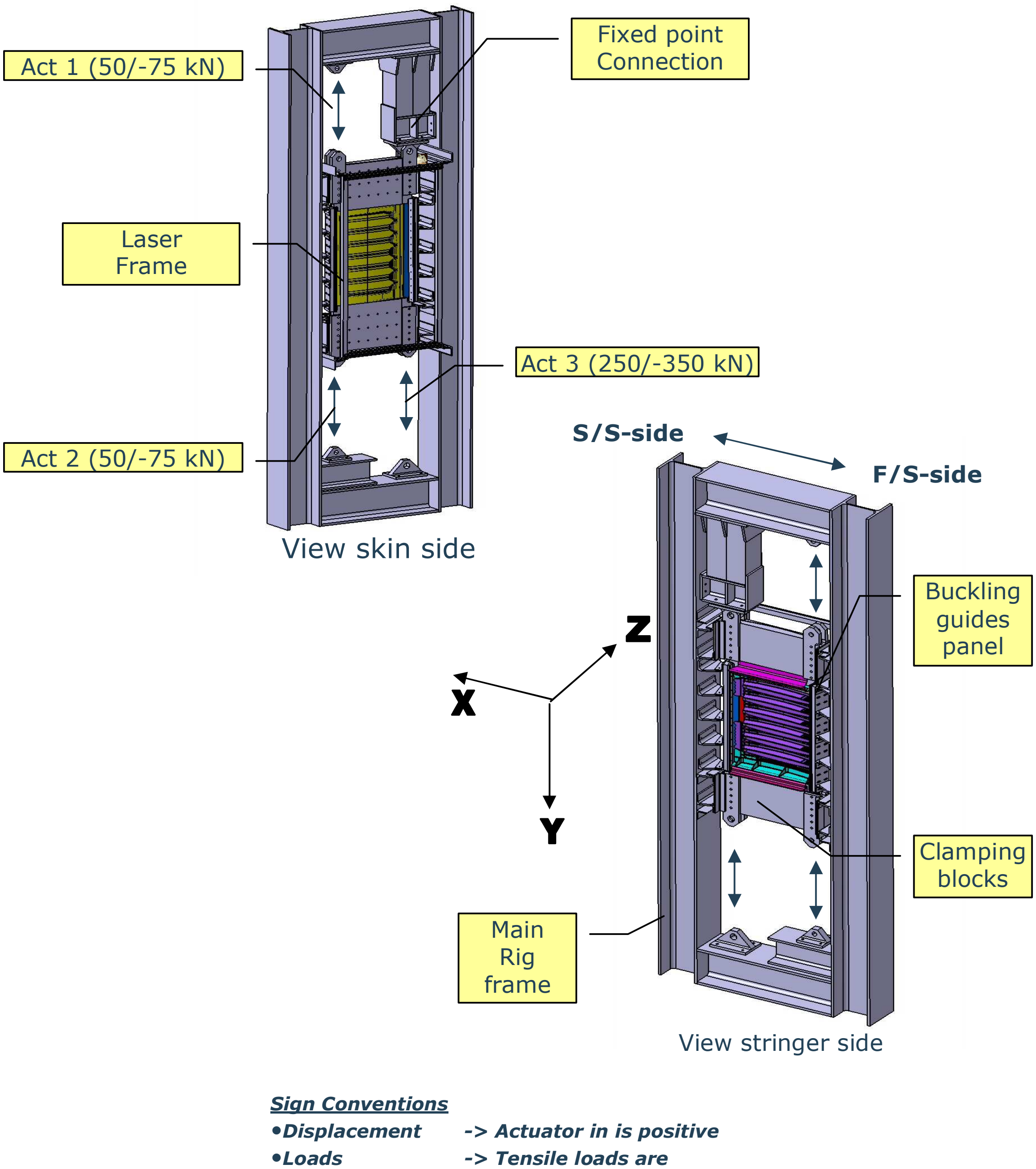


Figure 26 Overall view of the test rig and test specimen, as built up in the NLR Test Hall

