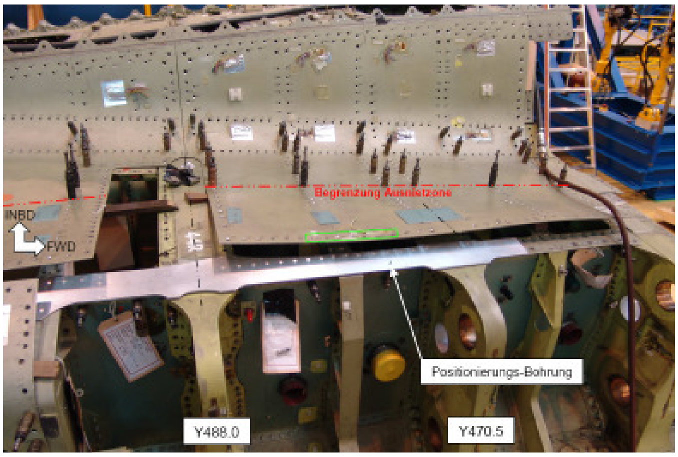
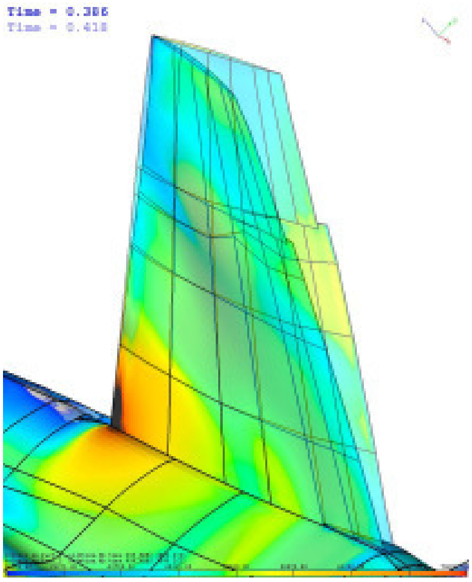
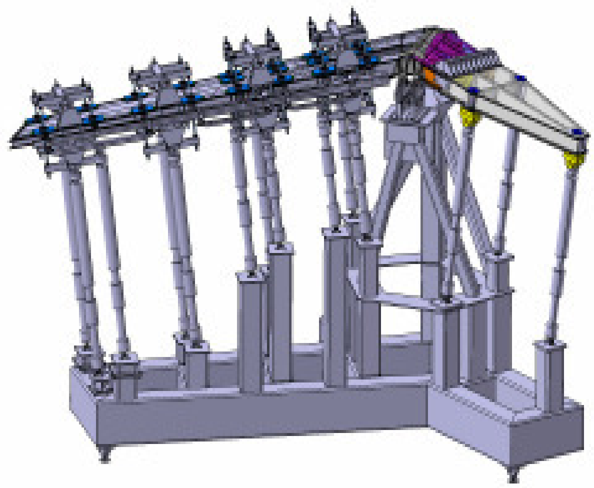
RUAG
Aerospace Defence Technology

Review of Aeronautical Fatigue Investigations in Switzerland

May 2007 – April 2009

KOR-TA2009-0101

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SUMMARY

The Swiss review summarizes fatigue work in Switzerland. It includes main contributions from the RUAG Aerospace AG (RA), CFS Engineering, M@M Mandanis GmbH, SMR, 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 31th ICAF Conference
Rotterdam, 25 and 26 May 2009

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 2007 till April 2009. The various contributions to this review come from the following sources:

- RUAG Aerospace AG; Structural Engineering, Fatigue Engineering, Aerodynamic Engineering
  A. Albrecht, U. Bartlome, A. Bertsch, B. Bucher, S. Büsser, A. Gehri, M. Giger, M. Godinat, M. Guillaume, C. Huber, R. Imhof, I. Kongshavn, C. Kunz, J. Lussi, R. Nebel, M. Nievergelt, R. Meyer, P. Pelloquin, G. Peikert, R. Renggli, B. Schmid, P. Stephani, A. Uebersax, R. Wagner, S. Zehnder
- Pilatus Aircraft Ltd; Structural Engineering, Stans
  D. Romancuk, L. Schmid, M. Gottier
- CFS Engineering, Lausanne
  J. Vos
- M@M Mandanis GmbH, Kriens
  G. Mandanis
- SMR S. A. Engineering & Development, Biel
  T. Ludwig, S. Merazzi

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 PC-12 Life Extension Program

D. Romancuk; Pilatus Aircraft Ltd.

The Pilatus PC-12 is a single engine turbo prop aircraft certified in 1994 as a FAR Part 23 normal category business and multipurpose aircraft. Currently, the fleet comprises of more than 800 aircraft in 34 countries, which have accumulated a total of more than 2 million flying hours.

The fatigue certification of the PC-12 was performed according to safe-life requirements. In 2007, a fatigue life extension program (LEP) was initiated due to fleet leaders approaching the initially certified life limit of 20’000 flying hours or 27’000 flight cycles.

The first of two stages of the LEP was completed in 2008. It comprised an extension of the safe-life limit based on the tear down inspection results of the full scale fatigue test (FSFT) article, fleet experience such as fleet usage (flown hours and flight cycles) and fleet findings (fatigue cracks, impact damage, corrosion, etc.) and based on a re-evaluation of assumptions for the flight mission and loads. This resulted in an update of the fatigue substantiation, which covered a certified life limit of 25’000 flying hours or 30’000 flight cycles. The fatigue substantiation is based on an analytical approach using crack initiation analyses in combination with the FSFT results.

The second stage of the LEP is currently in progress and planned to be completed by the end of 2009. It consists of a further fatigue life extension, which is based on damage tolerance (DT) principles. A supplemental structural inspection document (SSID) will be developed. The intention of the SSID is to supplement the existing maintenance program and to cover fatigue and corrosion aspects beyond the current life limit. The SSID defines inspection intervals, methods and procedures. This program will have a limit of validity (LOV), i.e. it will support continued safe operation up to 50’000 flying hours or 60’000 flight cycles. The major elements for the development of the SSID are:

- Review of fleet experience.
- Review of modifications (factory options, design changes, service bulletins, repairs, STC’s).
- Inspection definitions based on DT analyses and non-destructive inspection (NDI) procedures.
- Development of a supplemental corrosion prevention and control program (CPCP).
- Development of a DT repair concept.
- Review of the continued airworthiness process to make sure that the SSID is kept up to date.
- Review of non-structural systems with respect to aging problems.

Figure 4.2.1: PC-12 during take-off (left) and FSFT article during tear down inspection (right)

4.3 PC-21 Fatigue Monitoring System

L. Schmid; Pilatus Aircraft Ltd.

The Pilatus PC-21 is a single engine turbo prop aircraft certified in 2006 as a FAR Part 23 acrobatic category aircraft serving as a military advanced training aircraft. The service life corresponds to 15'000 flight hours and 27'500 landings.

The PC-21 was certified to damage tolerance requirements. The fatigue approach to certification consisted of the following major elements:

- Master Design Spectrum (MDS) development: A deterministic design spectrum consisting of primary, advanced, and tactical training missions was created. The severity of the MDS vertical acceleration spectrum was shown to be equivalent to FALSTAFF.
- Full-Scale Fatigue Test (FSFT): 2 lives of durability and 1 life of damage tolerant testing were simulated, followed by residual strength testing and tear-down inspections.
- Analytical Evaluation: The analyses conducted before the FSFT were used to pre-assess the airframe. After the FSFT, the test results were evaluated analytically in order to define inspection intervals (see specific paragraph).
- The aircraft inspection program is based on damage tolerance analysis and uses advanced non-destructive inspection technology.
- The Fatigue Monitoring System (FMS) is included in the dedicated Health and Usage Monitoring System (HUMS), which is also used to monitor systems performance.

The FMS is certified using guidance of AC 27 MG 15, because no regulations exist for such systems on FAR 23 aircraft. The fatigue usage is monitored by means of strain gauges located on major structural assemblies and expressed by so-called Fatigue Indices (FIs). Initial results of the operational fleet are promising and indicate a data capture rate in excess of 90%.

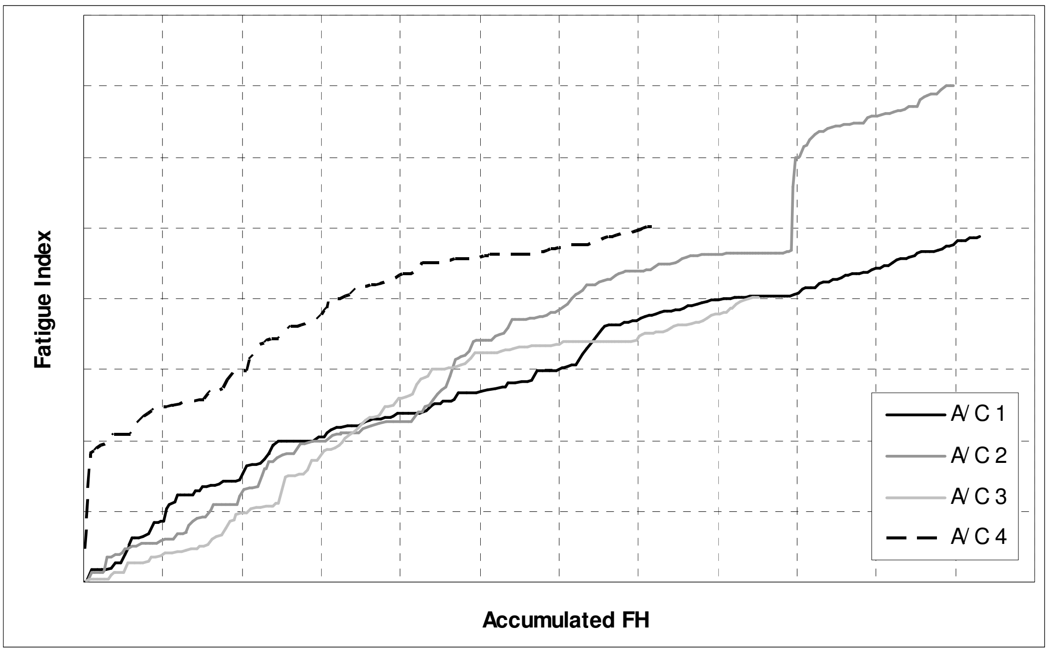


Figure 4.3.1: Typical output of the PC-21 Fatigue Monitoring System for four sample aircraft

4.4 PC-21 Definition of Inspection Intervals

M. Gottier; Gottier Engineering, Consultant for Pilatus Engineering Ltd.

The design of the PC-21 a/c is performed on an analytical approach by means of the Crack Initiation (CI) analysis and the material data, which were specifically determined for the PC-21 project. The required design life is 60'000 flight-hours, i.e. a safety factor of at least 4 had to be demonstrated.
In addition, Crack Growth (CG) analyses were performed for the most critical locations. A minimum safety factor of 2 was demonstrated for this design phase resulting in a target CG life of 30'000 hours.
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.

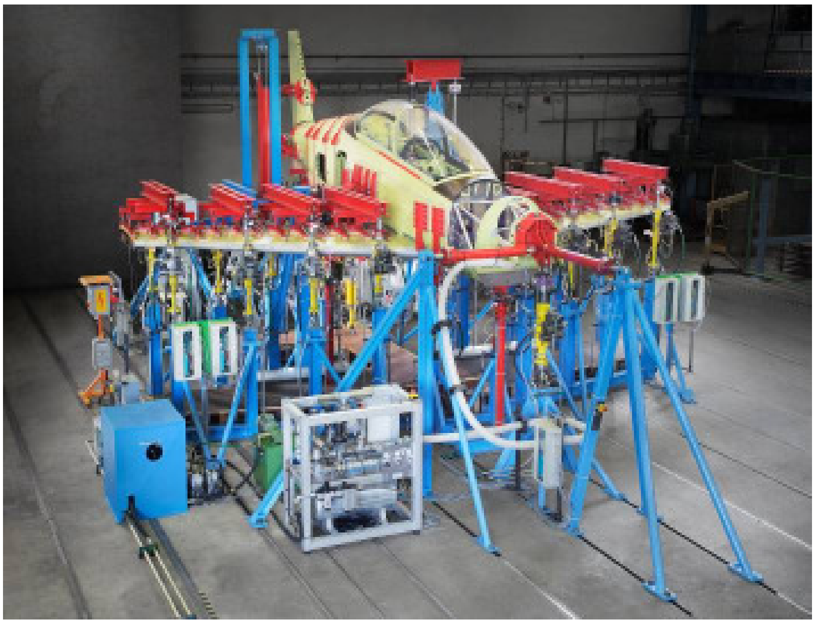


Figure 4.4.1: PC-21 FSFT (left) and PC-21 a/c usage (right)

4.5 Calculation of Buffeting Loads using unsteady Fluid Structure Interaction

M. Guillaume, A. Gehri, P. Stephani; RUAG Aerospace AG, J. Vos; CFS Engineering, G. Mandanis; M@M, T. Ludwig, S. Merazzi; SMR

For the validation of the F/A-18 Swiss Redesign a full scale fatigue test was carried out at RUAG Aerospace AG. Only few relevant fatigue load cases for the entire airplane were obtained from The Boeing Company in St. Louis, the F/A-18 Original Equipment Manufacturer (OEM).

