Review of Aeronautical Fatigue Investigations in Switzerland

April 2009 – February 2011

KOR-TA2011-0099

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SUMMARY

The Swiss review summarizes fatigue work carried out in Switzerland. It includes main contributions from the RUAG Switzerland Ltd, Division Aviation, CFS Engineering, M@M Mandanis GmbH, and Pilatus Aircraft Ltd. This document later forms a chapter of the ICAF conference minutes published by the conference host nation. The format of the review reflects ICAF requirements.

Prepared for the presentation at the 32th ICAF Conference
Montreal, 30 and 31 May 2011

4.1 Introduction

The present review gives a brief summary of the work performed in Switzerland in the field of aeronautical fatigue, during the period from May 2009 till February 2011. The various contributions to this review come from the following sources:

- **RUAG Switzerland Ltd; Engineering Services: Aerodynamic & Structural Department**
  M. Guillaume, A. Gehri, P. Stephani, B. Bucher, S. Büsser, S. Frei, A. Gassmann, M. Giger, C. Huber, I. Kongshavn, J. Lussi, B. Schmid, D. Schmid, A. Uebersax, M. Wyss, S. Zehnder

- **Pilatus Aircraft Ltd; Structural Engineering, Stans**
  L. Schmid, D. Romancuk, D. Quaranta

- **CFS Engineering SA, Lausanne**
  J. Vos

- **M@M Mandanis GmbH, Kriens**
  G. Mandanis

The work of related topics on the Swiss fatigue investigations supported by the above mentioned people is gratefully acknowledged.

A part of the preparation of this review is sponsored by the armasuisse. This fond is gratefully acknowledged.

4.2 Pilatus PC-12 Life Extension Program

D. Romancuk; Pilatus Aircraft Ltd

The Pilatus PC-12 is a single engine turboprop aircraft certified in 1994 as a FAR Part 23 normal category business and multipurpose aircraft. The fatigue certification of the PC-12 was performed according to safe-life requirements. In 2007 a life extension program (LEP) was initiated due to fleet leaders approaching the initially certified life limit of 20’000 flying hours or 27’000 flights.
The first of two stages of the LEP included a moderate extension of the safe-life limit based on existing design data. The second stage of the LEP used damage tolerance requirements. A supplemental structural inspection document (SSID) was developed. The intention of the SSID is to supplement the existing maintenance program, to cover fatigue and corrosion aspects beyond the existing life limit. The SSID defines non-destructive test (NDT) inspection intervals, methods and procedures. It has a general limit of validity (LOV) of 50’000 flying hours or 60’000 flights and, for some components, specific LOV’s that are lower than the general LOV. The SSID will support continued safe operation up to the LOV. As guidance material for certification EASA AMC 20-20 was used.
The major elements for the development of the SSID were the review of fleet experience, the review of fleet usage, the NDT inspection definitions, crack growth analyses based on tear down inspection results of the full scale fatigue test (FSFT) article, the development of a supplemental corrosion prevention and control program (CPCP), the development of a damage tolerance based repair concept, the review of non-structural systems with respect to aging problems, and the review of modifications (factory options, design changes, service bulletins, repairs, STC’s).
EASA certification was achieved in 2010. Third country validation (FAA and TCCA) was completed at the beginning of 2011. The challenges encountered during the program included the substantiation with respect to the change of the certification basis from safe-life to damage tolerance and the concurrent implementation in the field.

4.3 PC-21 Advanced Damage Tolerance Analysis

D. Quaranta; Pilatus Aircraft Ltd

During the development of the PC-21 a Full Scale Fatigue Test (FSFT) was performed. During the FSFT "fatigue critical" locations have been identified and, in some cases, the test evidence drove structural improvement (redesign) activities. The redesigned structures, not tested, have been assessed analytically after the FSFT and an inspection regime was defined. The analyses were basically done with the built-in crack growth models of AFGROW. Where no models were applicable, compounding techniques have been employed. In those locations where the structure being assessed shows a high level of complexity, in terms of both geometry and internal loads, the usage of AFGROW could be done accepting a high level of conservatism, eventually resulting in tight inspection regimes. In 2010 Pilatus reassessed selected PC-21 fatigue critical locations, aiming to relax the inspection regime by means of improved analysis techniques, with reduced conservatism. As a result, some inspection thresholds have been relaxed up to a factor of three. Two analysis improvements have been introduced:

- Stress Intensity Factors (SIFs) calculation
- Improved method for spectrum processing in the Crack Growth (CG) Life calculation

Stress Intensity Factor calculation
The analysis programs used as platform for the new analyses are Franc3D and Franc2D/L, originally developed at the Cornell University, NY, USA. Through collaboration of Pilatus with Fracture Analysis Consultants, Inc. (FAC), these two programs have been successfully integrated with Nastran. Franc3D and Franc2D/L are based on FEM and employ the following method for calculating the Stress Intensity Factors (SIFs):

- Use of quadratic elements with quarter point nodes at the crack front (related element shape functions and mapping between parametric and Cartesian spaces allow capture of the crack induced singularity)
- Use of J-Integral (Franc2D/L) and generalized M-Integral (Franc3D) for calculating mode I, mode II and mode III SIFs (KI, KII, KIII)

The entire analysis process consists in the following iterative, automatic, process:

- Feeding Franc3D (or Franc2D/L) with the Nastran model of the part under assessment (usually a sub-model integrated in the global FEM)
- Introducing crack(s) in the models (the mesh is automatically updated by Franc3D (or Franc2D/L)
- Running the analysis of the global FEM with the "cracked" model (Nastran solver used)
- Calculating SIF and consequent crack extension and trajectory
- Updating the FEM (re-mesh) with the new crack configuration
- Running the analysis

At the end of the process a SIF history of crack fronts is obtained, which is one of the principal inputs for the sub-sequent Crack Growth Life curve calculation. See Figure 1 for an example analysis using Franc2D/L and Figure 2 for an example analysis using Franc3D.

Spectrum
In general the airframe time history of events is composed by a sequence of load conditions which induce, instant by instant, a distinct stress field. The calculation of the CG Life is performed by invoking the "multiple time history" procedure, which basically consists in the following algorithm:

- Definition of the basic load conditions (i.e. manoeuvres), which are the building blocks for the time history of events
- Extraction of partial load time histories: A generic load condition is given by the combination of applied loads. For every load "channel" a time history is extracted.

- Calculation of the SIFs (in the “cracked” model) related to the partial load conditions.
- In the Linear Elastic Fracture mechanics (LEFM) the superposition of effects is applicable. For every “instant” of the global time history of events the total SIFs are calculated.
- The material NASGRO model is integrated instant-by-instant taking into account the “mixed mode” solution (effective K given as a combination of KI, KII and KIII).

The algorithm described above has been devised by Dr. Paul “Wash” Wawrzynek (FAC) and Domenico “Nico” Quaranta (Pilatus).

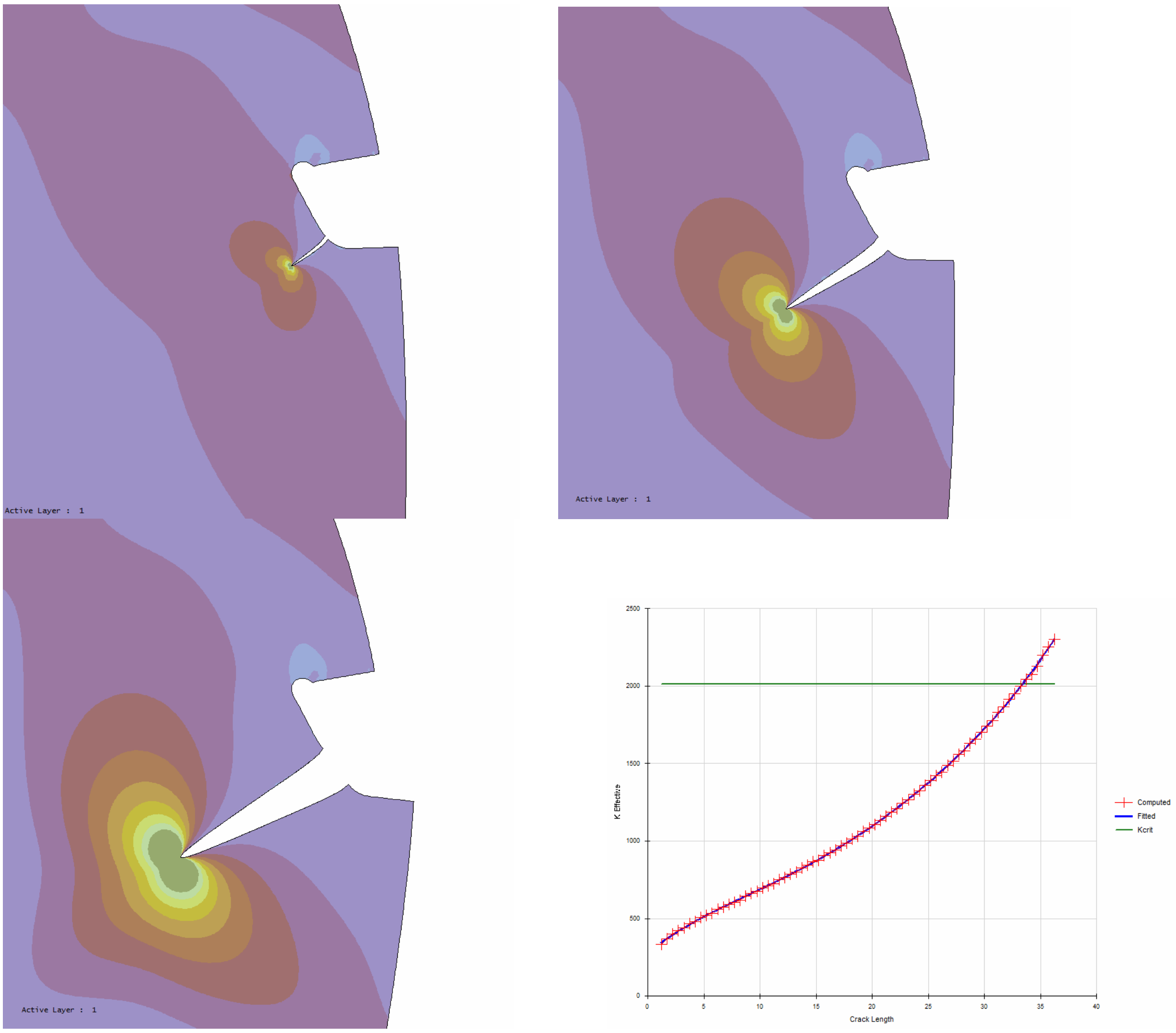


Figure: Extract from a frame stringer cut-out crack propagation analysis with Franc2D/L

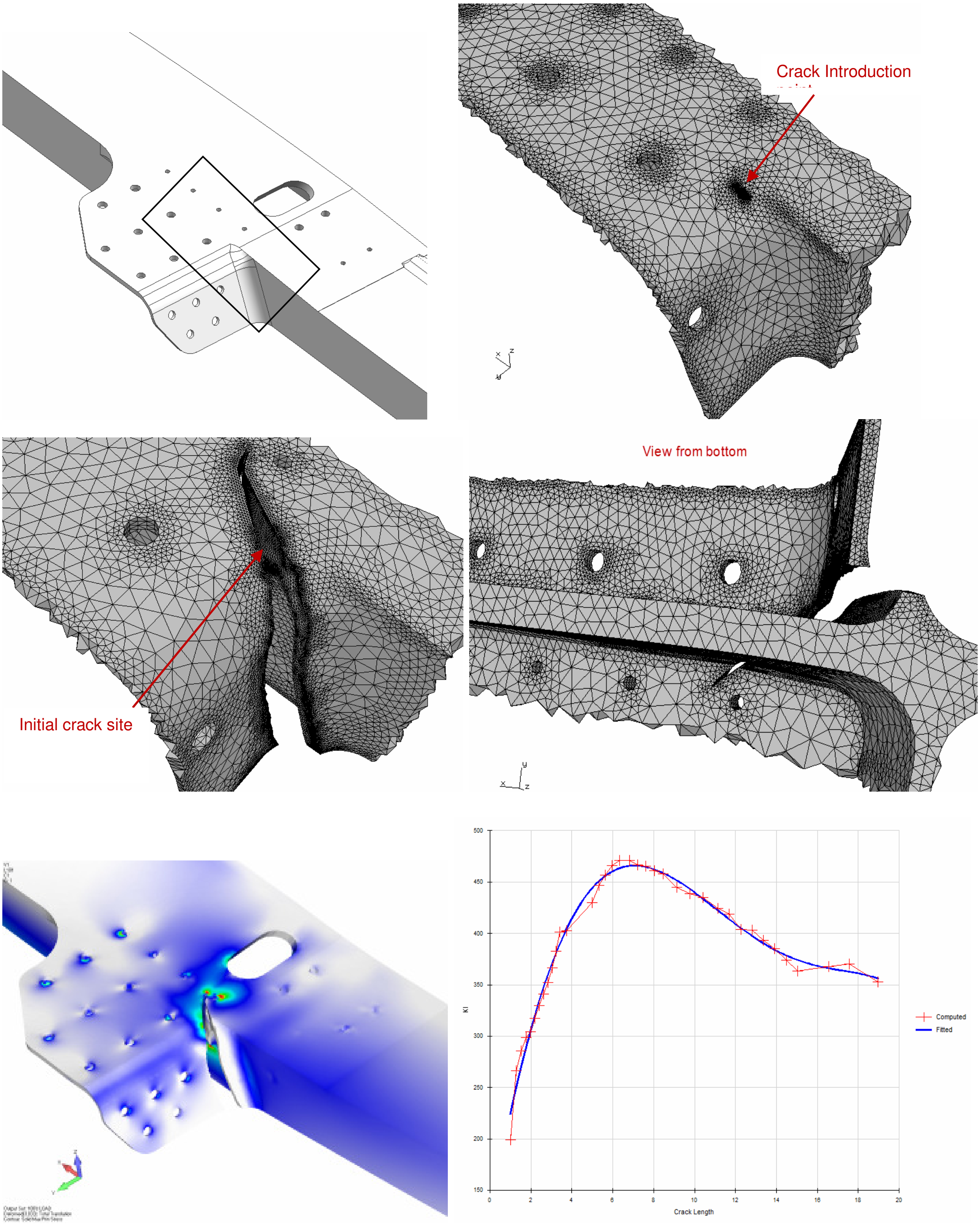


Figure: Extract from a fuselage longeron detail crack propagation analysis with Franc3D

4.4 RUAG Computational Fluid Dynamic Loads Activities

M. Guillaume, A. Gehri, P. Stephani; RUAG Switzerland Ltd, J. Vos; CFS Engineering SA, G. Mandanis; M@M GmbH

Introduction

Only few relevant fatigue load cases for the entire F/A-18 airplane are obtained from The Boeing Company in St. Louis, the F/A-18 Original Equipment Manufacturer (OEM).
This situation pushed RUAG to look for methods to generate aerodynamic loads for the F/A-18 and as a result a large investment was made in the development and implementation of steady and unsteady Computational Fluid Dynamics (CFD) calculations. To take the structural response into account a Fluid Structure Interaction (FSI) tool is developed. This effort provides RUAG with the ability to predict component loads to be applied on the structure for steady state and buffeting fatigue analysis. In addition, it is a tool which permits to better understand the complicated flow field over the entire F/A-18 full flight envelope and to check some of the load cases delivered by the OEM.
The calculations of the F/A-18 flow field are carried out using the NSMB Structured Multi Block Navier Stokes Solver. NSMB was developed from 1992 until 2003 in a consortium composed of four universities, and industry partners as SAAB, Airbus France and RUAG with CFS Engineering.

The F/A-18 Mesh Generation

The most time consuming process in a CFD simulation is the generation of the grid. This involves different steps. First (if required) the CAD surface needs to be cleaned up, then a multi block topology needs to be set-up, and finally the mesh is generated. The latest mesh for the F/A-18 fighter is generated by RUAG Center of Aerodynamics in collaboration with Mindware, using ANSYS ICEM CFD software. The half model mesh has 3377 blocks and 14.5 million cells see figure below.

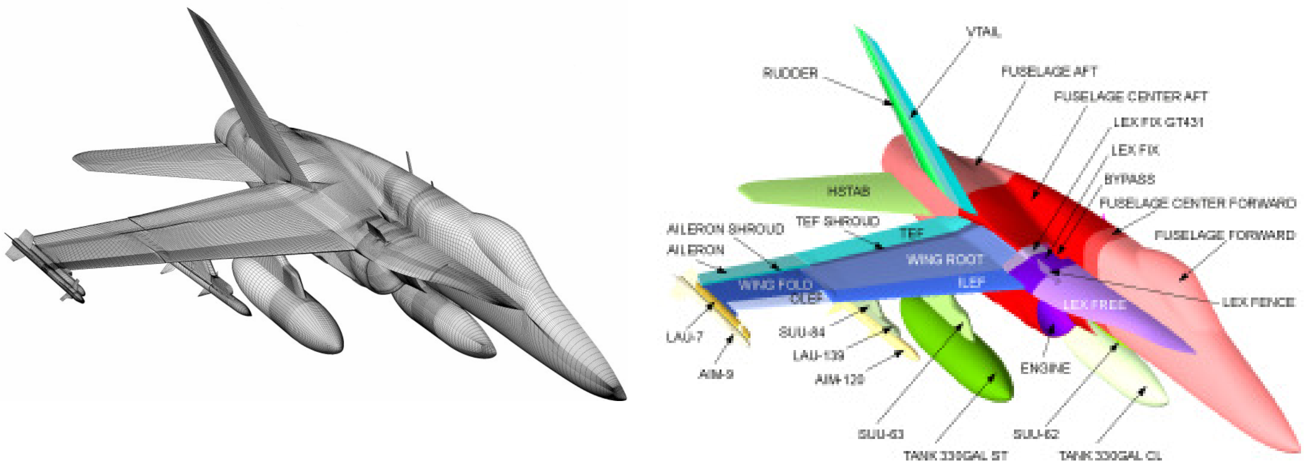


Figure: Detailed of the F/A-18 mesh (half model) and component loads areas.

The aerodynamic Loads Extraction Method and Loads Calculation

To permit the calculation of the aerodynamic loads on different aircraft a post processing program is developed that computes the aerodynamic loads on each aircraft component.
In all calculations discussed here it is assumed that the aircraft is perfectly symmetrical and only symmetrical load conditions are considered. Consequently only one half of the aircraft is used in the calculations.

The component loads in the F/A-18 US Navy F4 flight data base are based on reliable measurements and are used to validate the CFD calculation. 15 load cases are simulated with the half mesh model, taking into account the real control surface positions. Also 12 Swiss load cases from the ASIP study are calculated for comparison.

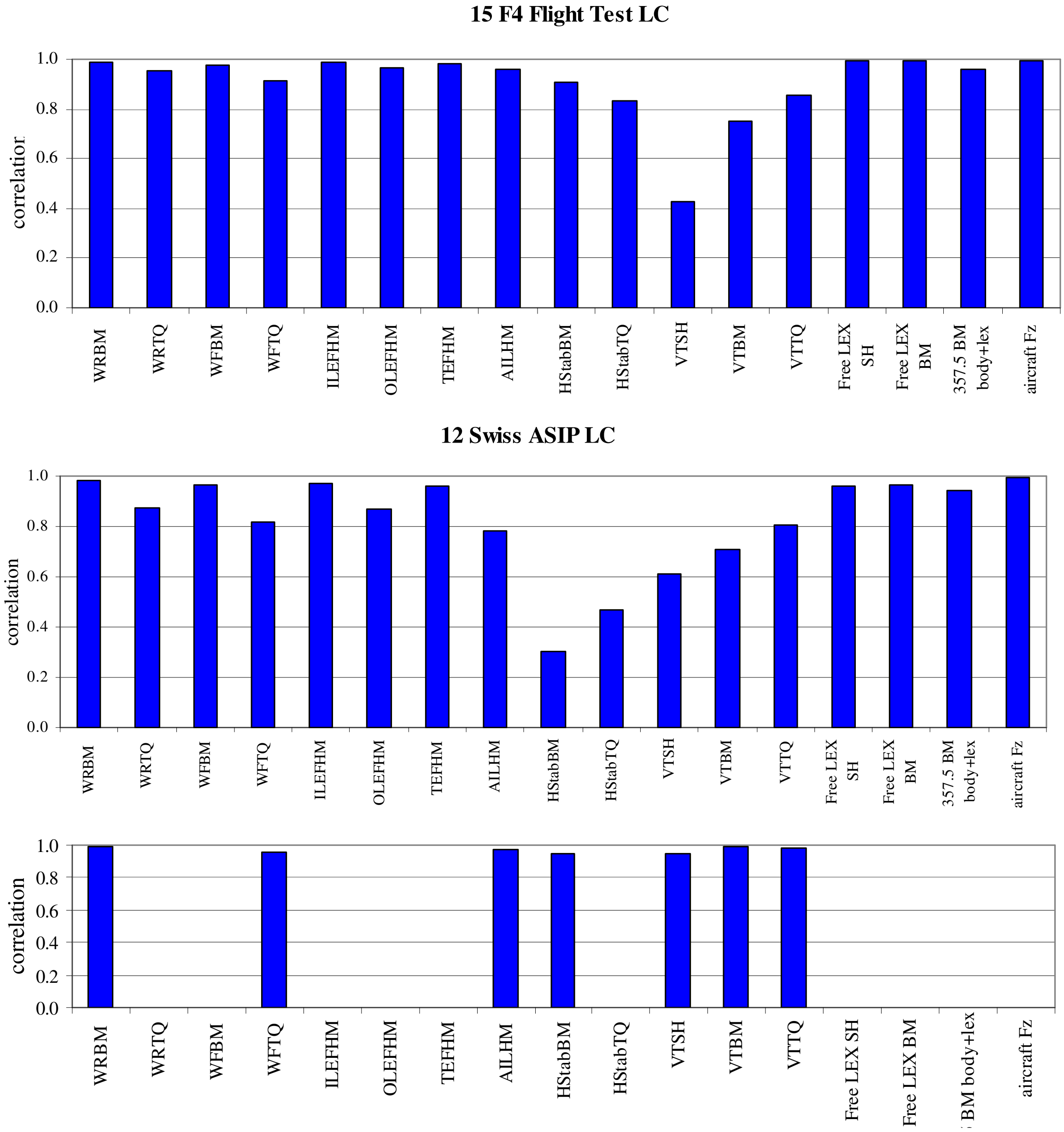


Figure: Comparison of CFD results with flight loads data and Swiss ASIP load cases

The CFD results for the F4 load cases (see figure above) match very well compared to the flight test data. However for the ASIP load cases the CFD data shows less good agreement compared to the Boeing loads. This could be explained by the fact that the Swiss ASIP loads were inter- and extra-polated from the F4 database and therefore these loads do not represent real flight data.