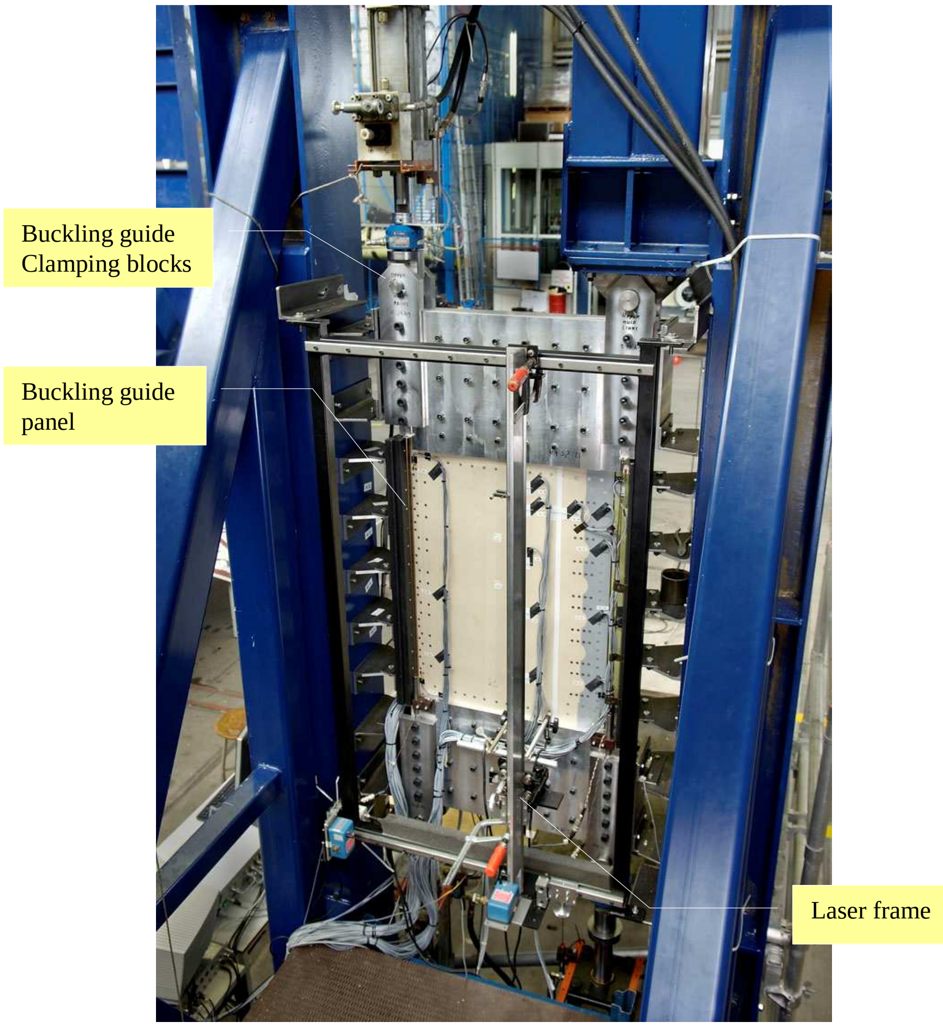


Figure 27   Test rig view from skin side

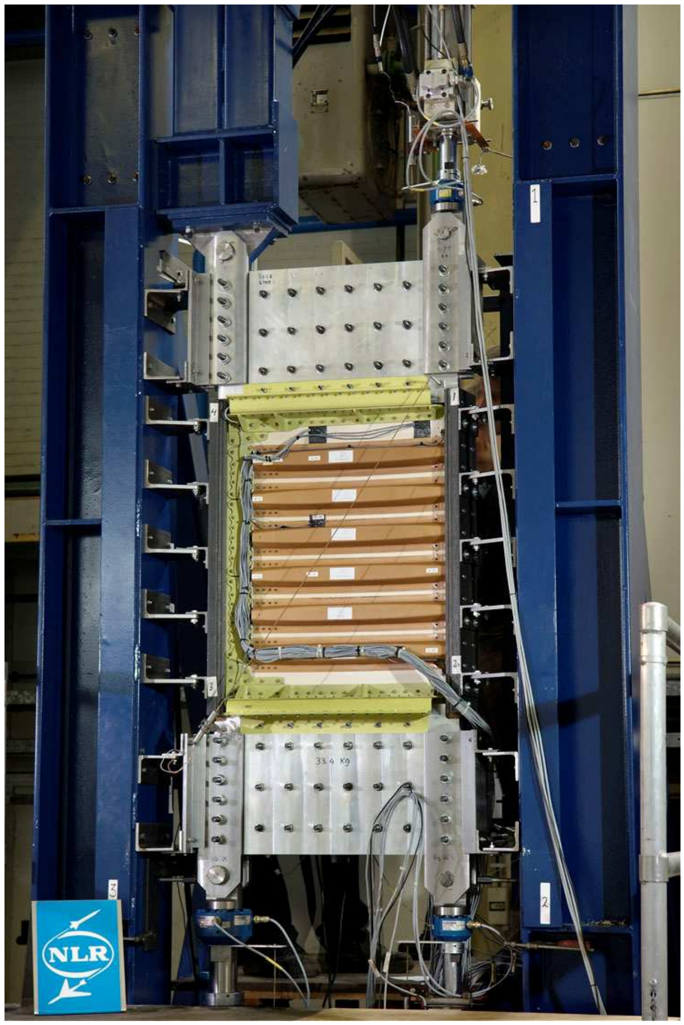