This situation pushed RUAG Aerospace AG to search for methods to generate independently aerodynamic loads for the F/A-18 and as a result a considerable investment was made in the further development of the Computational Fluid Dynamics (CFD) solver NSMB (Navier Stokes Multiblock Solver; inhouse code), and the application of this solver to steady and unsteady simulations.
To take the structural response of the structure into account a Fluid Structure Interaction (FSI) tool was developed. In a first step only static aero-elastic deformation was considered, in a second step the tool was extended to permit dynamic fluid structure interaction simulations for the analysis of the F/A-18 Vertical Tail buffeting. This novel tool permitted to better understand the complicated flow field over the entire F/A-18 full flight envelope and to check some load cases delivered by the OEM. The FSI tool follows the concepts developed by Farhat et al. in which the structural part is solved using a modal integration with the Newmark method. A semi-implicit coupling approach is used between the fluid and structural mechanics solvers in which the structural mechanics solver is called in the inner-loop of the dual time stepping method used by the CFD solver.
The unsteady aero-elastic coupling tool was validated using the data from the well known AGARD445.6 wing. The results were in full agreement with the experimental data.

The NSMB flow solver includes a re-meshing procedure since 2004, which was improved in 2005 and 2006. The initial re-meshing procedure was based on the Volume Spline Interpolation method. In this method the new coordinates are computed using the surface displacements at a number of so called prescribed points. The main short coming of this algorithm is the large computational requirements (CPU time and memory), in particular for complex geometries such as the F/A-18. For this reason other methods were implemented and tested, as for example the Trans Finite Interpolation (TFI) method. In this method first the mesh on the block edges is generated followed by the generation of the mesh on the block faces. In the final step the volume mesh is generated from the mesh on the block faces. Bunching laws are all taken from the original un-deformed grid. Although this method is straightforward to implement and computationally fast it led to problems in regions where the geometry is not moving due to the fact that the implemented TFI method was not exactly the same as the TFI method from the mesh generator. This resulted in problems when using the coupled CFD-CSM method for the F/A-18. For example the calculations showed a sudden increase in pressure in the gap between Aileron and Trailing Edge Flap, which propagated down stream until it arrived at the Vertical Tail where the high pressure completely deformed the structure leading to a crash of the calculation.

The breakthrough was to use the TFI method on the displacement of the edges. The advantage of this method is that if the displacement of the edges is close to zero, the displacement in the volume will be close to zero as well, and as a result the original coordinates are unchanged. The second advantage of this procedure is that it is independent of the method used to generate the original coordinates.
The re-meshing method can be summarized as:

- compute the displacement of block edges using Volume Spline Interpolation
- use 2D Trans Finite Interpolation to generate the displacement of the coordinates on the block faces
- use 3D Trans Finite Interpolation to generate the displacement of the coordinates in the volume
- sum the coordinates and displacements to obtain the new mesh

The implementation of this algorithm in NSMB solver was straightforward, and removed the problems encountered with the coupled CFD-CSM calculations for the F/A-18 fighter.
Today, running in parallel, it takes between 10 to 25 CPU seconds to generate a new mesh for the F/A-18 geometry, and this mesh has sufficient quality to continue the calculation.

Unsteady Coupling Strategy
The unsteady coupled CFD-CSM calculations all employed the same numerical approach, used before when performing unsteady calculations without the movement of the Vertical Tail.
The time integration was made using the dual time stepping method of Jameson in which the outer loop is used for the advancement in time, and an inner time-stepping loop to solve the equations at each time step. The inner time-stepping loop employed the same numerical method as far the steady state calculations. The Detached Eddy Simulation (DES) approach was used to model the unsteady turbulence.

Three strategies for the CFD-CSM coupling procedure were investigated:

- 1 MI: the coupling is made only each outer time step
- ALL MI: the coupling is made each inner time step
- STD MI: the coupling is made the inner time steps which yields an integer when taking the square root (inner time steps: 1,4,9,25, 36, etc.)

No differences could be observed for the AGARD445.6 wing when comparing the STD and ALL MI results indicating that for this calculation is not need to perform the CFD-CSM coupling each inner time step. However, performing the coupling only each outer time step appeared to be insufficient.

The structural model used for the F/A-18 Vertical Tail was the same as prepared for transient analysis. The initial version of the structural modal included 100 mode shapes, of which only the first 5 mode shapes (¨Eigenmodes¨) were used for these calculations.

Transonic Load Case (M = 0.9)
The load case has the following parameters:

Mach	Alt feet	AoA deg	LEF deg	TEF deg	HSTAB deg
0.9	5’000	0.0	0.0	0.0	0.0

This load case is not in the buffeting environment and so only a small well damped displacement will be expected.
The coupled CFD-CSM calculation was about twice as expensive as the uncoupled calculation. Time spend in the re-meshing algorithm was between 4 and 30 seconds.
The run was interrupted after 0.3 seconds real time, which took 28 days to compute and generated close to 700 Gbytes of data.
To analyze the movement of the Vertical Tail, 5 points were selected which are shown on figure 4.5.1. Points 1 and 2 are on the top of the Vertical Tail, points 3 and 4 near the gap between Rudder and Vertical Tail, the other points are near the trailing edge.
Most dominant is the movement in direction of the outboard or the inboard side, there the initial perturbation is clearly visible when starting the coupling calculation. This initial perturbation is followed by an oscillatory movement of the Vertical Tail, which is damped out in time. It can be observed that the asymptotic position to which the Vertical Tail seems to move is not the unperturbed position, but one which is slightly displaced.
The unsteady simulation without CSM coupling for this load case showed a steady flow on the Vertical Tail, which is confirmed by the fully coupled calculation. The initial perturbation at the start of the coupled simulation is damped out in time, indicating that there is no mechanism which maintains an unsteady flow behaviour.

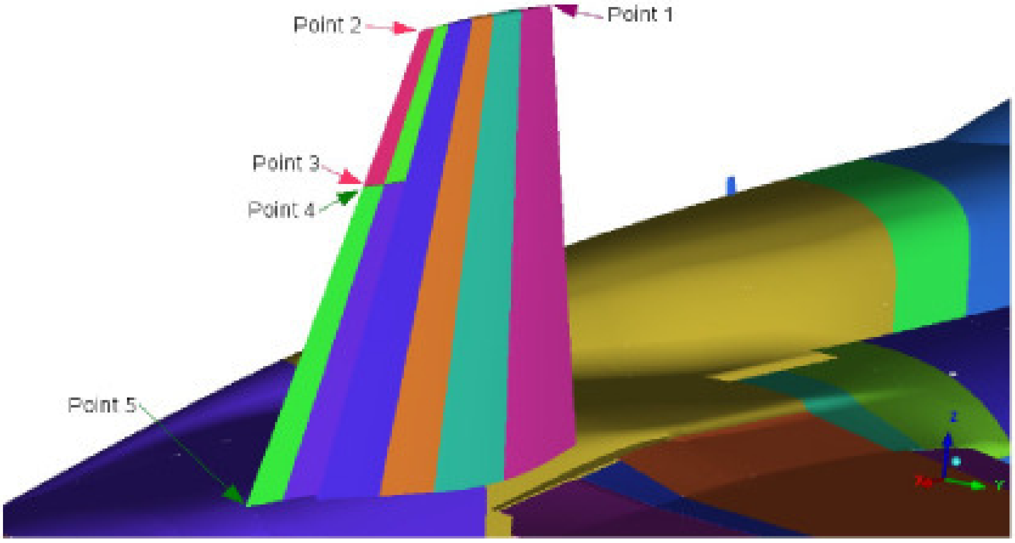


Figure 4.5.1: Five Selected points at the vertical tail

Buffeting Load Case C2S825

The load case C2S825 is indeed affected by severe buffeting based on the flight test results performed by The Boeing Company for the assessment of dynamic impact on the Vertical Tail. The load case is characterized by following parameters:

Mach	Alt feet	AoA deg	LEF deg	TEF deg	HSTAB deg
0.7	15’000	26.6	26.0	0.0	-6.1

As for the transonic load case 5 points were selected to show the movement of the Vertical Tail. The DES variant of the k-ω Menter Shear Stress model was used for the calculation, and 4 seconds of real time was simulated. Figure 4.5.2 shows the movement of point 2 (located on top of the vertical tail) in the y-direction as function of time, and one can observe important movements which are not damped in time. Figure 4.5.3 shows the position of the Vertical Tail with the pressure contours on top at 2 different time instances. The bending and torsion of the vertical tail can be clearly observed.

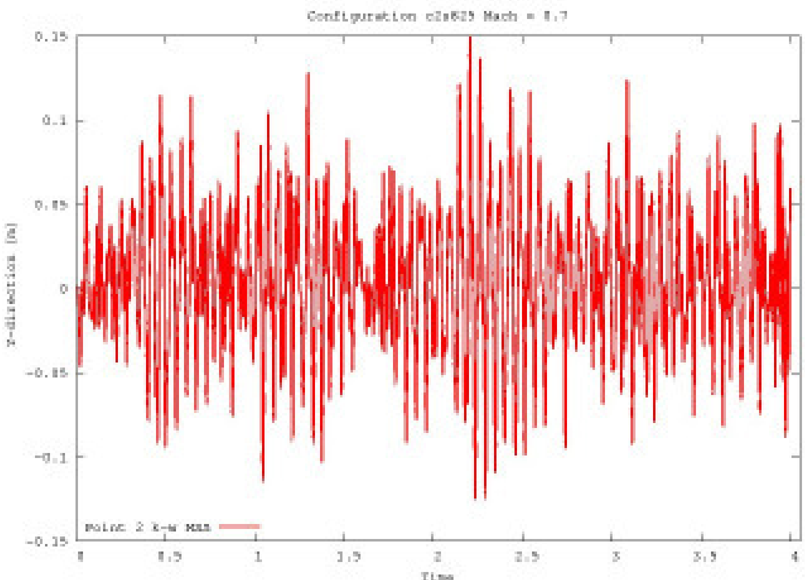


Figure 4.5.2: Movement of point 2 as function of time (4 sec), load case C2S825

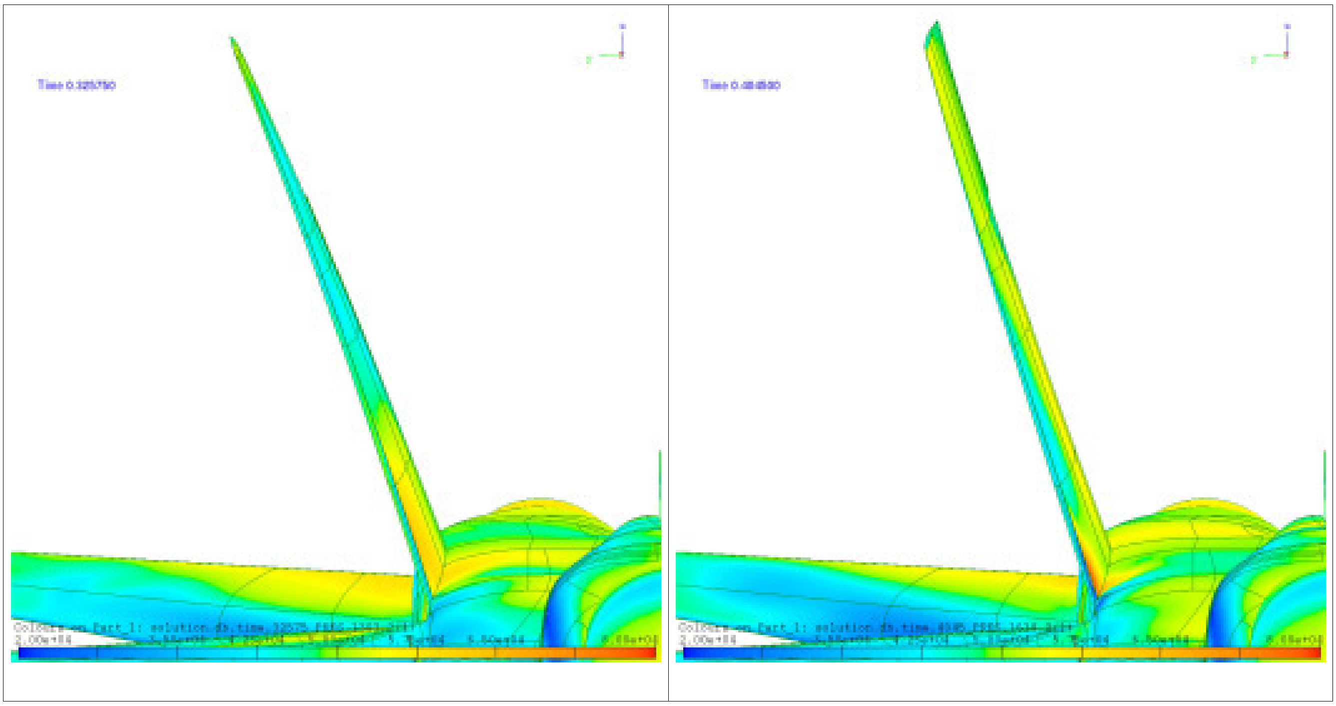


Figure 4.5.3: Pressure on vertical tail and position of vertical tail at 2 different times.