If only loads with an AoA below 10° are considered the results match much better especially for the vertical tail compared to the F4 flight database. The vertical tail and the horizontal tail are influenced by unsteady loads which are not correctly treated by steady RANS calculation. In summary we can say that the NSMB solver predicts the F/A-18 US Navy flight data loads quite well using the RANS approach.

The Swiss load case C1S825 corresponds to an 8.25 g steady state manoeuvre at an angle of attack AoA = 15.9°. At this condition the wing tip deforms due to the high loads up to 0.5m , and one can expect that this change in wing shape will influence the flow over the wing, and thus on the aerodynamic loads. To investigate this effect an iterative CFD calculation on a flexible F/A-18 wing (with control surfaces) is made. Four iteration steps are needed to reach the equilibrium between aerodynamic and structural forces. During this simulation the fuselage, horizontal stabilizer, vertical tail and rudder are considered as rigid.

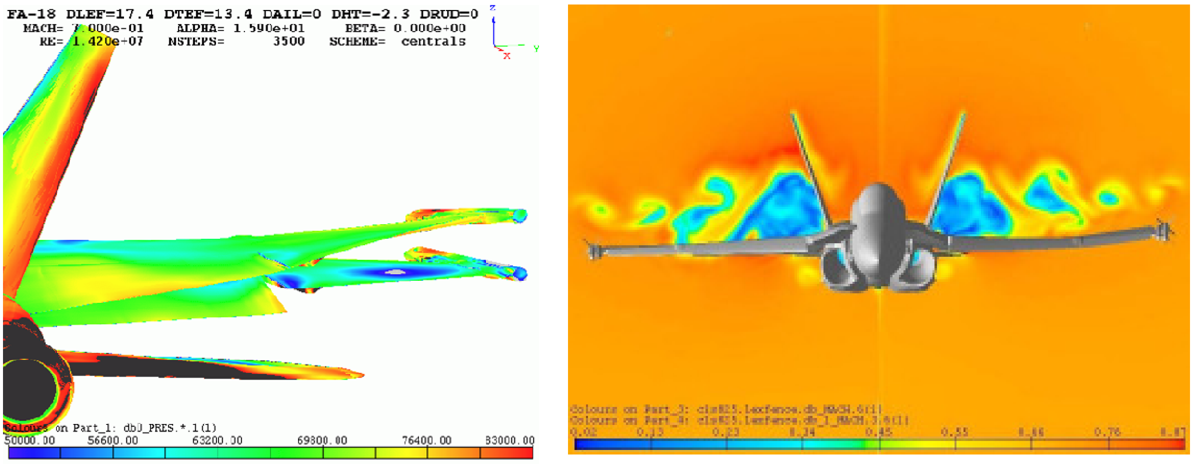


Figure: Deformed wing and comparison of mach contour

The deformed wing is shown in figure above. Note that the missile remains almost parallel to itself; hence we are facing a more or less pure bending deformation mode. It can also be seen that the size of the gap between TEF and aileron is reduced by the deformation of the wing. The differences of the flow field due to the deformation of the wing is clearly visible in the Mach contour plot, see figure 3.
The first iteration produces a large deflection. The second and subsequent iterations bring only small corrections, and the third and fourth iterations are almost identical. The wing bending and the torque moment is calculated for the undeformed (no FSI) and the deformed (FSI) wing shape. The wing bending moment along the span for the flexible wing is reduced due to the redistribution of loads from the outer wing to the inner wing. As a consequence the torque moment is increased along the elastic axis which is explained by the change of twist angle (downward) under loading.

Unsteady State Calculations for the F/A-18 Vertical Tail

Unsteady simulations are made for the F/A-18 fighter using different aircraft configurations and free stream conditions. All calculations are carried out using the dual time stepping method, in which the outer loop is used for the advancement in time, while in the inner loop a pseudo time stepping method (time marching) is used to solve the equations at each outer time step. The Detached Eddy Simulation (DES) variant of the k-ω Menter Shear Stress turbulence model is used to model the unsteady turbulence. All unsteady calculations are started from a converged steady state calculation for the same free stream conditions. An outer time step of 2.5 10^{-4} seconds is used, and the pressure and deformation vector were saved each outer time step. For the aero-elastic calculations the fluid structure coupling is executed each inner time step corresponding to a square number (inner step 1, 2, 4, 9,…). Initial tests show that this is sufficient and computationally less expensive compared to carrying out the coupling each inner time step.
A simple structural model of the Vertical Tail used for the aero-elastic calculations was prepared by RUAG.
For the F/A-18 vertical tail only the first 4 or 5 mode shapes are important (bending and torsion moments), and it was decided to use only the first 5 modes in the coupled calculations.

Several F/A-18 conditions/load cases are computed, especially Swiss load cases, and a subsonic case for which experimental data were reported in the literature.
A Swiss buffeting load case corresponds to an 8.25 g manoeuvre at Mach=0.7. The calculation without FSI coupling shows significant differences in the averaged unsteady aerodynamic loads compared to the loads obtained using a steady calculation, in particular on aft fuselage, vertical tail, rudder, trailing edge flap, aileron and horizontal stabilizer. This is a clear indication that unsteady flow effects are important on these components of the aircraft.
A first coupled simulation is made for this load case running up to 0.5 seconds real time. As for the transonic load case 5 points are selected to show the movement of the vertical tail, see figure next page. Compared to the transonic load case the movement of the vertical tail is much more prominent, up to 13 cm in Y-direction compared to a little more than 2 cm.

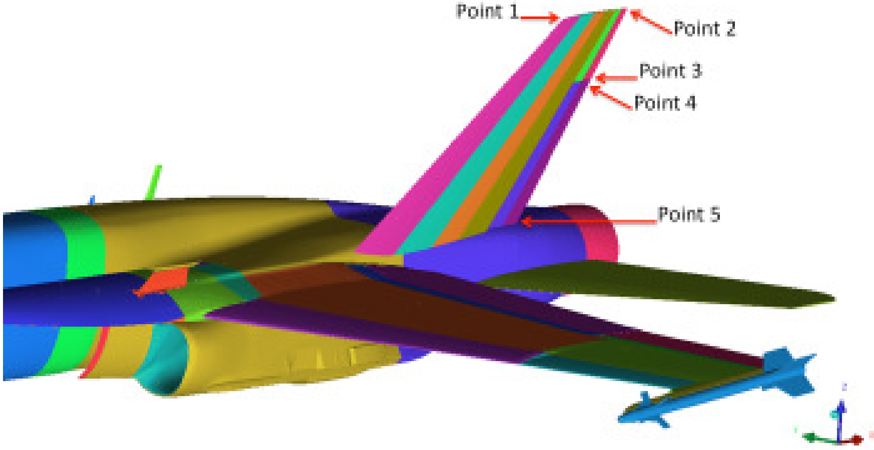


Figure: Five selected points which were used for dynamic data analysis.

To compare data transformed from the time domain to the frequency domain the simulation of 0.5 seconds is not enough. Therefore a second coupled calculation is made running up to 4 seconds real time, which is later used for fatigue spectrum data.

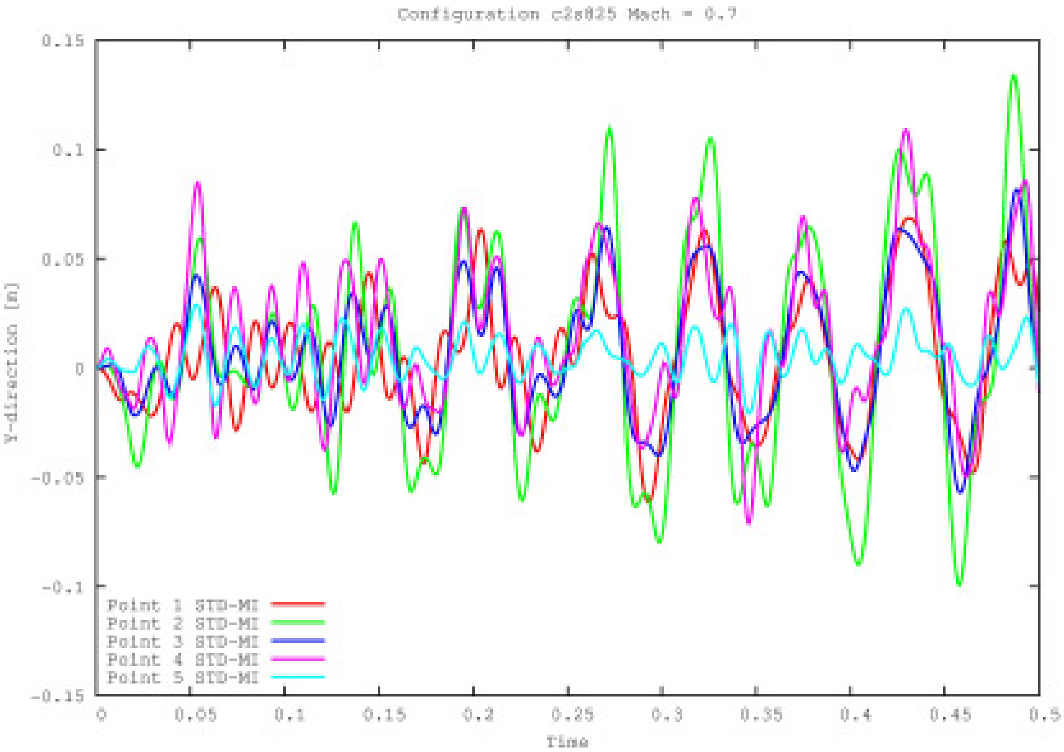


Figure: Movement in Y direction for the 5 selected points for 0.5 sec.

L. A. Meyn published in 1996 results of buffeting measurements on a Full Scale F/A-18 in the wind tunnel at NASA Ames Research Center Ref. [1]. The test conditions were Mach=0.15, SL at different angles of attack, the following data was analyzed, see table below.

Mach	Alt [feet]	AoA [deg]	LEF [deg]	TEF [deg]	HSTAB [deg]
0.15	0	var	0.0	0.0	0.0

Table: Specification of the subsonic load case.

Interesting results can be found in the Ref. [2], where the RMS-Pressure Coefficient (1) on the 45% chord, 60% span of the vertical tail from CFD-computations, wind tunnel test and flight tests are published. The Figure below shows the result of our CFD calculation.

$$\text{RMS-Pressure Coefficient} = \frac{1}{Q}\sqrt{\frac{1}{N}\sum_{1}^{N}(p_i - \bar{p})^2}$$

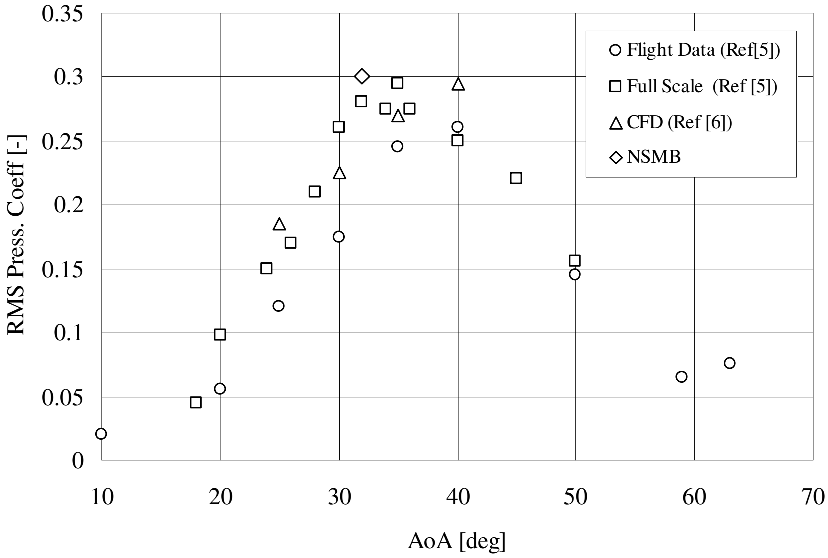


Figure: RMS pressure coefficient at 45% chord and 60% span of the vertical tail.

Simple Fatigue Analysis for Vertical Tail Buffeting

To determine the buffeting loads at the vertical tail root at FS557 stub location an engineering approach is used. In order to quantitatively take profit of the results of the CFD-FSI-calculation on the F/A-18 the vertical tail surface is divided into 54 panels and the pressure and the motion at each panel centroid is extracted.

The differential pressure p_i over the 54 panels as well as their displacement in time gives us the ability to determine the dynamic loading of the connection between the vertical tail and the fuselage, see formula below:

$$\vec{M}(t) = (BM(t), PM(t), TQ(t)) = \sum_{1}^{54} \vec{r}_i x (A_i \vec{p}_i(t) - m_i \vec{a}_i(t))$$

With BM, PM and TQ the bending, pitch and torsion moment at the root of the vertical tail, A_i the panel surface, p_i panel pressure, m_i lumped mass and a_i the acceleration. The most interesting aspect for RUAG is the fatigue damage caused by the vertical tail buffeting. In order to check the plausibility and the feasibility of a fatigue calculation based on the F/A-18 CFD-FSI, the peak-valley sequence in a structural location of the stub 557.5, which is the forward connection beam vertical tail to fuselage, has been calculated.

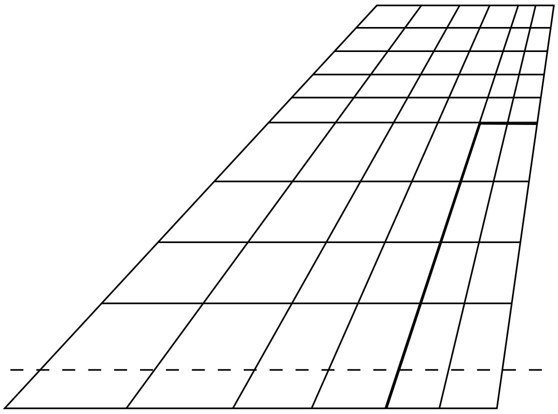


Figure: Vertical tail with 54 panels used for analysis.

The strain survey measurements accomplished during the Swiss F/A-18 Full Scale Fatigue test delivered us the coefficient A and B of equation below and on this way the relation between the strain at gauge position SF6001 (see figure below) and the moments BM and TQ.

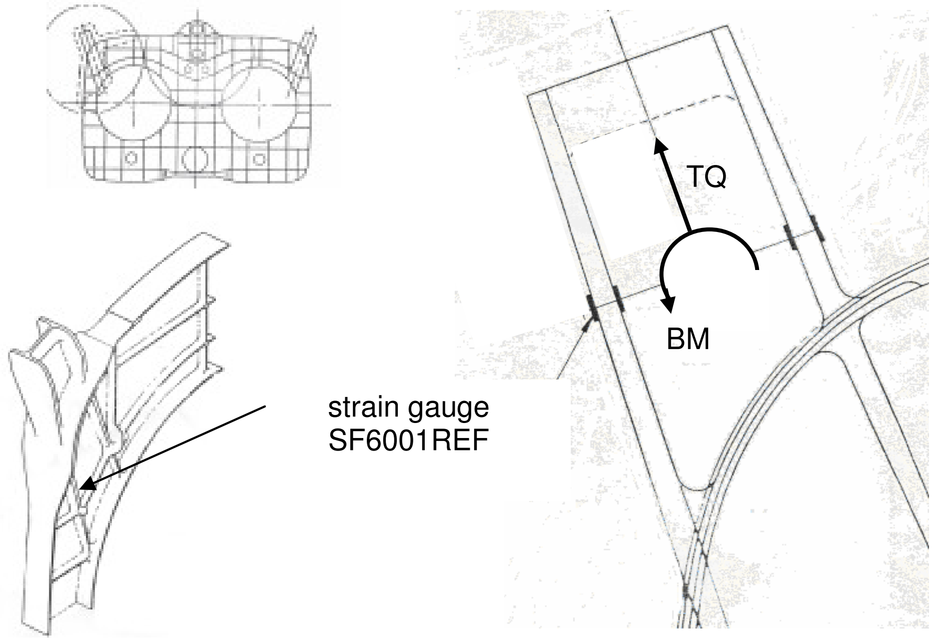


Figure: Gauge position at the vertical tail stub at FS557 (first stub out of 6).

A part of the ε_c(t)-function for the buffeting peak valley sampling can bee seen in the next figure.

$$\varepsilon_c(t) = A \cdot BM(t) + B \cdot TQ(t)$$

In order to evaluate the buffeting effect on the global fatigue damage this peak-valley sequence has been inserted in the test spectrum called MES used for the Full Scale Fatigue Test, considering that buffeting occurs when the angle of attack AoA exceeds a limit value of 20°. Such a mixed spectrum of maneuver and buffeting loads is presented on the next page (second Figure).

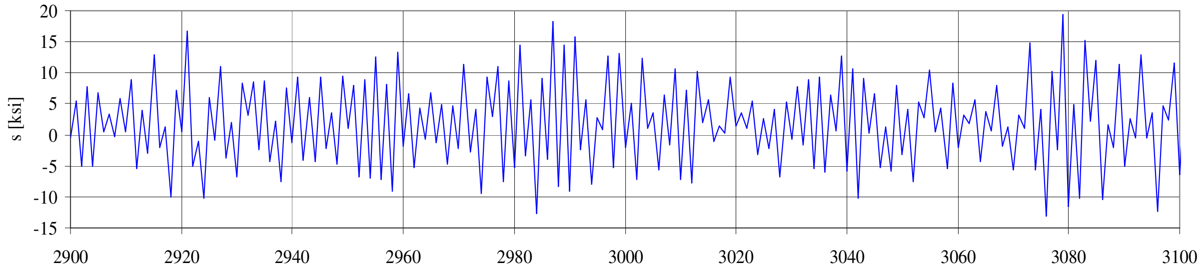


Figure: A sample of the $\varepsilon_c(t)$-function for the buffeting peak valley sampling for 2.5 sec.

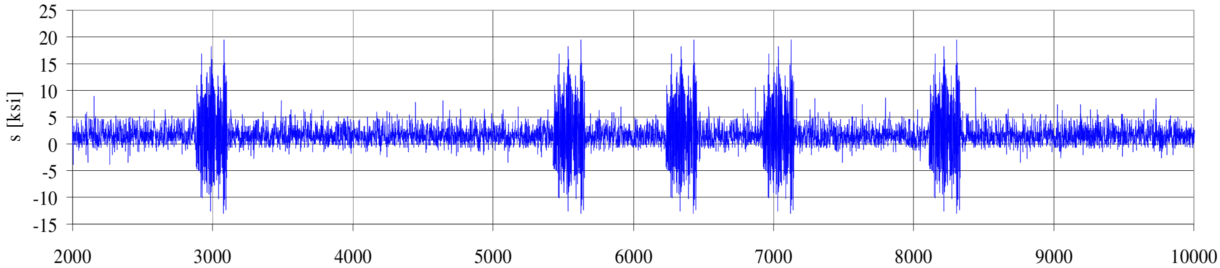


Figure: Mixed Spectrum with AoA limit for buffeting above 20°.

This set of spectrum has been used for the crack initiation life calculation (Boeing CI89 software) in order to evaluate the effect of the buffeting on the stub at F.S.557.
To produce realistic conditions the following parameters have been chosen for this calculation:

Neuber Notch Approach
Material: AL7050-T74
Flight Hours Number for Spectrum 200
No Prestain Material Data
Equiv. Strain Equation Smith-Watson-Topper

The Fatigue life curves for the onset of buffeting above AoA of 20° (green line) is compared with the no buffeting data (blue line / direct Swiss Full Scale Fatigue Test data). The fatigue life for buffeting cycles at the stress level of KtDLS 60 Ksi can be reduced by a factor of 30. The red line represents the fatigue curve due to the buffeting sequence only without manoeuvre loads, which occurs every 20 hours or 10 time in the Swiss design spectrum (sequence of 200 flight hours).

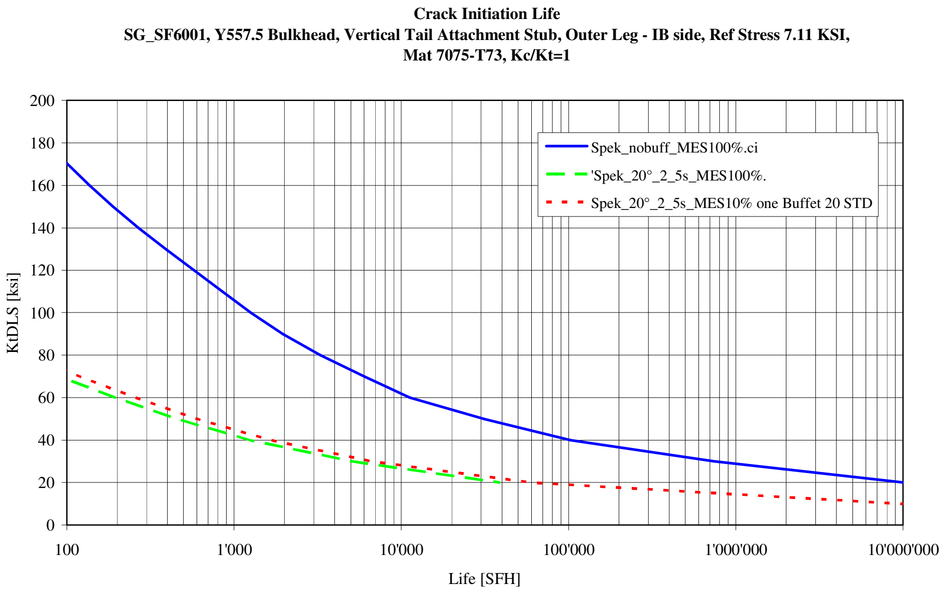
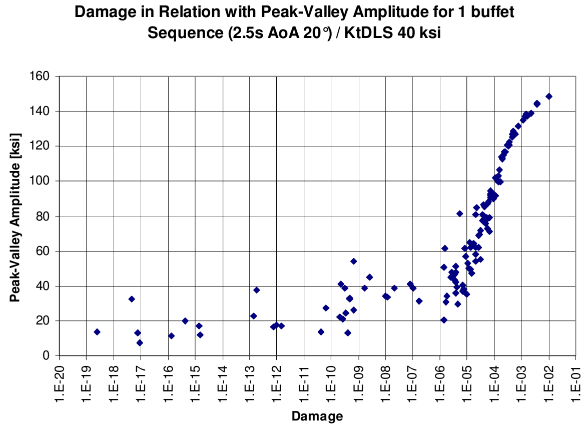


Figure: Fatigue life curves for MES and MES plus buffeting

In a more detailed study the fatigue damage rate of the buffeting cycles is more carefully analyzed (see figures below). The fatigue damage is based on the cumulative Miner rule using strain life material data. The investigation is done for the stress level KtDLS of 40 Ksi and 60 Ksi. The fatigue damage is mainly caused by the buffeting cycles only. Significant damage accumulation occurs at a peak/valley amplitude higher then 40 Ksi which belongs purely to buffeting loads. So the fatigue damage is primarily caused by the buffeting cycles.



