Review of Aeronautical Fatigue Investigations in Switzerland

May 2005 – April 2007

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SUMMARY

The Swiss review summarizes fatigue work in Switzerland. It includes main contributions from the RUAG Aerospace (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 30th ICAF Conference
Neapels, 14 and 15 May 2007

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 2005 till April 2007. The various contributions to this review come from the following sources:

- RUAG Aerospace; Fatigue Engineering, Aerodynamic, and Full Scale Fatigue Test Project Team
- Pilatus Aircraft Ltd; Structural Engineering, Stans
- CFS Engineering, Lausanne
- M@M Mandanis GmbH, Kriens
- SMR S. A. Engineering & Development, Biel

The work of related topics on the F/A-18 military aircraft was supported from the stress engineering team by M. Godinat, A. Del Don, R. Meier, P. Pelloquin, and T. Stehlin. There effort is gratefully acknowledged.

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

4.2 Fatigue investigation of the SIWA fin's at the F/A-18 wing fold

M. Guillaume, J. Vos, A. Gehri, G. Mandanis

The F/A-18 has a wing fold titanium rib which is fatigue critical. During most Swiss missions a SIWA (AIM-9P) is mounted at the wing tip. In an aerodynamic computational fluid dynamics (CFD) project the unsteady behaviour of the F/A-18 was investigated in detail. Fully Navier Stokes calculations on structured grid with 14 million cells and 3000 blocks were done with a SIWA with fins and without fins, see figure 4.2.1. The pressure on the outer wing was different, also some differences on the horizontal tail were observed, see figure 4.2.2. At the wing fold the wing fold the bending moment was calculated using the inertia data and the aerodynamic data. Four different load cases from steady state maneuvers were analysed for the SIWA with and without fins. On average the bending moment was significantly reduced for the configuration without fins. Based on the fatigue initiation software CI89 from Boeing the crack initiation life was 3 to 6 times longer depending on the design limit stress at the critical location (120 to 140 Ksi) for the configuration without fins at the SIWA (figure 4.2.3). This first result should be checked with a more detailed loads analysis. A first trend showed that the fatigue life can be probably increased at the wing fold for missions without any fins for the SIWA at the wing tip. With a very reliable CFD tool first load trends can be computed and used for a fatigue assessment at critical locations. With todays CFD technology we can study the impact of the flow behaviour on the structure.

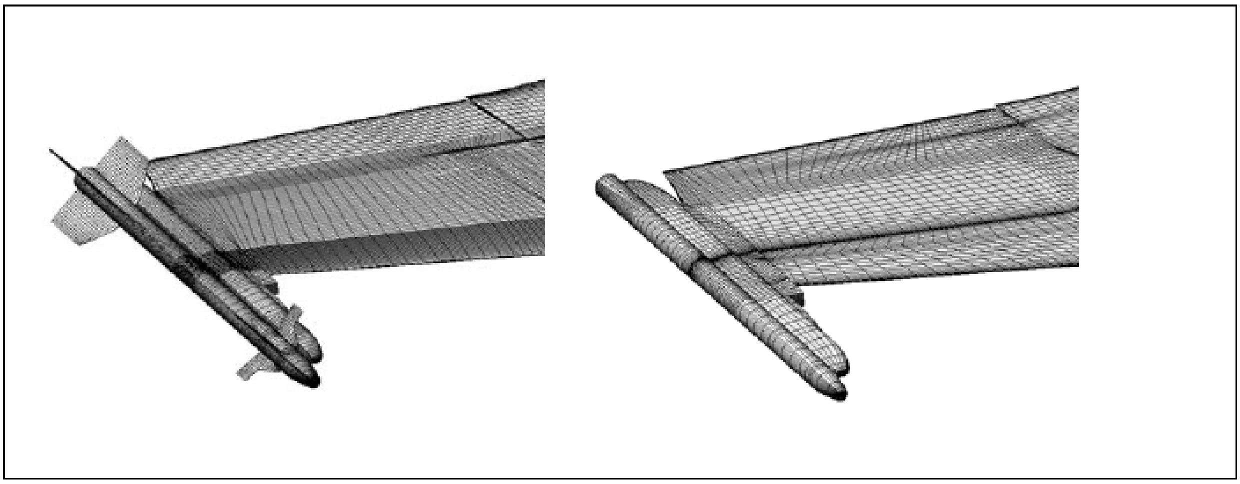


Figure 4.2.1: Outer wing surface mesh including SIWA with and without fins.

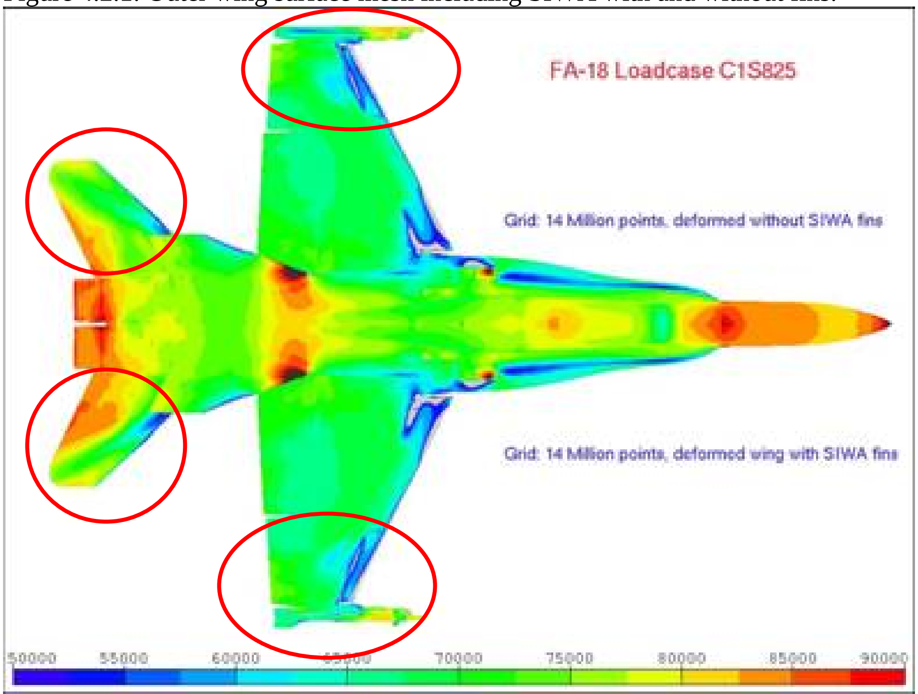


Figure 4.2.2: Pressure distribution with (lower half) and without fins (upper half)

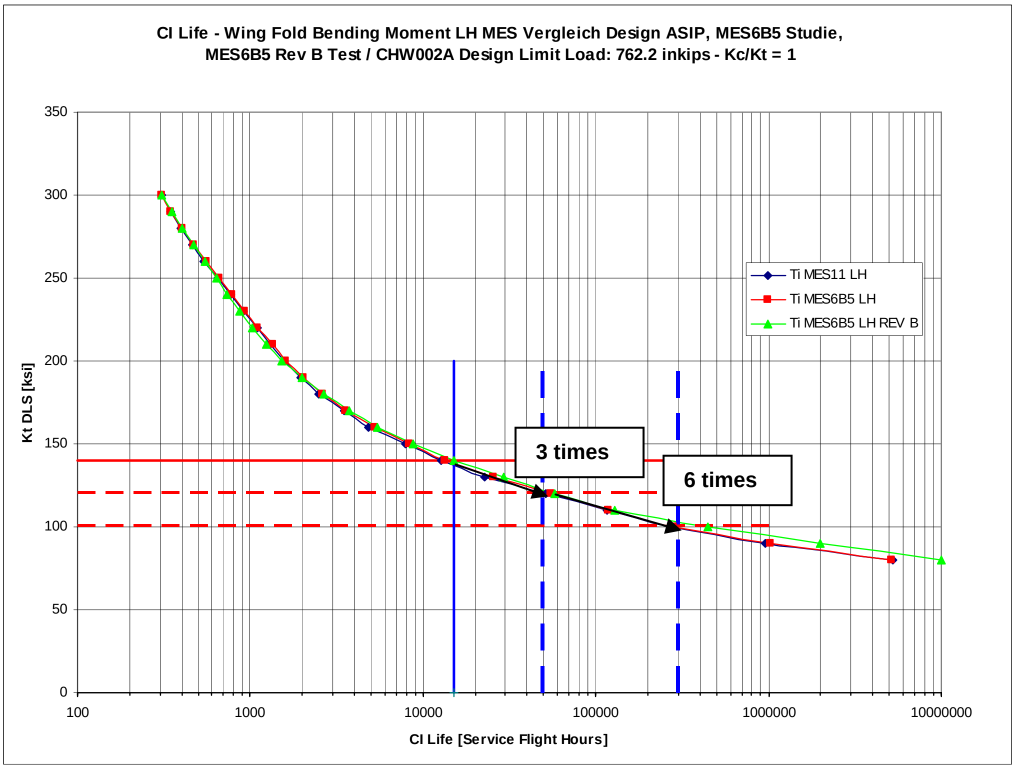


Figure 4.2.3: Influence on crack initiation life with and without fins for 140 & 120 Ksi.

4.3 Development for unsteady buffeting analysis on the F/A-18

M. Guillaume, J. Vos, A. Gehri, B. Bucher, S. Merazzi, T. Ludwig, G. Mandanis

In 2005 RUAG Aerospace developed together with CFS, SMR a fluid-structure interaction (FSI) between fluid and the structural grids for the F/A-18 wing. This FSI tool was successfully applied for load calculations on a deformed wing. Even for a wing of span of 12 meters the deformation has an important influence on the component loads especially for higher G maneuvers. The buffeting on the several components on the F/A-18 is well known. The impact of detached vortex induced fatigue damages on the vertical tail and on the hinges of the F/A-18 trailing edge flap and aileron depending on the AoA and the dynamic pressure. Several structural refurbishment programs were initiated from the US Navy by the Boeing Company in St. Louis. All Swiss F/A-18 will have the ECP574 for the trailing edge and the aileron hinges. This upgrade is running since two years and will be soon completed.
During the ASIP study the dynamic environment for the Swiss design was analyzed by Boeing. The conclusion was that the dynamic spectrum is as severe as the US Navy spectrum but the Swiss design maneuver spectrum was 3 times more severe. After close to ten years Swiss usage the time spent in damaging AoA and dynamic pressure ranges is much more severe. As result the impact on the fatigue life has to be studied in more detail.
RUAG Aerospace has done a lot of unsteady computational fluid dynamics calculation using there NSMB code with DES simulation and the the Spalart-Almares turbulence model. A real flow of 0.5 seconds was calculated for several maneuvers. The unsteady flow behaviour in the aft fuselage could be confirmed. The figure 4.3.1 shows a 8.25G steady state maneuver at two different time settings. The difference in the pressure in aft fuselage is clearly visible.

