Review of Aeronautical Fatigue Investigations in Switzerland

April 2015 – March 2017

Document: ZAV-KOR2017-012

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SUMMARY

The Swiss review summarizes fatigue work in Switzerland. It includes main contributions from the Zurich University of Applied Sciences (ZHAW) and RUAG Switzerland Ltd. (RUAG Aviation). This document 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 35th ICAF Conference

Nagoya, 5th and 6th June 2017

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 April 2015 till March 2017. The various contributions to this review come from the following sources:

- Zurich University of Applied Sciences (ZHAW); School of Engineering, Institute of Material Processing & Centre for Aviation

- RUAG Switzerland Ltd., RUAG Aviation; Structural Engineering

All the interesting contributions are gratefully acknowledged, especially the effort of Markus Gottier (Gottier Engineering), Gregor Peikert (ZHAW), and Andreas Uebersax (RUAG Aviation).

The financial support from the Swiss Federal Office of Civil Aviation (FOCA) is gratefully acknowledged for the activities of the Ageing Aircraft project. Personal thanks to Mr. Rolf Meier from FOCA for supporting the project and the Zurich University of Applied Sciences, ZHAW in Winterthur.

4.2 F/A-18C/D Structural Refurbishment Program (SRP)

RUAG Aviation

A complete airframe fatigue test of a Swiss F/A-18D was carried out between 1998 and 2004 in the RUAG facility in Emmen. The results of this test, combined with the data of shared experience from other F/A-18 operators, culminated in the development of a structural refurbishment program that will improve the operational availability of the F/A-18 in the long term.

The fatigue test – simulating 10'400 flight hours

It has been more than 20 years since the Swiss Air Force procured 34 F/A-18 Hornets. An additional F/A-18D airframe was acquired and, on behalf of armasuisse, an Swiss Air Force F/A-18 design flight spectrum was tested on the full scale fatigue test.

The Swiss Air Force flight spectrum required for the F/A-18 is more severe than that of other air forces. Therefore, to fulfil the structural requirements, armasuisse commissioned Boeing to install structural modifications and reinforcements during the production of the aircraft. A fatigue test was then required to validate the improvements. The full scale fatigue test was set up to demonstrate twice the foreseen life cycle of 5'000 flight hours. Upon completion of the tests in 2004, the test airframe was thoroughly dismantled and examined for damage.

Figure 1 **A total of 10'400 flight hours were simulated in a full scale fatigue test running from 1998 until 2004.**

Life cycle management

The structure of the Swiss F/A-18 is designed according to the safe life philosophy with additional damage tolerance requirements for fracture critical parts. In principle, this means that fatigue damage is not to be tolerated in any form on the airframe. As a consequence, even the smallest crack on the aircraft must be repaired, and problem areas must be monitored periodically using non-destructive testing (NDT). Using the results from the fatigue test, important information was gained about critical areas of the structure and the anticipated outlay for NDT was reduced. From the same test results, specific life-extending modifications were developed for the SRP 1 and SRP 2 programs, identifying potential critical locations and preventing damage. The modification of critical areas of the F/A-18C/D structure strengthened the airframe and ensured the airworthiness – and safety – of the fleet.

SRP 1: a first selection of critical locations are modified

Multiple modifications were consolidated within a structural refurbishment program. These modifications addressed a limited number of locations that cracked early in the fatigue test, concentrating mainly on the midsection of the fuselage. The fleet retrofit is quite advanced and the final three F/A-18s are awaiting its SRP 1 implementation.