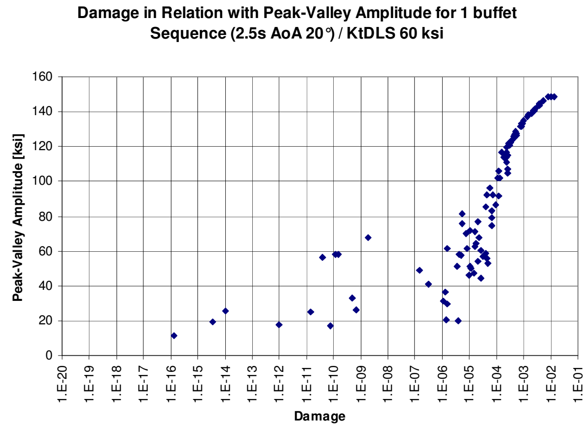


Figure: Damages of the buffeting sequence

Miner Damage sum of 40 Ksi for maneuver spectrum (200 SFH) = 2/1000
Miner Damage sum of 60 Ksi for maneuver spectrum (200 SFH) = 17/1000

Miner Damage sum of 40 Ksi for buffeting cycles (200 SFH) = 120/1000
Miner Damage sum of 60 Ksi for buffeting cycles (200 SFH) = 780/1000

The Miner damage sum for the buffeting cycles at 40 Ksi and 60 Ksi is at least an order of magnitude higher as for the maneuver cycles. This clearly demonstrates the severity of fatigue life due to buffeting.

The above simple method will allow a dynamic analysis using buffeting loads for the Swiss usage spectrum. With the help of FSI/MI unsteady coupling tool buffeting load cycles can be calculated to generate a full dynamic spectrum for fatigue analysis.

4.5 Collaboration with RUAG/CFS/Finflo for Computational Fluid Dynamics

M. Guillaume, A. Gehri, P. Stephani; RUAG Switzerland Ltd, J. Vos; CFS Engineering SA

Introduction

In 2005 we set up a collaboration with RUAG Aerodynamics Center in Emmen, CFS Engineering in Lausanne, and FinFlo in Helsinki. The goal was to improve the Navier Stokes Solvers used in Switzerland (NSMB) and Finland (Finflo) for doing Computational Fluid Dynamics calculations on the F/A-18. In the long term the research should offer an alternative to expensive flight tests for performance and load measurements.

The first step was to harmonize the data formats and models for doing first calculations.

The meshes were exchanged and calculations on each solver with the different meshes were done. The results were compared with pressure plots, friction line plots over the entire airplane. Several differences were observed which lead to improvements on the mesh and the calculation strategy.

In the Swiss mesh the gaps between slats and flaps was closed were the Fin's modelled the gaps. On the figure below the differences on the pressure contours is visible due to different meshes and solvers. After checks on the airplane we decided to close the gap at the slats (inboard and outboard leading edge) and do a study for the shroud and flap area. Both meshes will be improved and small deviancies will be corrected and once more harmonized to do better comparisons.

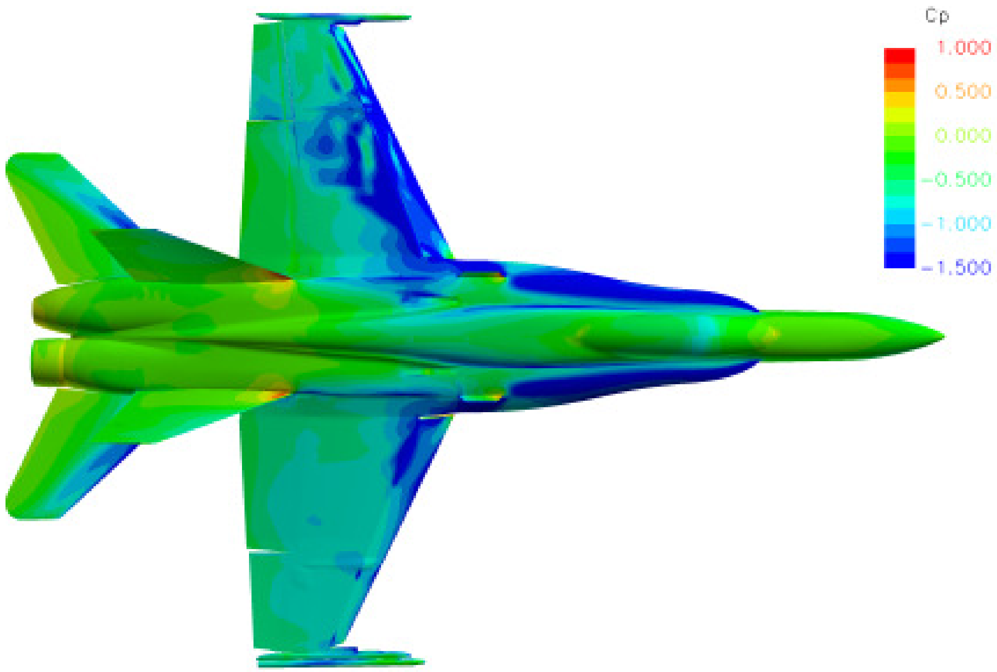


Figure with pressure contour, top Finflo solver and mesh, bottom Swiss solver and mesh

In October 2009 we decided to calculate aerodynamic loads at the reference locations used in the Swiss ASIP study. The list of reference locations was extended based on the results gained during the Swiss F/A-18 Full Scale Fatigue Test, see figure on the top of the next page.

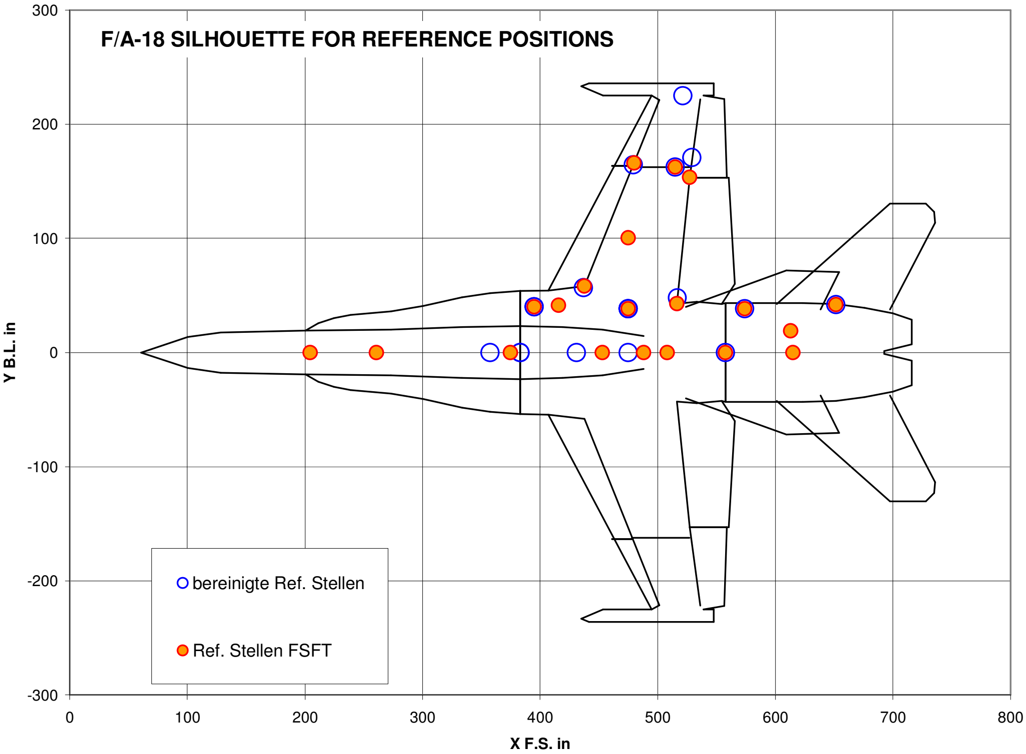


Figure of reference locations used for component loads calculations

A first review of the calculated component loads showed surprisingly good results. Especially at wing root and wing fold the shear, bending moment and torque match very well. More differences were observed in the fuselage which could be improved after clarifying the integration of loads from the origin of the coordinate system.
The program for the coming two years is to study also unsteady flows at the outer wing area for future supports of the fleets with advanced and reliable method. Therefore also finish MINIHOLM flight test data will be used for comparison.

The Swiss already processed first unsteady calculations for high angle of attack with detached flow using DES simulation approach, see figure on the next side.

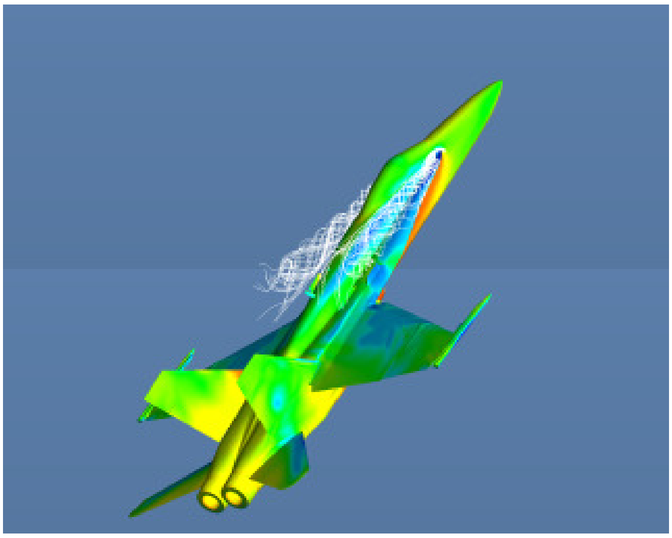


Figure advanced unsteady results over the whole airplane

4.6 F/A-18 Structural Integrity: ILEF OLM

D. Schmid, I. Kongshavn, S. Büsser; RUAG Switzerland Ltd

Introduction

During the Swiss F/A-18 Full Scale Fatigue Test (FSFT), cracks were found in the left and right hand Inner Leading Edge Flap (ILEF) spar after only 3000 Design Spectrum Flight Hours. Although the FSFT was representative enough for this location to ascertain that a problem will most probably develop in the fleet, it was not representative enough to use the test results directly to predict when the damage will occur in service. Two main uncertainties remained after the test, namely whether the design spectrum was actually representative of the usage spectrum at this location and the influence of the ILEF deflection angle on the loads, as the ILEF was fixed at 0° during the whole test.
It was decided to perform an operational loads measurement to remove these factors. The details of the calibration, instrumentation and flight test phase were reported in the 2007 Swiss ICAF Review. In this article, the focus will be on the data evaluation part of the program.

Overview

As the data acquisition flights needed to be performed on a certain aircraft which has already data acquisition equipment installed and as this aircraft was only available for a limited amount of flights it was unfortunately not possible to perform a long time flight program such that the data could directly be used to create a usage spectrum. It was therefore decided to perform a certain amount of flights with specific mission profiles such that the data could be used to train a neural network which takes flight parameters such as altitude, mach number, nz and AoA as input and gives the ILEF load data as output. As these flight parameters are recorded for each aircraft of the fleet in normal operation, it would then be possible to create ILEF load data for operationally used aircraft. To avoid having to calculate the data for each aircraft individually, it was decided to select an “average” aircraft whose flight data during a certain period would be used to create a usage spectrum block for the whole fleet.
Several road blocks had to be overcome during the project. First of all, a method to calculate the stress at the critical location from the recorded strain data needed to be found. In the beginning it was planned to use the data from the ground calibration test together with FE results to find a function which maps the strain gage data to ILEF load data such as hinge moment, out of plane lug bending moment and hinge loads. The neural network would then be trained to output the ILEF loads. They would then be used on a detailed FE model to calculate the stresses at the critical location for each flight phase.
The outcome was that the data did not show a good correlation between measured strains and lug loads. This approach was therefore abandoned and it was decided to use the detailed FE model to find a direct transfer function between the strains and the stresses at the critical location.

Data Preparation

The measured data was cleaned of all data points recorded while on the ground and those produced by malfunctioning sensors. The flight parameter data was then synchronized with the recorded strain data.

Stresses at critical location from Strain Data

As described in the 2007 Swiss ICAF Review, several calibration tests were performed on the instrumented ILEF both while on and off aircraft. These calibration tests were simulated in the FE model as well. Using both the simulated and measured data it was possible to calculate a strain to stress transfer matrix for the critical location.

$$[\sigma] = [\varepsilon] \times [T]$$

To simplify the calculations, the highest loaded node of the FE model was used as the target node.
The end result was a timestamped sequence of stresses at the most critical location.

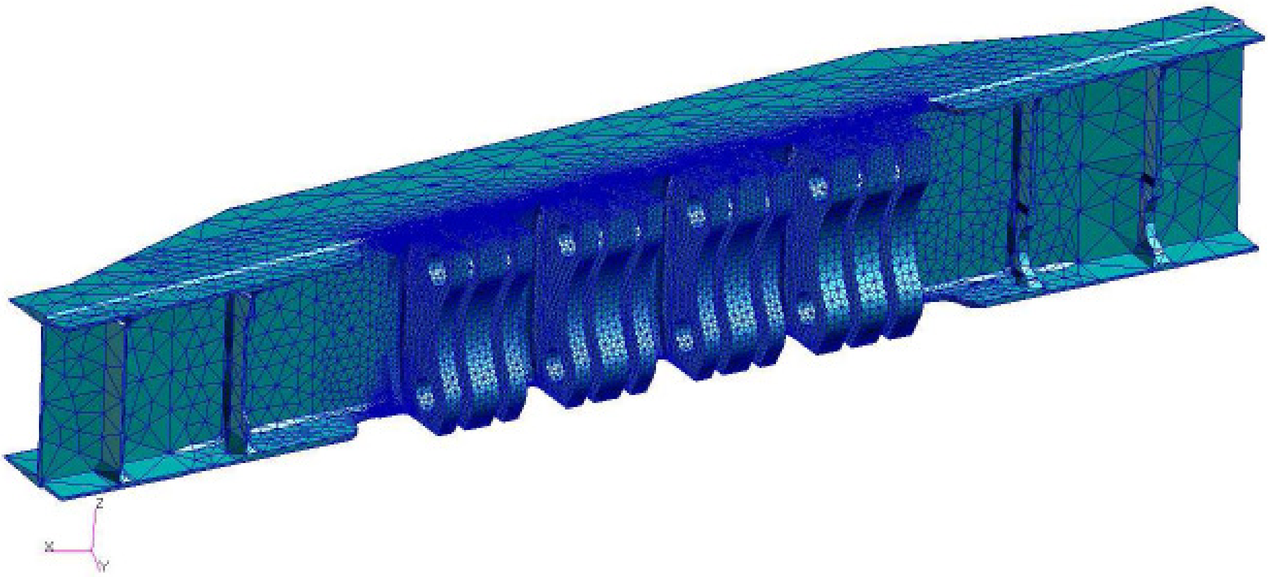


Figure of Detailed Spar FE Model

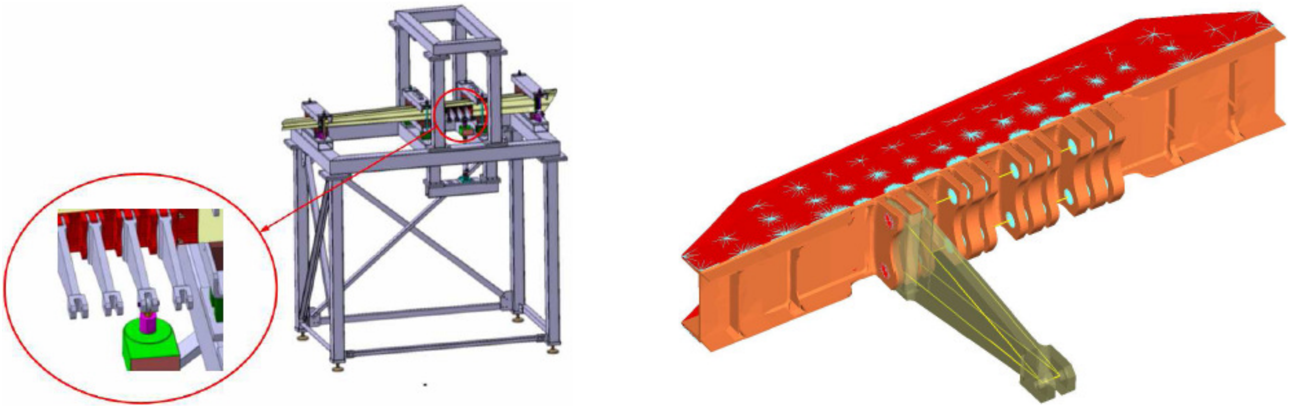


Figure of Calibration Rig and FE model of Calibration (with CAD overlay)

Training the Neural Network

Neural networks can be used for many different tasks such as sorting, decision making and system control. In this context, neural networks are used to approximate an unknown function, namely the mapping of flight states described by different parameters to the stress at the critical location. The advantage of neural networks is that they can, depending on the size of the network and with some limitations, describe any function without any prior knowledge about its structure. The only prerequisites are a sufficient numbers of data points as input and the corresponding function value as output, i.e. if

$$y = f(\vec{x})$$

Is an arbitrary function, then no knowledge about f is needed except a sufficiently large number of input / output pairs [x,y]. The neural network algorithm "trains" the network using a feedback loop which updates the internal parameters such that they produce the desired output.

The input parameters for the ILEF neural network are shown in the following table and the figure below shows a schematic of the neural network.

Parameter Name	Description
SIWA	Value depends on whether a missile is mounted on the wing tip or not
IEWTCL	Weight of fuel in centreline tank
dTemp	Ambient Temperature
Nz	Aircraft vertical axis acceleration
longStick	Longitudinal Stick position
latStick	Lateral Stick position
rollRate	Roll Rate
yawRate	Yaw Rate
pitchRate	Pitch Rate
Alt	Altitude above Sea Level
Mach	Mach Number
Cas	Calibrated Air Speed
TotWt	Total Weight
CGmac	Centre of Gravity position in relation to mean aerodynamic chord
Q	Dynamic Pressure
AoA	Angle of Attack

Input Parameters for Neural Network

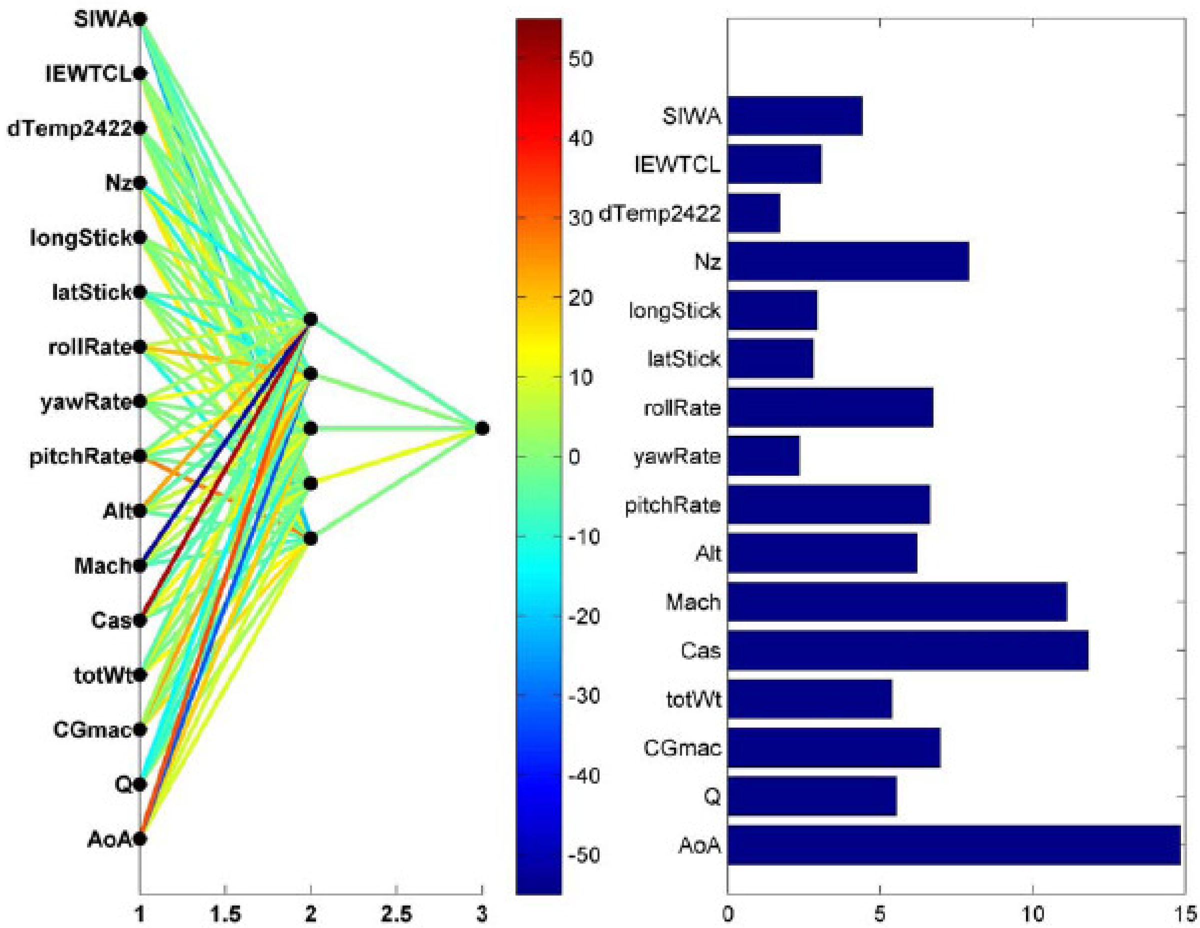


Figure of neural network schematic showing influence of each parameter on output

The Matlab neural network toolbox was used to create the network and train it. The bar graph in the figure shows that, as expected, Angle of Attack, speed and acceleration (Nz) have the largest influence on the result.
As the neural network is basically a black box, it is important to judge the quality of the output, essentially how well the network approximates the principles underlying the data. The figure below shows the regression plot. Additionally the R-value is calculated and gives a numeric indication of fitting quality. There is no absolute criterion which denotes a boundary between good fit and bad fit, it strongly depends on the application. As fatigue calculations are very sensitive towards stress levels, a very good fit (R-value > 0.95) is important.