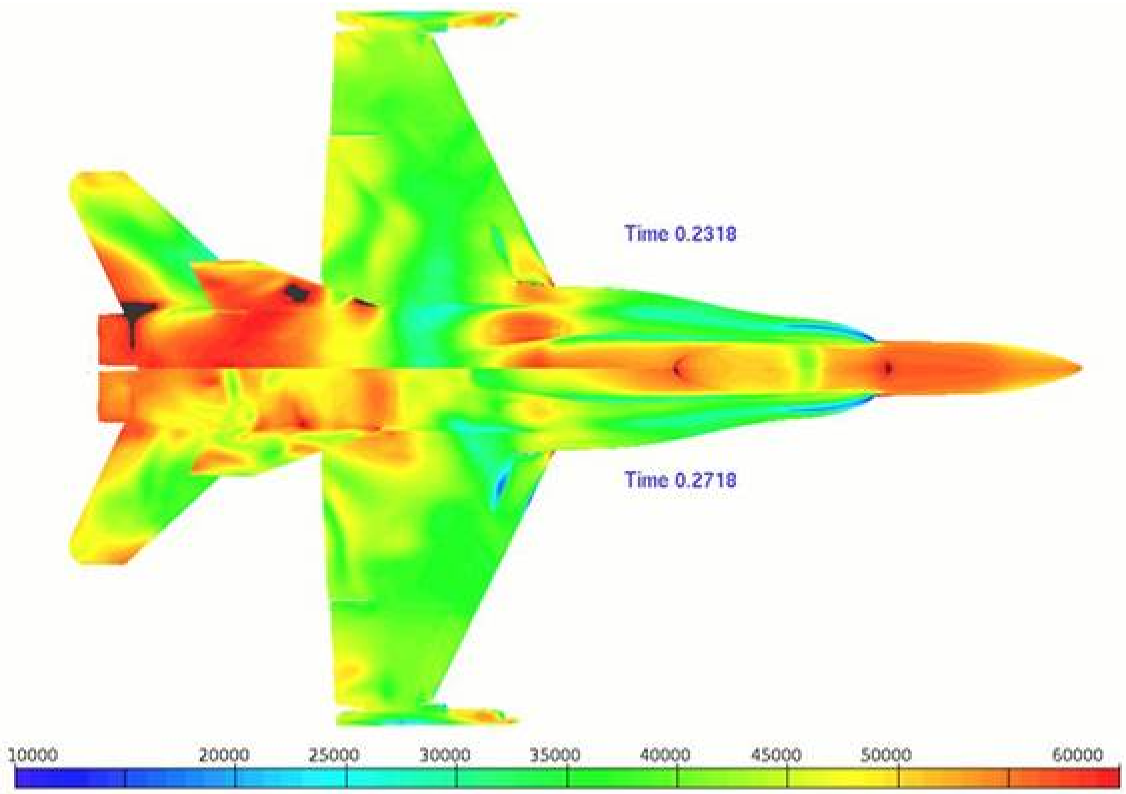


Figure 4.3.1: Pressure distribution at 0.23 sec (upper) and 0.27 sec (lower)

An unsteady aero-elastic software tool to investigate the dynamic at the vertical tail is under development. The method of Farhat is applied for the unsteady fluid-structure interaction. A dual time stepping implicit modular integration algorithm based on Newmark approach was implemented in the NSMB solver. The process was validated with experimental data of the AGARD445.6 wing, see figure 4.3.2. A grid smoothing subroutine is under development to improve the method for complex structures as the F/A-18 vertical tail.
The goal is to develop a buffeting fatigue spectrum for critical locations at the vertical tail.

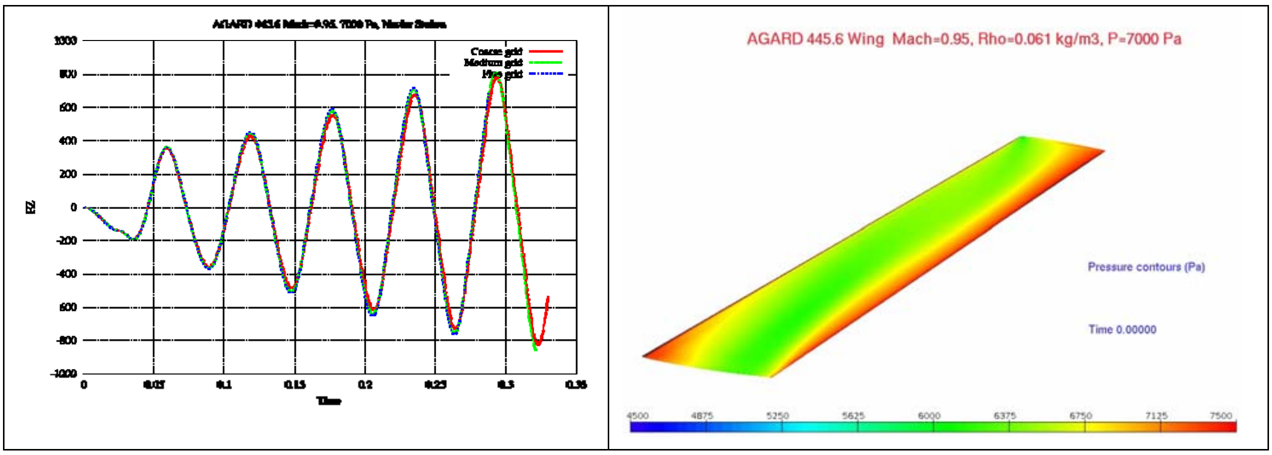


Figure 4.3.2: Unsteady simulation of AGRAD445.6 wing at a pressure of 7000 Pa (critical flutter behaviour)

4.4 Simplified buffeting spectrum for the F/A-18 vertical tail

M. Guillaume, J. Vos, A. Gehri, B. Bucher, G. Mandanis

For the forward stub at FS557 at the vertical tail a dynamic peak valley sequence was developed using the strain gauge data from Swiss F/A-18 full scale fatigue test. The fatigue test spectrum consists only of steady maneuver loads. The buffeting was not addressed. Influence coefficient at the strain gauge location for bending moment and torque were calculated from the test. Using the data from the unsteady computational fluid dynamics calculation (figure 4.4.1 & 4.4.2) together with inertia data base (bending moment & torque) transient stresses could be determined with the influence coefficient approach. For the Swiss maneuver design spectrum a study was done for the buffeting at the vertical stub FS557. The onset of buffeting was studied for AoA angles greater 18°, 19°, and 20° and for the associated dynamic pressure ranges. The transient cycles were then added to the maneuver stress spectrum. A variation of time spending in the buffeting regime was considered. The selected time was 5, 10, and 20 seconds. Therefore the number of cycles was considerable increased.
Using the crack initiation software CI89 from Boeing the crack initiation life was decreased from 10’000 SFH (63 Ksi = design goal) to 1770 SFH, see figure 4.4.3. Due to buffet the durability life could be decreased by factor up to approximately 5. For critical buffeting locations higher safety factors should be considered. Detailed unsteady aero-elastic coupling together with flight test data will deliver more information for the assessment of the structural integrity on the F/A-18 vertical tail due to the Swiss usage.

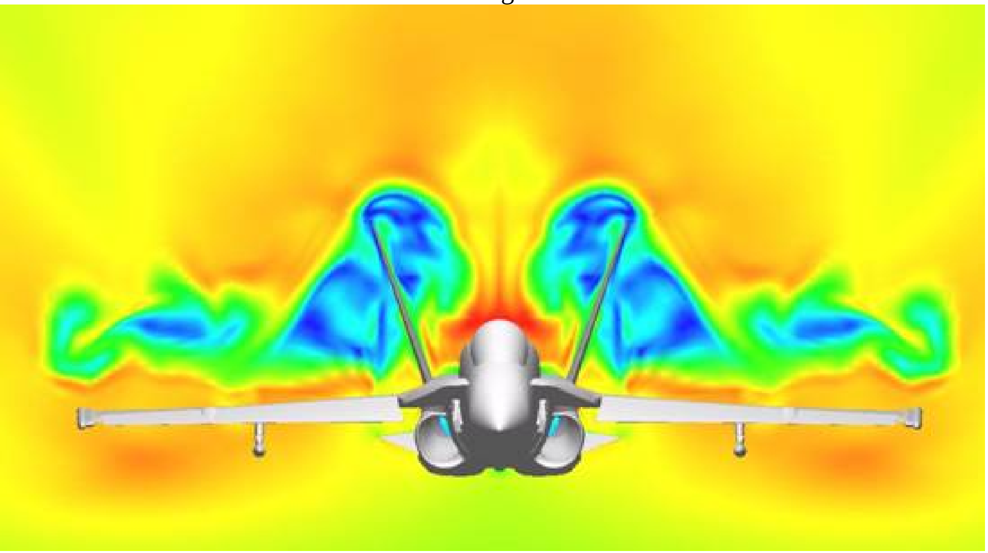


Figure 4.4.1: Pressure distribution in the plane at the position of the vertical tail

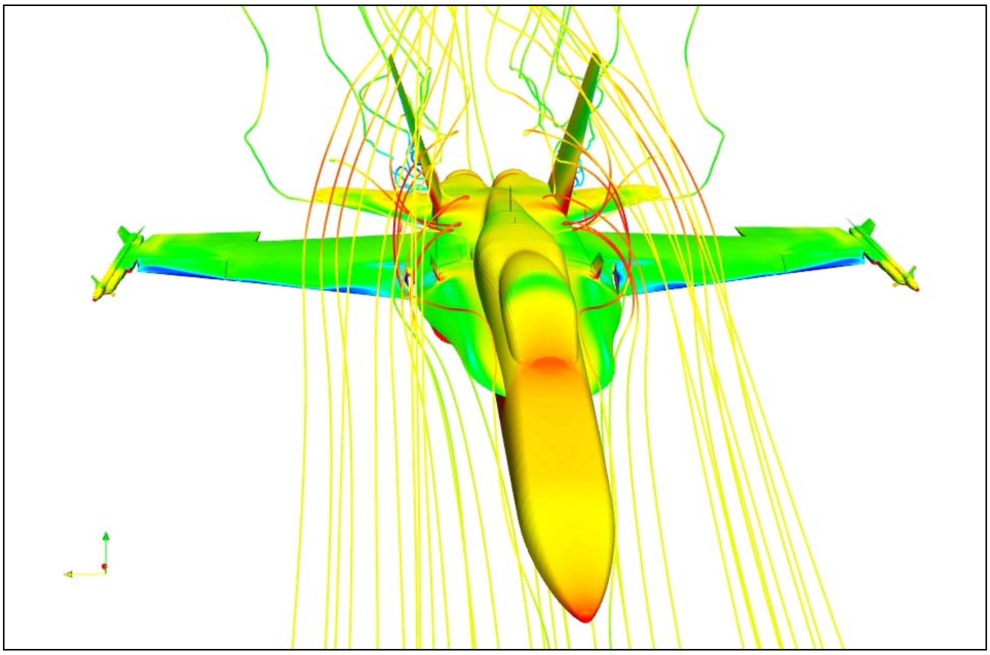


Figure 4.4.2: Streamlines showed the dynamic affected vertical tail (buffeting)