Figure 28 Test rig view from stringer side

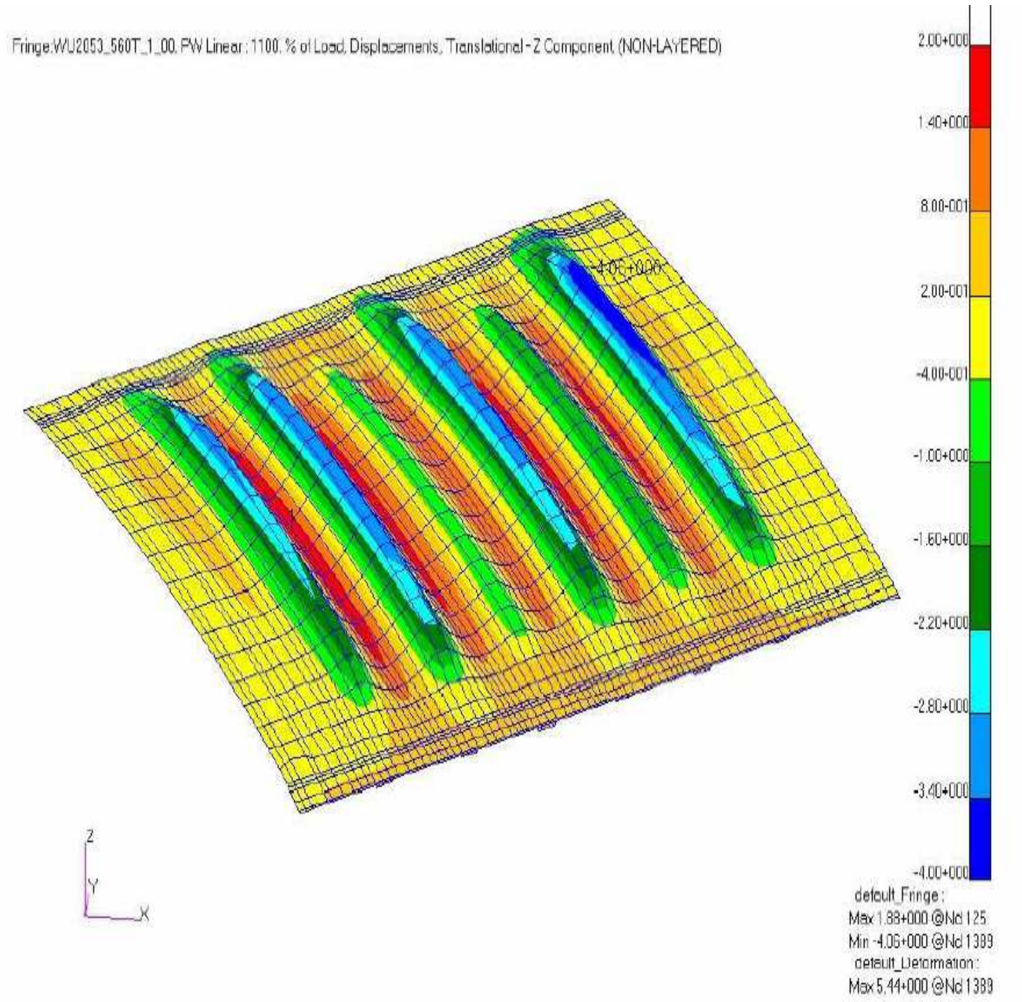


Figure 29 FEM out of plane displacements plot during a Wing Up Bending load case