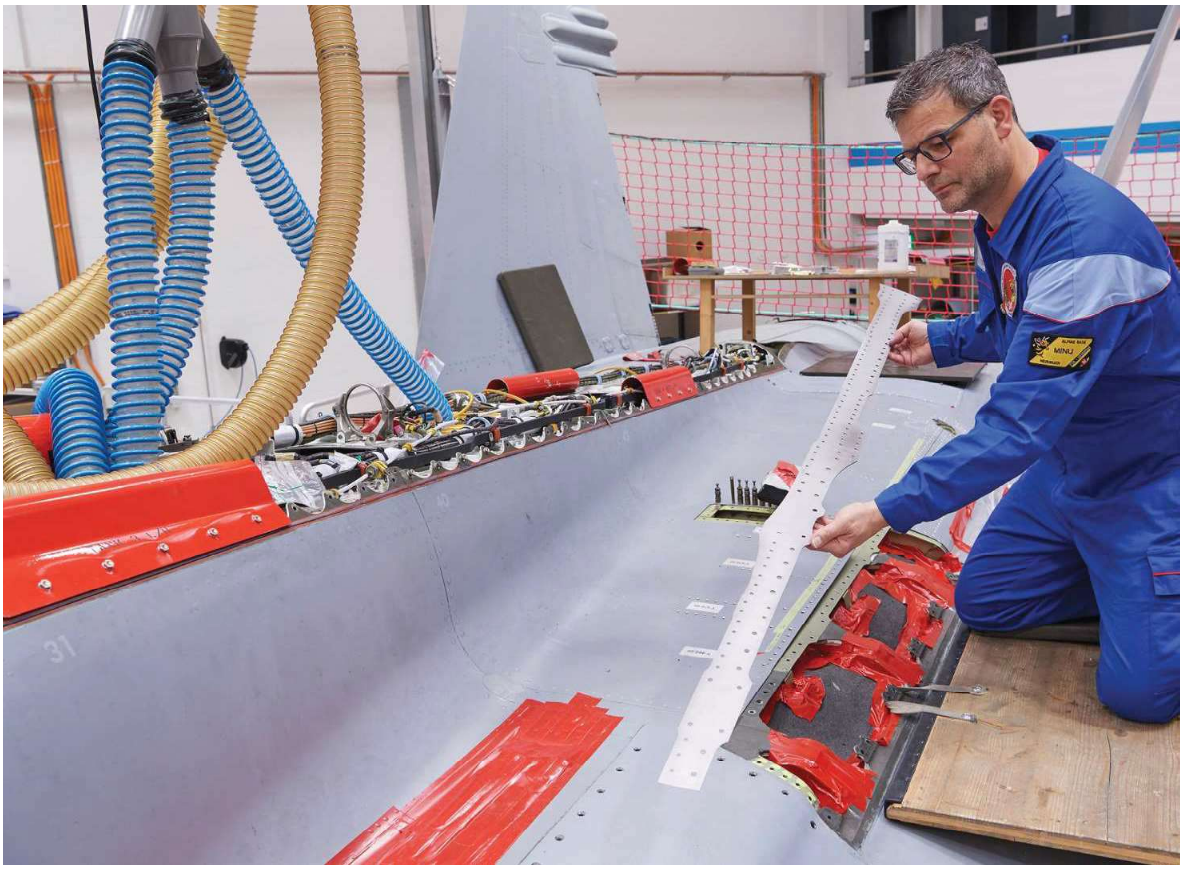

Figure 2 Installation of a new left-hand upper outboard longeron strap during SRP 1.

SRP 2: measures for extending the platform's service life

Further analyses, based on the results of the full-scale fatigue test (1998-2004) as well as other criteria, have made it clear that the implementation of SRP 1 is insufficient to satisfactorily achieve the anticipated service life of 5'000 flight hours. In addition, the "F/A-18 Service Life Extension Program" actually increases the service life requirements to 6'000 hours. As a result, armasuisse has commissioned RUAG to develop a further structural refurbishment program, SRP 2. This development is currently progressing at full speed. The program is significantly more extensive than SRP 1. Around 200 structural locations need to be assessed, inspected and modified where necessary. According to current plans, the program should be implemented from 2018 on.

Following the successful verification and validation of the developments on the first aircraft, the entire fleet should be modified from 2019 until 2022.

4.3 Non-Standard Life Improvement Techniques

Andreas Reber, Raphaël Rigoli, Luzian Michel, Mark Weber, Marcel Koch, Ingrid Kongshavn, Beat Schmid / RUAG Aviation, Michel Godinat / armasuisse

As part of a structural refurbishment program, several holes of a highly loaded structure need to be modified to increase their fatigue life. To do so, split sleeve cold expansion or an interference fit bolt are considered as feasible techniques. Both techniques are commonly used to improve fatigue life. They are investigated in detail concerning their suitability to the herein presented case which features special and problematic conditions. The material has to expand in the weak S-T grain direction, material data shows a low ultimate strain allowable and therefore the risk of static material failure during cold expansion of the hole. The hot spot stress magnitude is at such a high level that it could affect the commonly used life improvement factor for both considered techniques. The interference fit would be achieved by a special radial lock fastener, which requires a proof of the life improvement factor. To cope with these issues, several tests are planned and running, split sleeve cold expansion process tests on one hand and life improvement factor fatigue tests on the other hand.