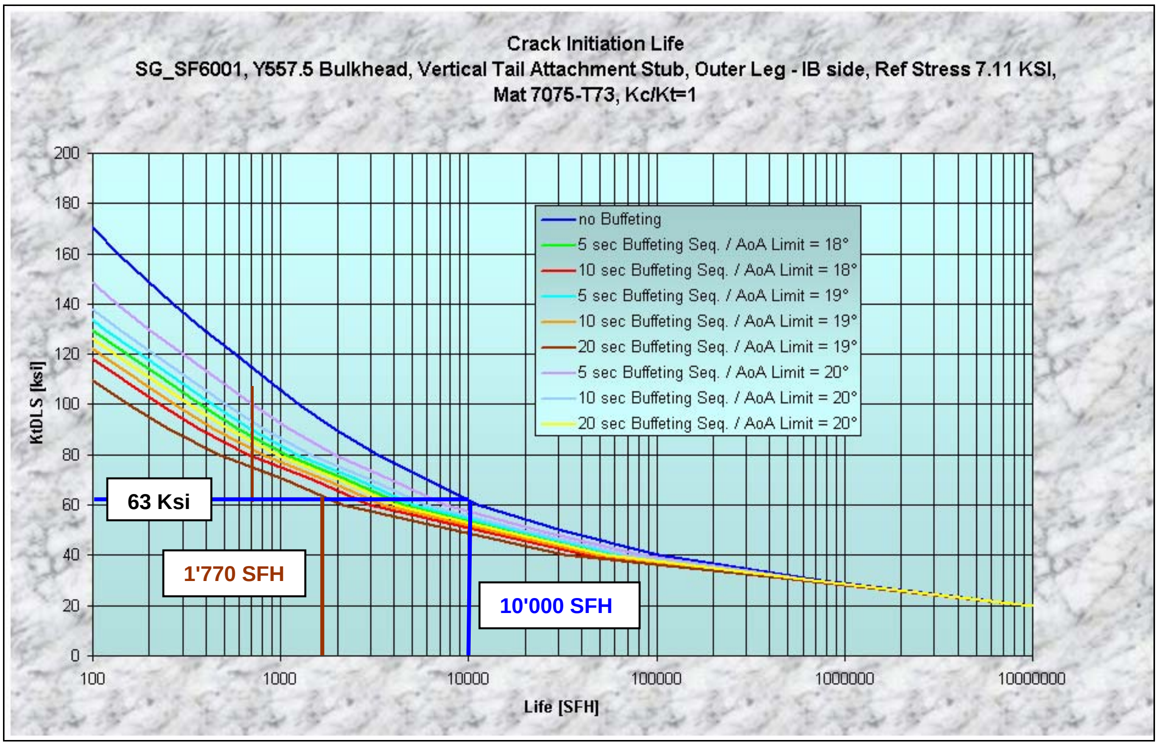


Figure 4.4.3: Crack initiation life curves for the variation of AoA and buffeting time

4.5 F/A-18 Tear Down Inspection

M. Guillaume, I. Pfiffner, R. Jaccard, L. Schmid, M. Gottier, G. Mandanis

The F/A-18 full scale fatigue test reached 10'400 SFH by end of September 2004, for test schedule see figure 4.5.1. First results of the test were reported in the last national review in May 2005.

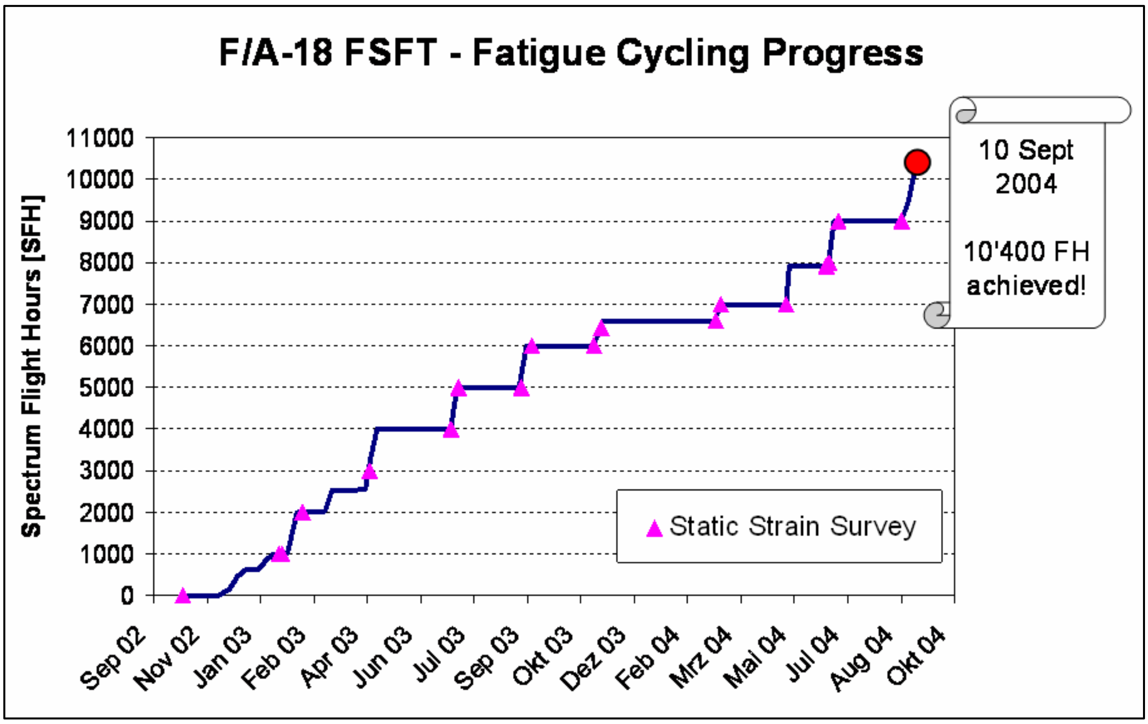


Figure 4.5.1: Test schedule for the Swiss Full Scale Fatigue Test (FSFT)

In spring 2005 the test set-up was disassembled. The outer wing was dismounted before the test article was removed from the test rig. Afterwards the inner wing was removed from the center fuselage.
Detailed non destructive inspections were done in the forward fuselage section. No failures were found in the forward fuselage. This result was expected because the forward fuselage was not a critical component based on the Swiss ASIP study. No redesign was necessary.
Also the free leading edge extension showed no cracks.

The first action in the tear down was the disassembly of the splice section FS374 were the forward fuselage is connected to the center fuselage. The splice section was modified for the Swiss due to stiffness changes from the aluminum to the titanium longeron in the center fuselage. No cracks were found at this location.
In a final inspection in the forward fuselage section in the area which was not accessible during testing no damage was found.

Aft fuselage and vertical tail

In the next phase the vertical tails were disconnected from the aft fuselage. The vertical tails were used as dummies to introduce the maneuver loads in the aft fuselage through the six stubs. No buffeting cycles were simulated. In the holes at the stubs no damage was found.

The aft fuselage section was de-spliced from the center fuselage section at the splice FS557. In the longeron between the splice and the speed brake attachment some damage was observed. The inboard plate of a twin plate splice (RH side) at the section FS574 was fully cracked, see figure 4.5.2. The outboard bracket was cracked approximately 45 % of the cross section.

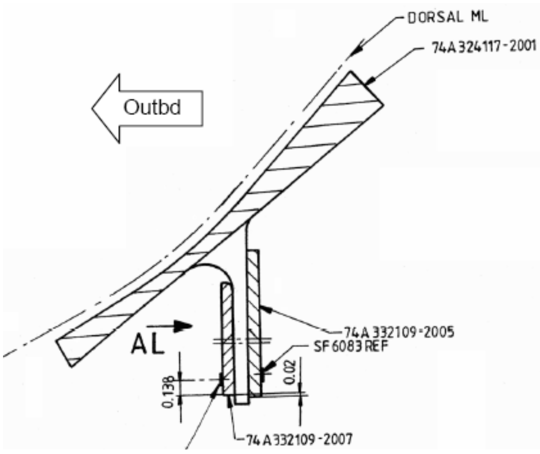


Figure 4.5.2: Sketch of twin plate splice and one piece of the full cracked plate

The aft fuselage section showed the following damages during the teardown inspection:

- Several cracks (approx. 0.04 to 0.3 inch) at “milson” holes and one from fillet radius (approx. 0.4 and 0.6 inch) in the lower outboard longeron between FS615 and FS657 on the LH & RH side were observed.
- Corner cracks (approx. 0.2 to 0.4 inch) at the L/H & R/H keel cover between FS590.5 and FS598.
- Small crack (< 0.1 inch) at hole in the L/H skin at section FS574.5.
- Small cracks at the holes in R/H skin between FS557.5 and FS574.5.

The fracture critical locations in the aft fuselage showed any damage.

Center fuselage

The center fuselage consists of three major sections: forward, center, and aft fuselage.

Forward section:

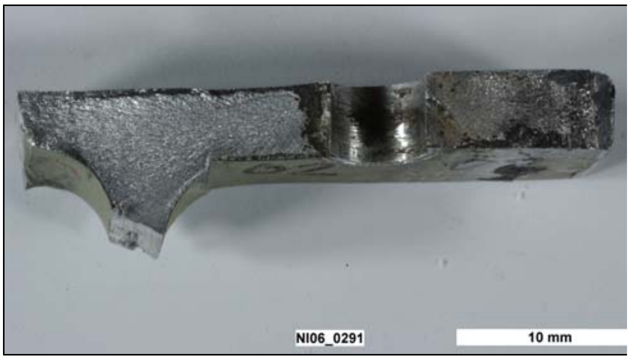
- Small cracks at the two holes were found in the upper skin on both sides, the affected holes are not the some on LH & RH side.
- Several cracks were observed in lower skin between FS 383 and FS395on the LH & RH side, already at 1000 SFH first cracks were found, a patch was installed, this patched changed the stiffness in this area.