Conclusions

Component loads for structural and fatigue analysis were calculated using the RUAG in-house CFD solver NSMB. The calculated loads were in good agreement with the flight loads data.

The interaction of the aerodynamic pressure over wing with the structural stiffness is important and must be considered for the loads calculation using fluid structure interaction.

With today's computer performance an unsteady state CFD calculation brings more information into buffeting and flutter behaviour of modern airplanes. With the NSMB unsteady capabilities real flow field for 4.0 seconds over the F/A-18 were processed, see figure 4.5.3.

The buffeting impact for the F/A-18 Vertical Tail was simulated using unsteady aero-elastic coupling algorithm. This method uses implicit conventional serial staggered coupling in the inner loop. An efficient re-meshing procedure within the flow solver is important for complex geometries such as a F/A-18 fighter. The CPU time and memory is still very high but can be managed with today's computer capacity but further improvements are needed.

The tool chain for CFD-CSM unsteady coupling was validated by the experimental data for AGARD445.6 wing from the literature.

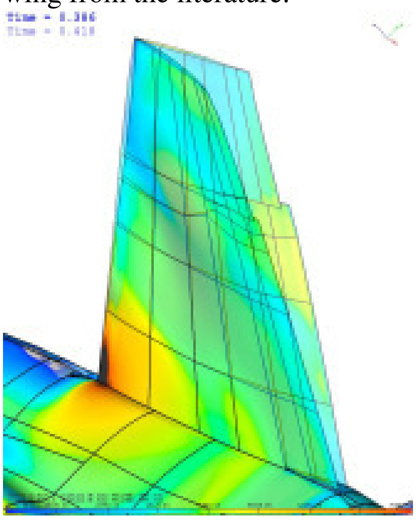


Figure 4.5.4: Movement for Load case C2S825 at two time steps, at the tip 150 mm were observed

First successful calculations for buffeting load cases were done for the F/A-18 Vertical Tail. These results will be used to develop a fatigue spectrum for the Swiss F/A-18 usage under buffeting environment condition to assess the structural integrity.
The buffeting and flutter should be addressed in an early design phase of a modern airplane. RUAG Aerospace AG CFD dynamic fluid structure coupling tool may provide an early answer to these problems.

Collaboration with Finland

In the frame work of a collaboration with Finland calculations are made for the F/A-18 fighter to compare results. Experiences and results of CFD calculations are exchanged with FinFlo AB, and regular meetings are organized. Current collaborative work is focusing on the influence of wing tip missiles, for first results see figure 4.5.5.

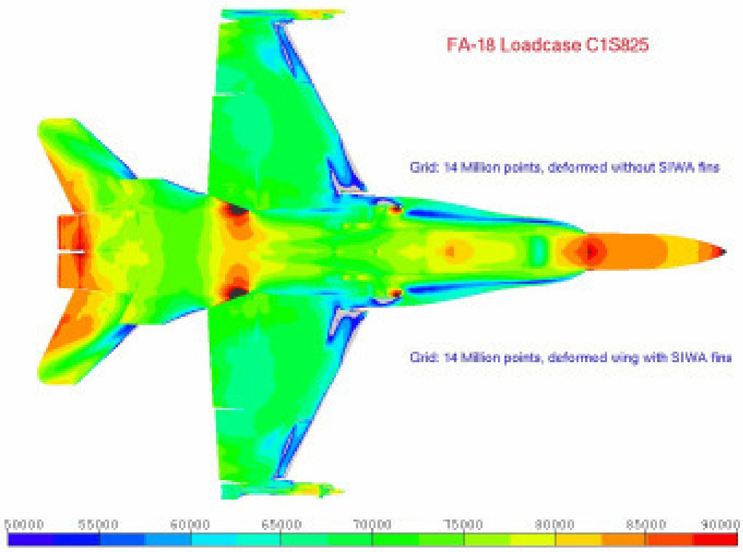


Figure 4.5.5: Influence of SIWA fins at the wing tip, see also difference of pressure distribution at the horizontal tail

4.6 Fatigiue Crack Growth Investigation of 7149-T73511 Aluminium Extrusion

M. Guillaume, R. Jaccard; RUAG Aerospace AG

Introduction

Fatigue crack growth rate (FCGR) testing was conducted on a 7149-T73511 aluminium extrusion (heat treatment per spec P.S. 15500 & P.S: 15500.8) using four geometries (center crack, corner crack, edge crack and compact tension). Both steady-state and variable amplitude tests were conducted in laboratory air at a temperature of 20°C and a relative humidity of 40%. The stress ratio varied from –1.0 to 0.65 for the steady-state tests. Variability among the different sample geometries was greater than expected. It was suspected that the main source of variability in test results was due to global residual stress. This required that measurements of crack closure and $K_{residual}$ be included and were an important part of this investigation. The testing was performed in accordance with the requirements of ASTM E647-05 “Standard Test Method for Fatigue Crack Growth Rates”.

Sample Preparation

The material was selected from the longeron geometry of upper outboard longeron in the aft center section (FS498) of the F/A-18 which showed several cracks in tear down inspection. The figure 4.6.1 shows the layout of various test samples on the specimen.

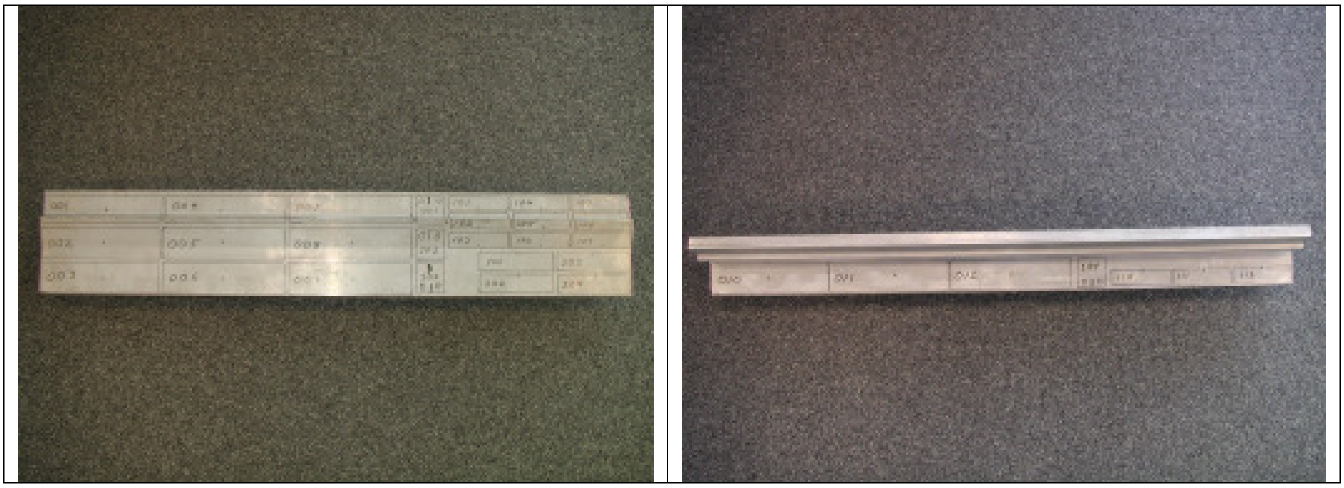


Figure 4.6.1: Layout of Test Samples

Steady State Testing

For the center crack and compact tension sample, the pre-cracking was done using a final constant K_{max} of ~4.0 to 5.0 ksi$\sqrt{inch}$ at a stress ratio of 0.1. The decreasing K tests (Region I) were started at 20 Hz and at crack growth rates of ~2-5 x 10-7 inch/cycle using a decreasing K-gradient of –6.0 1/in. Region II and III data were generated at 20 Hz using an increasing K-gradient of +6.0 1/in. Crack length verification and correction were done using the surface measurements. This information was used for the purpose of correlating the physically measured crack length with the compliance calculated crack length.

For the corner crack sample, the pre-cracking was done using a K_{max} of ~1.0 ksi$\sqrt{inch}$ at a stress ratio of –2.0. This pre-crack procedure provided an acceptable pre-crack at a maximum load less than that required for threshold at a higher stress ratio. Since the initial notch for the corner crack sample was triangular in shape, an equivalent crack was calculated based on the equivalent area of a semi-circular flaw. In each case, the samples were pre-cracked until the target crack length was indicated. At the completion of pre-cracking, the stress ratio was changed to the specified value and the test started at constant load and a cyclic test frequency of 20 Hz. If no crack growth was observed, or the rate of crack growth was less than 1 x 10-9 inches/cycle, the loads were increased ~5% and the test continued. Once sustained crack growth was observed, the loads were maintained constant for the duration of the test. No load shedding was performed during the test. Each test was terminated prior to failure so that the final crack length could be measured. The initial and final crack length measurements, along with the corresponding potential drop, were entered into the post-test analysis software to correct for any discrepancy between the physical crack length and the direct current potential difference method (DCPD) measured crack length.

Individual FCGR plots are showed in figure 4.6.2. These results including the linear correction factor applied to the normalized compliance or (DCPD) in order for the calculated crack length to match up with the measured crack length.

During testing, the special testing software calculated the effective stress intensity according to the ASTM opening load method. However, another methodology for determining the effective stress intensity was also used and is called the adjusted compliance ratio (ACR) method [1]. Crack growth rates are computed using a combination of the modified secant method and the seven point incremental polynomial technique. The first method is computed as follows:

$$da/dN = (ai+2-ai)/(Ni+2-Ni) \qquad (1)$$

and

$$aaverage = (ai+2+ai)/2 \qquad (2)$$

The second method smoothes the data but misses three points at the beginning and three points at the end of the data set. The combined method uses the seven-point method for the bulk of the data with the modified secant method used for the “missed” points in the beginning and end of the data set.

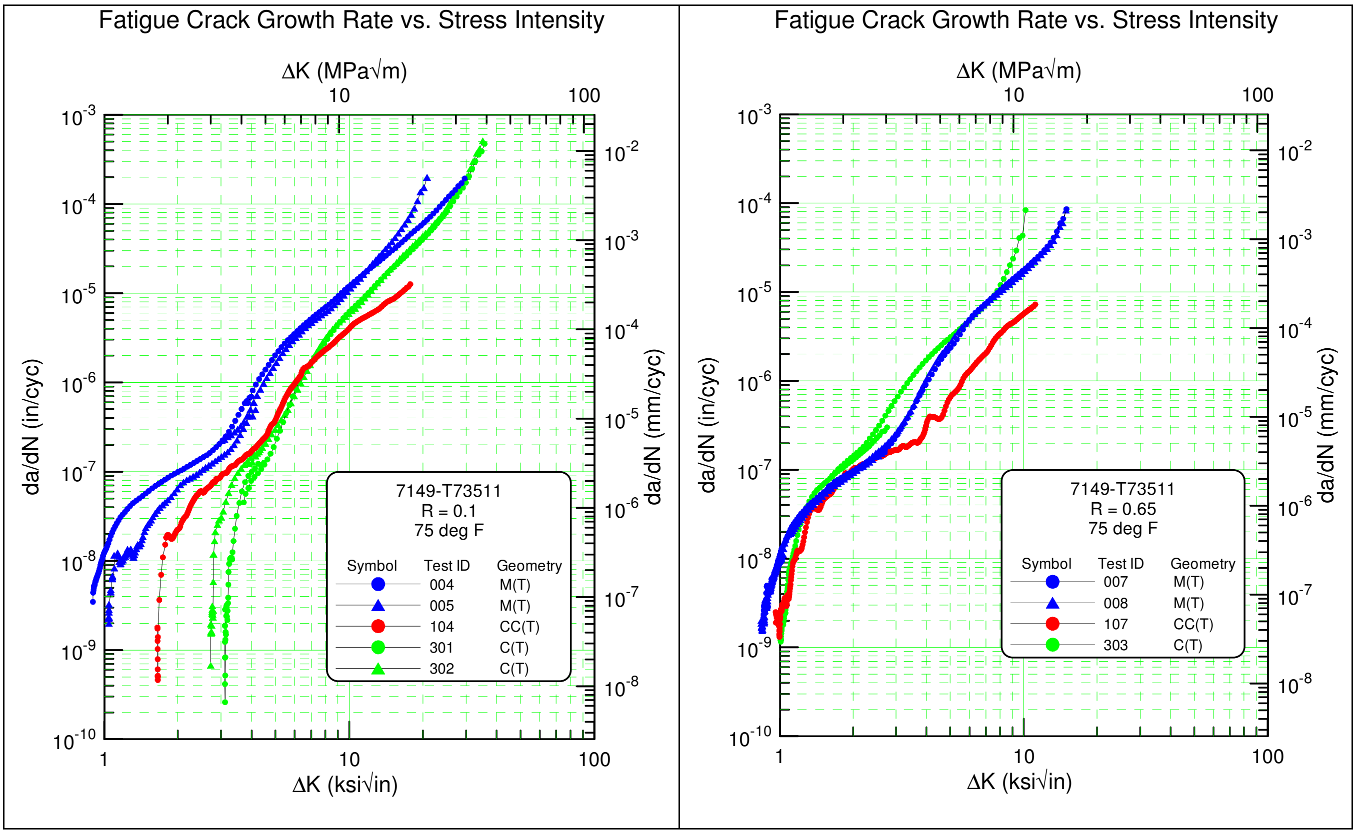


Figure 4.6.2: FCGR response for stress ratio 0.1 and 0.65 for center crack M(T), corner crack CC(T), compact tension C(T) specimen showing effect of sample type

The three different sample types give almost identical results at a stress ratio of 0.65. This is largely because, at a high stress ratio, all samples are closure free, whereas at a low stress ratio, compressive residual stress will clamp the crack shut and tensile residual stress keeps the crack fully open, even at minimum load.
The high R data are unaltered for the ACR method, but the R = 0.1 data have been shifted significantly to the left after correcting for crack closure.
In general, the ACR method does a reasonable removing most of the stress ratio effect by providing a ΔK effective curve free from remote closure effects.
An examination of the fatigue surface and visual crack length measurements revealed mostly uneven cracking associated with the low and negative stress ratio. This is most likely the result of residual stress. Furthermore, the compact tension samples had a tendency to grow out of plane.