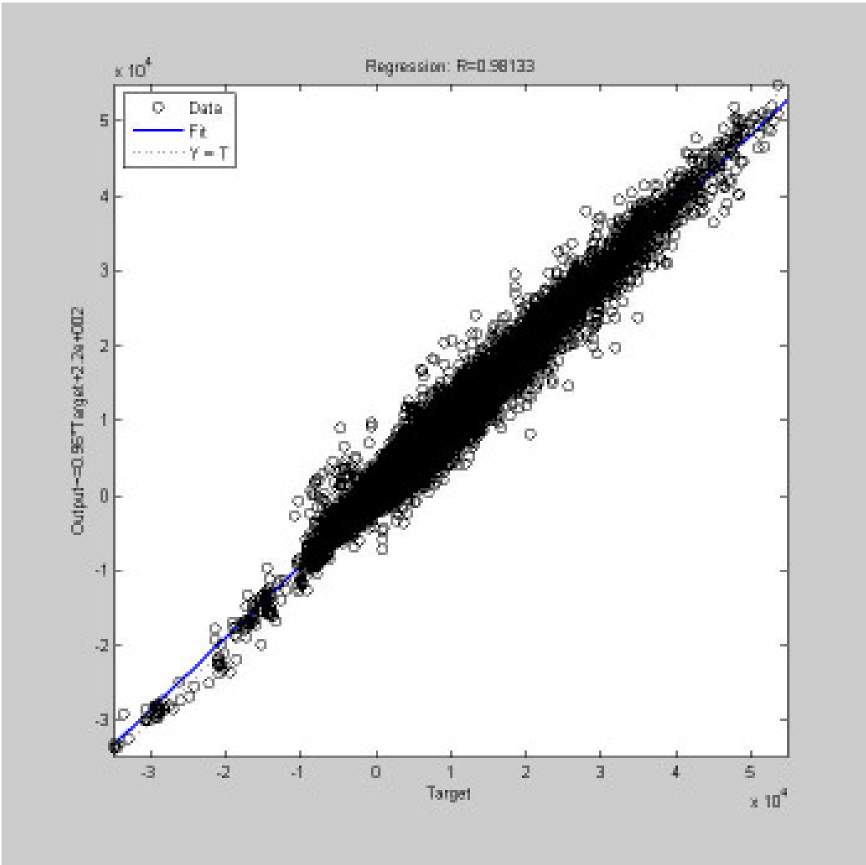


Figure of Regression plot of neural network results

Selecting a representative usage block
In the scope of the project it was not possible to calculate an individual stress spectrum for each aircraft over its total life. Instead a representative block of 200 FH of a single fleet aircraft was selected and used to create a spectrum block. The aircraft and flights were selected according to the following criteria:

- Source: Swiss F/A-18 C
- Configuration: No flights with external wing mounted stores
- Severity: Severe Usage, above fleet average judged on the basis of Wing Root Bending Moment FLE
- Data Quality: Good Condition of Fatigue Tracking sensors

On this basis, a total number of 214 flights in the years 2005 and 2006 were selected amounting to a total of 207.12 flight hours.

Spectrum Generation
The corresponding data of the 214 flights was input into the trained neural network and the output calculated. As reference load case, the manoeuvre generating the maximum simulated stress during the instrumented

flights was used. The manoeuvre is a steady state pull up with 7.5G at 0.9 Mach and 5000 ft. The corresponding stress was used to normalize the spectrum.
The spectrum was subsequently used in a crack initiation / growth calculation. The result showed that while problems can be expected during the service life of the aircraft, they are not as immediate as the test would have suggested.
A comparison of the measured flight test strain data with the measured FSFT strain data has also shown that the usage spectrum is less severe than the test spectrum which can be seen in the figure below. As the FSFT gage and the OLM gage were at the same location, both spectra were not referenced with the same load case but the same reference stress for an easier comparison.

Further Work
With the release of the fatigue life calculation report, the ILEF OLM project was successfully completed. Further possible actions based on the results are the calculation of a fleet ranking by calculating the spectrum individually for each aircraft over its service life and the discussion of preventative measures to be undertaken to insure fleet availability and flight safety in cost effective way.

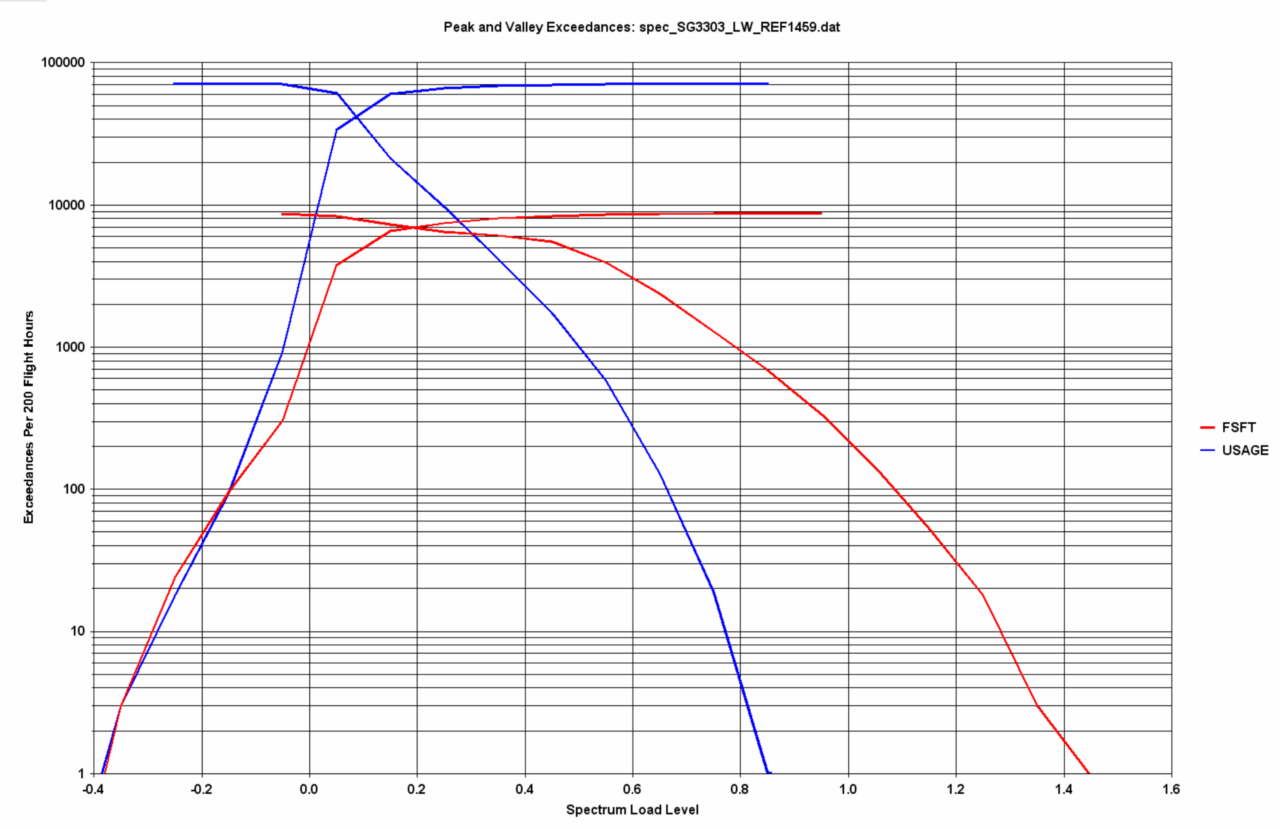


Figure of Spectrum Comparison between Usage and FSFT (Referenced with same strain value)

Acknowledgement
The work of this project is sponsored by armasuisse.

4.7 F-5 Wing Trunnion Fretting Fatigue Tests

A. Uebersax, C. Huber, M. Wyss, B. Schmid, A. Gassmann; RUAG Switzerland Ltd

Introduction

Fretting fatigue damages occur in an aluminum 7075-T73 wing rib of the wing-to-fuselage-joint, under the 17-4PH steel bushing's shoulder. A solution to avoid the fretting fatigue cracks in the highly loaded area is to be found. As a general redesign of the area is out of scope, improvements of the tribological conditions are investigated. To validate the behaviour of added anti-fretting compound, fretting fatigue tests are conducted.

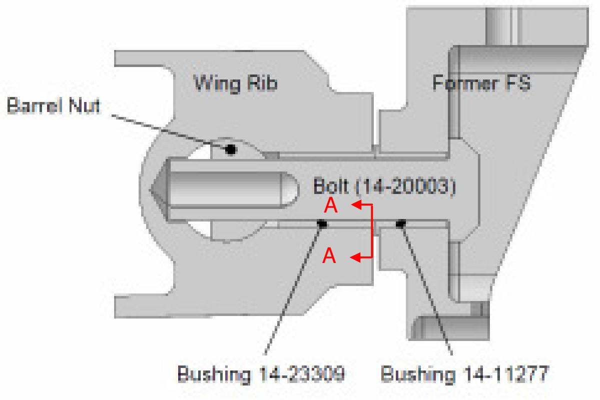


Figure 1 Structural situation: Pre-tensioned steel bolt transferring shear loads.

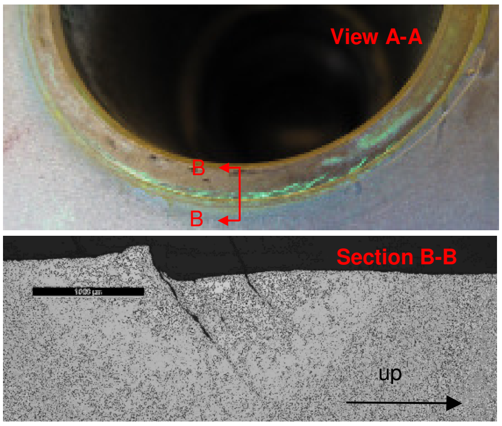
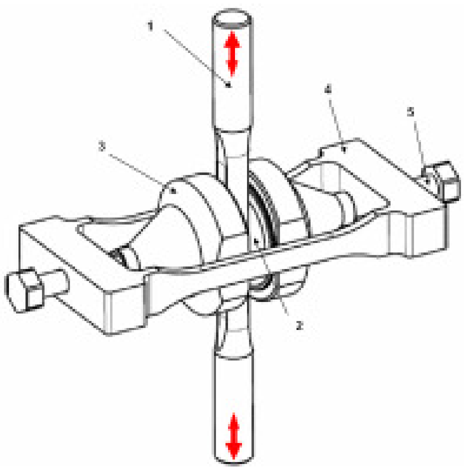


Figure 2 Penetrant crack indication after removal of the bushing (above) and sectioned cracks (below)

The goal of the fretting fatigue tests is to quantify the relative improvement in total life between baseline configuration and configuration with added anti-fretting compound. Most anti-fretting compounds are easier to apply on the handier steel bushing than on the aluminum rib integrated in the wing structure. Therefore the anti-fretting compound is applied on the bushing's shoulder.

Test Setup

As general standards for fretting fatigue tests are not available, the test layout was engineered by RUAG. The fretting coupon represents the wing rib and the fretting pad acts as the bushing's shoulder. The fretting pad was chosen to be of a bridge type and circular. It has exact the same radius and depth as the bushing's shoulder. Because of its circular shape, it benefits from a simple manufacturing that eases close tolerances and it can be used multiple times as it can be rotated. The normal force on the pads is produced by a normal force rig and tensioning screws. The force is transmitted to the pads through conical shaped pad rigs. The application of the normal force is controlled by two strain gages on the force rig, see Figure 5. Measurement of the relative slip amplitude between the test coupon and the fretting pad was not considered necessary for this test.



Caption of Figure 3:

1. Test coupon (Al 7075-T7351)
2. Fretting pads (Steel 17-4PH H1025)
3. Pad rig (Steel)
4. Normal force rig (Steel)
5. Tensioning screw (Steel)

Figure 3 Fretting fatigue test setup

Testing

Tests are performed on a single cylinder tension-compression test stand. In total, three baseline tests and four tests with the anti-fretting compound (thermally cured MoS_2 based solid film lubricant) are performed.

Every 600 [SFH] the circular fretting pad is rotated, so to expose a pristine steel surface coated with anti-fretting compound. This simulates the replacement of the steel bushing every 600 [SFH] on the aircraft wing.

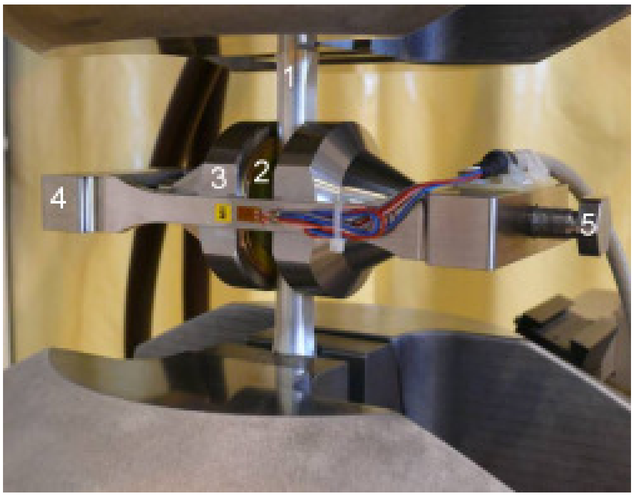


Figure 4 Testing on 250[kN] MTS Test machine

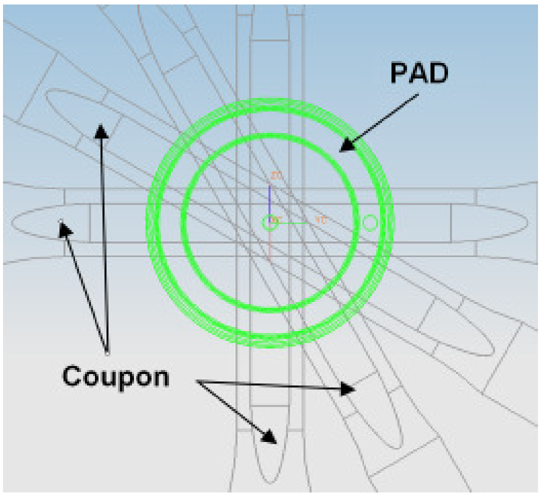


Figure 5 Axially symmetrical pads in different positions

- The normal force on the fretting pads is monitored throughout the test.
- The coupons are non-destructively inspected by eddy current probe after 1200 [SFH] and every subsequent 600 [SFH] for cracks.

Test Results

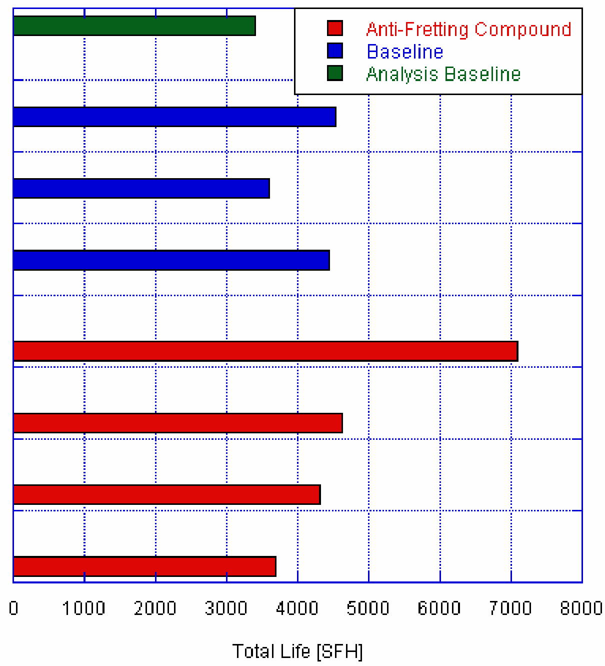


Figure 6 Total life of tested coupons

- The analysis under-predicted the total life slightly, see Figure 6.
- All coupons failed at exact the same and predicted location at the outer diameter of the fretting pad, see Figure 7 and 8.
- The reason for the remarkable longer life of one of the coupons with anti-fretting compound could not be determined, even not by fractography.

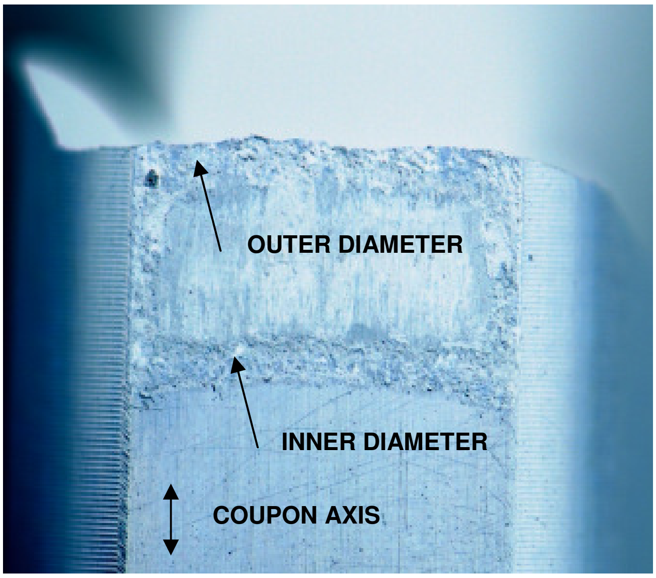


Figure 7 Detail of coupon contact surface.

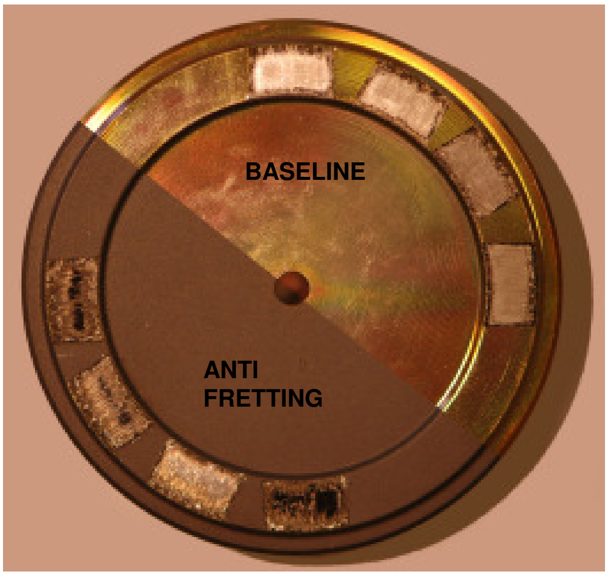


Figure 8 Baseline und anti-fretting contact surfaces after testing. Both surfaces were worn blank.

Conclusion

The application of the MoS_2 based solid film anti-fretting compound does not result in significant longer total lives. The anti-fretting compound does not withstand the high contact pressure combined with the relative fretting movements and is worn away.

Nevertheless, the chosen test procedure proves to produce reliable results (relatively small scatter) and is straightforward.

Further testing with different anti-fretting compounds will be performed.

Acknowledgment

The work of this project is sponsored by armasuisse.

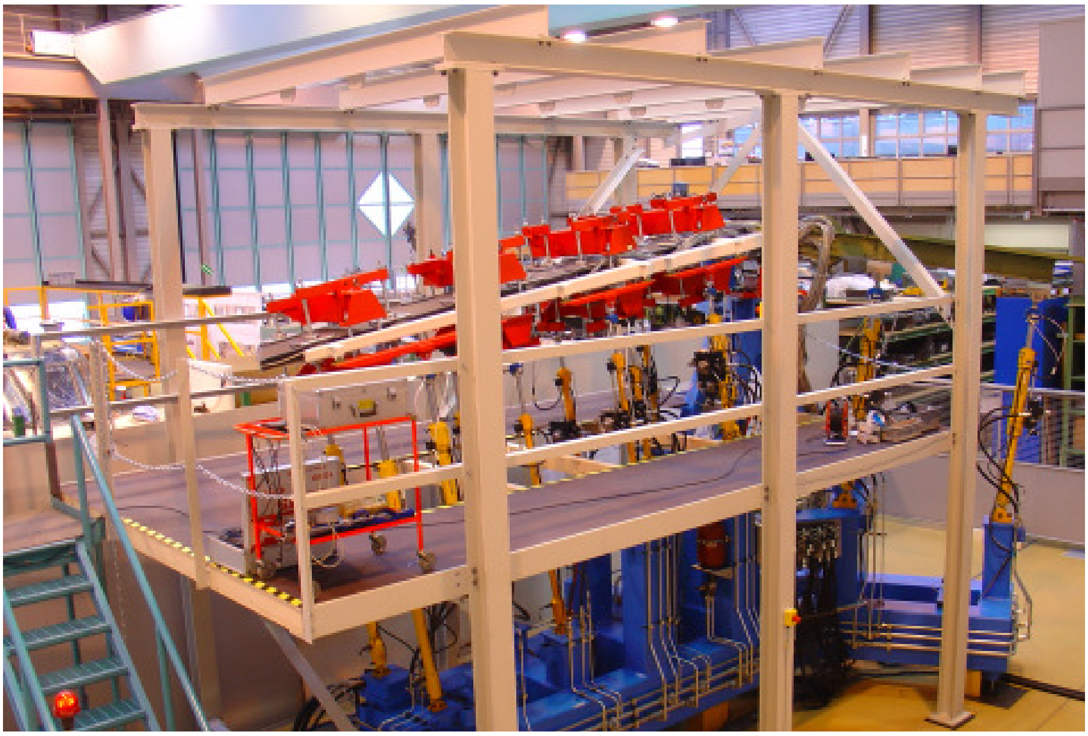
4.8 ALCAS - Advanced Low Cost Aircraft Structures

B. Bucher, M. Giger, B. Schmid, Josef Lussi, Stefan Frei, S. Zehnder; RUAG Switzerland Ltd

ALCAS is a European Commission co-funded integrated project which was started in 2005 and will be completed by the end of April 2011.