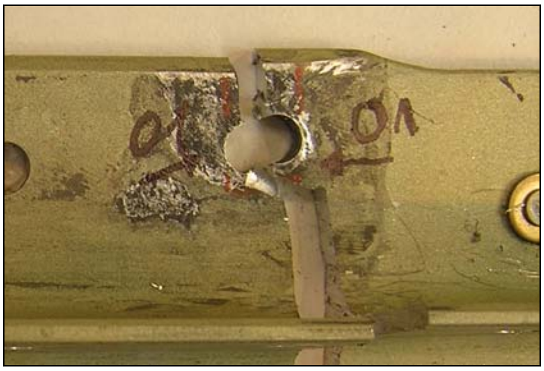
- Four small cracks were found in the lower flange (LH & RH side) at the former FS395
- Small corner cracks on both sides (but different locations) were observed in the diverter longeron and the diverter stringer between FS395 and FS453.

In the teardown inspection in the forward fuselage only very small cracks were observed at non critical locations. Therefore the residual strength of the structure is of no concern for the Swiss design spectrum.

Center section:

- Several small cracks were found in the upper floor segment of the dorsal deck between FS453 and FS557.5. These very small cracks are of no concern compared to the test results of the US Navy ST16 test.
- A crack in the strap radius between two holes in lower section of the duct former FS478.5 was found, also the splice plate showed a crack in this area, damages at different location on both sides. Also the engine skin duct showed cracks between holes at the lower section FS478.5.
- Some more cracks were found during the tear down inspection in the upper outboard longeron. A crack was observed during fatigue cycling on LH side at 7’000 SFH at FS499. The crack was initiated at the down standing leg of the upper outboard longeron. Also on the RH side a few cracks were observed during the fatigue cycling in the section FS492.5 till FS503.2. These cracks were reported after 8’300 SFH. Some edge distances (e) were very low compared to the blueprint value of e/D = 2 (D = hole diameter). Furthermore the holes in the longeron should have Swiss interference fit fasteners which was not the case in test article. The upper outboard longeron was heavy loaded for the Swiss loads in the test section. One reason was surely due to the stiffness changes for reinforced center fuselage (titanium upper longeron and titanium bulkheads with beef up of structural elements). Up to now the much higher shear loads in center section for the Swiss compared to the US Navy could never be understood.





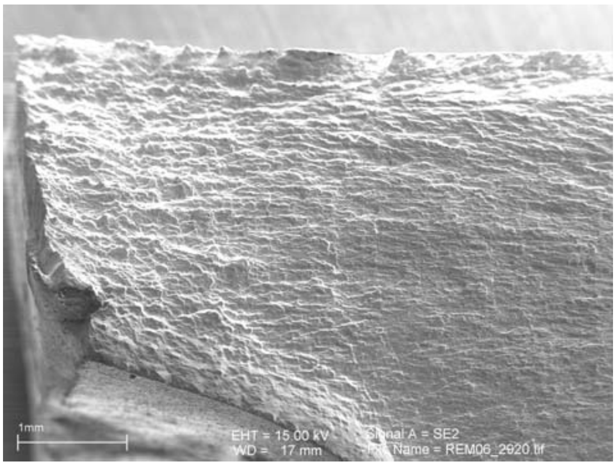
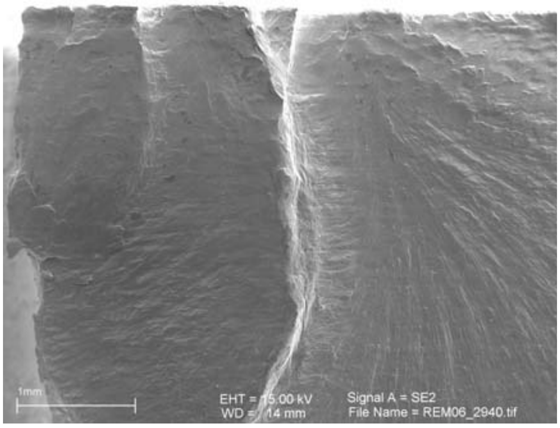


Figure 4.5.3: Longeron R/H, cracks started from hole, REM picture hole L/H side



Longeron L/H (lower side), short edge distance, crack started between hole and edge; REM picture upper side

A detailed FEM analysis on the outboard longeron showed a high stress at the edge of FS489. During tear down inspection a crack was found there but the Fraktographie (figure 4.5.3) showed that the crack initiation occurred not at the predicted highest stress area by finite element analysis, see figure 4.5.9. The material (7149-

T76 Extrusion) of longeron seems also have some internal residual stresses which may lead to early cracks. The cracks in the outboard longeron may be initiated at approximately 5’000 SFH.

- No cracks were found the in titanium bulkheads in the center section. These bulkheads are the most fatigue critical parts on the F/A-18. They are of major concern for all the other operators of the F/A-18. A major structural refurbishment program is on going for the US Navy (replacement of center barrel section).

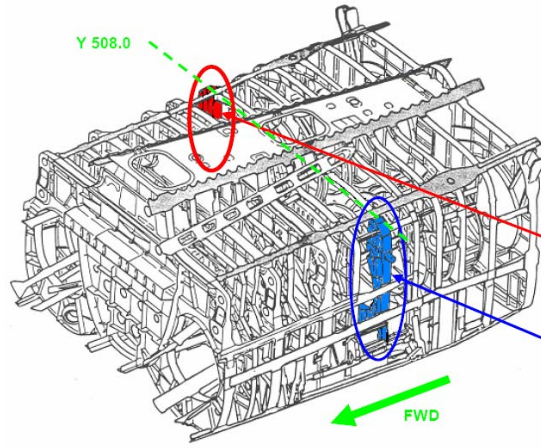
Even some cracks were found in the aft fuselage of the center section and at the dorsal deck, the test article survived 10’400 SFH which means that the structural integrity is not a major concern. The detailed Fraktographie at selected areas confirmed that the crack origin can not only be identified by detailed FEM analysis. More carefully the material of the test article needs to be checked and compared with the fatigue material data used in the ASIP study. The crack initiation and the crack growth data may not fully represent the material of the real structure. Furthermore loads assessment of the ASIP loads process (SPECGEN) on the center fuselage may be also bring more understanding.

In February 2007 the teardown inspection on the center fuselage was completed. All parts were safely stored for future investigations if requested.

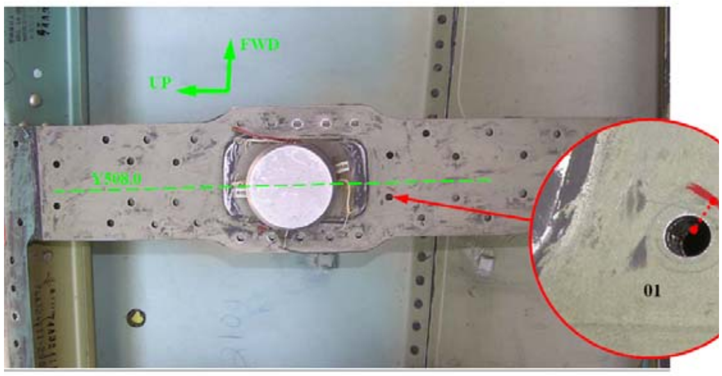
Summary of damages in the center fuselage

The critical locations based on the Swiss FSFT were evaluated in detail. Total 37 parts of the center fuselage showed cracks after 10’400 SFH. For 18 parts of 37 parts the cracks were observed during the tear down inspection.
The only fracture critical (FC) part was the former center section at FS508, see figure 4.5.4. Only a small crack at the hole was found at LH side during the tear down inspection.

Figure 4.5.4	**Crack in Hole of FORMER R/H, FUSELAGE CENTER SECTION, Y508.0 P/N**



FM-TDI0066

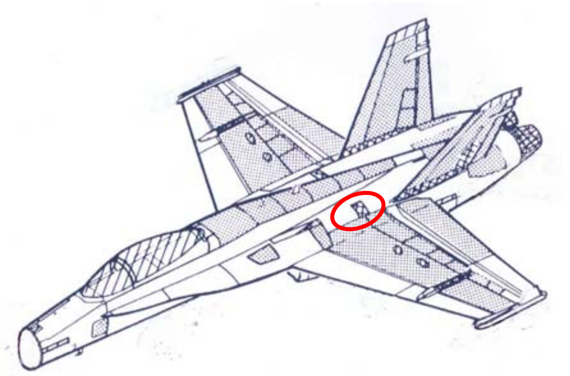


The following maintenance critical (MC) parts showed cracks in the test:

- Upper outboard longeron on both sides (LH & RH), see figure 4.5.5
- Floor segment upper section FS453 to 557.5 only LH side, see figure 4.5.6
- Titanium upper longeron only on the LH side, see figure 4.5.7
- Floor fuselage center section between FS383 and FS453 only LH side, see figure 4.5.8

Figure 4.5.5 — **Several Cracks in horizontal and vertikal Webs, Section FS 488 to FS 544 Upper Outboard Longeron**

FM-FSFT0106, FM-FSFT0108, FM-FSFT0139, FM-TDI0013 FM-TDI0042, FM-TDI0046 FM-TDI0052



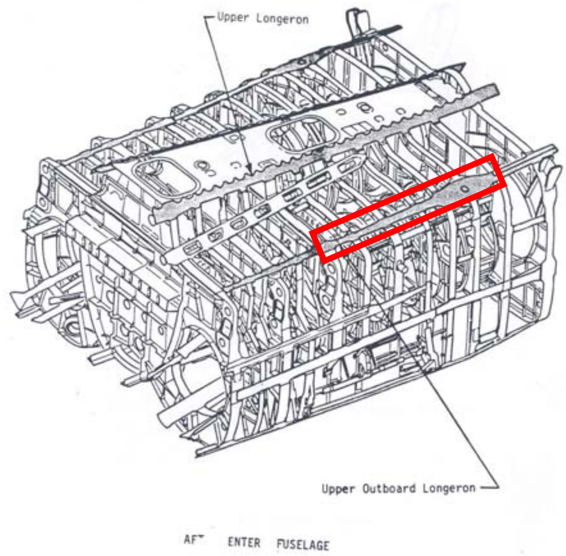
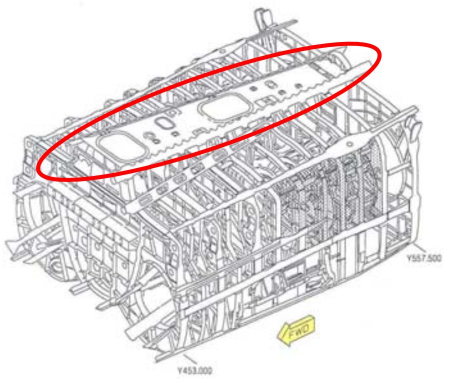