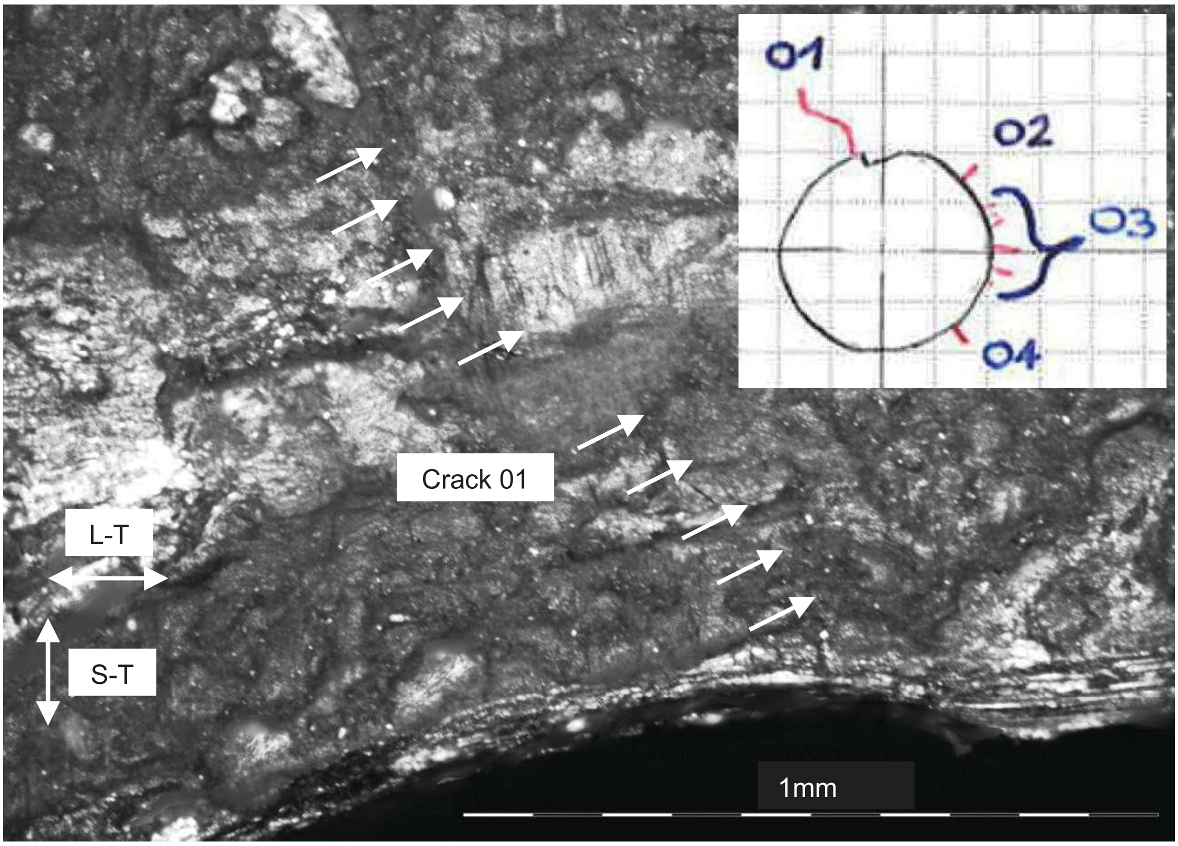

Figure 3 Topview on cracks around the split sleeve cold expanded hole with longest crack 01 in detail.

The split sleeve cold expansion process test applies the cold expansion process at different strain levels and varying split sleeve orientations. The test structure is set up very similar to the original structure (i.e. material, grain orientation, thickness, hole size, edge distance, hole spacing and split sleeve orientation). First test series did not reveal complete material failure, even at a strain level considerably higher than the strain level defined in the process specification. Shear tears as shown in Figure 3 were nevertheless identified but by now assumed to develop only in the more brittle surface layer due to surface treatment. More detailed crack inspection is planned to verify that assumption.