$K_{residual}$ Determinations
Additional center crack and compact tension samples were tested using a constant ΔK test procedure and the methodology outlined in reference [2, 3] to determine the $K_{residual}$ profile as a function of crack length. A stress ratio of 0.1 was used and different levels of K_{max} ranging from 4.0 to 25.0 ksi$\sqrt{inch}$ were also used to characterize the behaviour over a range of crack growth rates. A comparison was made to the standard crack growth rates with evidence that residual stress played a major role in the center crack and compact tension test results. The linear correction factor was applied to the normalized compliance in order for the calculated crack length to match up with the measured crack length.

Variable Amplitude Testing
Three spectrum types were conducted on three sample geometries (center crack, corner crack and edge crack) to investigate variable amplitude behaviour.
Spectrum 1 is the baseline spectrum, of the Swiss Full Scale Fatigue Test at the upper outboard longeron FS498, and represents 200 SFH. Spectrum 2 consists out of 25 repetitions of spectrum 1 and some additional marker cycles after each repetition. The Block Spectrum is based on spectrum 1. The elements of the block spectrum shall be applied at random. The sequence of spectrum 1 and 2 is max peak to front. The sequence in spectrum 3 is randomized. (the terms are taken from the AFGROW Software). The AFGROW software was used to estimate the stress multiplication factors yielding a CG Life of about 4100 SFH.
These tests were conducted using either compliance or DCPD to monitor the crack length. The tests were started at the specified stress level without initiating a pre-crack. The 3D compensation feature of the analysis software provides endpoint levels within 0.1% of the full magnitude of the spectrum [3]. The average loading rate is constant, resulting in a test frequency inversely proportional to the amplitude. A damage parameter is tabulated for each data point and is an indicator of the agreement between the target load and the actual load for each endpoint. A value of 1.000 ± 0.002 represents excellent agreement. Individual FCGR plots are showed in figure 4.6.3 to 4.6.5 including the linear correction factor applied to the normalized compliance or DCPD in order for the calculated crack length to match up with the measured crack length.

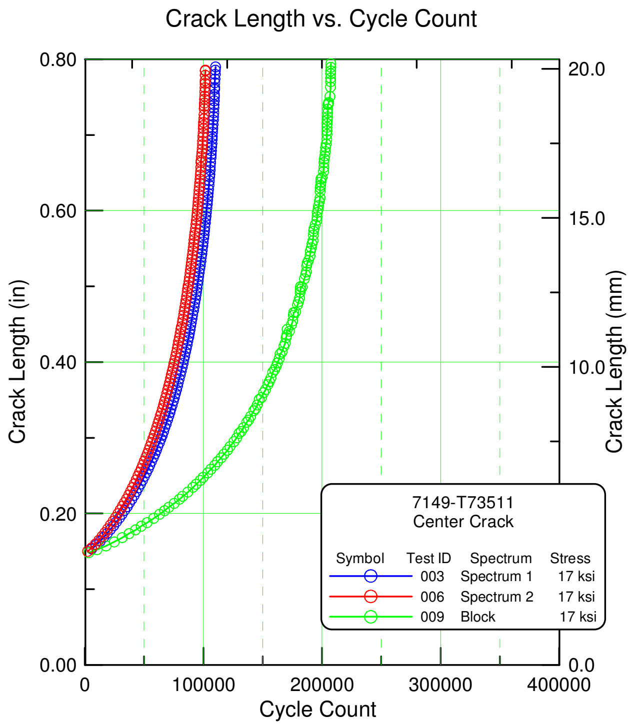


Figure 4.6.3: Crack length versus cycle count comparing spectrum types for the center crack sample

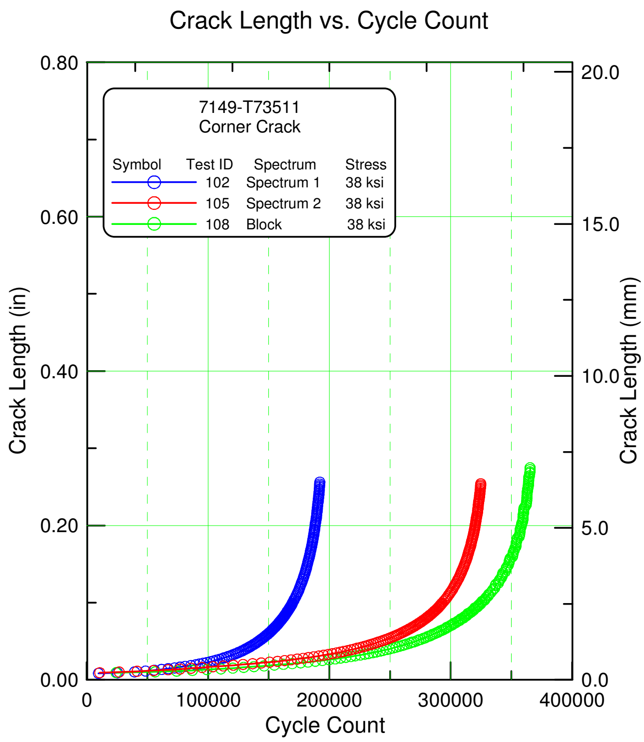


Figure 4.6.4: Crack length versus cycle count comparing spectrum types for the corner crack sample

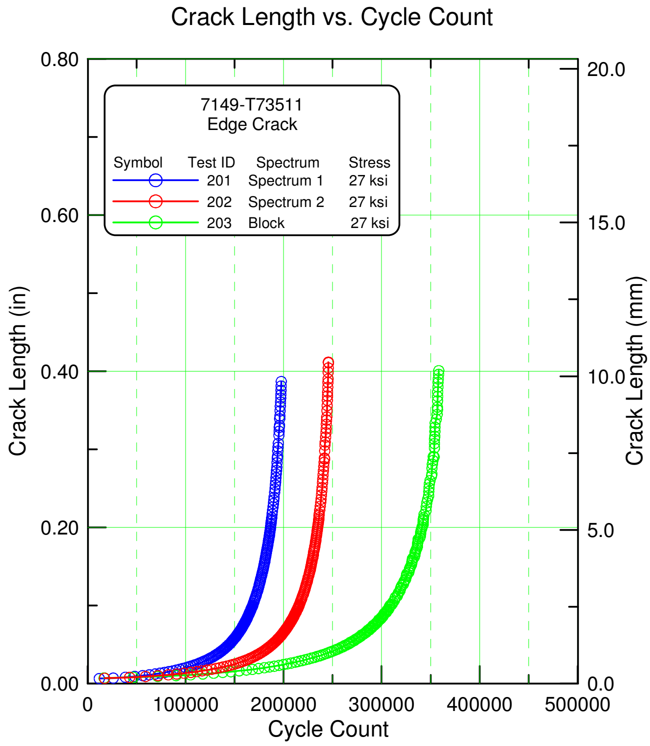


Figure 4.6.5: Crack length versus cycle count comparing spectrum types for the edge crack sample

Even the AFGROW software showed for all three spectrum a CG life of 4'100 SFH the experimental results differs for each sample geometry, see table 4.6.1.

Sample Typ	Spectrum	Test Life	AFGROW
Center Crack	Spectrum1	3831 FH	4100 FH
Center Crack	Spectrum2	3530 FH	4100 FH
Center Crack	Block	7207 FH	4100 FH
Corner Crack	Spectrum1	6675 FH	4100 FH
Corner Crack	Spectrum2	11276 FH	4100 FH
Corner Crack	Block	12694 FH	4100 FH
Edge Crack	Spectrum1	6868 FH	4100 FH
Edge Crack	Spectrum2	8538 FH	4100 FH
Edge Crack	Block	12450 FH	4100 FH

Table 4.6.1: Results for the three spectrum and the 3 different specimen; test life and AFGROW calculation.

All samples exhibited even cracking suggesting that residual stress was less of an issue with the variable amplitude tests than with the steady-state tests.
The results showed that residual stresses, specimen type and load interaction effects play an important role in the fatigue crack behaviour.
Fractographic investigations showed that striations are present in each specimen and each spectrum down to a few µm, see figures 4.6.6 till 4.6.8. The fractographic morphology can be analyzed in every detail which concludes that crack growth is present down to the smallest size. The concept of fracture mechanics could be extended to small scale but the response function e.g. stress intensity factor may not be a trivial function, the origin of the initial discontinuity will vary considerable. Further research will be needed to find an engineering approach to handle crack growth for small scale up to macro crack growth using well kwon fracture mechanic principles. Important will be the separation of the crack closure from the residual stress effects.
The material treatment is very important for conclusions to the real structure, the same processes had to be applied to gain real results and not ¨academic¨ laboratory results.

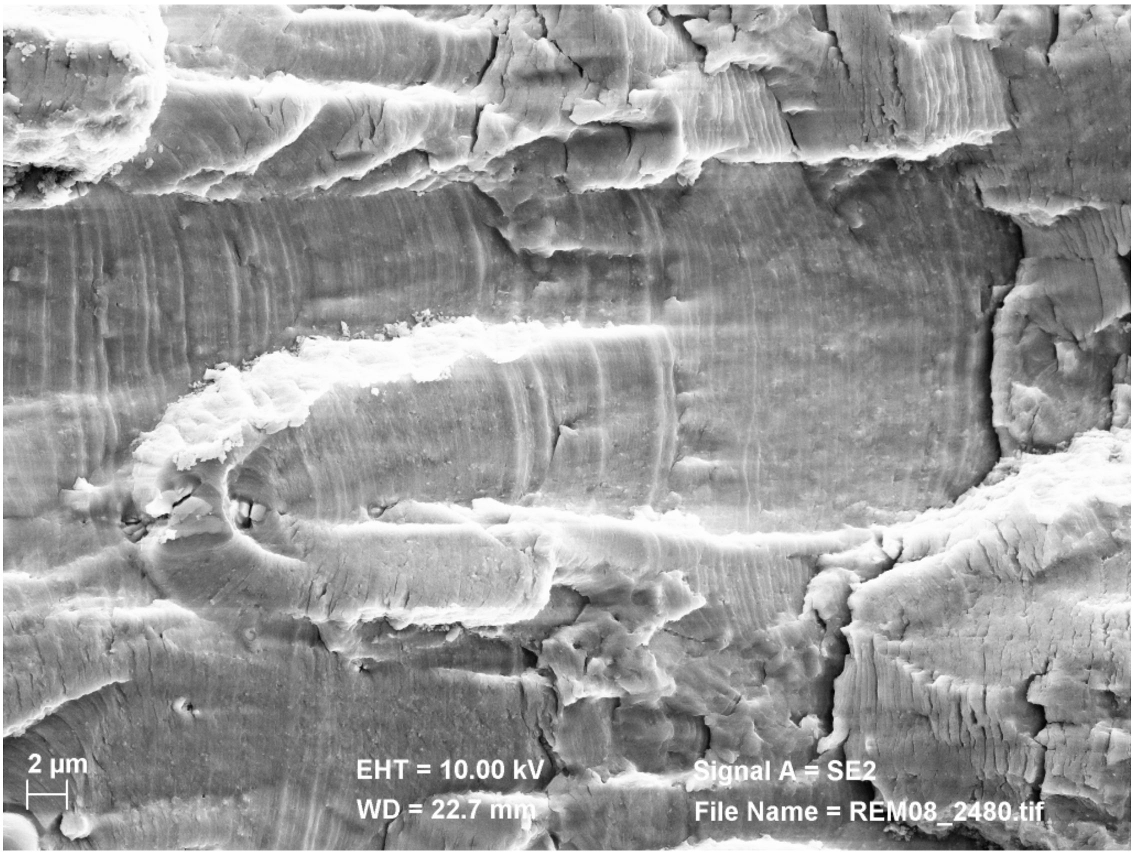


Figure 4.6.6: REM fractography of spectrum 2 and center crack sample (striations at level of 2 µm)