Figure 4.5.6 — **Cracks and low Detection Signal in Floor Segment UPR, Y453 to Y557.5, Aft Section Y544 to Y557.5**



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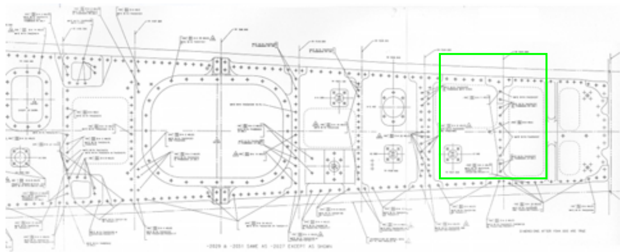
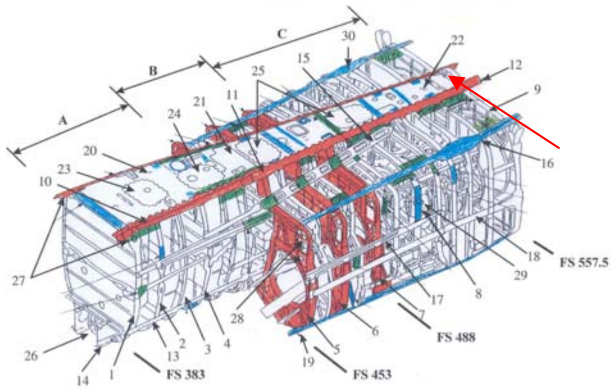


Figure 4.5.7 — **Low Detection Signal only with ET in Hole at Door # 55L, Zone FS 550.00 at L/H Dorsaldeck Longeron**



FM-FSFT0157

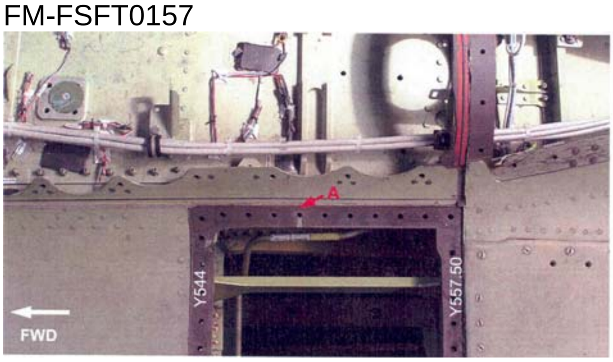
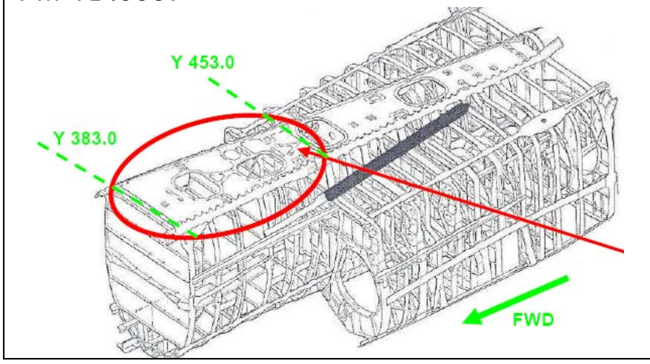
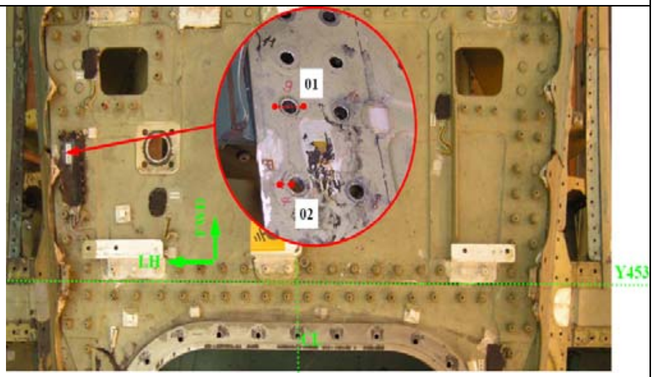


Figure 4.5.8	**Cracks in FLOOR SEGMENT, CENTER SECTION, Y383 - Y453**
FM-TDI0067 	

The remaining 32 parts are classified as non structural.

A total of 65 Swiss FSFT and 50 TDI ¨Fehlermeldungen¨ were created to document the damages. Similar damages were observed in the US Navy ST16 test. In most parts during the ST16 test more severe cracks were observed.

To study the stresses and strains at the critical locations in the center fuselage the already existing Finite Element Model (FEM) was refined and checked in detailed with Swiss drawing set, see figure 4.5.9. The measured strains in the test were also used and compared with strains from the FE model.

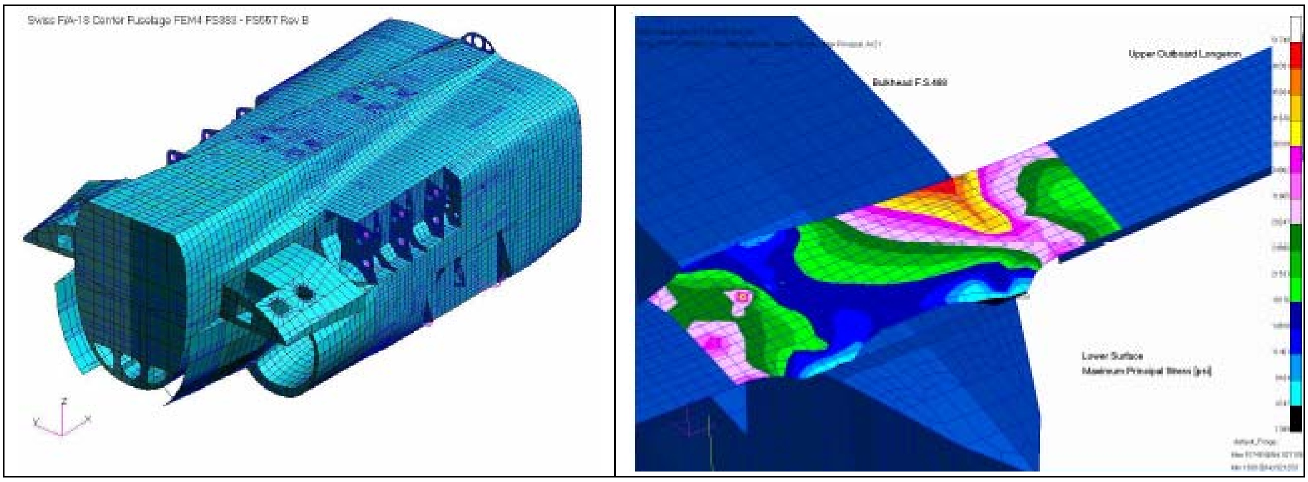


Figure 4.5.9: Detailed Swiss FE model used for analysis with details in the upper outboard longeron at FS489

Outer and inner wing

The tear down inspection on the wings started in March 2007. First the upper composite wing skin was removed from the outer wing. As a surprise a large crack was observed running in spar direction on the third spar. The crack was only observed on the RH wing, see figure 4.5.10. This anomaly has to be investigated in more detail.
Afterwards the inner wing will be tear downed. All areas will be inspected in detail. The tear down will be completed by end of 2007.

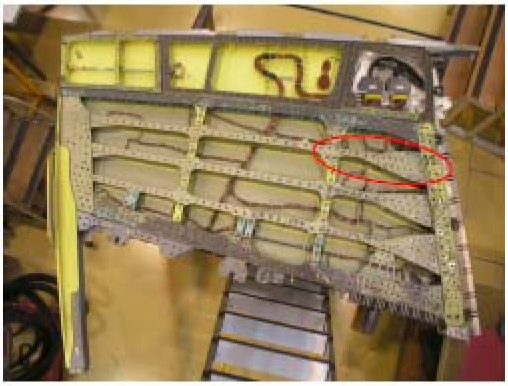
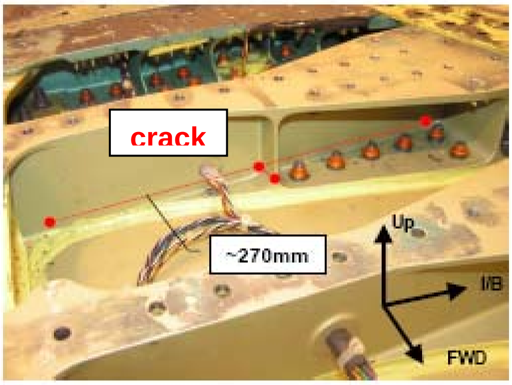



Figure 4.5.10: Observed crack anomaly at the RH side of Swiss FSFT (outer wing)

All the damages of the Swiss FSFT are documented on data base which is linked to the test results of all the US Navy tests. The data base is arranged in that way that the configuration is the leading data. The Swiss configuration is unique therefore most results of the test are only valid for the Swiss.

4.6 Swiss F/A-18 Inboard Leading Edge Flap (ILEF) Operational Loads Measurement (OLM) Program

S. Büsser, B. Bucher, M. Nievergelt

Introduction

During the Swiss F/A-18 Full Scale Fatigue Test (FSFT), cracks of considerable size have been detected in both the left and right ILEF after only 3000 Design Spectrum flight hours. Such early cracking was not expected, as this location was improved by Swiss-specific modifications. Although the ILEF was primarily used as a load introduction component for the Inner Wing during the FSFT, and the loads were not intended to fully represent expected usage loads, the FSFT results are representative enough to consider the damages as fleet relevant. In addition, similar ILEF damages have been found in other fatigue tests (i.e. FT01/ST16/FT93), as well as in fleet aircraft of other users. Based on these findings, the cracks in the ILEF spar were rated as a Major Structural Issue for the Swiss F/A-18 fleet.