The life improvement factor fatigue test consists of 3 to 5 coupons for non-life improved holes as well as for life improved holes. Both techniques, cold expansion and interference fit bolt, are tested to show the life improvement factor. The test coupons are again set up very similar to the original structure. That time load spectrum and stress magnitude are considered as well. Some of the coupons were manufactured out of an existing aircraft structure (from a fatigue test), they are therefore to a high degree representative. To ease eventual quantitative fractography and to validate the coupon design concept, preliminary coupons were tested successfully. Two different marker band concepts were tested and revealed different advantages and disadvantages. One marker band concept features an overload. The overload creates crack retardation for the non-cold expanded holes and will probably reduce the advantageous compressive stresses at the cold expanded holes. These circumstances will influence the provable life improvement factor but provide a very good readability of the crack surface. The other concept features a constant amplitude marker band. The constant amplitude marker band will probably show a negligible influence on the life improvement factor as its magnitude is rather low compared to the maximum peak of the load spectrum. Unfortunately the constant amplitude marker band is harder to read. To select a marker band concept, a tradeoff or combination will be necessary.

The coupon design concept was partially validated with 4 preliminary coupons. All cracks appeared in the desired hole section, see Figure 4. The adhesively bonded and interference fit bolted load application section showed no evidence of a crack.

Figure 4 Preliminary coupon design concept and validation, second hole of the row cracked under cyclic loading.

Meanwhile the test program will be continued as planned, based on the results of the preliminary tests.

4.4 Hail Damage Investigation

Ingrid Kongshavn, Andreas Uebersax, Florian Oggier, Christoph Kunz, Josef Lussi / RUAG Aviation

An investigation was performed on F/A-18 aircraft which had suffered damage during a hail storm, to compare the NDT results using ultrasound and radiography, to a destructive investigation using PATTI Jr testing and micrographs. The pitch and catch method was chosen as it is preferred over through transmission for inspection of curved surfaces (dents). X-ray phase contrast imaging (XPCI) was also used to detect fibre failure.

The Bondmaster NDT results of the inspection indicated 'core unbonds' and the radiography results indicated crushed core. The NDT results classified the damage as being more severe than the classification based only on the dent dimensions, position and pattern. It was therefore necessary to better understand the damage that was being detected by NDT.

The NDT signals with vs without top coat were similar in the undamaged area and significantly different in the damaged area. This suggested that there was some kind of bond degradation between the top coat and the skin in the area of the dent.

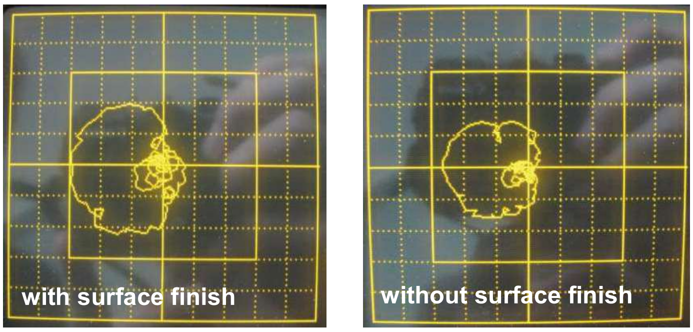

Figure 5 Bondmaster signals with and without top coat in undamaged area.

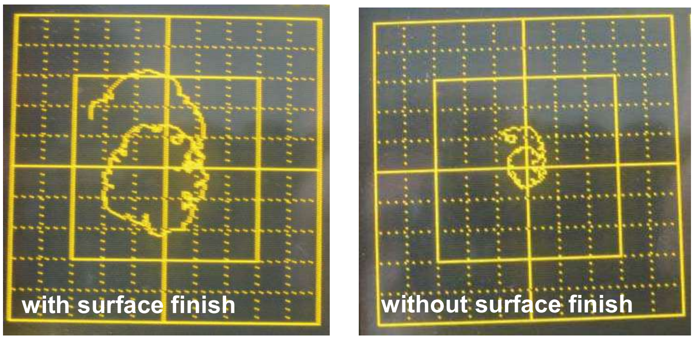

Figure 6 Bondmaster signals with and without top coat in damaged area.