RUAG's task in the current ICAF reporting period includes structural static and dynamic testing of the carbon fibre reinforced composite Validation Article (VA) which is based on a Falcon 7X Horizontal Tail Plane architecture. For the purpose of ALCAS, the design details have been changed however, in order to investigate different materials and manufacturing methods. Very important aspects are the damage tolerance, no growth behaviour of barely visible impact damages (BVID's) and inspectability using ultrasonic NDI methods. The test set up features the following elements:

- 14 low-friction servo-valve controlled hydraulic actuators
- 280 bar hydraulic pressure supply
- Independent A/B channel INTERFACE load cells
- MOOG/FCS control and monitoring system
- HBM data acquisition system using 230 data channels
- Safe Emergency Unloading



The tests on the VA include the following activities:

1. Initial inspection of artificial damages and BVID's using ultrasonic NDI methods
2. 8 shake down Limit Load Cases at 30% and at 70% load level * LEF (Load Enhancement Factor)
3. 8 Limit Load Cases up to 100% Limit Load * LEF + NDI between each load case
4. Fatigue cycling using 20 elementary load cases which are combined to 10036 test load cases and repeated 8 times to represent one aircraft life
5. 8 Limit Load Cases up to 100% Limit Load * LEF + NDI between each load case
6. 8 Ultimate Load Cases up to 150% Limit Load * LEF + NDI between each load case
7. Overload for one symmetric load case to rupture or max. 2x Limit Load * LEF

4.9 Research Activities in Structural Integrity

I. Kongshavn, R. Zehnder, M. Wyss, B. Schmid, C. Huber; RUAG Switzerland Ltd

Introduction

Following the Quantitative Fractography (QF) seminar given by Dr. S.A. Barter (DSTO) at RUAG in 2008, RUAG started to develop in-house QF capabilities.

In a first step, necessary equipment was evaluated and procured. To start crack growth measurements and for a better understanding of spectrum, coupon tests with different marker bands were carried out. As test spectrum the F/A-18 Wing Root Bending Moment (WRBM) Spectrum was used, this spectrum was also used in the Swiss Air Force F/A-18 full scale fatigue test.

Equipment

A high powered optical microscope was evaluated and procured to observe the crack surface in detail. So far, five different objectives magnifications between 25x and 500x are available. An internal light as well as an external source with two optical fibre gooseneck lamps are available. Up to a magnification of 200x also dark field is available.

A five megapixel photo camera is mounted directly on the microscope and is used for documentation purposes. The camera is connected directly to a work station and all settings can be controlled by software interface. The camera is calibrated to the microscope such that measurements from pictures can be done.

In case of crack surface measurements, a high resolution X-Y stage is used. The stage can be moved manually over a travelling range of 100 x 80 [mm] with an accuracy of 40 ± 2.5 [µm] and a resolution of 0.05 [µm].

Coupons testing

High Kt dog bone coupons out of Aluminium 7050-T7451 with a bare surface were used. The dimension of the main cross section was 40 [mm] width, 6.5 [mm] thick with a hole of 6.5 [mm] diameter. All tests were done with open-hole coupons.

As a baseline, the Swiss Air Force F/A-18 WRBM spectrum was used. The same spectrum was used during the Swiss F/A-18 full scale fatigue test. Four different kinds of marker loads, i.e. underload, overload, reordering and constant amplitude (CA) were added to WRBM spectrum.

Underloads and overloads were designed in accordance with reference [QF1]. Both marker loads are composed of five cycles with R=-0.88 with a peak of 90[%] maximal spectrum load. For overloads an additional 120[%] peak was added and for underloads an additional 90[%] peak. For the reordering spectrum, the five biggest peak loads were put close to each other. In the case of the CA marker band spectrum, a 200[FH] CA Block was designed with the same severity as a 200[FH] WRBM block. The final test spectrum contained 5 WRBM blocks and one CA marker band block.

A total of 10 coupons were tested on a single cylinder tension-compression test stand. The test results can be seen in Figure 1. Further testing is planned as only two coupons per marker type (statistical significance) and no baseline spectrum coupons (influence of marker bands on life) were tested in the current run.

Crack growth measurement
Several measurement were performed for each tested coupon (except CA200[MPa] and CA220[MPa]). For comparability and to build up a widespread know how, most measurements were done by two different people. Especially over- and underload coupons showed a very good repeatability. Overloads are easy to identify for a wide range and produce the best readable marker striations, see Figure 3. On nearly every measurement, an exponential behaviour of the crack growth curve could be shown (or linear slope in a log-linear plot), as an example a measured crack growth curve is shown in Figure 2. Within the 5CA marker load loaded spectrum, each WRBM spectrum block could be distinguished between two marker bands, see Figure 4. This behaviour confirms that WRBM is 'selfmarking' and mesurements on samples from full scale fatigue tests should also be possible.

Conclusion
QF has shown its strength in measuring crack growth after finishing tests. With under- and overload spectra , a very high repeatability could be shown.
A lot of time and training was necessary to perform the measurements on the marker band spectrum tested coupons and for the next step, measuring crack growth from coupons without marker bands, an even bigger effort will be necessary.

Outlook
RUAG continues to work with Quantitative Fractography on coupons without marker bands and will try first steps in measuring and interpreting cracks from Swiss Air Force F/A-18 full scale fatigue test.

Acknowledgment
Special acknowledgment goes to Dr. Simon Barter and Lorry Molent from DSTO Australia. The visit from Dr. Barter at the end of 2010 was very helpful and we would also like to thank him for the quick answers by e-mail.

References

[QF1] S.A. Barter, L. Molent and R.J.H. Wanhill, Defence Science and Technology Organisation, DSTO, Melbourne Australia; National Aerospace Laboratory, NLR, Amsterdam, The Netherlands; ICAF 2009

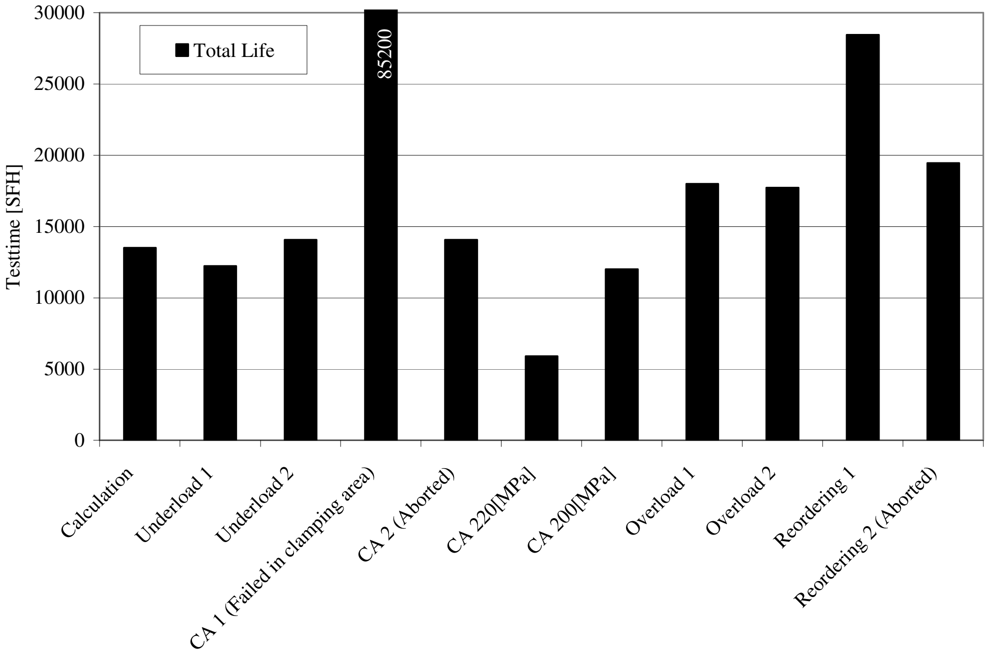


Figure 1 Total test life of coupons, loaded with different marker band spectra.

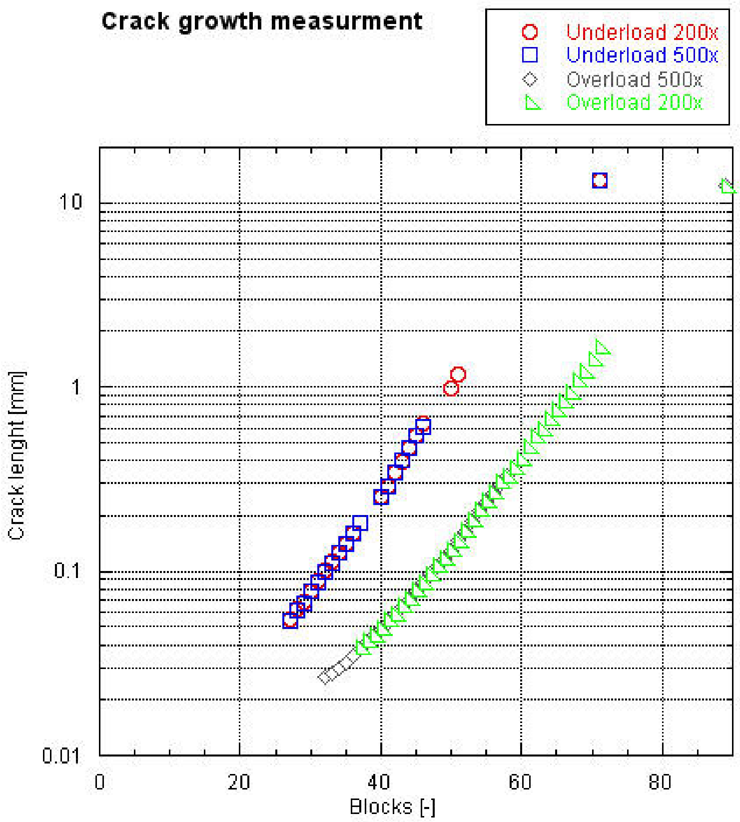


Figure 2 Crack growth measurement from coupons loaded with over- and underload marker band spectra.

Figure 3 Detail small crack growth overload marker band spectrum

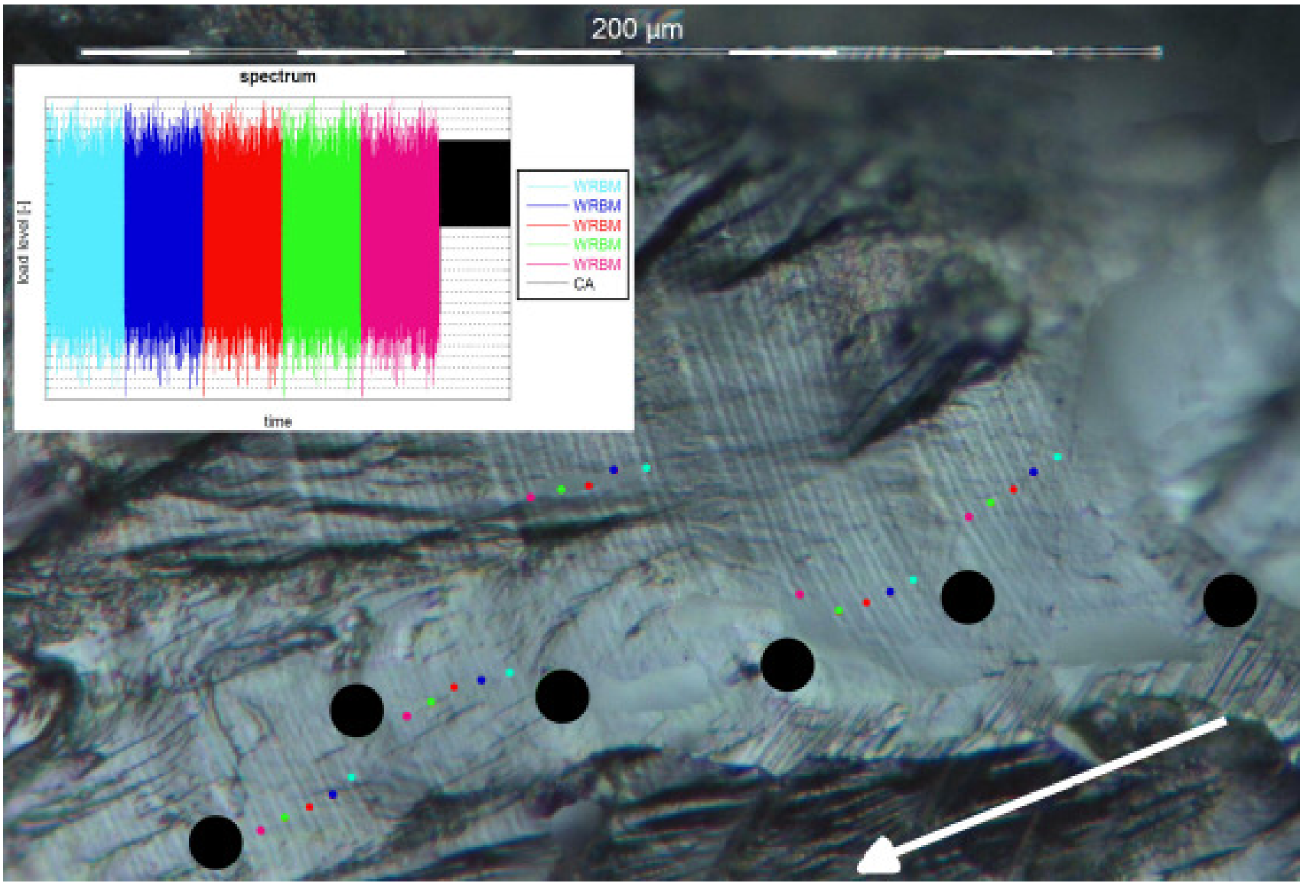


Figure 4 5CA spectrum: Five blocks WRBM spectra (small coloured dots) in between one block CA loading (big black dot). → WRBM is self marking

5.0 100 Years Swiss Aviation; Review of Full Scale Fatigue Tests

M. Guillaume; RUAG Switzerland Ltd, G. Mandanis; M@M GmbH

The Fatigue History Simulator developed by F+W Emmen [3]

1952 F+W Emmen decided to develop a so called ‘Fatigue History Simulator’, because the Swiss Airforce was aware of the fatigue problems, that would concern the new fighter aircraft and trainer projects. In 1952 no facility was available in the market to determine the safe life time of an aircraft. The essential property of the new test facility is the absolute proportionality of the load distribution and the genuine sequence of loading.

On that time the fatigue testing of an aircraft was not usual, industry pulsators were used for fatigue testing on aircraft components applying an oscillating loading. No specialist of general engineering was available to develop and design the test rig, consequently F+W Emmen had to start itself with the work. It is important to mention here that J. Branger (first Swiss national delegate of ICAF) was the iniator, the moving spirit and at last the father of the fatigue testing in Switzerland. During this time Switzerland had the opportunity to join Holland, the U.K. and Sweden in the ICAF and to share know how and experiences in relation with the fatigue of aircraft structures.

The requirements for this simulator can be expressed in 3 main requirements:

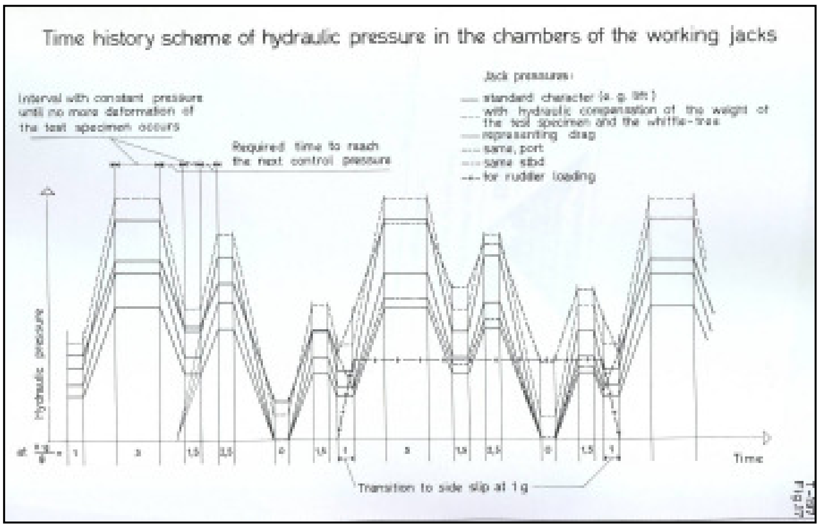
1. All applied load on the structure are controlled by a single command
2. The test facility must allow every loading history
3. The test facility must control itself automatically

The fatigue history applied on the simulator is an extract of the service history which contains every element which is significant for the fatigue life of the aircraft structure. The force are produced hydraulically and transmitted by mechanical means to the test specimen, which jacks which are working in tension as well as in compression.

A sophisticated system of a control piston connected by an adjustable scale beam with a servo piston allows to maintain and to control the jack forces, and by shifting the fulcrum of the scale beam to alter the pressure ratio within wide limits.

The oil pressure of 180 bar is delivered through a high performance pump with 120 litters p.m.

The next figure shows an example of fatigue loading upon the time.



Three curves show the working pressure in 3 jacks. The lapse of speed is depending on the time until the test specimen under load has reached its elastic equilibrium. The time which is needed for the pressure build up and drop is dependent from the performance of the load maintainer valves and the connecting pipes to the jacks, and also from the pump performance.
The sequences of the load are written on a 8-holes tape with optic identification allowing to transmit 256 orders. 67 orders are required for manoeuver loads, ground loads and the switching to different weights and load cases, the rest is needed for special load conditions like engine thrust, wheel spin etc.

In the conclusion of the report J. Branger states:
'We believe that with the simulator any fatigue history of any aircraft can be simulated, and all the happening follow each other in the genuine sequence which is – until the contrary has been proved, as important as the number and the magnitude of the events .'

This idea to develop a fatigue facility dedicated to the full scale fatigue test of different aircrafts was very rational and interesting. The reality has nevertheless showed that for different reasons each aircraft full scale fatigue test needs a test rig and an overall facility, which are developed ad hoc for the specific test.

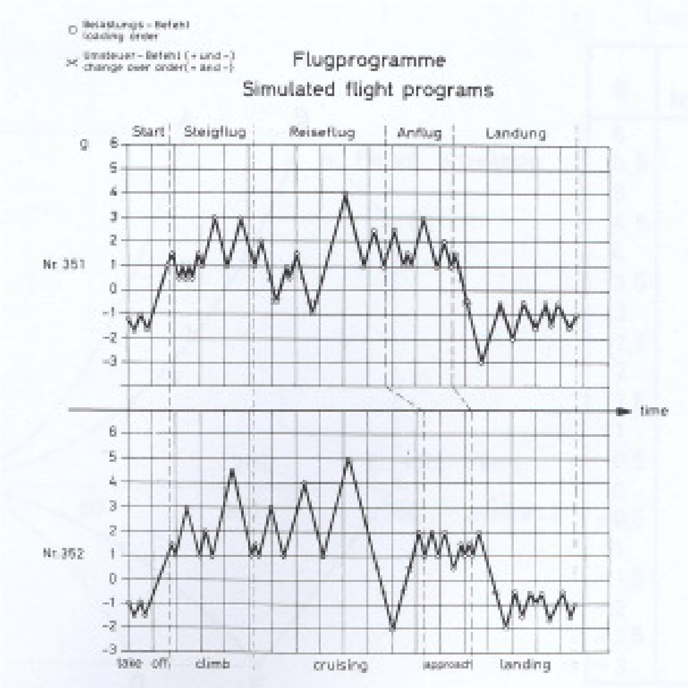
The Full Scale Fatigue Test on the P3 Trainer Aircraft [4]

At the end of the fifties the Swiss air Force ordered for the training of the pilots the P3 aircraft of the PILATUS AG in Stans. The following clear specifications were required:

2500 hours safe-life	
1800 times	+5 g
5600 times	+4 g
10400 times	+3 g
3700 times	-2g
370 times	-3g

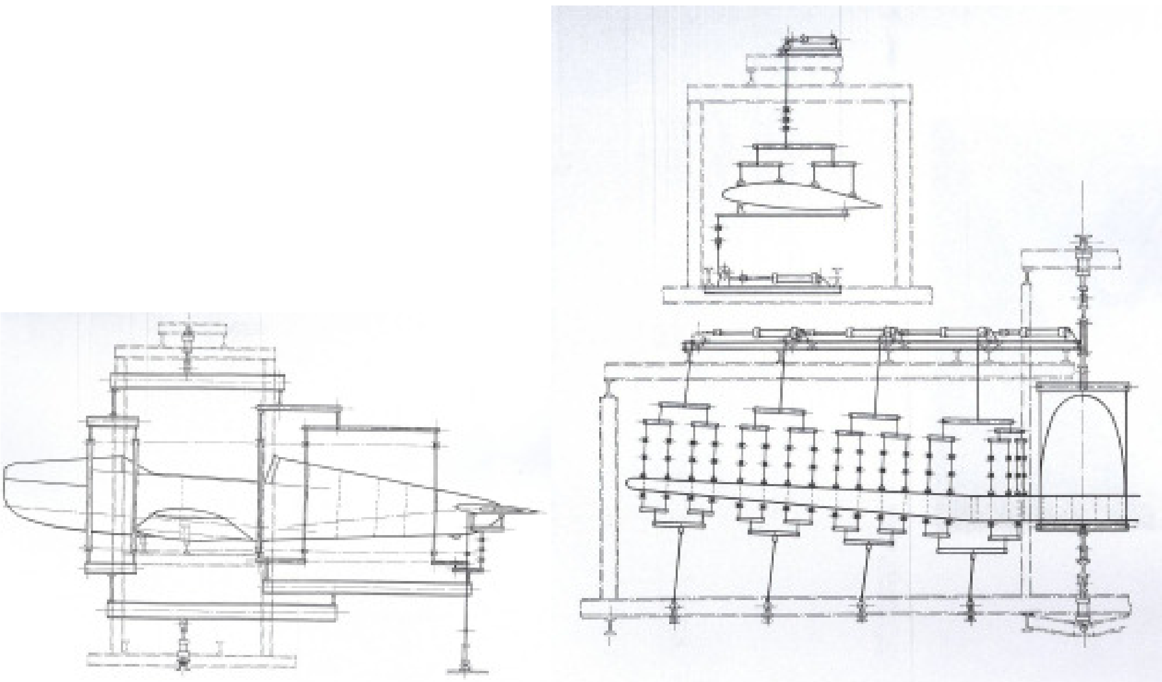
At this time the 'fatigue history simulator' of F+W Emmen was ready for testing, and thus the P3 was the first test item. After different preparations and some preliminary run the real test began in August 1960.

24 different partial programs, based on v-g-h records, describing take-off, climb, different types of flight, descent and landing, have been combined to generate 1056 different flights, containing about 23'000 cycles. The next figure shows 2 examples of flight.



To set up the sequence of flight the ballot box methos has been used, it means that one flight after the other has been drawn randomly like in a raffle.

For this test the rig applied the forces through 19 jacks, 16 for the wing in a lying position (see the next figure) and 3 for the fuselage. The chordwise force application is different for the positive and the negative loads, matching on this way the specific pressure distribution on the wing as well on the fuselage. The load applying points have been located along each rib. No side loads were simulated at that time.
With this test rig and its equipment within 24 hours 72 real flying hours could be simulated.