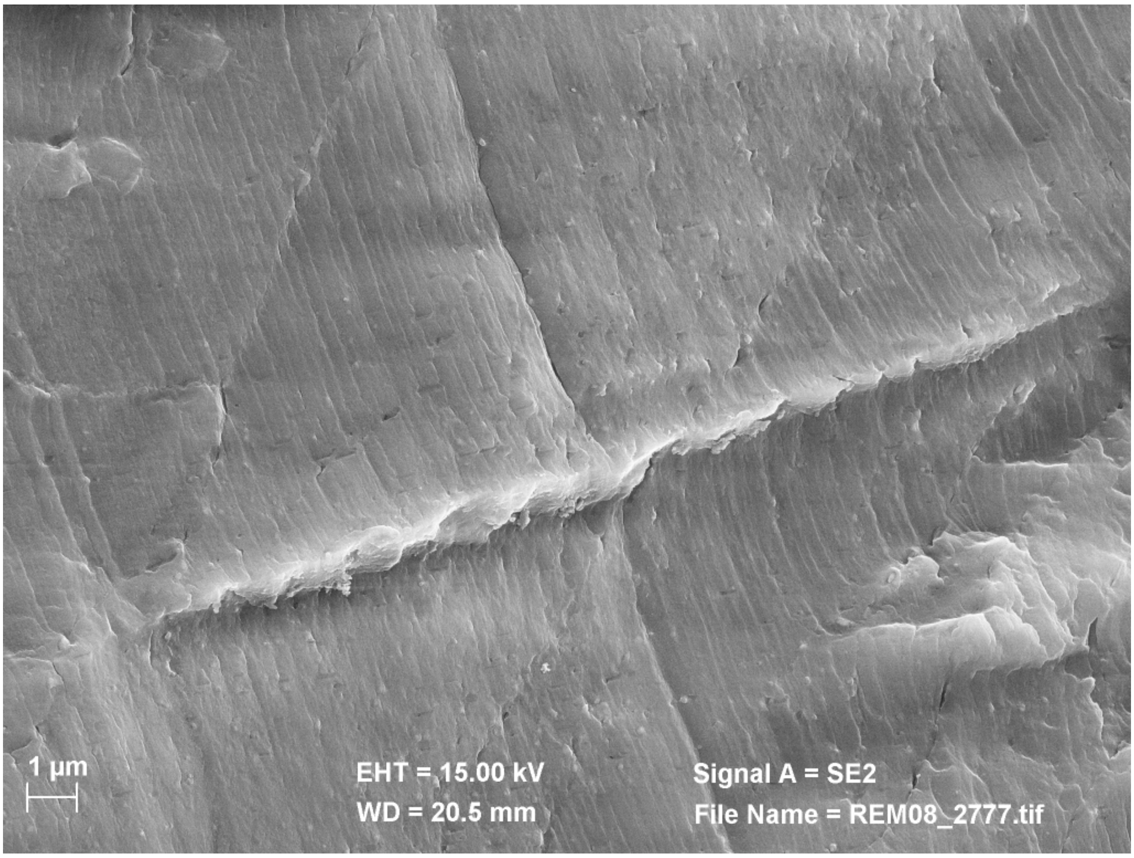


Figure 4.6.7: REM fractography of spectrum 2 and corner crack sample (striations at level of 1 µm)

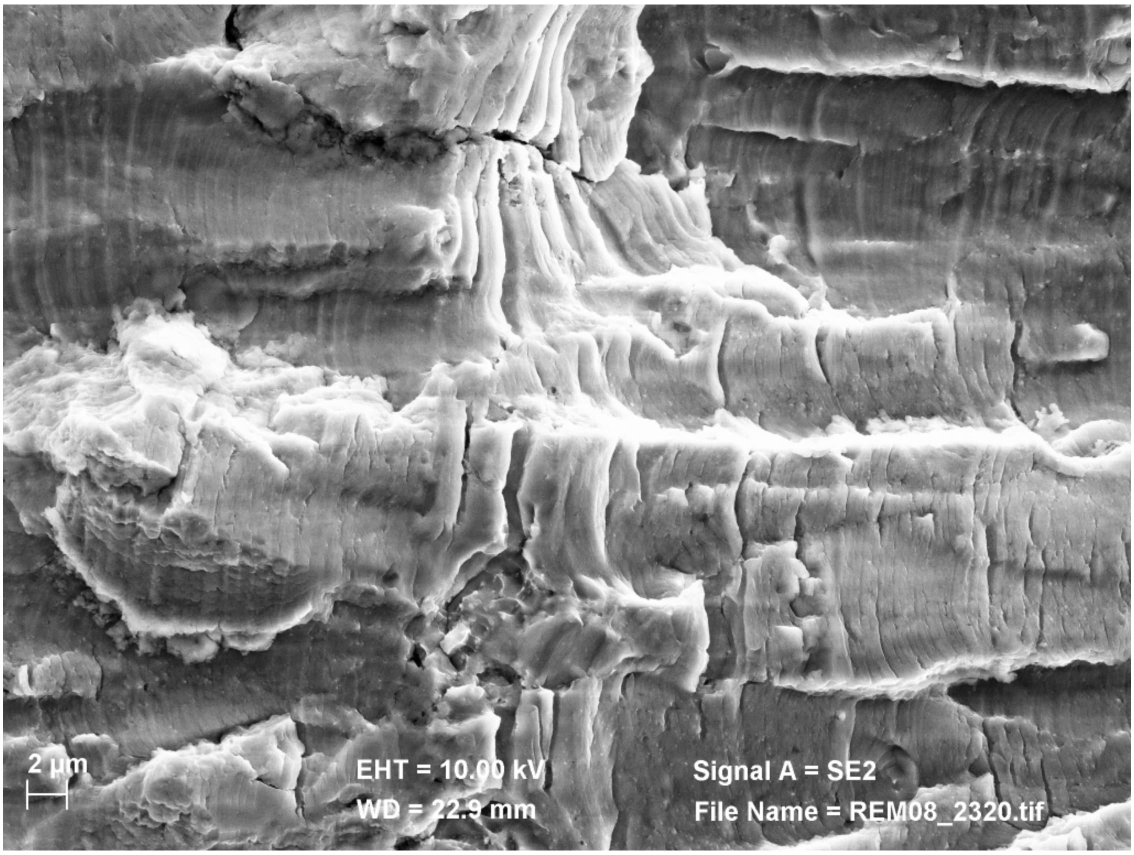


Figure 4.6.8: REM fractography of spectrum 2 and single edge corner crack sample (striations at level of 2 μm)

4.7 Structural Refurbishment Program

M. Nievergelt, R. Wagner, M. Godinat, P. Pelloquin, R. Meyer, B. Bucher, C. Kunz, I. Kongshavn, R. Nebel, R. Renggli, S. Büsser, A. Albrecht; RUAG Aerospace AG

Introduction

Starting in 2009, the Swiss F/A-18 Center Fuselage Structural Refurbishment Program (SRP) entered Phase B, installation of the prototype, having successfully completed the engineering analysis and technical data package in Phase A during 2007/2008.

The goal of the refurbishment program is to prevent cracking in parts of the center fuselage which require access to the fuel tank, where cracks were found in the Swiss Full Scale Fatigue Test (FSFT) and in other F/A-18 fleets. Repairs and replacements which require removal of the fuel bladder are labour intensive and costly. Furthermore, the bladders may become brittle in time and are more susceptible to being damaged during removal. The goal of the program is therefore to combine the refurbishment of all structural parts which require fuel tank access in a one-time Center Fuselage Structural Refurbishment Program. In particular, the program addresses the upper outboard longeron (UOL), fuel barrier web (FBW), fuel cell floor (FCF), strap door 44, side skin (SS) and crease longeron (CL).

The urgency of the program is primarily due to the UOL, in which cracks were detected in the Swiss FSFT, and for which fatigue lives of less than 1'000 flight hours have been predicted based on crack initiation (CI) and crack growth (CG) analyses. At the end of 2006, the fleet leader had already accumulated more than 1'400 flight hours.

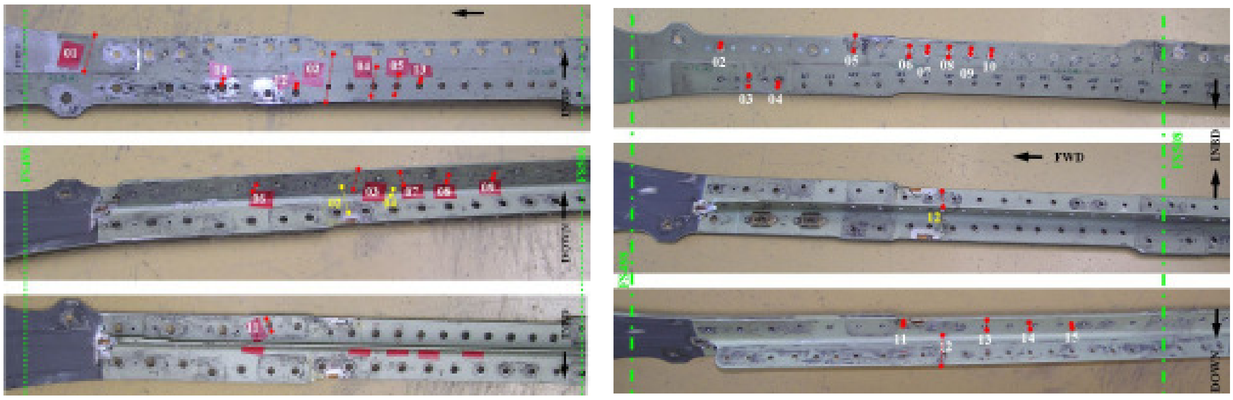
Despite the tight time constraints and the technical challenges of improving an already optimized design, Phase A of the program was successfully finished as planned by the end of 2008. The modifications are currently being installed in the prototype and the first article and fleet retrofit (Phase C) will be implemented in 2010.

RUAG Aerospace is conducting the Swiss F/A-18 Center Fuselage Structural Refurbishment Program on behalf of armasuisse and the Swiss Confederation.

Upper Outboard Longeron

C. Kunz, P. Pelloquin; RUAG Aerospace AG

During the Swiss F/A-18 full scale fatigue test (FSFT), cracks of up to 53 mm were discovered in the down standing leg of the right upper outboard longeron (UOL), first observed during the 7'000 hour inspection. During the final tear-down inspection after 10'400 hours, many fastener holes connecting the UOL to the skin and the fuel barrier web were also found to be cracked. After 7'000 hours of testing an aluminium strap was installed on the right hand UOL to be able to continue the test.



L/H-side R/H-side

Figure 4.7.1: FSFT cracks in the UOL between FS 488 and FS 508

For the structural refurbishment of the UOL for the fleet aircraft, a new strap was developed. Finite element analysis (FEM) and hand calculations were used to justify this modification. To determine the loading, the coarse center and aft fuselage model FEM4 was utilized. These loads were then applied to another detailed FEM. In the detailed FEM every fastener is modelled in the area of interest.