When the PATTI Jr tests were performed (a flat-wise adhesion tension test performed directly on the skin), the skin did not separate from the core and thus there was no indication of core unbond. The micrographs also confirmed that no core unbond was present. They showed the presence of small amounts of matrix cracking, delaminations between the plies of less than 3.5 mm and some paint unbond within the dented area. Some waves in the core next to the skin were always observed which represent the extent of the 'core crushing'. The XPCI radiographs showed possible indications of a very small amount of fibre failure.

The dented skin has an elliptical shape from the hail impact. The ratio of the dent depth to the smallest diameter (minor axis) of each dent was plotted, based on the assumption that this ratio is the critical one, in that it determines the minimum local bend radius of the laminate. The data suggested that a threshold value of this dent depth to diameter ratio lies between 0.012 – 0.016. Above this threshold, some damage is incurred inside the laminate.

It was concluded that a core unbond indication from the Bondmaster does not necessarily correspond to a true core unbond (complete separation of the core and skin). In this case, the Bondmaster signal was most likely being altered from the ‘crushed’ core, paint unbond, interply delamination, matrix cracking and fibre failure.

Minimally invasive core stabilisation

Gregor Peikert, ZHAW

In cases of small impact damages such as from hail, it can be difficult to determine the true extent of the damage from the NDT signals. Further, classifying this damage within the guidelines of the repair manual is not always straight forward, as some core winkling or crushing is inevitably present, which would move the damage into a higher damage classification that may require repair. Two repair scenarios are currently being investigated which would fix or stabilize the crushed core without damaging the skin, restricting further damage growth and leaving the option to perform a repair later on.

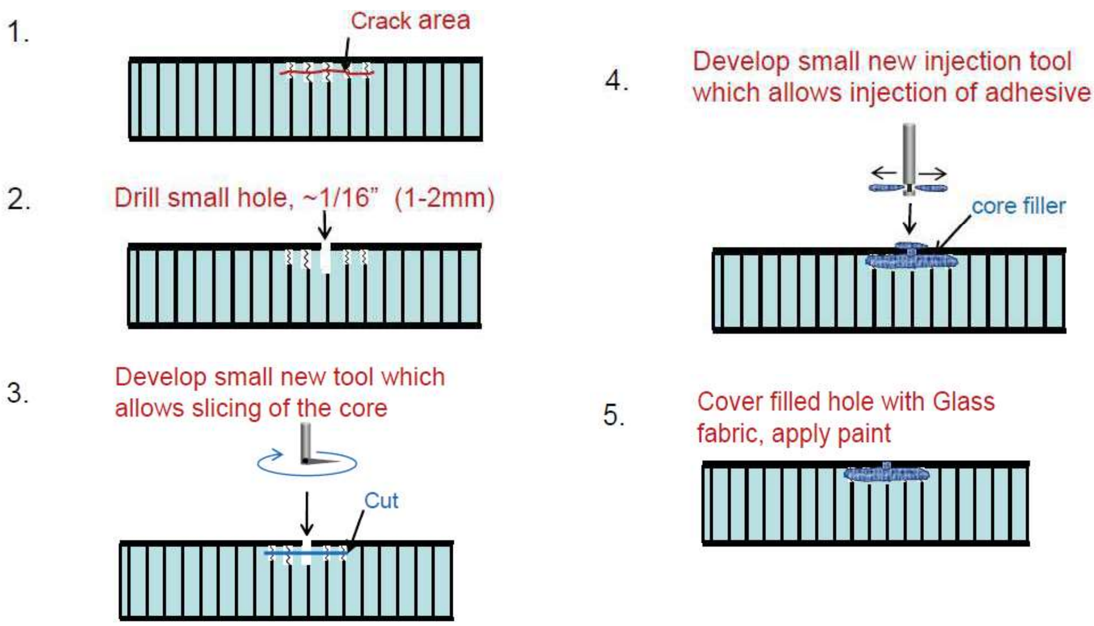

Figure 7 Sliced Core Repair.