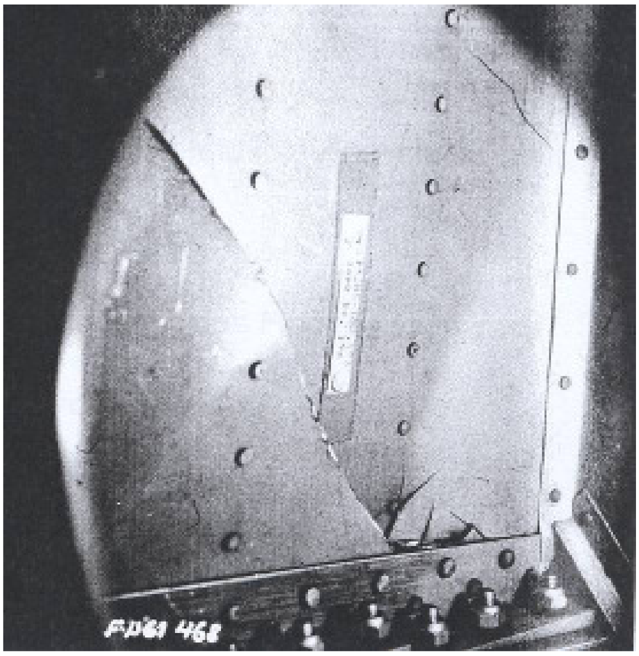
With this arrangement for the P3 test all jacks have acted in tension.

A simple inspection program has been set up:

1. the structure was inspected visually for defect every day, one day representing 72 flight hours.
2. each reduction of the stiffness of the structure is reported by the wing tip cut out switches at +6 and -3 g due to higher deformation
3. all load peaks from -0.5 to 0.5 g all zero passages were counted and registered
4. the load of 9 selected jacks were measured by precision strain gauge.

For the full scale fatigue test the last aircraft of the production has been selected. By test start the first aircraft has already 500 flying hours, and some loose rivets have be found due to overloading but sure not due to fatigue. After 1000 hours fatigue testing a fretting damage occurred. A repair has been done on the test article as well on the aircrafts of the fleet. After 2500 test hours the authority decided to extend the test up to 5000 hours after a static overload test of 7 g. At the end of 5000 hours an ultimate test (factor of 1.5 on max service loads) up to 9 g has been done. The specimen failed at 8 g whilst the design ultimate load target was 9g.

Despite very careful inspecting before this ultimate load test, the fatigue damage which leads to failure was not remarked before. Finally one of the hydraulic people found by chance in a hidden corner a damage which can be termed as a classic example for a fatigue failure (see next figure). The failure occurred by shear in a web plate of the main spar in both outer wings, at the attachment to the inner wing. A check of the stress calculation of this part showed, that the web and the riveting did not reach the ultimate factor of 1.5. On all service aircraft the allowed limit was thus reduced temporary to 5 g, and it was possible to strengthen it before an accident occurred.



The test simulated two required safe-life and revealed the weak points of the structure reaching on this way the original goal. The original solution of the lying jacks in order to reduce the height of the test rig worked very well, but didn’t set the standard for future tests. In the most full scale fatigue test the jacks are working vertically.

The Full Scale Fatigue Test of the DH-112 Venom Aircraft [5]

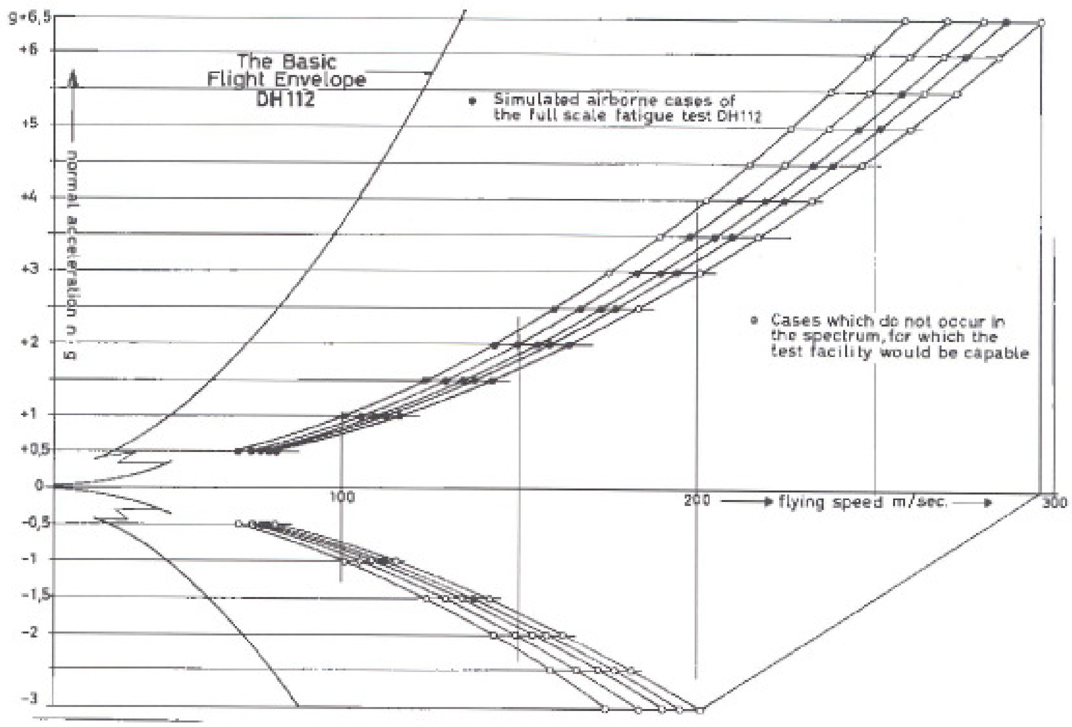
Astonishingly the Venom aircraft was originally guaranteed for a safe life of 500 hours. The full scale fatigue test has been initiated to elongate the life to 1000 hours, conforming to the aim of Swiss Air Force. This test has been carried out on the Fatigue History Simulator developed by F+W Emmen some years before.

To solve this task, to double the safe fatigue life, the following terms were set up:

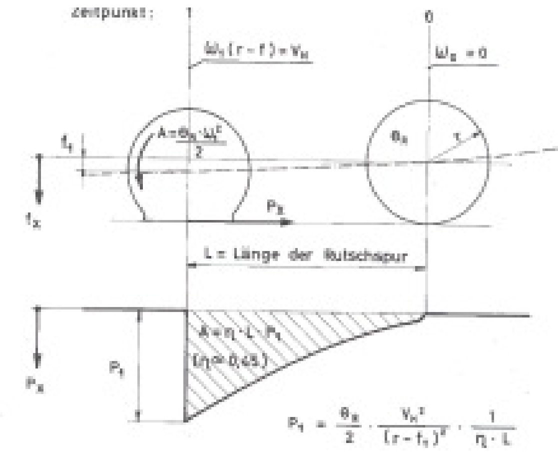
1. The load of the aircraft in service had to be ascertained.
2. These load conditions must be represented in the full scale test without restriction
3. All reinforcements, modifications and replacements juged necessary for the 'flying safety' of the full scale test structure must also be introduced on all service aircrafts.
4. Incipient cracks and their propagation must be observed on the full scale test. On that basis the periodical intervals for inspection of the service aircraft will be done.
5. All aircrafts of this type must be put into service in such a way that all are fatigued in the same manner, forming one population only.

The loading of the wing and the fuselage bodies have been achieved in a classical way conforming to the experience collected with the P3. The introduction of air and inertia load is located at each crossing of a rib with a stringer by mean of 2 or 3 Jo-bolts. In this way the test specimen is neither weakened nor stiffened. During flight some pressure prevails within the wing as the fuel tank have a venting system by ram air. This pressure is also simulated in the test.

The load distribution over the supporting area was measured in the transonic wind tunnel of F+W Emmen. The conditions at M=0.68 were chosen as standard. The simulated air cases are shown in the next figure, see below.



A big attention has been given to the landing gear load at landing due to the sudden rotation of the wheel.



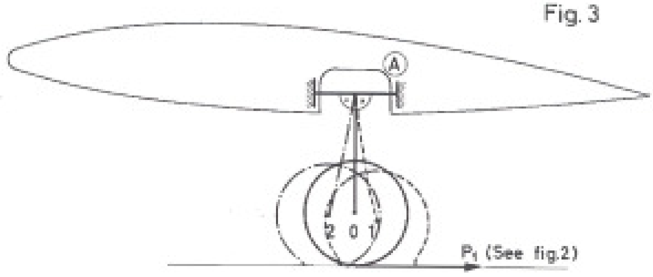


Fig. 3

This load occurs on each landing and this is very important for fatigue. This variable oscillating stress has caused a fatigue defect. Thus a repair and reinforcement of the area had to be made.
Thanks to the fatigue meters that have been mounted in the aircraft very early we had excellent statistical data available. Each case where the limit load was exceeded could be investigated. Also vertical acceleration at landing and taxing could be recorded. The great number of short brake application during taxing surprised us and were very important for the fatigue of the undercarriage.

To prepare the 'History Loading' the following principles have been applied:

The spectrum has been divided in 5 identical periods of 200 flying hours representing each 350 flights. 9% were made at gusty weather conditions, the rest at normal weather.

The aircraft weight has been varied in 5 groups: 5/5 to 1/5 fuel carriage

189 flights with 5/5 fuel carriage at take off
172 landings with 1/5
10 landings with 2/5
72 landings with 3/5

150 flights with 4/5 fuel carriage at take off
140 landings with 1/5
7 landings with 2/5
3 landings with 3/5

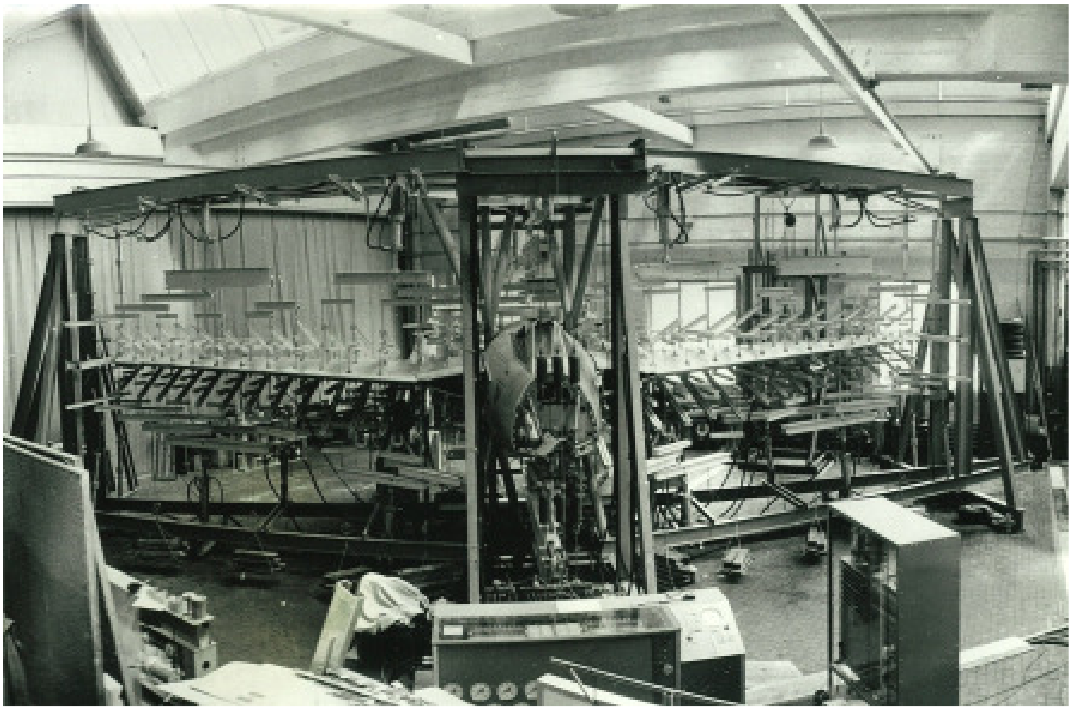
11 flights with 3/5 fuel carriage at take off
7 landings with 1/5

4 landings with 2/5

For the take-off two different taxing length with two different braking forces have been considered. For the flying air case 5 flights with normal weather and 6 with gusty weather have been simulated. A great number of landings have been chosen conforming to the next table:

		Touch – down – speed [km/h]			
Character		180	200	225	245
Soft	10%	1%	7%	2%	-
Normal	70%	6%	46%	13%	5%
Medium	17%	2%	11%	3%	1%
Hard	3%	1%	2%	-	-
Total	100%	10%	66%	18%	6%

The sequence of the flight types has been found to the ‘Monte carlo’ method as for the P3 aircraft. The 350 flights (200 hours) have been stamped on a telex tape which contains 90’000 orders on a length of 235 m. The speed of the Venom test is such that the whole tape (200 flying hours) could be run in 5 days.



Swiss Venom DH-12 test test up at F+W in Emmen during 1962 till 1964.

After 472 hours the wing was reinforced in the same manner at rib 2 as it has been done on all Venom aircraft in service between 400 and 500 hours. 20% of the aircraft had at this position fatigue cracks due to undercarriage loads.

At 1472 hours the starboard fuselage/wing attachment fitting was replaced on the fuselage bay a fitting from an aircraft which showed a very interesting fatigue crack. The reason is an unfavourable coincidence of extreme tolerances at a position where the design must be considered as a little bit unfortunate. The propagation of this crack could be observed during the test without any risk. Since the failure of this fitting would be fatal an improvement had been designed and the improved fitting introduced in the test specimen at the same time as the cracked fitting, but symmetrically to the other. The crack at this time had a length of 0.25’’. At 600 hours later the propagation stopped at 0.4’’, as the crack crossed the weak area of the fitting. At 5000 hours it was still at 0.4’’. From that we learned that it is not necessary to replace this fitting on the service aircraft although a failure would be fatal. Alone this experience saved 25% of the money spent for the whole full scale fatigue test.
At 5000 hours the test has been stopped and the specimen exposed to a 9 g overload (90% of ultimate load). It did not fail. The dismantled wings showed some failures, but all easy to repair.

For the Inspection the following method has been used:

After 200 hours a strict external visual inspection has been made. After 1200 hours a servicing of the aircraft is made on which the wings are removed. For a further insight into the wing 4 television cameras which can be switched to a screen during the test have been installed. After 200 hours the static whole wing deformation at 3 g is measured. After 1200 hours the noise within the wing is recorded by a very sensitive tape recorder. Further inspection is done by strain gages.

This test, which has simulated the usage spectrum in a high level of detail, has found out, like for the P3, the fatigue critical points of the airframe, and this result could be applied very usefully to the fleet.

The Full Scale Fatigue Test of the Swiss Mirage III S

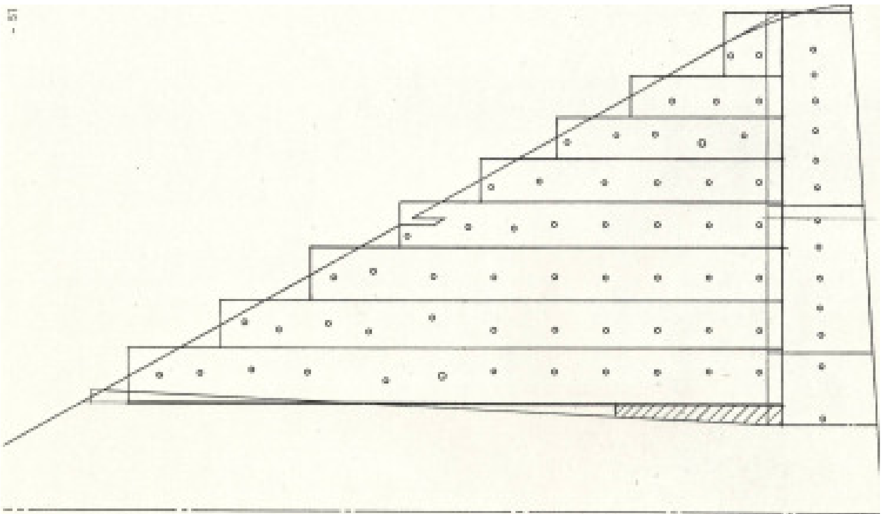
The Mirage III S full scale fatigue test took place in F+W Emmen from 1976 to 1985 and brought essential knowledge in the fatigue behaviour of the aircraft in relation with Swiss usage spectrum. Compared to previous full scale fatigue test this one was in an order of magnitude more complex for two reasons:

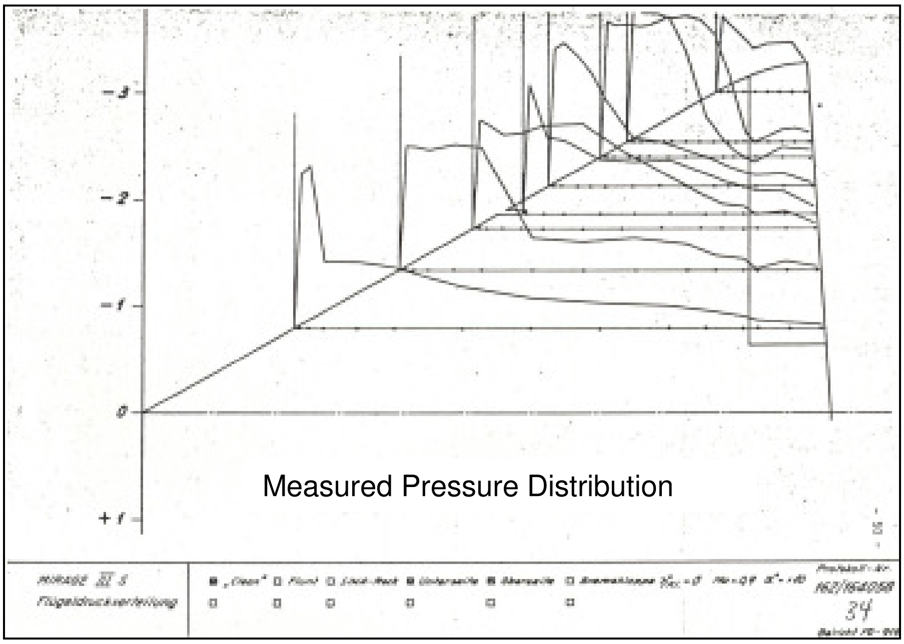
1. the Mirage III S is a heavy sub- trans- and supersonic aircraft
2. the wing has a delta shape whose lift behaviour is very different from a conventional wing

It is also remarkable to note that this project has been achieved in a total independent way. Practically all the data necessary to determine the different forces to be applied through the test rig in a large set of points of the airframe have been elaborated in the company. The aerodynamic loads have been generated through tests in the wind tunnel facilities of F+W Emmen. In the next paragraphs we want to give some details to illustrate the very intelligent, efficient and technical perfectly correct method to come to realistic load forces. All this procedure has been mostly achieved on paper sheets in a time where engineers didn't used the computer every day.

Loading of the wing

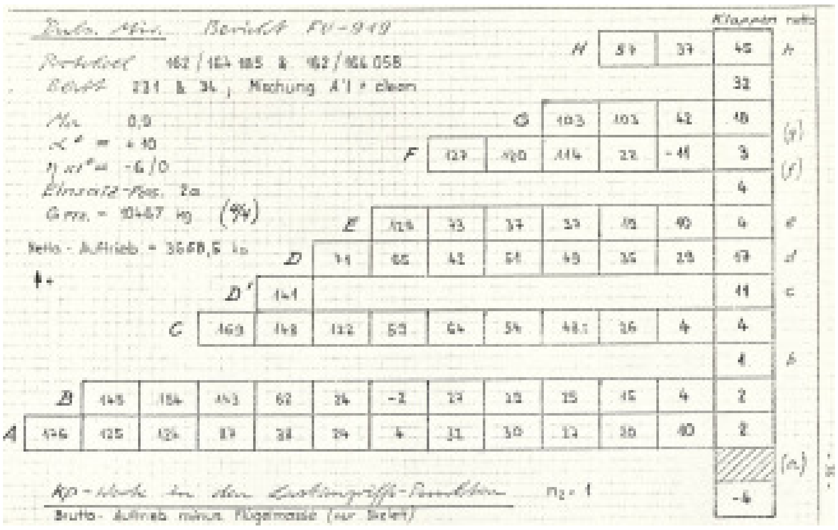
The wind tunnel measurements have been achieved in the transonic wind tunnel facility with a 1:50 full model and a 1:25 half model (see next pictures). On the upper and lower side of the delta 69 pressure measuring points have been distributed documented in the next figure. The distribution in seven rows A to H for the selected points have been kept for the whole load calculation process. Of course the control surfaces were adjustable. On this way a large range of load cases with different angles of attack and Mach number have been investigated.



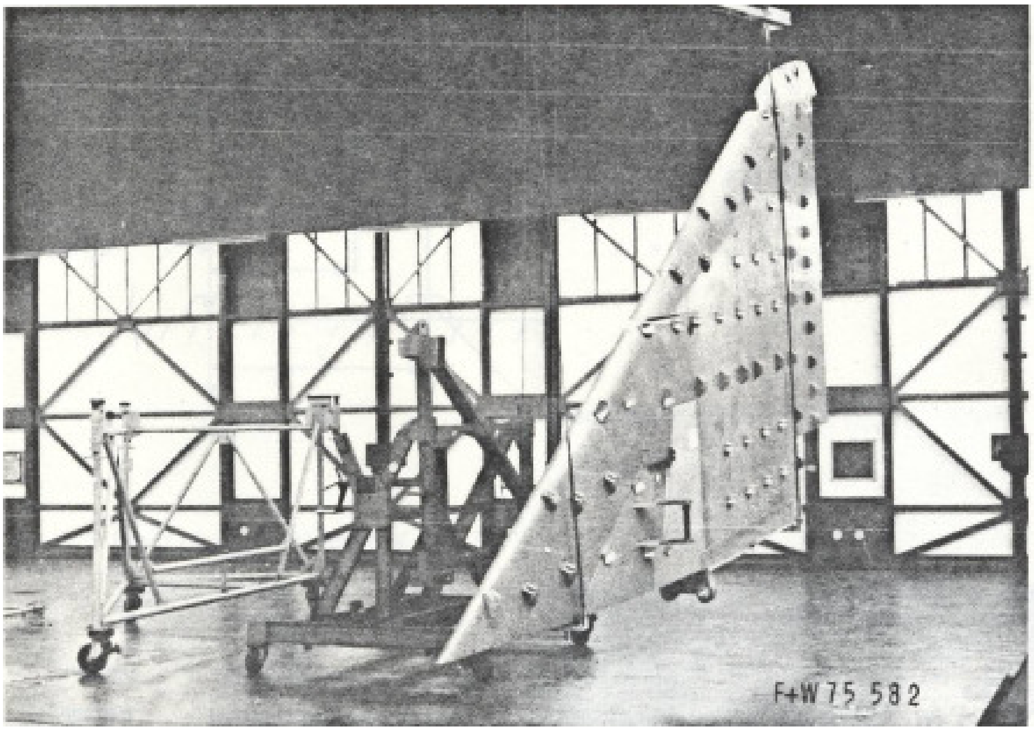


Measured Pressure Distribution

The break down of the wing masses was given for 60 grid points. With some interpolation they could calculate for every load case for each panel displayed in the next figure the inertial force and determine the net force, lift minus inertia, acting on the airframe.