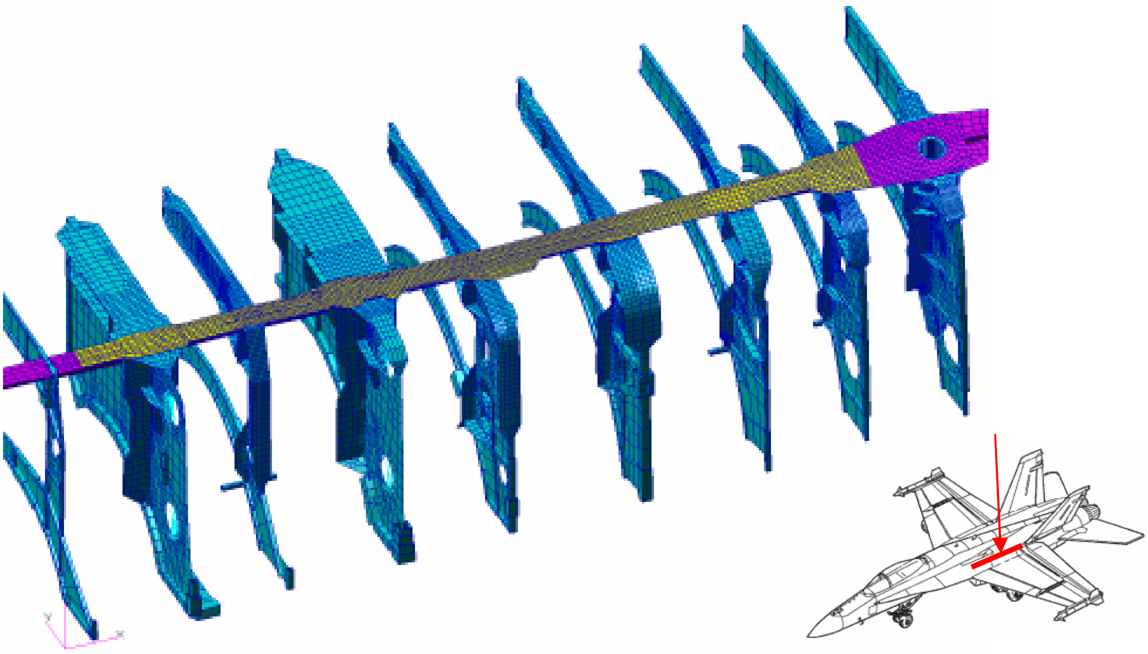


Figure 4.7.2: Detailed FEM shown without skins

After considering different design solutions and going through several optimization loops, a satisfactory solution for both static and fatigue criteria has been found. It consists mainly in the installation of a 63 inch long steel strap. The strap is attached to the UOL using the existing fastener locations. At highly loaded fastener holes interference fit fasteners are installed. In some locations the fastener holes are cold worked. For most of its shape, it follows the geometry of the longeron and it has tapered ends. The strap will be installed between the UOL and the skin.

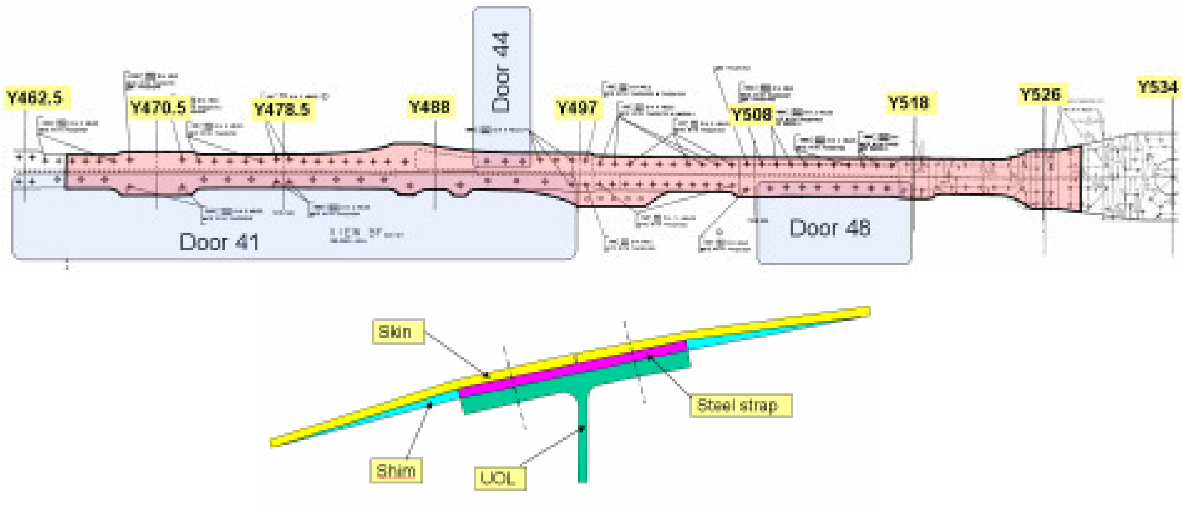


Figure 4.7.3: Steel strap to be installed at the UOL

For the static strength 3 load cases were analyzed. The smallest calculated margin of safety is 0.02. The design goal for fatigue was to demonstrate 2 design lifetimes of durability including life improvement methods (cold working, interference fit fasteners) based on the design spectrum. The lowest calculated crack initiation life for the refurbished UOL and the surrounding area is 10’160 flight hours. The UOL refurbishment with steel strap therefore fulfils the design criteria of 10’000 flight hours CI-life.

A feasibility study of the strap installation at the full scale fatigue test structure was made to define the accessibility (areas where the fasteners have to be removed), a procedure to transfer the fastener pattern to the strap and the shim definition.

Figure 4.74: Feasibility of the strap installation

Fuel Barrier Web

R. Nebel, I. Kongshavn, P. Pelloquin; RUAG Aerospace AG

An overview of the center section of the fuel barrier web (FBW), which was addressed in the SRP program, is given in Figure .

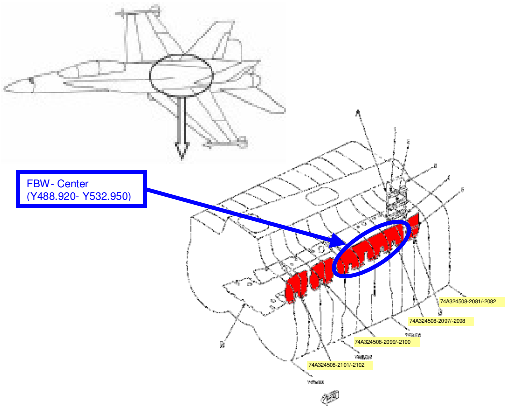


Figure 4.7.5: Overview of the fuel barrier web, center section, modified in the SRP program.

During the Swiss FSFT, cracks of up to 290 mm were found in the chem.-milled radii of the web, starting at a 5’000 SFH inspection, at the following locations:

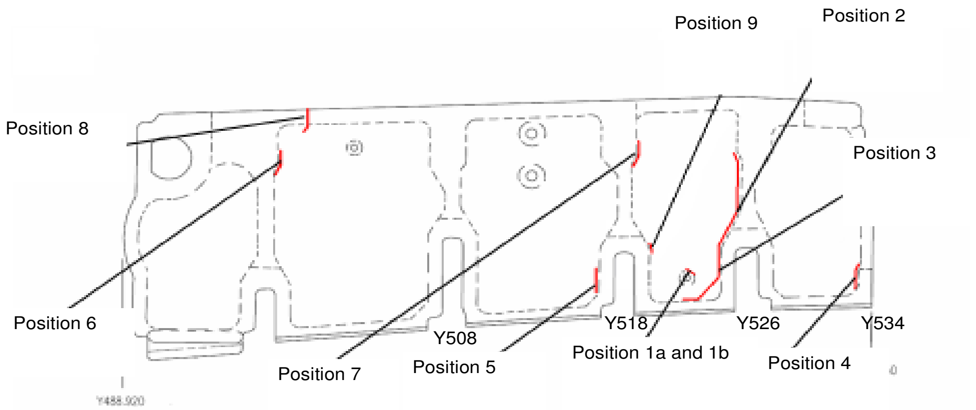


Figure 4.7.6: Cracked locations in the Swiss FSFT of the fuel barrier web.

The cracking is primarily caused by buckling of the web, between F.S. 508 and F.S. 534, as shown in Figure . It must be mentioned that in the Swiss FSFT the fuel tank was empty and not pressurized. The buckling was confirmed by a non-linear post-buckling finite element model analysis of the fleet configuration, which included fuel pressurization. In the fatigue analysis, a non-linear spectrum modification was performed using the FE results for different reference load cases and tank pressures. The reference load case for design was a 9g steady state pull-up manoeuvre. Since 50% of events in the Master Event Sequence represent asymmetric manoeuvres, a sensitivity study was also performed to assess the influence of fuel inertia during asymmetric manoeuvres.

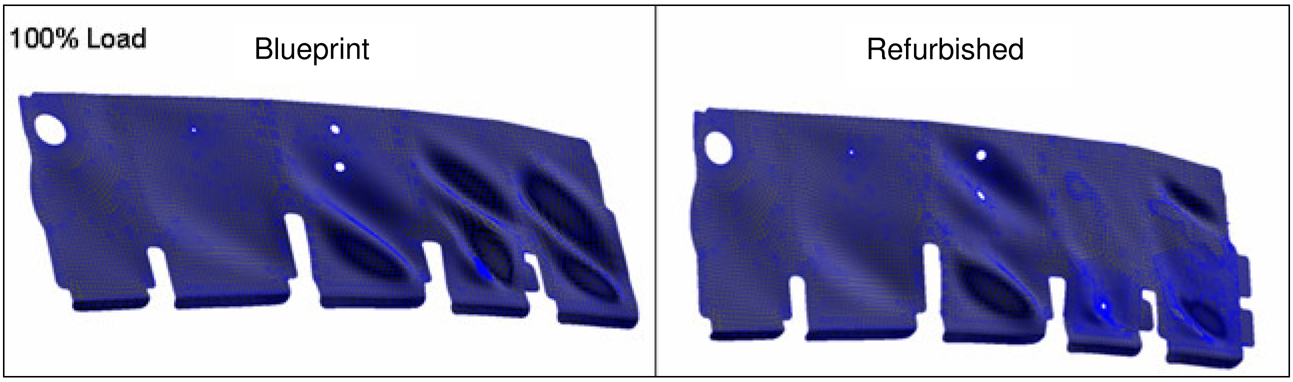


Figure 4.7.7: Buckling in the fuel barrier web; original (Blueprint) and refurbished configuration (same magnification).

A good agreement was obtained between the analytical CI lives and those observed in the FSFT. An analysis of the original fleet configuration (blueprint) confirmed that the short CI lives observed in the FSFT are also relevant for the fleet (see Figure). Critical fastener locations joining the web to the UOL at F.S. 508 to F.S. 526 had lives starting at 3’800 SFH. The critical chem.-milled edges of the web between F.S. 518 and F.S. 526 had analytical lives starting at 4’500.

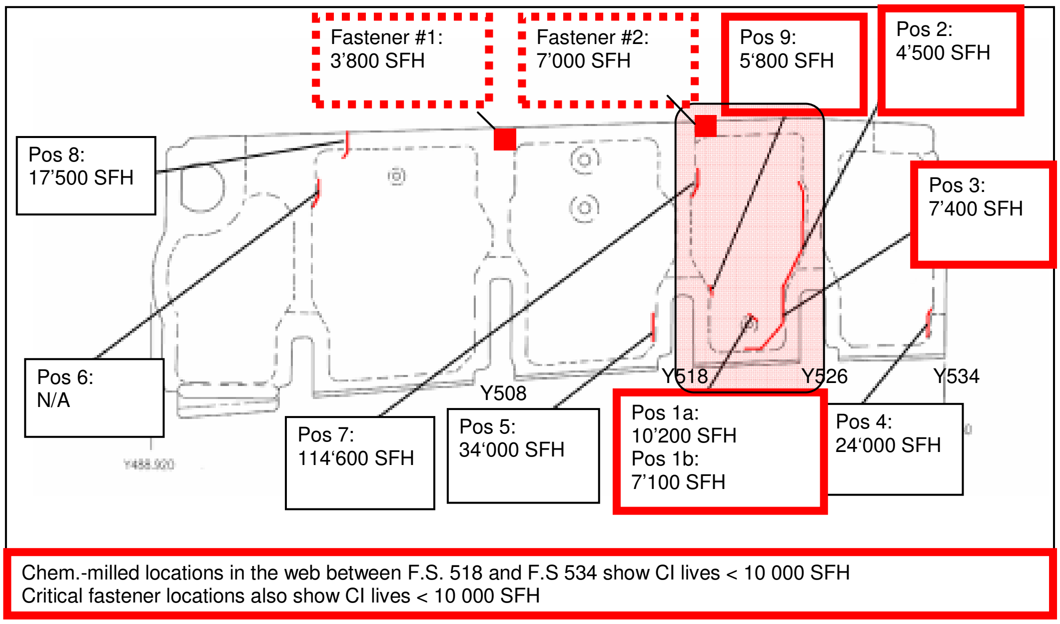


Figure 4.7.8: CI lives of critical fastener and chem.-milled radii in the Blueprint analysis of the FBW.

The refurbishment consists of the application of an aluminium doubler between F.S. 518 and F.S. 534 and a stiffener across the bay F.S. 508 – F.S. 518 (see Figure). The doubler and stiffener are made of 7075-T76; the FBW is made of 7075-T62. The aim of the stiffener is to limit buckling in the web by reducing the size of the shear panel, without attracting too much load. For the bays between F.S. 518 and F.S. 534, a stiffener did not reduce the buckling sufficiently and a doubler was designed in order to reduce the shear stress in the web. Life improvement methods such as cold working and interference fit fasteners were also used.