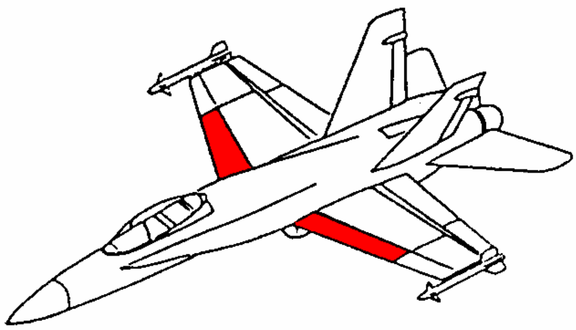


Figure 4.6.1: Inboard ledading edge flap (ILEF)

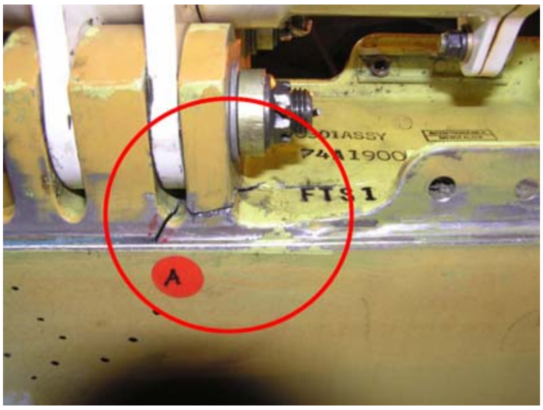


Figure 4.6.2: Cracked FSFT ILEF

The structure of the Swiss F/A-18 is designed for 5000 Swiss Design Spectrum flight hours. The goal of the Swiss F/A-18 ILEF OLM program is to determine whether the ILEF in the original configuration will meet this requirement, or whether a preventive structural modification or a component replacement is required. In order to answer this question, it is necessary to obtain more representative data for life prediction of fleet usage. It is therefore necessary to first capture

all loads that contribute to cracking, determine their influence on the local stress at the critical location (root cause analysis) and then to generate a local stress spectrum at the critical location. For these reasons armasuisse has tasked RUAG Aerospace to perform an ILEF OLM program.
The ILEF OLM program is on track will be completed by the end of 2007.

Instrumentation

The left hand ILEF of the instrumented flight test aircraft has been equipped with 17 strain gages and 3 additional strain gages were applied to the inner wing front spar. Some of the strain gage locations on the instrumented flight test aircraft are identical to those on the FSFT article, allowing a direct comparison between the two data sources. In addition to the strain gages, 3 temperature sensors have been installed on the ILEF spar. The temperature data will be used to account for the temperature effects on the strain gage measurements.
The instrumented aircraft J-5001 and its original data acquisition system (gun bay pallet, ALBUS) have been used to capture the flight parameters from the aircraft data bus, as well as from the 23 dedicated analogue channels.

Calibration

To better understand the relationship between the applied loads and the measured strains and to determine their transfer function, a ground calibration of the ILEF was performed. The calibration took place during a regular inspection of the instrumented flight test aircraft. It was required that the ILEF calibration must not cause any additional down-time.
A total of 21 calibration load cases were applied. Two calibration load cases were performed “on aircraft” with the ILEF installed. The aircraft was jacked up and calibration loads were reacted by the a/c weight only. The loads were applied with hydraulic jacks and special load introduction equipment was manufactured. Other load cases were performed “off aircraft” whilst the ILEF was retained by a calibration rig.

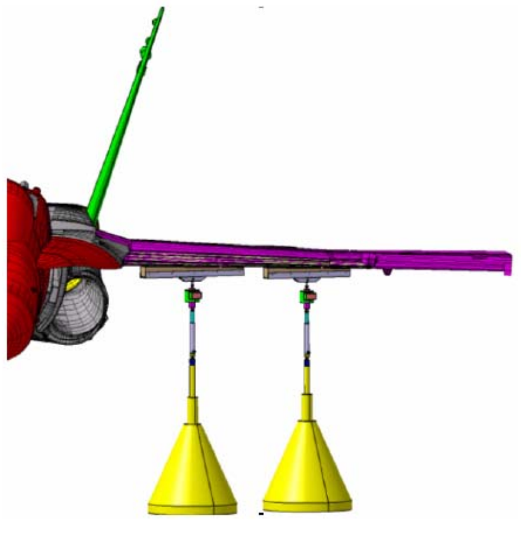


Figure 4.6.3: On a/c calibration

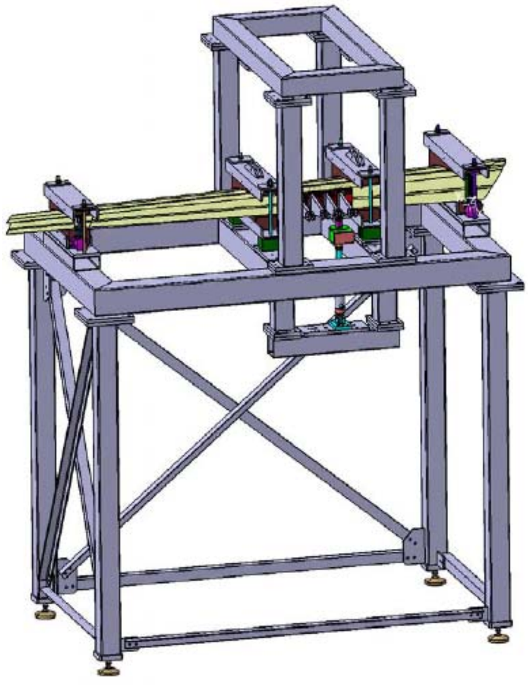


Figure 4.6.4: OFF A/C calibration

To gather the strain gage data during the calibration, the data acquisition unit of the a/c was used. The jack forces were monitored and recorded with a separate mobile data entry system.

Flight Testing

A dedicated ILEF OLM flight test program was developed. The main focus of the program was to test the fleet usage envelope of the Swiss F/A-18 and the corresponding flight sequences. Typical maneuvers and the corresponding Points In The Sky (PITS) were identified and selected for flight testing. The external stores configuration of the wing was changed between the flights and some flights were performed with clean wing as well as with heavy external stores mounted.
Additional flight maneuvers from the Swiss FSFT, the Swiss ASIP Study and some fatigue load cases that were applied to the ILEF Finite Element Models were also integrated into the program.
The test flights were performed by the test pilots of the armasuisse flight test department and the flight test phase was completed by the end of February 2007. The data acquisition system worked reliably and very valuable data was collected during the flights.

Flight data evaluation

To determine the functional relation between the flight parameters (Normal acceleration, Speed, Altitude, etc.) and the ILEF flight loads represented by the strain gage readings, the method of the Artificial Neural Network (ANN) will be used. The input values to the ANN are common flight parameters from the regular F/A-18 Fatigue Tracking System (FTS), as recorded by each aircraft during the flight. This allows the calculation of the ILEF flight load sequence for basically any flight by feeding the recorded FTS data to the ANN.
As the flight data collected with the instrumented flight test aircraft during the test flights will only be used to develop and train the network, a separate set of Swiss F/A-18 FTS data is currently being prepared to represent the average usage severity. This data set will be processed with the ANN to derive the Swiss ILEF usage spectrum.
A detailed Finite Element Model of the ILEF spar has been developed. The calibration load cases were applied on the FE Model and the measurements were compared with the results of the simulation. In general the comparison showed a good correlation. With the use of the FE Model, the procedure to calculate the material stress at the critical location as a function of the strain gage readings was defined. This procedure will then be used to determine the stress at the critical location.

4.7 Assessment of F/A-18 Buffet Issues

S. Frei, M. Nievergelt

It has been established that the buffet spectrum of the Swiss fleet is more severe than in design. For this reason, several aspects of Swiss buffeting are currently being investigated.

Based on a review of relevant F/A-18 buffeting reports, potential critical locations have been defined on the vertical tail and aft fuselage stubs. The most critical locations are the structural parts of the 62% and 77% spar of the vertical tail, as well as the last two aluminum aft fuselage stubs Y580 and Y590.

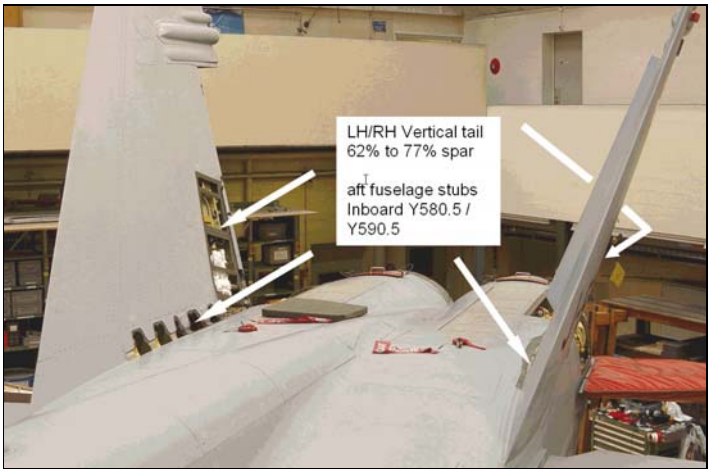


Figure 4.7.1: F/A-18 Buffeting NDT-Inspection

These components have been NDT-inspected in the Swiss fleet leader (based on an AoA>10° criterion). No cracks were found. Investigations are also underway to investigate Swiss F/A-18 buffeting from measured in-flight data.

Another approach used to examine the severity of buffet loading in the Swiss fleet, has been the development of the dynamic spectra of buffeting affected parts. (Procedure according to the MDC buffeting analysis 1991/1992). The most interesting results are obtained from a relative comparison of the design spectrum, the test spectrum from the FT97/FT98 and the current Swiss fleet usage spectrum.

4.8 F/A-18 Center Fuselage Structural Refurbishment Program (SRP)

M. Nievergelt

Based on early cracking found during the Swiss F/A-18 Full Scale Fatigue Test (FSFT) as well as findings in other F/A-18 fleets, Switzerland has set up a list of Major Structural Issues that do not fulfill the full life requirement and consequently require corrective actions. Safety of flight is ensured by a safety by inspection philosophy. However, on condition repairs tend to cause unplanned need of resources as well as delay. In order to ensure economic maintenance and optimal fleet availability, it was decided to take preventive actions that can be planned and managed.

The refurbishment of structural parts in the area of the center fuselage often requires access to the fuel cavities and consequently the removal of the fuel bladders. Removing the fuel bladders and other fuel subsystems is labor intensive. In addition, removal of the fuel bladders often causes damage to the bladders which may have become brittle in time. Therefore, it was decided to capitalize on synergy effects, i.e. to combine the refurbishment of all structural parts that are prone to early cracking and require fuel cavity access into a once-in-a-lifetime Center Fuselage Structural Refurbishment Program. The most important parts of this category are the Upper Outboard Longerons (UOL), the Fuel Cell Floors (FCF) and the Fuel Barrier Webs (FBW).

For the UOL and the FBW, the application of doublers is envisaged. The currently installed FCF made of chemically milled aluminum sheets will be replaced by titanium sheets. For all components, local life improvements such as cold working of holes are considered in order to further enhance the fatigue behavior and to compensate for increased bearing loads at fasteners transferring load into doublers and stiffened structures.

The urgency is mainly driven by the UOL for which fatigue lives of less than 1000 flight hours have been predicted based on crack initiation analysis. As of the end of 2006, the fleet leader accumulated more than 1400 flight hours. The production of a prototype is planned for 2008, the fleet refurbishment for the period of 2009 to 2011.