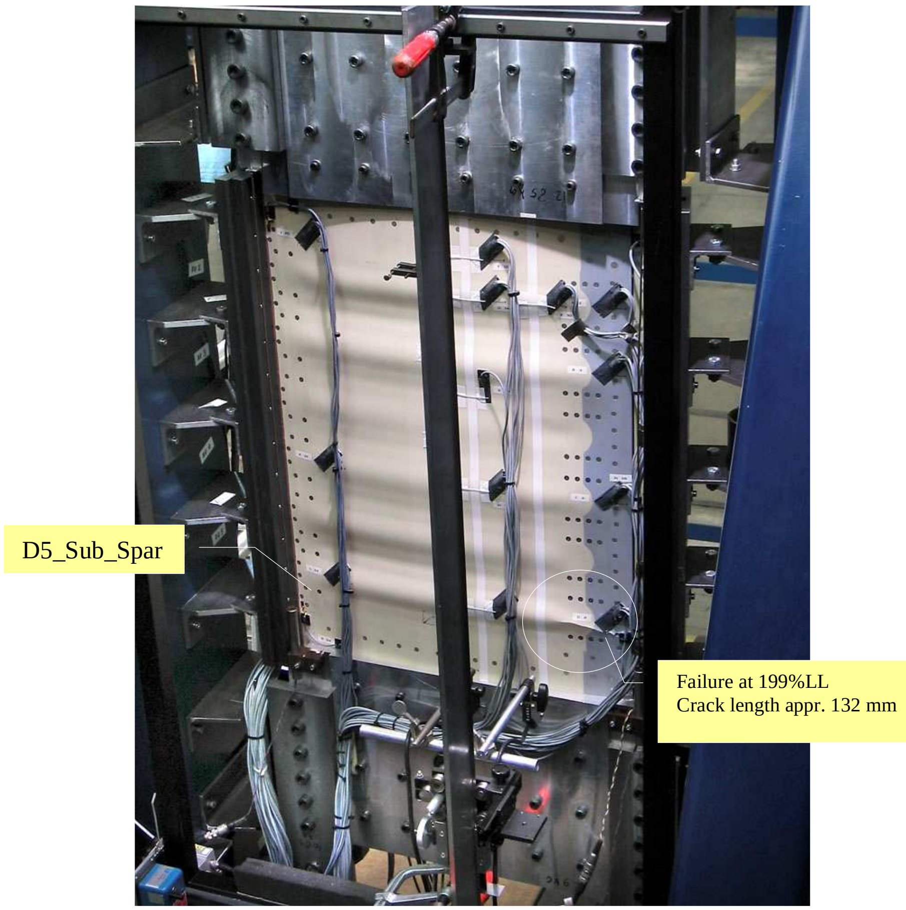


Figure 30   Failure of static test specimen at lower front spar location under 199% Limit Load