In the next picture the load application points on the upper side of the right wing are shown, the similarity with the panel distribution is clearly visible.



The break down of the typical flight for the Swiss usage is represented in the next table.

Flight		**Configuration**	**% Distribution**
Mach 2	11000 m		1.5 %
Mach 1.8	12000 to 20000 m	without	3.5%
Mach 1.6	11000 m	supersonic tank	5%
Mach 0.9 to 1.2	higher than 9000 m		3%
Mach 0.9	higher than 6000 m	with	65%
V_A 1200 km/h	in 1200 m	supersonic tank	19%
V_A 500 to 800 km/h	betw. 2000 and 6000 m		3%

One can see that the transonic domain is the most important and that two essentially different configurations are considered. Therefore the point in sky M = 0.9 at H = 23’000 ft with the two configurations were selected for the Swiss test.

Loading of the Fuselage

For the fuselage the aero as well as the inertia load have been considered although the inertia is clearly more relevant. The fuselage has been divided in load sector similarly like the panels of the wing using the wind tunnel pressure measurement and the break down of the fuselage mass sectors.

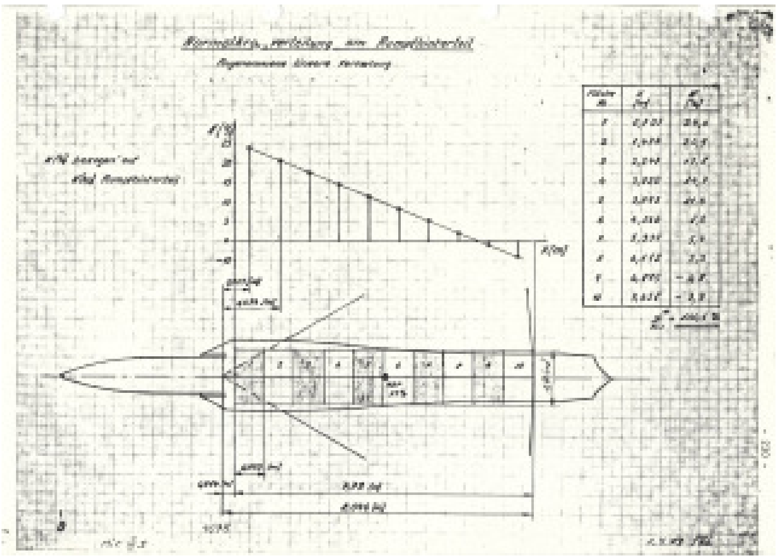
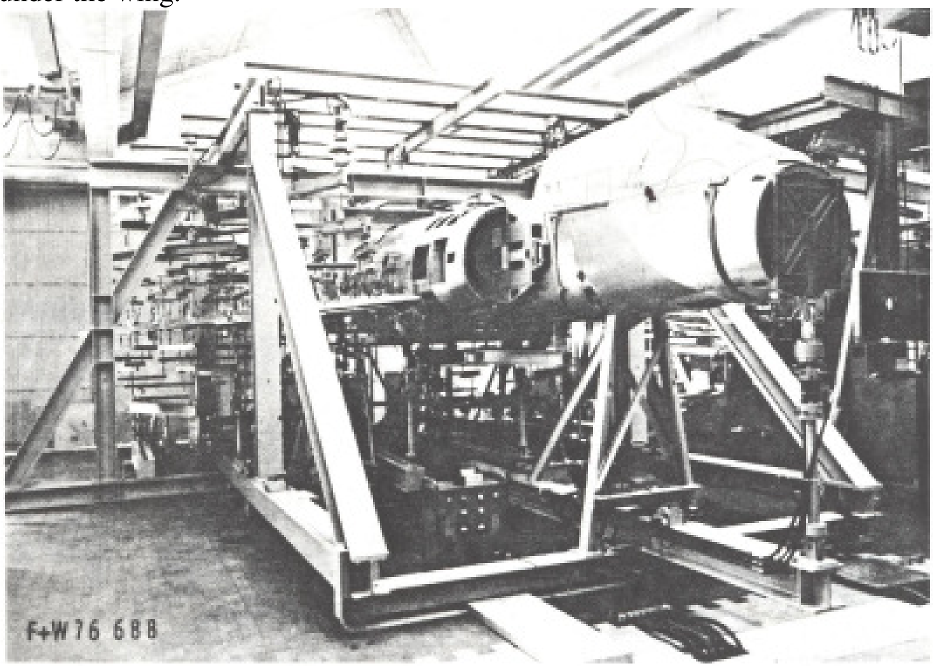


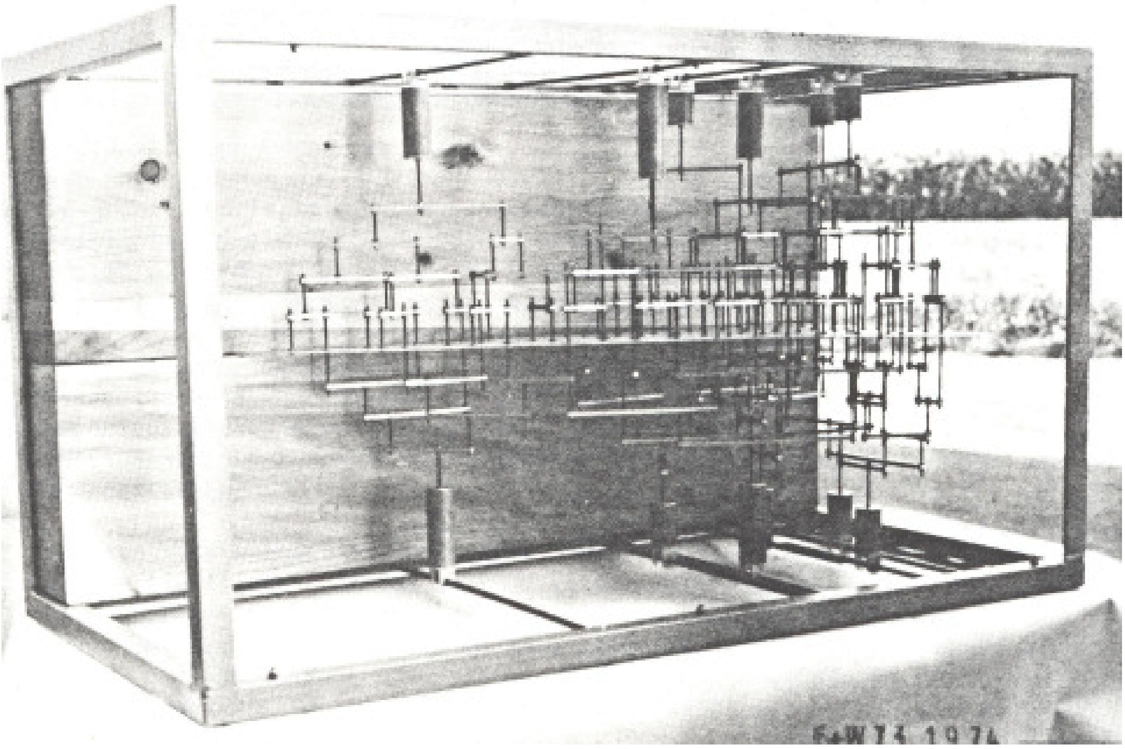
Figure with wind tunnel pressure measurement and the break down of the fuselage mass sector

The Test Rig

Finally the test rig could be realised in a compact shape (see the next figure) with a high density of whiffle trees over and under the wing.



For the load application on the wing up to four level whiffle tree have been used, working only in tension mode, as one can see in the next mockup picture.



It is interesting to observe the widely symmetrical jack distribution over and under the wing. Totally 38 jacks have been used for the fatigue test article, with 20 jacks only for the wing.

The Test Results

The test started in 1976 and ended in 1985 simulating of 21085 flight hours for the fuselage, 4832 for the wings. At this time happened the first crash of the wing due to a rupture of the main spar. Another already flown Swiss wing has been mounted on the left hand side, and an Australian RAAF wing with 2192 flight hours on the left hand side. Between 1986 and 1994 the test article and the test rig have been reused to achieve the full scale fatigue test of the canard, the KAWEST upgrade program, developed by F+W Emmen.

The full scale fatigue test allowed to discover a crucial weakness at the main spar due to different drilled holes (slan rivets and tooling hole) where cracks have been initiated very soon during the test. A refurbishment has been undertaken to suppress this weakness in the fleet. Here again no more argument are needed to demonstrate the usefulness of the full scale fatigue test on an aircraft.

Damage recording
During the test including tear down inspection 725 deficiencies were documented. Later during post analysis the deficiencies were categorized in Maintenance Critical Part (MC) and Fracture Critical Part (FC).

Assembly	Total	MC Part	FC Part
Wing	701	401	300
Fuselage	24	16	8
Total FSFT	**725**	**417**	**308**

Table with summary of ¨Note of Deficiencies¨

Swiss F/A-18 Full Scale Fatigue Test [6]

The Swiss Air Force requires for the F/A-18 C/D an operational life of 30 years with a service life of 5’000 flight hours (FH). The Swiss design spectrum is generally three times more severe than the US Navy F/A-18 design spectrum. In an early phase of the procurement Boeing, St. Louis IL, performed an aircraft structural integrity study (ASIP) based on MIL-HDBK-1530 to assess the criticality of the structure. As a result of this study the Swiss F/A-18 had to be redesigned to meet the structural requirements. For the Swiss a crack initiation life of 10'000 FH for fracture and maintenance critical parts had to be demonstrated. The Swiss also wanted a crack growth life of 10'000 FH for fracture critical parts to be demonstrated, according to US AIR FORCE damage tolerance philosophy based on the MIL-STD-87221. Overall Boeing has done 833 analyses. The Swiss F/A-18 is reinforced in the center fuselage by material change from aluminum to titanium for the three carry-through bulkheads and the dorsal longerons. Furthermore, in the center fuselage und the inner wing structure a lot of beef up and interference fastener und cold working was introduced.
Based on tests for the US Navy the Swiss F/A-18 was only cleared for a usage of 2'000 FH. The redesign has to be qualified for a service life of 5'000 FH of Swiss design usage.
In 1998 RUAG Aerospace at Emmen, Switzerland, was assigned to perform a full scale fatigue test. The test should start no later then in 2003 and 4'000 FH should be simulated by end of 2003. RUAG Aerospace teamed with German company IABG, Munich, in 1999 to develop a modern and efficient test set-up.

The full scale fatigue test program was split into four phases:

Pre-Test Activities (1998 – 1999)
The test article (Effectivity FTS1) consists of a complete F/A-18 D (double seated) airframe. The final assembly took place at RUAG Aerospace. Overall 600 strain gauges were already applied during production of the major components by Boeing. Additional 400 strain gauges were specified and applied at RUAG Aerospace.

The roughly 1'100 strain gauges are distributed over all components and positioned respecting different criteria:

- Criticality of the location (Fracture and Maintenance Critical Part)
- Location already chosen in earlier tests such as FT01/ST16/FT93 for the US Navy
- Location which was modified for the Swiss configuration
- Location with non-conformances from production and assembly

Eight displacement transducers on the wing and six on the fuselage were specified to measure the deflections during the static strain survey.

Some components were omitted in the test, because they are not part of the qualification of the Swiss redesign. These components were replaced by dummy structures for load introductions or for load reaction purposes. These components are:

- Nose radom
- Nose and main landing gear
- Drag brace
- Gun mount
- Fuselage centre pylon, forward & aft attachment
- Missile launcher, L/H & R/H, forward & aft attachment
- Arresting hook
- Engines, L/H & R/H
- Horizontal stabilizer, L/H & R/H
- Inboard and outboard wing pylon, L/H & R/H
- Wing tip missile launcher, L/H & R/H

All non-structural doors and covers such as equipment bay doors, dorsal deck covers, wing and fuselage fairings, flap and aileron shrouds were omitted in order to gain better access to the test structure for inspection purposes during fatigue cycling.
During the early phase of the pre-test activities a comprehensive internal loads FE model was developed from a lot of sub models used during the ASIP study. This FE model was used for the validation of the structural integrity due to load introduction and for the prediction of the global structural displacements. This data was also used to specify the hydraulic test equipment (hydraulic force and stroke, valve capacity).

The Swiss master event spectrum (MES) from the ASIP study was slightly simplified for the test. The number of point-in-the-sky/configuration combinations was reduced from 25 to 6. Especially supersonic flights at high altitude and heavy attack configurations were omitted. These maneuvers occurred very rarely in the MES, and are not part of the actual Swiss usage.
The goal was to keep the crack initiation life within ±10% for 8 selected critical locations in the center fuselage and the wing. The test spectrum called MES6B5 consists only of 2009 different load cases compared to the 3107 load cases for the original Swiss MES.
According to the rule of thumb, a ±10% change in stress would result in a -50/+100% change in fatigue life. Therefore, a ±20% change in fatigue life amounts to an appropriate -2.5/+3.3% change in the stress of the structure. Considering the fact that the test equipment for the F/A-18 FSFT could control the test loads only approximately within ±3% and that the load cell accuracy was within 2% of the full capacity of the load cell, the 10% life criterion was very acceptable for the simplification of the test spectrum.

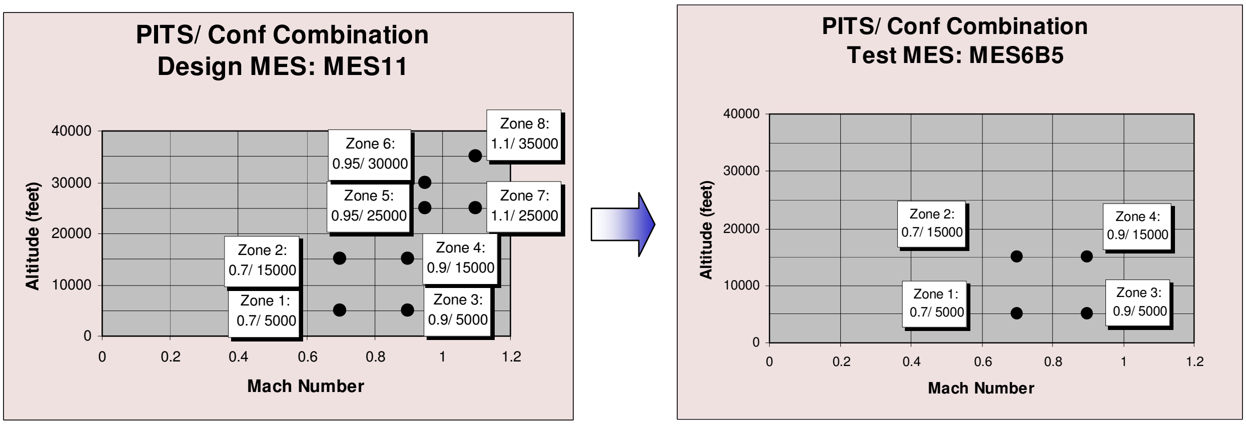


Figure: LH side original Swiss MES with 8 point-in-the-sky, RH side Test MES with 4 point-in-the-sky

One block consists of 200 FH which corresponds to 300 flights. Three center of gravity positions were applied in the MES. The weight of the aircraft which depends on the configuration and the fuel was kept constant during a flight. During 2/3 of the flights cockpit pressure was applied in the cockpit area. For the landing a simple sink speed spectrum for land based usage was included in the MES. For the test the landing cycles were not applied only a weight on wheels load case was simulated. A study demonstrated that the landing cycles had no impact on the structural redesign for the Swiss.

Together with IABG a first test concept was developed. The loading of the wing and the center fuselage was required to be very accurate. Therefore quite an effort was made to define an optimal actuator layout. The goal was to match the vertical bending moment along the fuselage and along the wing reference line for 30 selected fatigue load cases with an accuracy of 0.5%. For the whiffle tree layout the fatigue damages associated with the point-in-the-sky/configuration was used to scale the levers and determine the point of average load introduction.

Test Set-Up (2001 – 2002)

The detailed test set-up was developed together with IABG. All the test equipment was specified, manufactured and installed in the test hall at RUAG Aerospace, Emmen. The assembly of the test rig started in November 2001. In 2001 the actuator spectrum for 68 actuators was developed and approved for the test.

The test article was placed into the rig in spring 2002. The installation of hydraulics and the control system together with the data acquisition system was completed by summer 2002. The static and dynamic commissioning of the complete test set-up took place in autumn 2002. By the end of 2002 the test was ready for the fatigue cycling.

Fatigue Cycling (2003 – 2004)

In January 2003 the fatigue cycling was started. The test is operated by three shifts 24 hours a day. It takes about 5 days to simulate 1'000 FH. After 1'000 FH a detailed non-destructive inspection (NDI) program was done. In summer 2003 the test reached already 5'000 FH, i.e. the first service life, without any noteworthy failures. The test was completed in autumn 2004. The test report was delivered by the end of 2004.

Tear Down Inspection (2005 – 2007)

After the test completion of 10400 FH an in depth final inspection of the major structural assemblies was conducted. First a concept for the tear down inspection was developed to address all the critical zones of the ASIP study including the findings during fatigue cycling. An inspection plan with dismantle working records was prepared to ensure an efficient procedure. Tear down inspections were performed on the center fuselage and the R/H inner wing and the outer wings. All structural anomalies which were detected during the final inspection or during tear down inspections were recorded in ¨Note of Deficiencies¨. A data base was used to collect all the ¨Note of Deficiencies¨ based on part numbers including all the results from the US navy tests FT01/ST16/FT93.
All the parts were stored carefully for future investigations if necessary.

Test Set-Up and Equipment

Restraint System

The test article is supported by six single struts in a statically determined manner. On one hand they permit a safe mounting of the test article, and on the other hand they are used as load introduction during the test performance.
The restraint struts are connected to the nose and main landing gear dummy structures (3x Z-direction), to a load introduction fitting at the forward fuselage and to the arresting hook dummy (2x Y-direction) and the engine dummies (1x X-direction), see Figure 2. In case of an emergency shutdown the restraint system is designed to react the loads of the passively controlled unloading of the test article.

Test Rig and Inspection Platforms

The mechanical test rig consists of a number of steel structures supporting the loading system (68 hydraulic jacks and 6 restraint struts). All struts of the rig are connected by screws to special interface plates welded on the structural test floor surface. Every member of the support structure was positioned by a laser measurement system to ensure the correct position of the interface to the hydraulic jacks with a maximal deviation of 0.1 inch crosswise to the loading direction.
The three inspection platforms system (under wing, under the fuselage and along the airplane) allows for an easy accessibility to the entire test article from each side. The inspection platforms are also designed with respect to the test hall infrastructure. The inserted safety railings of the upper inspection platform can be easily removed to allow a quick integration or removal of the complete test article with the existing hall crane.

Load Introduction System

The loads exerted by the hydraulic jacks in the test are either distributed via whiffle trees to several load introduction points or introduced directly by fittings or dummy structures attached to the test specimen.

- Fuselage 20 jacks
- Wing box and flaps (L/H, R/H) 16 + 16 jacks
- Leading Edge Extension (LEX; L/H, R/H) 3 + 3 jacks
- Empennage (vertical, horizontal) 4 + 6 jacks

Altogether 44 jacks are equipped with whiffle trees each distributing their load to up to four load introduction points. For stability reasons it is necessary to take into account the correct degree of freedom for the loading system for the motion during the entire test performance.
The following load introduction devices are used:

- 15 different dummy structures (fuselage, wings, horizontal tails)
- 6 different load introduction fittings (fuselage, wings)
- 48 shear pads (center/aft fuselage)
- 522 tension/compression pads (wings, flaps, LEX)
- 4 contour boards (vertical tails)

Control & Monitoring and Data Acquisition System
For efficient fatigue cycling an automatically operating digital control and monitoring system is used. It was specified by IABG and delivered by FCS Control Systems. In total 77 control channels were implemented to fulfill the following tasks:

- Closed loop force control of all 68 hydraulic jacks
- Control of jacks in "normal fatigue mode", "strain survey mode" and "maintenance mode"
- Measurement of the reaction loads at the 6 struts of the restraint system
- Pressure control for the cockpit section
- Transformation of the numerical loading program
- Ensure the safety concept of the FSFT to prevent overloading of the test article
- Perform complete start-up, fail-safe and shutdown procedures

The data acquisition system was supplied by HBM, Germany. It was designed to measure 1'720 channels during the static strain surveys and to measure approx. 300 channels simultaneously during the fatigue test with a sampling rate of 10 Hz. The measurements can be initiated via an interface connection (CANBUS) between the control & monitoring and the data acquisition system.

The test speed was optimized to simulate the loads within an accuracy of approx. 2%. The simulation of one block representing 200 FH takes about 18 hours. Efficient fatigue cycling is achieved by operating the test in three shifts. That way 1'000 FH can be simulated in less than one week.

Strain gauge Measurements
Strain Survey:
Every 1'000 FH a strain survey was conducted. The strain survey load cases consist of 6 load cases of the spectrum block (symmetric (8.25g, -2g) and asymmetric (1.0g, 6.25g) conditions and the cockpit pressure condition. Only 70% of the maximum load was applied with steps of 10%. The criteria for the load condition were max actuator load, max component load at reference location, maneuver at the edge of the envelope. The test time for the strain survey takes about 15 minutes. A strain survey tool was established to evaluate the measured data, strain gauge load vs. load level of each applied load case.

Continuous Campaigns:
All 7 fatigue tracking sensors (primary & back up) and additional 16 strain gauges at FSFT reference locations, with short crack growth life from ASIP study, with single load path, note of deficiency location (NOD),were recorded with 10 Hz continuously through out the test. All load cell forces at the actuators and at the struts were also measured. The data was used to process Crack Initiation CI89 runs and compare the CI curves with the data established at 1'000 FH when all 2'011 unique load cases were measured, and the fatigue sequence could be constructed
For four strain gauges at single load path the reference stress was in excellent agreement with the FEM calculation.