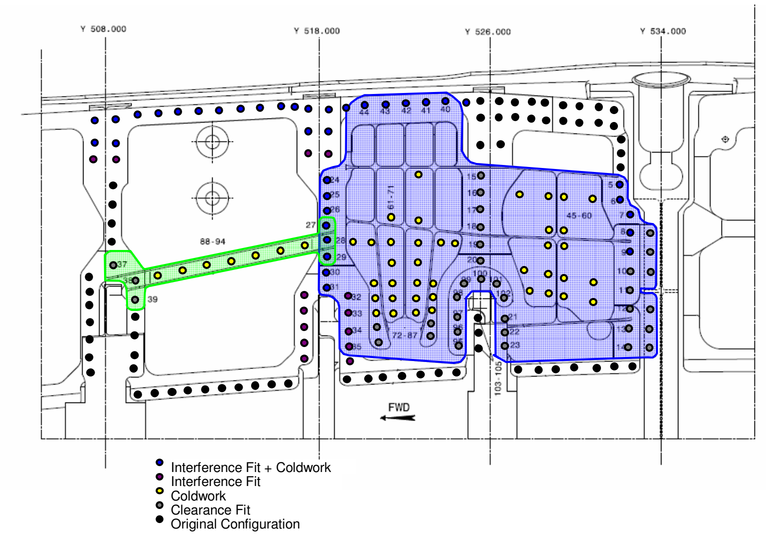


Figure 4.7.9 FBW refurbishment: addition of a doubler and stiffener between F.S. 508 and F.S. 534, with fastener life improvement methods.

Despite the increased stiffness in the FBW and adjacent structures, a refurbished design was found, which met static and fatigue requirements. The fatigue design goal was a 10’000 SFH CI life. For locations which did not meet this criterion, the goal was a total life (CI + CG) of 10’000 SFH.

Fuel Cell Floor

M. Godinat, C. Kunz, R. Renggli; RUAG Aerospace AG

The currently installed fuel cell floor (FCF) is made of chemically milled aluminium sheets. Cracks were found in other fleets (F/A-18 A/B/C/D) at the chemically milled edges. The first were found after 2574 hours. In the Swiss F/A-18 full scale fatigue test (FSFT) the fuel cell floors were not tested realistically because no fuel pressure loads were applied. Therefore no cracks were detected in these locations.
For the structural refurbishment an already available modification will be installed. The chem.-milled aluminium center fuel cell floors of tank 2 and tank 3 will be replaced with 0.04 in thick titanium sheet. Due to the material change from aluminium to titanium the stiffness of these fuel cell floors is increased and this results in higher loads.

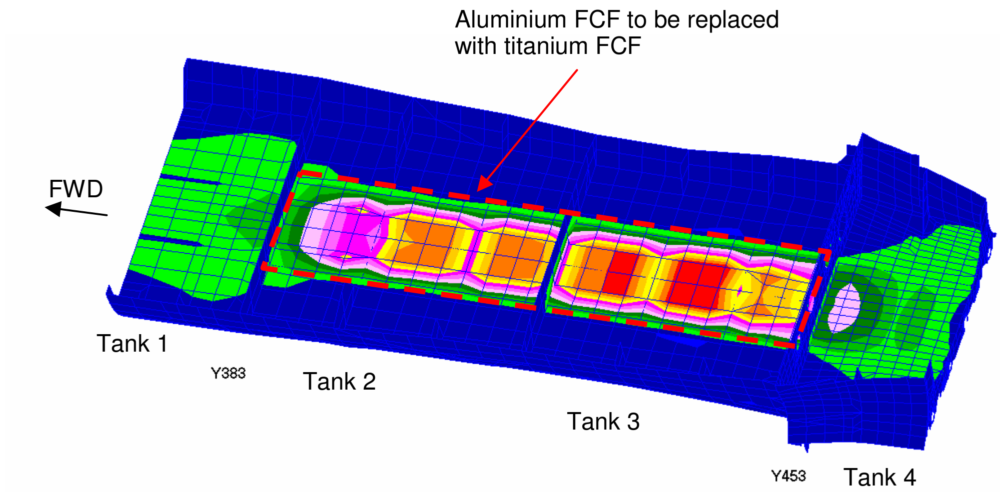


Figure 4.7.10: Increased Fuel Cell Floor loads due to the installation of the stiffer titanium floors (blue = no change of the load)

The analysis associated to the available modification covered the new FCF and parts of the surrounding structure. Additionally the FCF for tank 1 and tank 4 had to be checked for these increased loads. One location did not achieve the Swiss design goal of 10'000 flight hours CI-life. At the FCF of tank 4 the chemically milled edges are at the bottom and at the top surface. The chemically milled radius at the top surface has to be blended with a bigger radius in one location to reach the design goal.

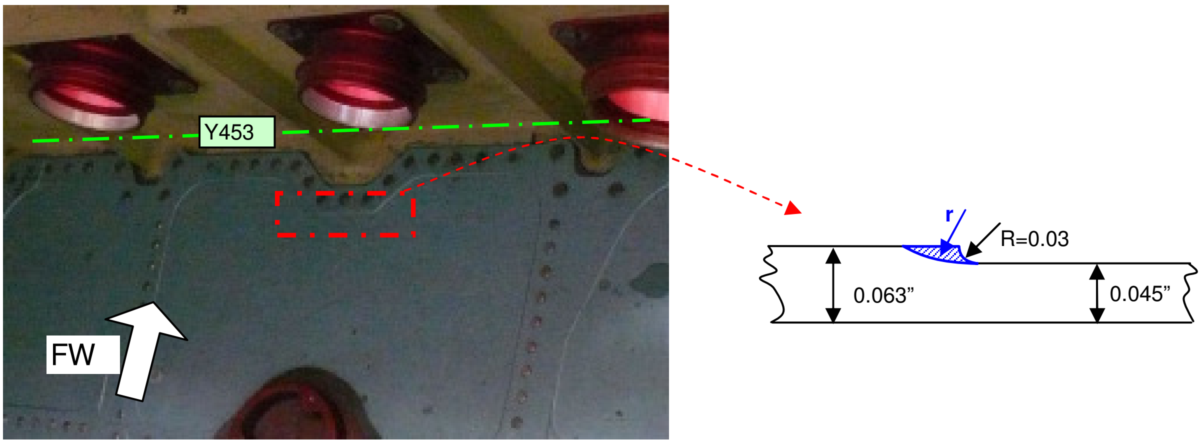


Figure 4.7.11: Area of the chemically milled radius to be blended to reduce the stress concentration.

Strap Door 44

C. Kunz, P. Pelloquin; RUAG Aerospace AG

After 6'600 hours of testing in the Swiss F/A-18 full scale fatigue test (FSFT), cracks were discovered at nearly every fastener hole of the strap at door #44. After 7'000 hours of testing the strap was replaced with a new identical strap and the test was resumed.

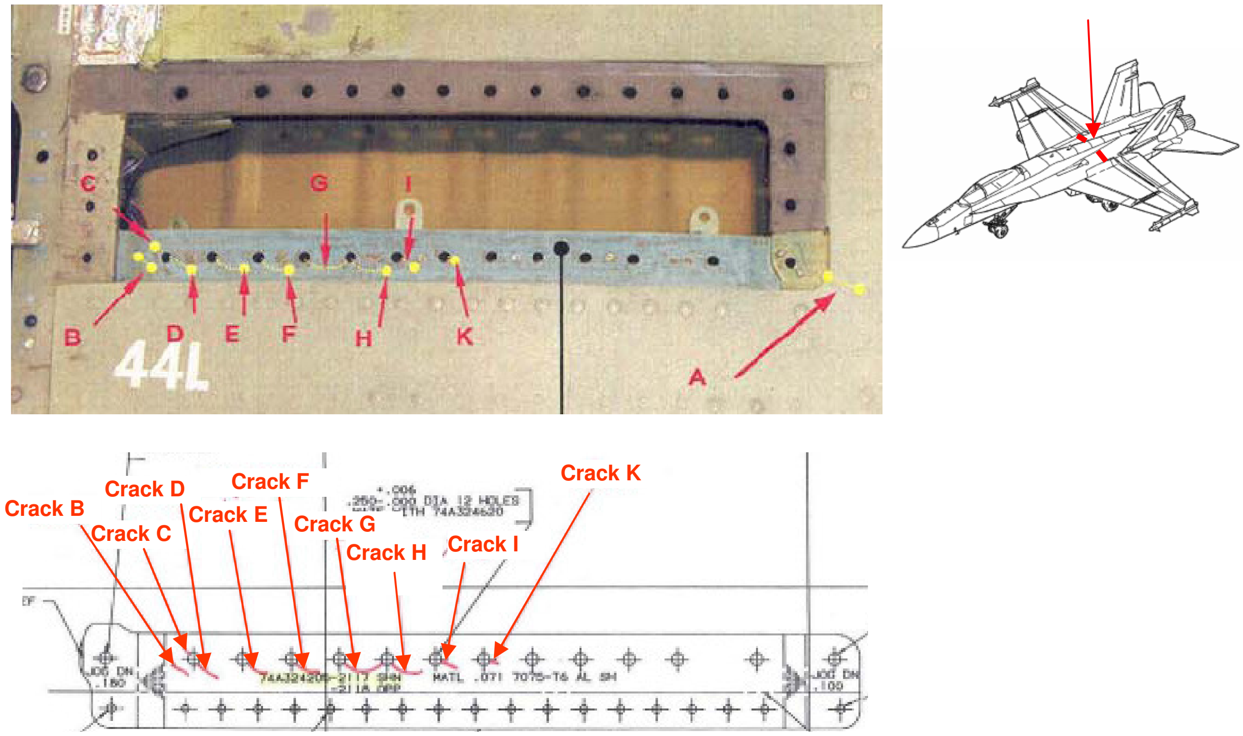


Figure 4.7.12: FSFT cracks in the strap at door #44

Since the strap is mounted between different components, the thickness should not change. Two solutions would have been possible for the structural refurbishment. First an aluminium strap with cold worked fastener holes or second a similar strap made from titanium. With both options the required design criteria of a crack initiation life of 10'000 flight hours can be met. As with the titanium strap the fastener holes do not need to be cold worked this solution was selected for the structural refurbishment.

Figure 4.7.13: Feasibility of manufacturing the titanium strap for door #44

Side Skin

R. Renggli, M. Godinat; RUAG Aerospace AG

Several cracks in the Side Skin at F.S. 383 were found during the Swiss FSFT. During testing, a crack was found in a chem.-milled radius at Position 1 at 1’820 SFH and during the tear-down inspection cracks were found at fasteners at Positions 2 to 5 (Figure).

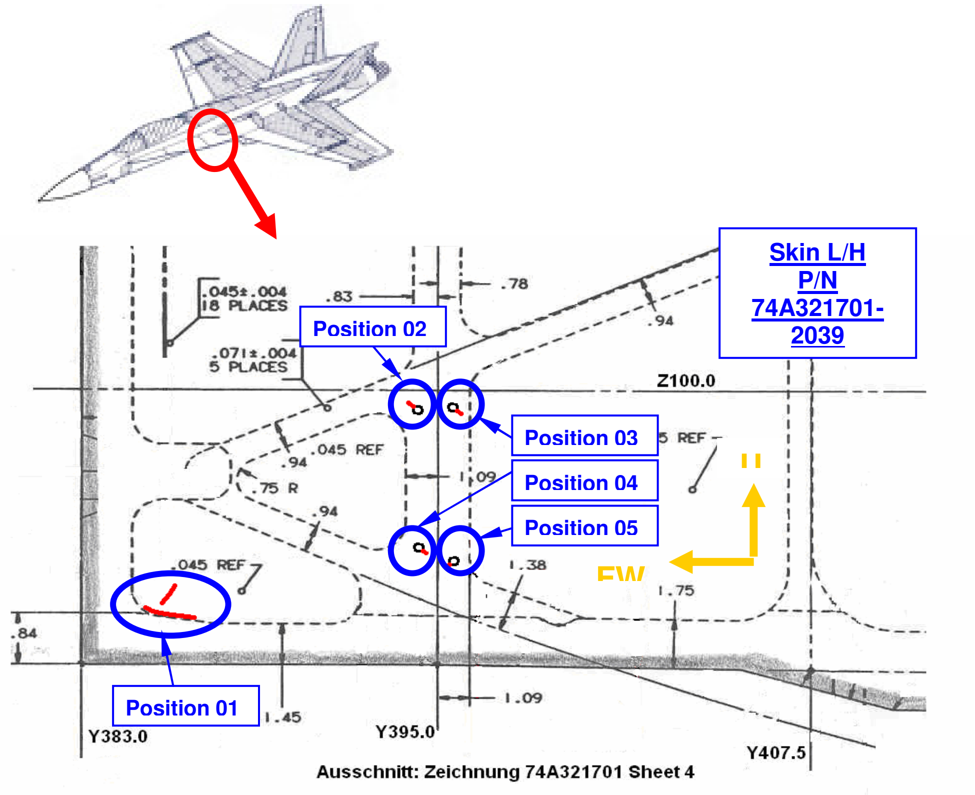


Figure 4.7.14: Swiss FSFT crack locations found in the side skin at F.S. 383.

During testing, significant buckling of the skin was observed. This buckling was analysed with a non-linear post-buckling FE model of the side skin for the test configuration, without fuel pressurization. For Position 1, a fractographic analysis indicated that the cracking initiated on the inboard side. A CI life analysis with a compression-dominated spectrum and reference stress for Position 1 confirmed the short CI life found in the test.