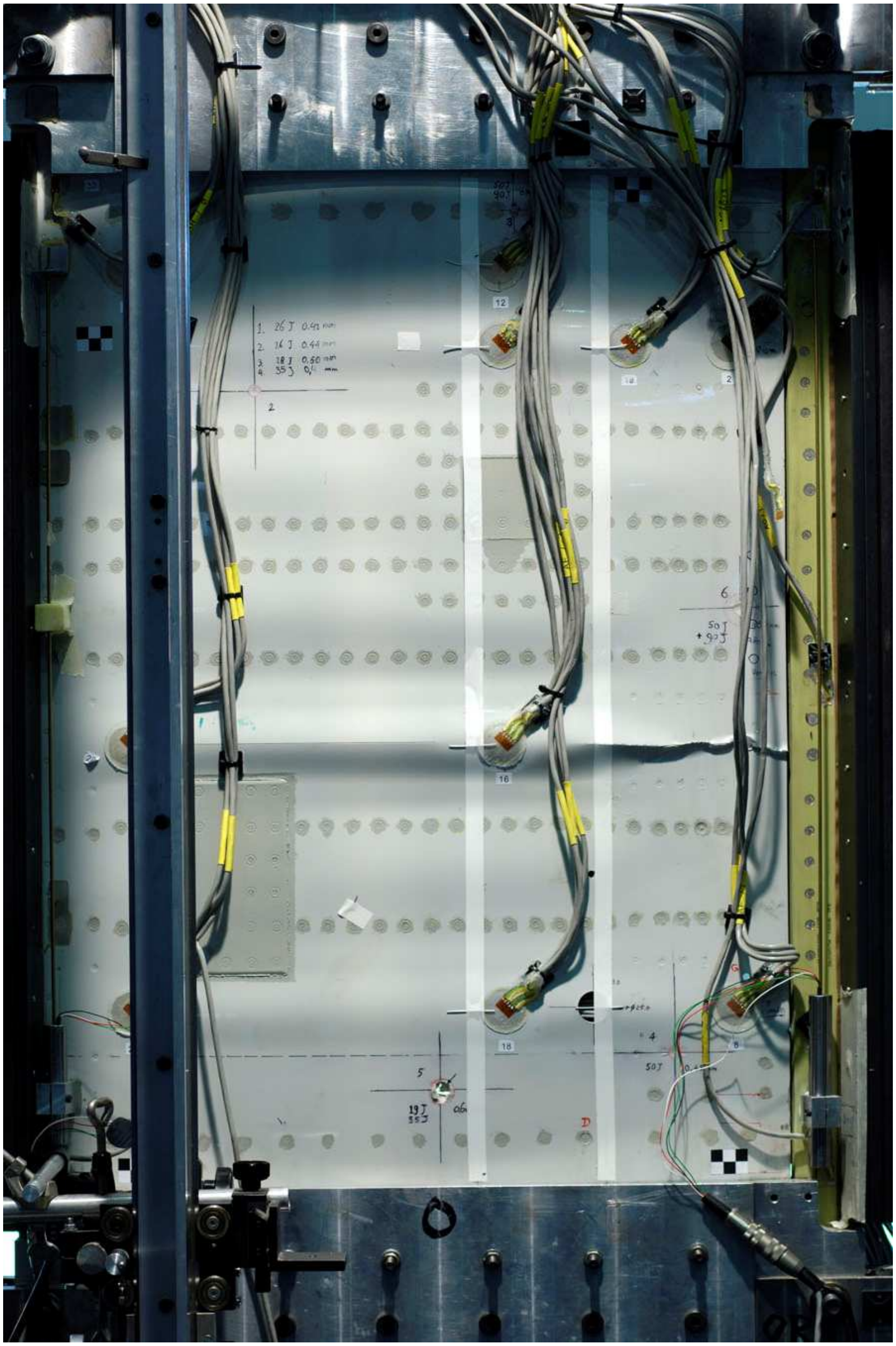


Figure 31 Failure of fatigue and damage tolerance test specimen at the front spar location under 190% Limit Load