Inspection Program

The inspection concept is based on the Swiss ASIP study performed around 1990. Due to the optimal accessibility the test article is inspected visually by qualified staff in daily walk-around inspections during fatigue cycling. Also the dorsal deck, the engine bay area, the engine inlet, and the bulkheads on the lower side are inspected for early cracks.

Major inspections are scheduled in intervals of 1'000 FH. After the removal of covers and selected bolts detailed visual inspections, eddy current inspections, and ultrasonic inspections are performed. On the upper wing surface some bolts are removed to get access to the inner wing structure for the video scope sensor. The critical locations at the closure rib, the kick rib, and at the spars and ribs can be monitored to get early information on potentially damaged structure.

After 5'000 FH (first service life) the test article was inspected during more than two months to assess the condition of the structure. This inspection was a major milestone in the test program. No fatigue damage was observed on fracture critical parts. The simulation of 5'000 FH corresponds to more than 16'000 FH on the US Navy test FT01/ST16/FT93.

At the end of a successful FSFT a residual strength test is required to demonstrate that cracks in the structure are not near criticality and thus the structure is still able to carry the residual strength load, which is general defined as 1.2 times the maximum spectrum load.

In the Swiss FSFT an alternative approach was adopted to demonstrate the residual strength capability at the end of the test; i.e. an additional 400 FH of spectrum cycles were applied on the FSFT article. These 400 FH were determined by specific crack growth analysis performed at four different locations of the structure.

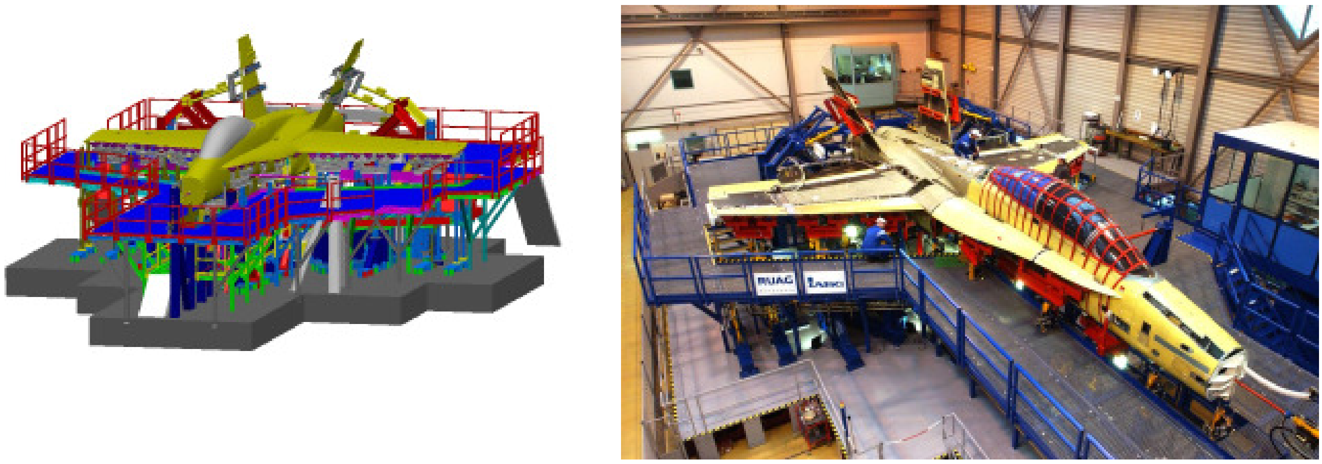


Figure with test set up (CATIA layout and photograph during cycling

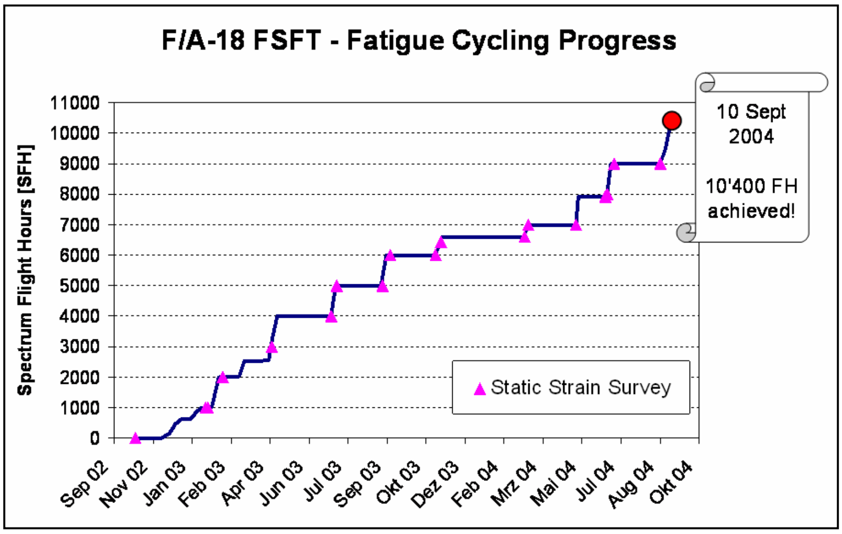


Figure shows the relatively short test time

Summary of Structural Deficiencies
During fatigue cycling and the tear down inspection only 7 Fracture Critical (FC) parts and 42 Maintenance Critical (MC) parts are affected by structural deficiencies. Overall 271 structural deficiencies were documented. Most affected parts were classified as non structural and absorbed most time of the inspection and analysis during testing. The distribution of deficiencies is listed in the table below.

Assembly	Total	MC Part	FC Part
Inner Wing	83	21	4
Outer Wing	31	7	2
Center Fuselage	127	7	1
Aft Fuselage	30	7	0
Total FSFT	**271**	**42**	**7**

Table with summary of ¨Note of Deficiencies¨

Pilatus PC-21 Full Scale Fatigue Test

The Pilatus PC-21 trainer (Figure 1) is a low-wing monoplane with a pressurized, stepped, tandem-seat cockpit. It is powered by a 1’600 HP turboprop engine. The aircraft is designed to satisfy the needs of the basic and the advanced pilot training. In addition to the high aerodynamic performance, it is equipped with a mission computer, which has more capacity than any other found in current generation training aircraft.

The primary structure of the aircraft is made of aluminum alloy in machined and sheet form (plate: 2124-T851, sheet: 2024-T3 and 2014-T42). The aircraft has to meet an operational load factor range of +8g and –4g for symmetric maneuvers. The required service life is 15’000 flight hours for the PC-21 design spectrum.

The Master Design Spectrum (MDS) consists of 3 types of missions (primary pilot training 50%, advanced pilot training 40%, tactical training 10%), 22 distinct design sorties, 36 unique types of maneuvers, and 69’129 events (i.e. combinations of flight parameters). The MDS represents blocks of 500 flight hours or 530 flights. Cockpit pressure is considered in the MDS in affected areas of the fuselage: 2 cycles from zero to operational pressure per flight. Landing loads are only considered by one particular load case at the end of each flight.

In order to certify the airplane to FAR 23, a full scale fatigue test was conducted (Figure 2). This full-scale fatigue test complied with damage tolerance requirements of MIL-81227.
Critical locations were defined based on the following criteria: on single load path, high load transfer of load paths, high stresses from FEM, large geometrical changes (high stress concentrations).
Summary of critical locations by analysis:

Wing 17 critical locations
Fuselage 15 critical locations
Horizontal Tail 7 critical locations
Vertical Tail 4 critical locations

Not having the detailed know-how or equipment for these complex tests, Pilatus awarded the contract for this work to IABG, a test institute based in the vicinity of Munich.

A representative test structure of all future series production aircraft – consisting of wing, fuselage, vertical tail, cockpit canopy and engine mount – was freighted to IABG, assembled locally and then rigged into a “torture chamber” made up of load rigs (shown blue in Figure 2), 24 hydraulic actuators to simulate manoeuvre loads (yellow) and a pneumatic system to simulate differential pressure in the cockpit. Load harnesses (red) distributed the cylinder loads evenly across the test structure. The monitoring system (in the foreground) ensured that the loads were applied on the correct cylinder in the correct sequence and at the right force. The test article was instrumented with 150 strain gauges.

Major Test Steps

- Durability Test of 2 x 5’000 FH
- Damage Tolerance Test of 1 X 15’000 FH
- Residual Strength Test with 1.2 x maximum load of specific load condition
- Tear Down Inspection

It took just 15 months see figure below to simulate the three service lives (equivalent to 45000 flying hours). The test spectrum is derived from the master design spectrum (22nd Symposium of ICAF, “Fatigue Development Program for the PC-21 Trainer Aircraft”). The spectrum represents a block of 500 flight hours. In order to shorten the test duration of the FSFT, the number of events of the master design spectrum was reduced to 58’035 events. This was done based on an analytical approach to insure that the two spectra create the same damage in the major structural parts, such as wing and fuselage.

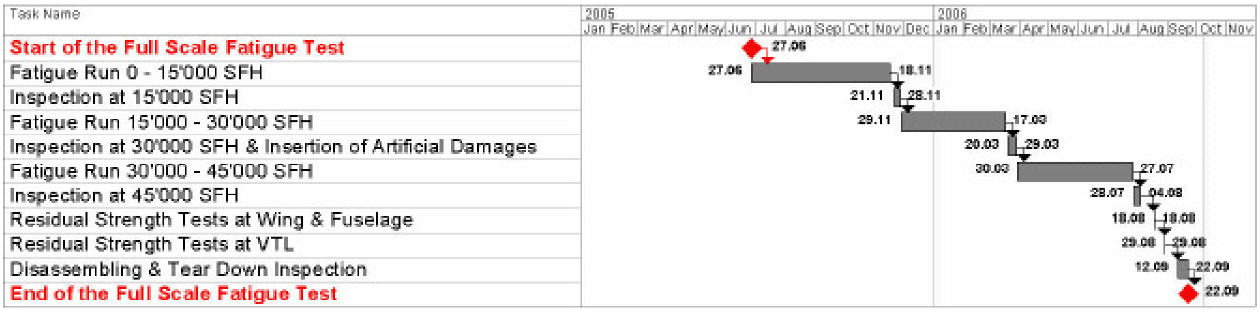


Figure of the test schedule

The simulation of 1’000 flight hours in the rig takes about 36 test hours. This is a test acceleration factor of 28.

The fatigue tests ran Monday to Friday for 16 hours every day, interrupted only by periodic inspections such as:

- Daily walk around inspection during test run: most damages were detected by this method because tension stresses open the cracks, making them more visible.
- 5000-hour inspection: after each block of 5000 hours the test was stopped for approximately two days for a visual inspection with a magnifying glass, or for non-destructive testing with Eddy-Current or Ultrasonic.
- 15000-hour inspection: after each block of 15000 hours the test was stopped for about a week and all access covers were removed for predominantly non-destructive testing with Eddy-Current or Ultrasonic.
- Tear down inspection: all bushes, bolts and rivets were removed in fatigue-sensitive joints and the joints were Eddy-Current tested.

The fatigue tests were conducted in three phases:

1. Durability Test: This test was designed to demonstrate the PC-21’s operational readiness and verify critical fatigue areas. The test structure was required to show no damage during 2 x 15000 simulated flight hours. Components that exhibited any signs of damage were modified. Major modifications were immediately incorporated into aircraft already in service and production. Our analysis models could be verified based on the inspection results and the measurement data of 165 strain gauges and 6 deflection transducers.

2. Damage Tolerance Test: After two tested service lives, 33 artificial damages were applied at safety-critical components. Notches were sawn into highly loaded fastener holes and lugs. The aim of this particular test was to show a slow crack growth and that in reality, any cracks will not run unstable until the next scheduled inspection, allowing for repair work.
3. Residual Strength Tests: At the end of the three simulated aircraft service lives, the test structure was put through a final round of static tests. Prior to these tests the artificial damages at the aircraft were exaggerated to such a degree that the faults were visible with the naked eye. For the 'grand finale', the loads – already at maximum level in reality – were then further increased by up to 20 % and the test structure was put through its paces one more time even though it was already battered by natural and artificial damage. It also withstood these trials.

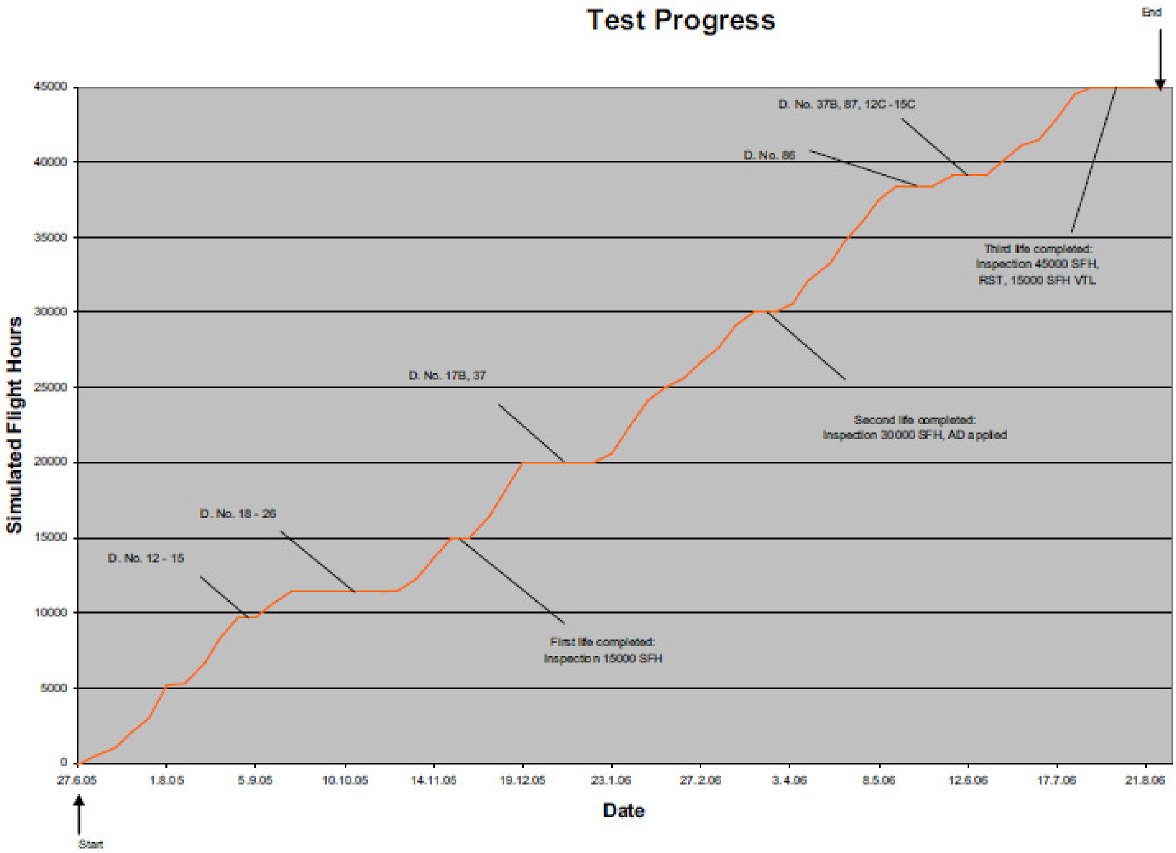


Figure with test progress and milestones (start on June 2005; completed in October 2006)

Summing up, the PC-21 passed the Full Scale Fatigue Test with excellent results, and the structurally weak points have now been identified and rectified. Appropriate adjustments have already been made to the series production aircraft ahead of the first aircraft being delivered to any of our customers. This will eliminate costly retrofitting during production or downtimes of our customer aircrafts.

After analysing of all test results we were able to define the inspection intervals and methods for the maintenance manual. We are delighted that the Full Scale Fatigue Test has helped to further maximise the reliability and cost-effectiveness of PC-21 maintenance work.

Figures of PC-21 and the test set-up at IABG in Ottobrunn

After completion of the FSFT, the test results of all critical locations were reviewed. For most locations the CI and CG analyses were adjusted to the FSFT results by means of a so-called “pegging procedure”. The following cases were considered:

Case 1: Crack detected at the end of FSFT
Case 2: No crack in durability test, but in damage tolerance test
Case 3: No crack detected at the end of FSFT
Case 4: Crack detected in durability test
Case 5: No crack detected at the end of the FSFT and long analytical CI life

The goal of this procedure is to define inspection intervals that are based on crack growth analyses adjusted to the FSFT results. In total, 150 critical locations were analyzed.

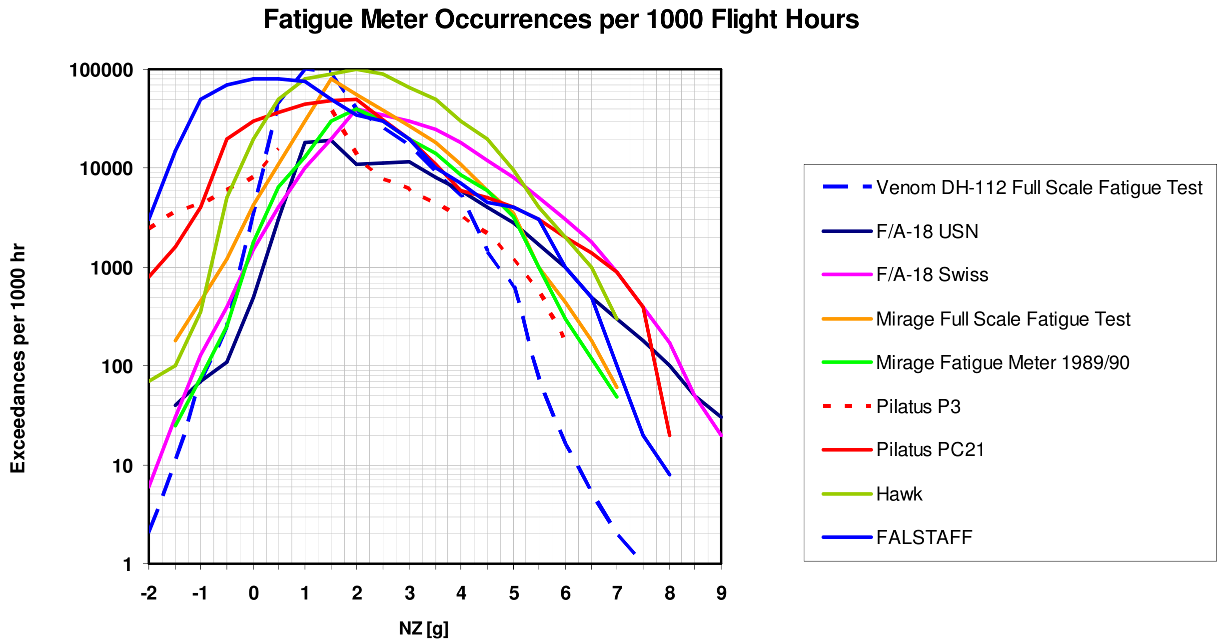
During the test 63 fatigue damages were reported. In the tear down inspection additional 6 damages were observed.

Conclusion

Over 60 years military aircraft evolution it is interesting to compare the occurrences measured through the fatigue meter on Swiss trainer and Swiss fighter aircrafts.

The diagram below shows that the trainer aircrafts exhibit similar spectra like the fighter of the same generation, at least in the positive normal acceleration ¨g¨ domain. Between the Venom, the Mirage and the F/A-18 generations a clear evolution of the occurrences shape can be observed. In the positive ¨g¨ domain the shape becomes unambiguously wider. For instance with an exceedance of 60 the Venom was flying 5.5 g, the Mirage 7 g and the F/A-18 flies 8.5 g. The Venom, which is the oldest fighter in this set, seems to have very narrow exceedance shape, showing clear limits in the negative g as well in the positive g domain.

Impressive too is the difference between the USN and the Swiss spectrum by the F/A-18. More than one g shift in the level of high occurrences characterises the severity of the Swiss flight conditions (app. 3 times more severe). Therefore the Swiss F/A-18 was redesigned to meet the fatigue requirements. This fact motivated RUAG Aerospace (before F+W Emmen) to undertake a complete full scale fatigue test, like described in a previous chapter, in order to ensure a safe life and an economical usage of the aircraft for the Swiss Air Force.



The Nz exceedance curves per 1’000 FH demonstrate the changes of technologies in the last 60 years:

- Higher Nz accelerations and more exceedances due to new materials with higher strength, fly by wire, higher thrust (advanced trainers, multi role fighters with much higher maneuverability)
- Longer durability life due to fatigue design and better materials

Considering the next table below one can remark an interesting evolution of the aircraft life and the technology of the full scale fatigue tests.

The aircraft life has been multiplied with a safety factor between 2 and 4 from one aircraft generation to the other. On the same time the test speed escalated significantly, so that at the end the test duration remained roughly the same. Only exception in this trend is the Mirage full scale fatigue test, where the unexpected crash of the left wing, which has happen very soon, interrupted the normal progress of the test. Overall four wings were tested, only the refurbished wing showed after 6286 flight hours only minor cracks, all others failed during testing.

Also interesting is the rising of number of the jacks between Mirage and F/A-18 test. A very precise loading of the control surfaces, inboard and outboard leading edge flaps, trailing edge flap, aileron and horizontal tails required 16 jacks, even when the Mirage with its much simple system of control surfaces needed only a few jacks.

Now a days the structure of the test article is monitored using a lot of strain gauges. More and more gauges are continuously recorded to get out more information and find early changes in the stress due to damages. An excellent access and close visual inspection to the test structure during cycling is very important.

Test	A/C Life* FH	Test FH	Test Duration Months	Test Speed h for 1000 FH	No of Actuators	No. of Gauges	Cycles for 200 FH
P3	2’500	5’000	14	333	19	--	18’400
Venom	500	5’000	24	600	24	--	39’000
Mirage	1’000	21’000	100	860	38	1’000	40’500
F/A-18	5’000	10’400	21	90	68	1’200	26’000
PC-21	15’000	45’000	15	36	24	150	27’650

* Original Design Life determined by OEM.

Due to more realistic testing and better inspection methods using several NDT methods more damages were found. Even with materials with higher strength and detailed fatigue guidelines and modern structural analysis capabilities the number of damages has increased considerably.
Even the redesign for the Swiss F/A-18 using the ASIP approach showed cracks at FC and MC parts. Al test showed cracks in the wing structure at the wing root area (spars, ribs). This area is the most loaded structure on an aircraft and requires careful detailed design and material selection.

Future tests with hybrid structures of metal and composite will be even more challenging than the presented tests in this review. More research will be needed to ensure the structural integrity with efficient full scale testing.

5.1 References

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