Illustrations:

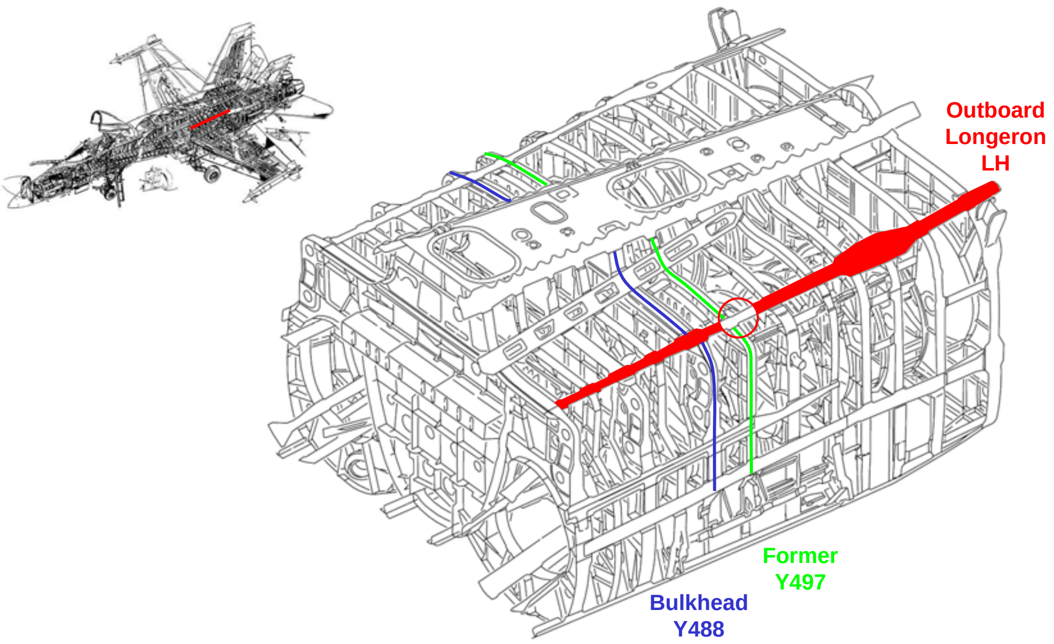


Figure 4.8.1: Upper Outboard Longeron

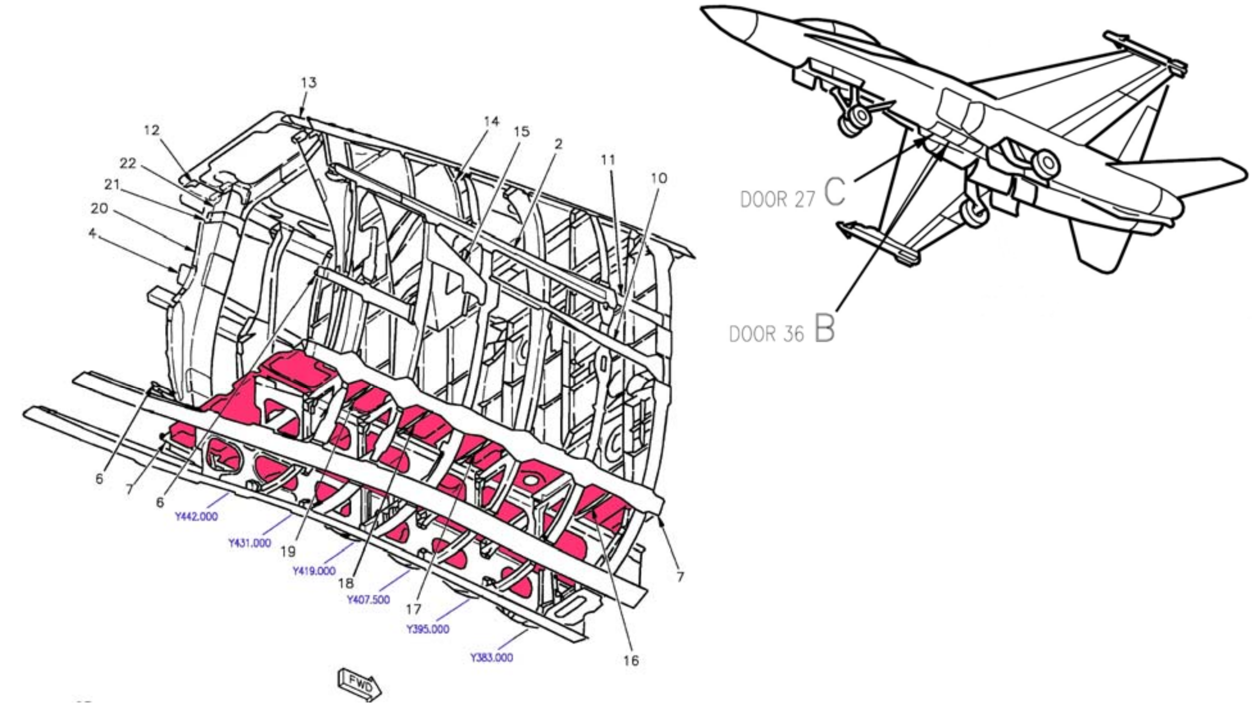


Figure 4.8.2: Fuel Cell Floor

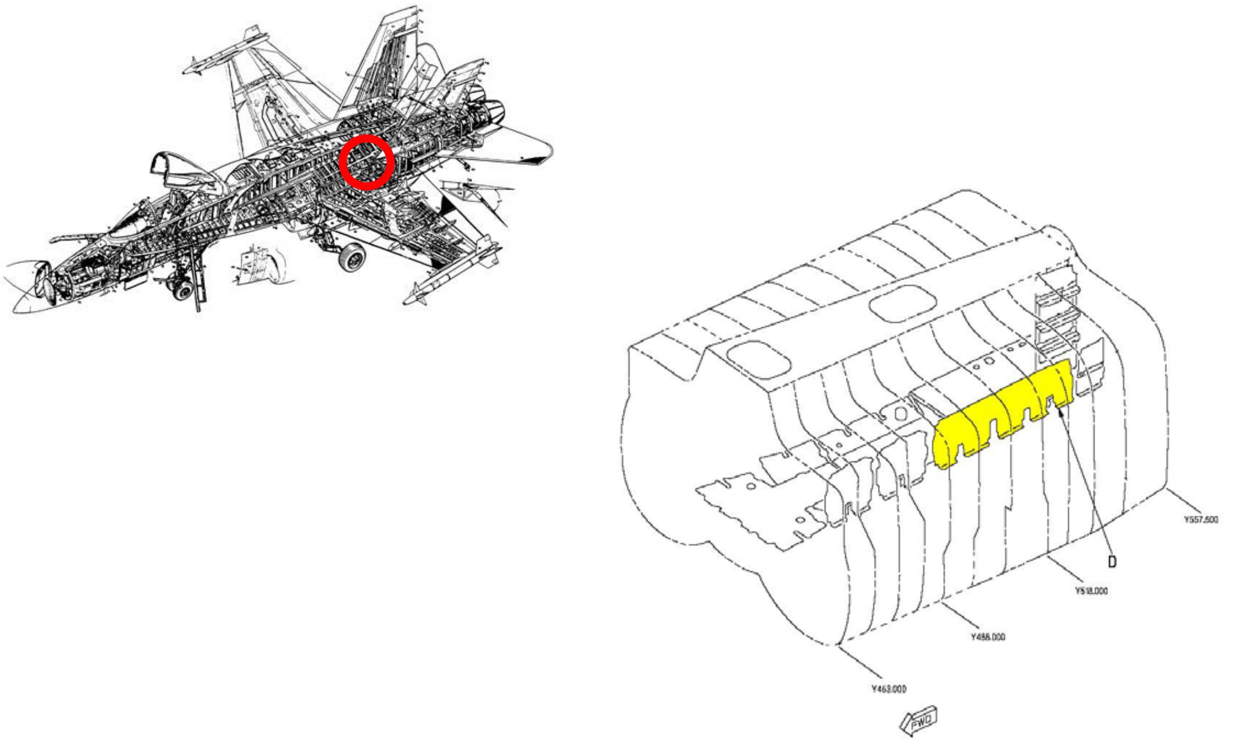


Figure 4.8.3: Fuel Barrier Web

4.9 PILATUS PC-21 Flap Drive System Endurance Test (FDSET)

B. Bucher, B. Schmid

Introduction

The purpose of the PC-21 Flap Drive System Endurance Test (FDSET) was to comply with FAR Part 23.701a and CRI-D-11, Issue 2. Pilatus Aircraft Ltd. Stans/Switzerland tasked RUAG Aerospace, (RA) to set up and perform this 6 life times/90'000 spectrum flight hours endurance test. At the end of the test, the Flap Drive System had to withstand one select design limit load case. The test was prepared between January and April 2005, and was completed in August 2005 at RA premises.

Description of the PC-21 Flap Drive System

All structural components of the flap drive system are mounted to the aircraft wing structure. The PC-21 flap system incorporates a one-piece flap surface on each wing. These two flap surfaces are driven by a hydraulic actuator, which is linked by push-pull rods and links. The flap movement is a rotation about the hinge lines.

Due to the limited space available in the PC-21 design, there is no independent mechanical interconnection possible between the movable flap surfaces. Therefore the flaps are considered as two individual movable surfaces according to FAR 23.701(b). Due to the dihedral shape of the wing, the hinge lines of the two flaps are not collinear. The layout of the flap drive system is a safe life design.

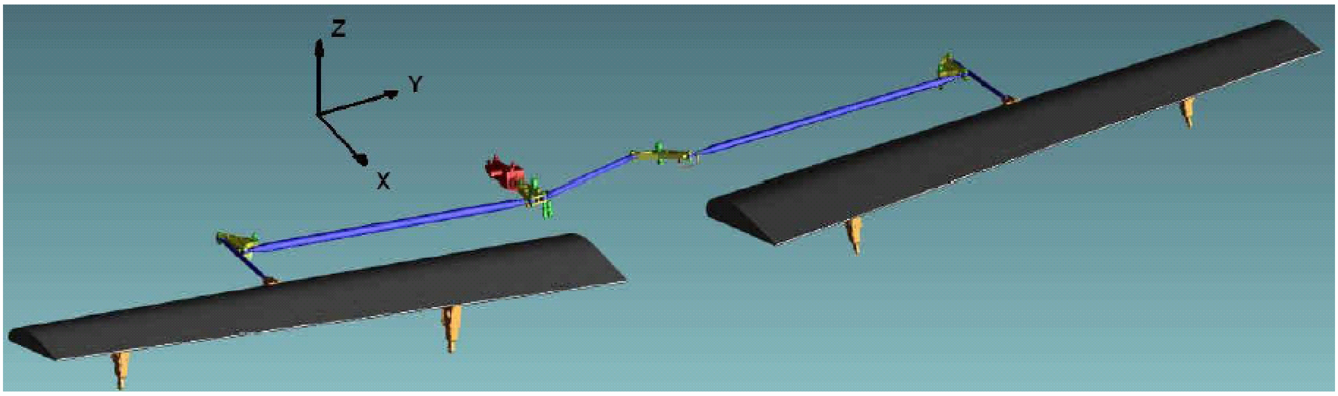


Figure 4.9.1: PC-21 Flap Drive System, Flap Actuator, Flaps and Flap Hinges

Description of the Test Set Up

The test set-up consisted of the test article (flap drive system), a mechanical test rig, a loading system, a flap actuator, two hydraulic systems, a control & monitoring system and a data acquisition system.