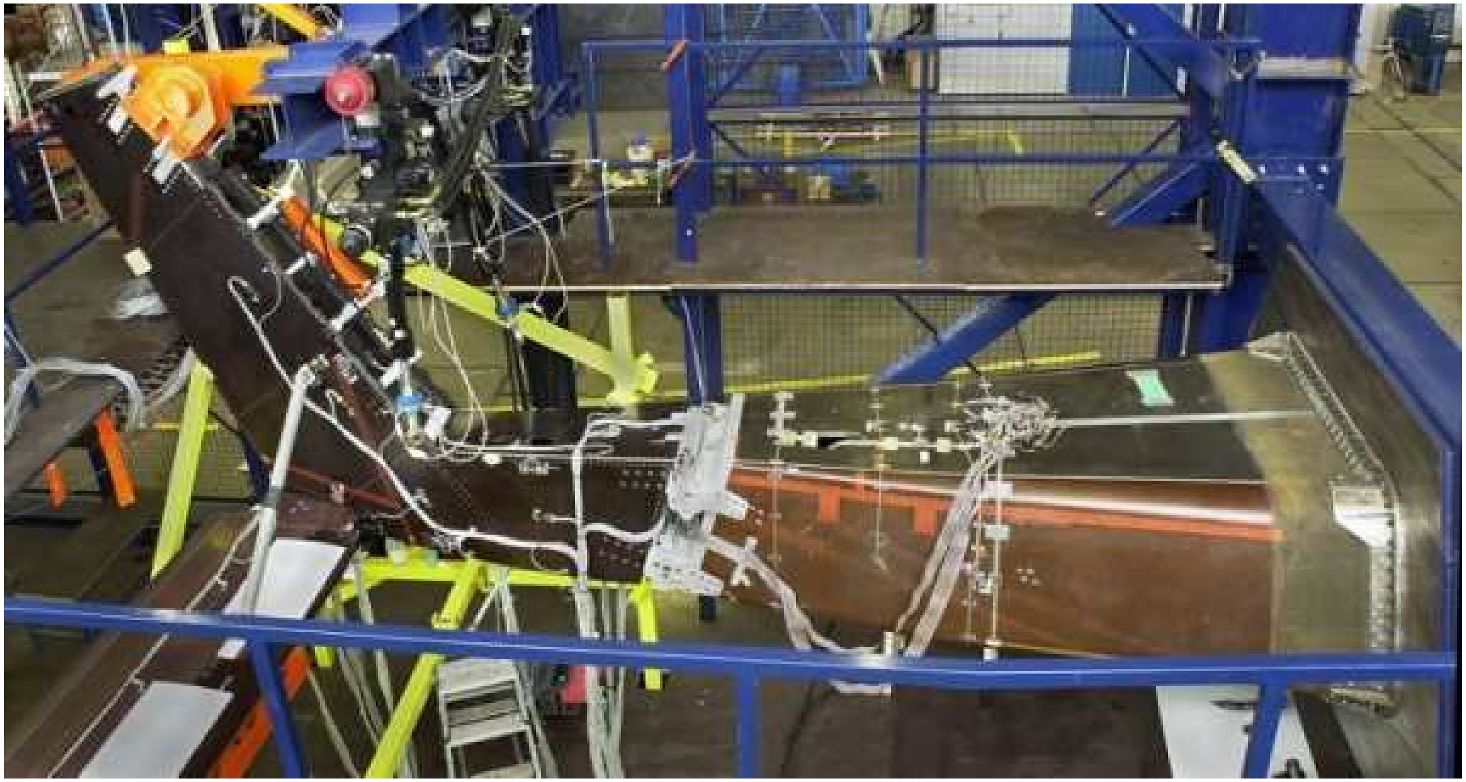


Figure 32 NH90 helicopter tail module fatigue test set-up in the NLR Test Hall