A non-linear FE analysis for the blueprint configuration indicated that the addition of fuel inertia and ultralage pressure delays buckling and reduces the depth of buckling at Position 1. As well, the buckling mode and critical locations for the skin are different to those of the FSFT configuration. There have not been any reports of cracks in a fleet aircraft by any other F/A-18 operators at these positions. It was concluded that no refurbishment is needed at this location to meet the required 10’000 SFH service life.

Crease Longeron

M. Godinat; RUAG Aerospace AG

Large fatigue cracks were found in the chem.-milled radii of the F/A-18 so-called ‘crease longeron’ during the inspection of some user’s aircraft (F/A-18 A/B/C/D). Fatigue cracks were detected in the left hand crease longeron of the Swiss F/A-18 full scale fatigue test as well. From 2001 to 2003, the OEM performed a detailed fatigue analysis. This study showed that the full life requirements (total life pre- and post-longeron severed) of the specific user can not be met at different locations. An external steel strap was then developed to achieve full life in all affected structural components.

RUAG reviewed the OEM analysis for the Swiss aircraft and questioned the need of the strap for the Swiss fleet. Some detailed fatigue analyses were performed for the crease longeron and surrounding area in order to prove that the installation of a steel strap is not required to meet the Swiss design criteria of 10’000 SFH full life. The intent of these analyses was to get some confidence in the ability of the aircraft structure surrounding the crease longeron to carry the loads even if the longeron is fully severed at FS 453.

4.8 Compression After Impact (CAI) Testing

B. Bucher, G. Peikert, M. Giger, I. Kongshavn, C. Kunz, J. Lussi, U. Bartlome; RUAG Aerospace AG

Within a RUAG Aerospace development project in the year 2008 compression after impact tests according ASTM D7137/D 7137/M were performed to investigate the behaviour under compressive loads of damaged carbon reinforced composites plates. The material used was a HexPly®8552/38%/AGP370/C with a lamina layup of [45/0/-45/90/45/0]s which resulted in a thickness of approximately 4.5mm. The 0 degree orientation is along the long edge (150mm) and the 90 degree orientation is along the short edge (100mm). Goal of the project was to establish all the relevant skills and hardware to define, perform and document compression after impact tests (CAI) based on standard ASTM or similar procedures. A batch of 8 coupons was tested dry and an other batch of 8 coupons was tested with a previous conditioning hot wet at 90°C. In order to implement the impact damage a drop weight tower was developed, build and successfully used to implement damages to coupons and ALCAS validation article components. The relevant guidelines were derived from ASTM D7136/D 7136/M. The drop tower allows for impact speeds of up to 6.4 m/s and impact energies of up to 113.38 Joules. Typical impact energies used to achieve barely visible impact damages for the coupons were about 30 Joules. Impacted coupons were ultra sonic NDI tested and the damaged area was protocolled with all the relevant test parameters. The test set-up also allows the determination of the absorbed energy in the structure. The damage mechanism was very similar for all coupons and the coefficient of variation for the net section strength was 2.2% for the dry coupons and 5.1% for the wet coupons. To back up the test results two fully parametrized Patran/Nastran FE models with an open hole representing the damaged zone were built. The first one consists of layered shell elements the other one consists of solid elements. The test methods are established and confirmed. Continuing test activities have the target to further improve the compression after impact test methodology (iCAI). Future tests will also involve the investigation of impact damage propagation under cyclic fatigue loads.

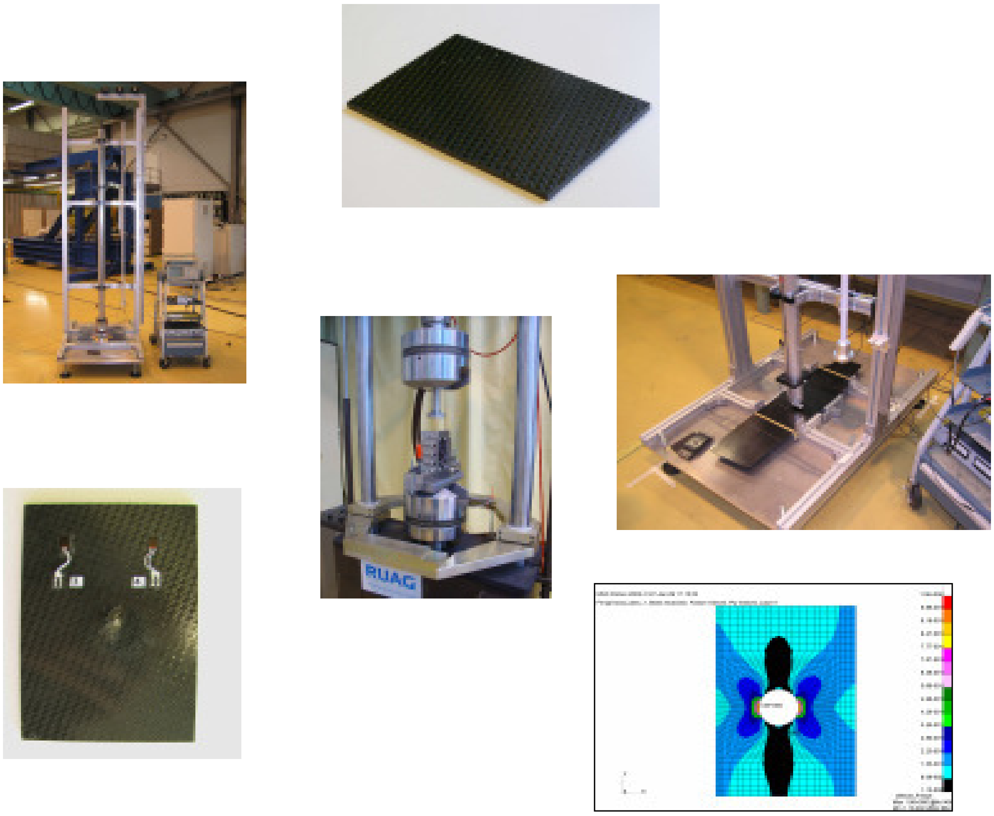


Figure 4.8.1: Compression After Impact test set-up and analysis

4.9 ALCAS - Advanced Low Cost Aircraft Structures

B. Bucher, M. Giger, R. Imhof, A. Bertsch, B. Schmid, J. Lussi, S. Zehnder; RUAG Aerospace AG

Objective

ALCAS is a European Commission funded integrated project with currently 19 Nations and 60 partners involved. The project was started 1 February 2005 and will be finished 31 January 2011. The objective is to reduce the operating costs of relevant European aerospace products by 15%, through the cost effective application of carbon fibre composites to aircraft primary structure, taking into account systems integration. The project will integrate and validate mature and new composite technologies through the design, manufacture and test of appropriate wing and fuselage assemblies that represent both airliner and business jet products.

RUAG Aerospace contributions

RUAG works in the business jet wing platform where Dassault Aviation (F) is the platform leader.
RUAG's contributions to the project involve design, analysis, manufacturing of validation article components and dummy structures (composite and metallic) and structural testing.
RUAG's contribution to fatigue and damage tolerance involves full scale testing of the composite validation article. Its architecture is derived from a typical business jet horizontal tailplane (HTP). The tests are scheduled end of 2009 and include a static, fatigue, damage tolerance and rupture test. The composite components contain select simulated manufacturing and BVID impact damage deficiencies which were implemented by RUAG according to the test plan. During the test it is foreseen to implement further impact damages to study damage behaviour and damage growth.

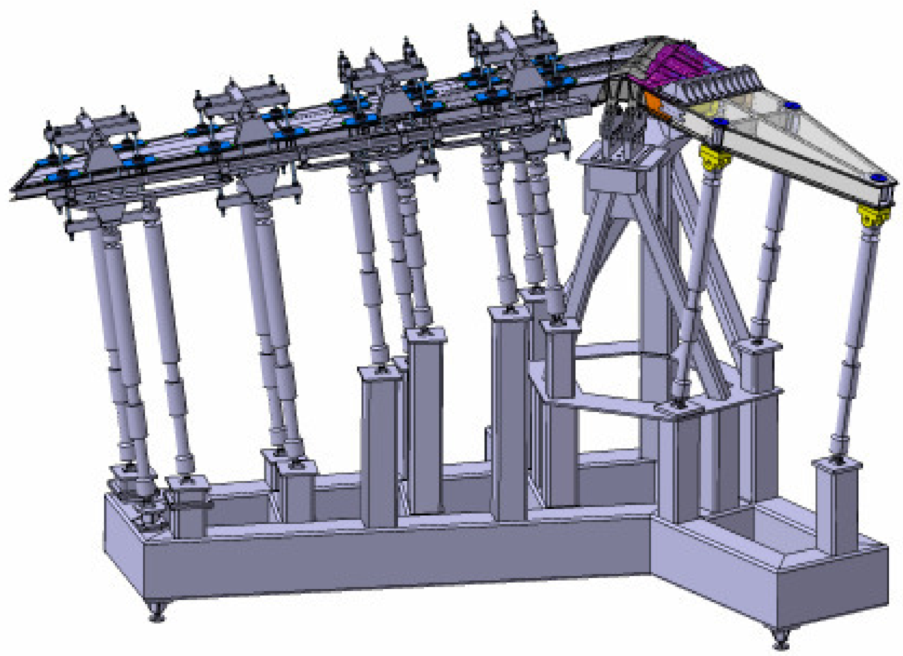


Figure 4.9.1: Full scale test set-up

5.0 Effect of Exfoliation Corrosion on Wing Structural Integrity

Andreas Uebersax, Cyril Huber; RUAG Aerospace AG; Guillaume Renaud, Min Liao; National Research Council Canada

A combination of static and crack growth analysis models were developed to evaluate the impact of exfoliation corrosion and grind-out repairs on remaining fatigue life and residual strength of a wing upper skin (Aluminium 7075-T651) under a fighter aircraft spectrum loading. For this purpose a total of 20 dog-bone coupons were modelled, analyzed and validated by test results. The tests were performed on specimens that were pristine, naturally corroded, artificially corroded and repaired by grind-out.

A local damage model, based on a soft inclusion concept, was used to model the corroded and grinded coupons. Results showed that the local damage models provided a good estimation of the performance reduction due to the exfoliation or grind-out.

The results from the local damage models were applied to a finite element model of an F-5E wing upper skin section. It was shown that for the analyzed wing section, the model could be used to define the most important structural strength criteria. This information is important when defining repair actions for corrosion damaged wing skins. It was also shown that leaving the corrosion in place gives an advantage in static residual strength, as well as in fatigue strength over grind-out repairs.

This project was performed in collaboration with the Institute of Aerospace Research (IAR) of the National Research Council Canada (NRC).

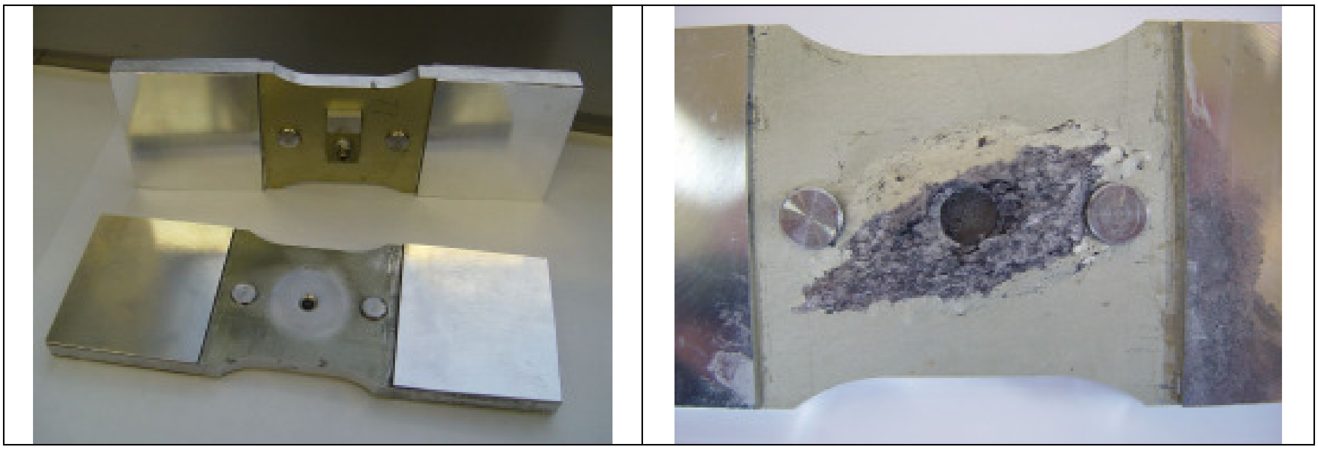


Figure 5.0.1: Layout of test samples in pristine and grind-out condition (left) and artificially corroded condition (right)

5.1 References

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[3] Donald, J.K., Lados, D.A., ¨An integrated Methodology for separating Closure and Residual Stress Effects from Fatigue Crack Growth Rate Data¨, Fatigue & Fracture of Engineering Materials – Special Issue Contribution (in press)