The mechanical test rig was a stiff mixed steel/aluminum design. All interfaces to the flap drive system to be tested were made of aluminum. The flap drive system was mounted to the rig with original aircraft fasteners. The same geometrical tolerances used for the aircraft were used for the mechanical test rig.

The loading system consisted of two loading actuators and load application levers. The load application levers rotated about the flap hinge lines. The loads applied corresponded to the required hinge moment defined in the flap spectrum. Two single grooved ball bearings were used for the rotating levers to minimize friction losses.

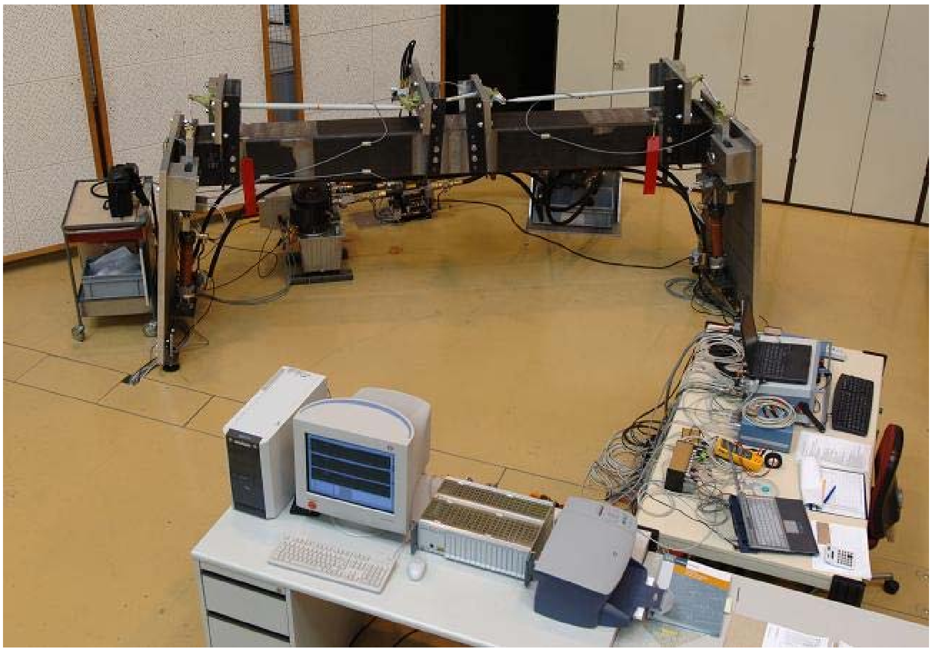


Figure 4.9.2 PC-21 Flap Drive System Test Set-Up, Control and Monitoring System, Data Acquisition System, Mechanical Rig with Loading Actuators and Flap Actuator Pump

The original aircraft flap actuator was used to set the flap positions to 0, 20 or 34 degrees. Reaction loads were changed simultaneously by the control and monitoring system while changing the flap position. For the flap actuator, a separate hydraulic system was used. The flap actuator was not part of the flap drive system to be tested.

The control and monitoring system surveyed the flap position indicator and the load cells mounted to each load actuator. Loads varied simultaneously with the flap position. The control and monitoring system ensured that the applied loads were within the required limits. It also ensured a controlled shut-down procedure for system failures during the test or in the case that load limits were exceeded.

To ensure proper operation of the 10 axial strain gages applied to the push-pull rods of the flap drive system, a shunt calibration was performed during commissioning.

The data acquisition system collected loads, flap position and strain gage data with the required sampling rate. The raw data was peak-valley filtered.

Conclusions

A total of 91’800 flight hours (510 FH per block x 30 blocks per life x 6 lives) were tested. This represents 6.12 lifetimes of aircraft design.

During the test, daily walk-arounds and periodical close visual inspections every 5’100 hours were performed. Periodical continuous measurements, strain surveys and design limit load tests were also performed. All pass/fail criteria were met during the test. After test completion, PILATUS performed close visual as well as non-destructive inspections of the flap drive system subcomponents.

The FDSET completed fatigue cycling and the final limit load test successfully without catastrophic failure of any structural element (push-pull rods, ball joints, levers, fittings, bolts, and fasteners) of the test article.

In addition, the test provided valuable information for the definition of a maintenance concept of the Flap Drive System ball joints.

4.10 Manipulation of Residual Stresses and their Impact on the Durability of Aluminum 7050-T7451 Structural Parts

A. Uebersax, M. Geering

Most structural parts are affected by residual stresses that are intentionally or unintentionally induced during the manufacturing process. Residual stresses are superimposed with the stresses caused by external loading. As a result, the mechanical characteristics of a structural part and, in particular, its durability will change significantly.

The overall goal of this research project is to determine the influence of a specific vibratory stress relief (VSR) process on macroscopic and microscopic residual stresses and to investigate the impact on the fatigue behaviour of aluminum 7050-T7451 specimens. In order to analyze the influence of the process, it is required to measure the residual stress state before and after the application of the VSR process. Aluminum alloy 7050-T7451 was selected, as it is a representative alloy for aerospace applications.

Residual stress measurements were performed by a Cut Compliance Method as well as by the Time of Flight Neutron Diffraction. Both techniques are used in order to gain specific results on one hand, and to be compared against each other on the other hand. Neutron diffraction was chosen because it is a non-destructive method that provides strain and consequently stress data within any chosen location, be it at the part surface or within the part volume. Furthermore, it provides indications on microscopic residual stresses, i.e. second and third level residual stresses. A total of four measurements were conducted on two specimens. One at a cold worked hole and one at a non-cold worked hole, each before and after applying the vibratory stress relief process.

The residual stress measurements before and after VSR-treatment show no significant alterations in microscopic and macroscopic residual stresses in both cold worked and non-cold worked specimens. Nevertheless, a significant difference in the fatigue behaviour was noted in particular between the VSR-treated and non-VSR-treated specimens with the cold worked hole. In order to gain a better understanding of the effect of the VSR treatment of the cold worked specimen, metallographic and fractographic investigations are set up.

4.11 Effects of Exfoliation Corrosion on Wing Structural Integrity

A. Uebersax

Exfoliation corrosion is commonly found in wing upper skins around fastener holes where it originates at the exposed grains in the countersink. To date, found corrosion has to be removed by local blending of the affected areas due to lack of alternative repair methods and lack of accurate methodologies to assess structural integrity taking into account the presence of corrosion. In case of the relatively large wing skin/panel affected by wide spread corrosion, the blending process is rather labor intensive. Consequently, the repair of a wing may no longer be economic.

Using the example of a fighter aircraft's upper wing skin, effects of exfoliation corrosion on the wing structural integrity are being assessed. For this specific situation, processes and engineering tools are under development in order to accurately assess the effects of corrosion presented. In case of the compression dominated upper wing skin, damage due to corrosion, loss of material due to blending processes may lead to structural instabilities (buckling) as well as plastic deformations. Fatigue behaviors are dependent upon the loading spectrum and the local damages. These effects are analyzed by finite element models as well as fracture mechanics modeling.

The modeling will be backed up by a number of static and fatigue tests. Specimens are cut from an upper wing skin fabricated from 7075-T651 aluminum alloy. Fatigue tests will be carried out using pristine, naturally (Fig. 2-1) and artificially (Fig. 2-2) corroded specimens, as well as locally ground out specimens to simulate removal of corrosion by blending.

The results of this project will be essential inputs to set up an effective 'Corrosion Management' allowing to replace the currently used "find-it-fix-it" philosophy and will offer more economic ways of dealing with existing corrosion.

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

Figure 4.11.1: Natural Exfoliation corrosion around fastener heads in aluminum wing skin

Figure 4.11.2: preliminary artificial corrosion tests in different configurations: with/without fastener and with modified fastener head

4.12 Full Scale Fatigue Test of the PILATUS PC-21 Aircraft

N. Rössler, D. Hänni; Pilatus Aircraft Ltd.

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. 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.

Figure 4.12.1: PC21 Trainer Aircraft

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. 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

Figure 4.11.2: PC-21 Full Scale Fatigue Test

monitoring system (in the foreground) ensured that the loads were applied on the correct cylinder in the correct sequence and at the right force.

It took just 15 months (Figure 3) 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"). It consists of 3 types of missions, 22 distinct design sorties, 36 unique types of maneuvers and 69'129 events. 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.

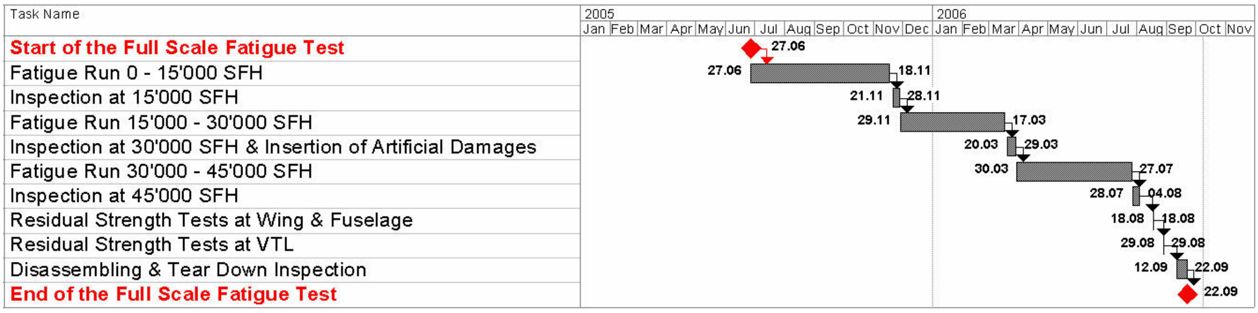


Figure 4.12.3: Time Schedule

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.

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.