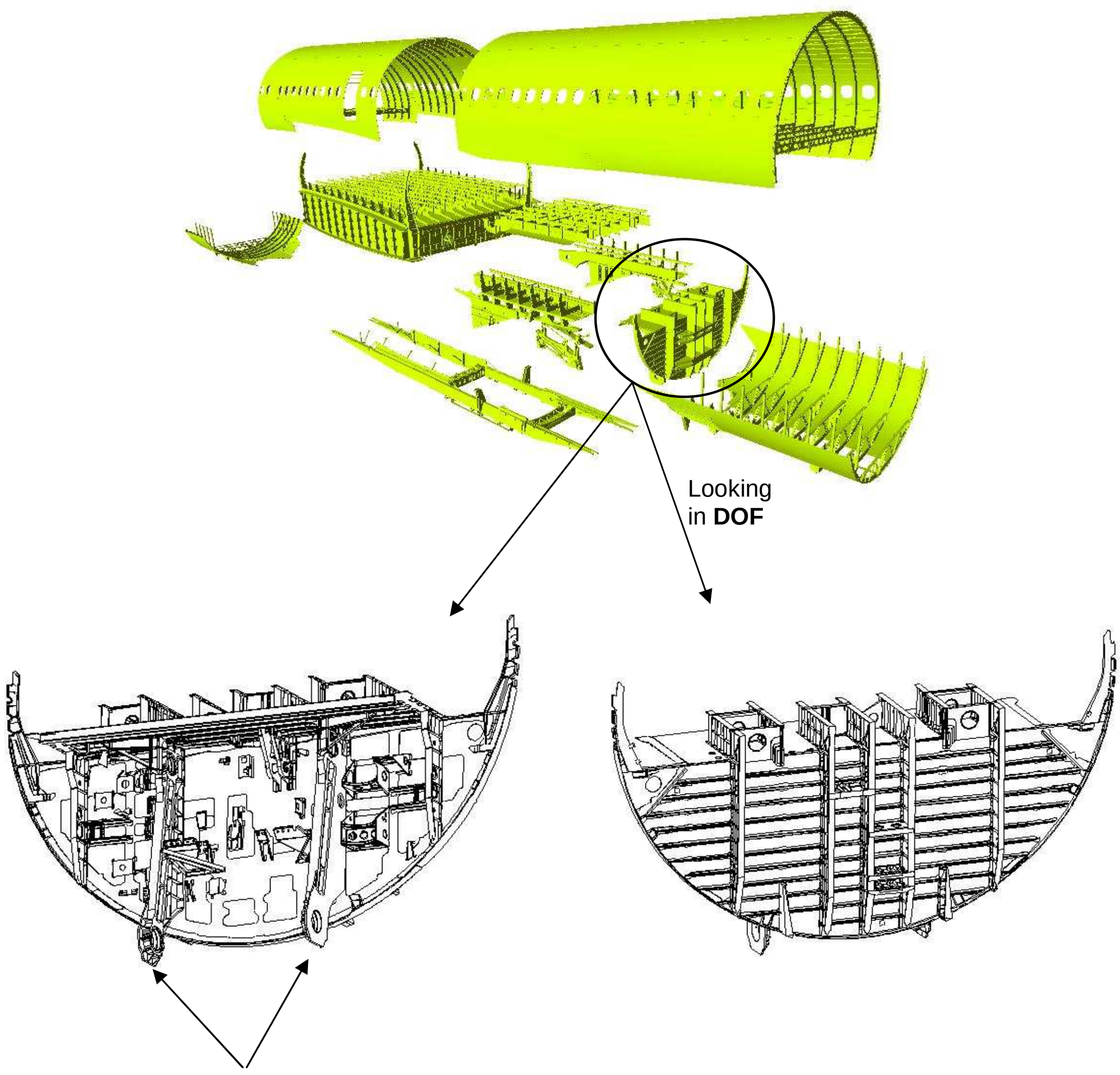


Figure 33   A340-500/600 Aft Pressure Bulkhead and Centre Landing Gear (CLG) brackets