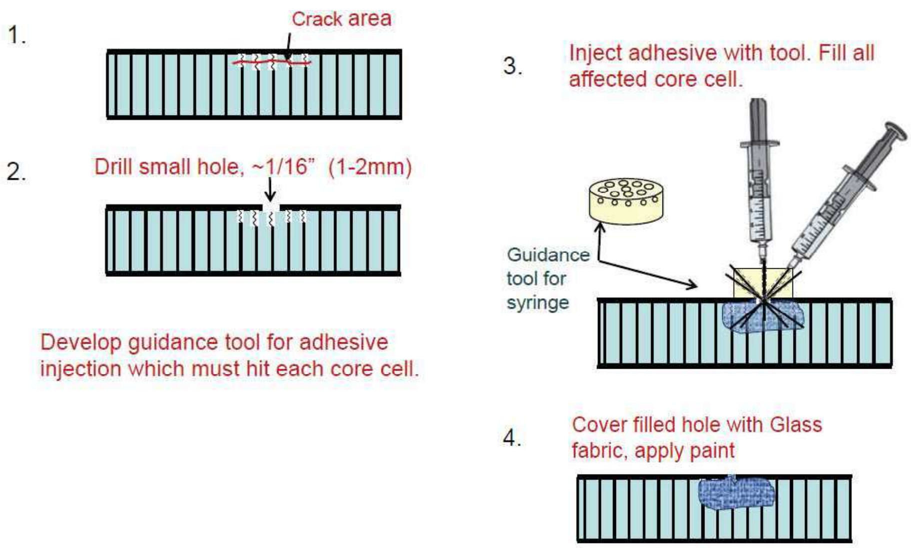

Figure 8 Syringe Repair with Guide Tool.

4.5 Stress Corrosion Crack Depth Estimation Based on Eddy Current Signal Strength

Andreas Uebersax, Raphael Zehnder, Christoph Kunz, Josef Lussi, Stefan Frei / RUAG Aviation, Cyril Huber / ZHAW

Eddy Current Testing (ECT) is widely used today to detect surface breaking cracks. Whereas the method is well suited to detect cracks, its use for the determination of the crack depth is still under investigation, especially if the cracking is due to Stress Corrosion Cracking (SCC).

In most cases stress corrosion cracking leads to parts replacement; nevertheless, crack size information can be of significant use to determine intermediate actions prior to parts replacement and can help to prevent unnecessary grounding of an aircraft.

The study evaluates the relationship of the eddy current signal strength with the actual crack depth of naturally grown stress corrosion cracks and compares them to signals of fatigue cracks and Electrical Discharge Machined (EDM) notches of reference standards. All tests were performed with shielded eddy current pencil probes, like they are widely used for aircraft inspections. The frequency was 350kHz which is generally used to detect surface breaking discontinuities in aluminium aircraft structures.

The investigated SCC were in-service cracks in a fuselage longeron of a fighter aircraft. The cracks developed in the end pad of the longerons, which were manufactured from aluminium 7075-T6511 stepped extrusions.

ECT signal strength was recorded every 2mm along the crack path on the surface for each of the 15 investigated stress corrosion cracks. The stress corrosion cracks were subsequently broken open and the actual crack depth was measured at the positions where the eddy current signal strength was recorded previously.

It is known that there is a difference in the eddy current signal strength between artificial notches and real cracks, which is even more pronounced for stress corrosion cracks. The investigation determines this relationship for cracks in the aluminium 7075-T6511 extrusion.

It was revealed that the ECT signal strength does not show an one-to-one relation to the stress corrosion crack depth. A differentiation of the eddy current signal strength between circumferential and radial cracks could be made. The circumferential cracks seem to give ECT signals partly comparable to fatigue cracks, whereas the radial cracks give much smaller indications compared to their depth. The radial cracks show a significantly lower eddy current signal strength than the reference standard EDM notches. The depth of radial cracks would be significantly under-estimated if it was compared to the signal of an EDM notched reference standard.

The difference of the ECT signal strength of SCC compared to the EDM notched reference standard, as well as compared to fatigue cracks, is assumed to occur because of the different morphology of the different discontinuities and their respective electromagnetic behavior. The same assumption is stipulated and discussed for the ECT sensitivity towards the crack depth of circumferential and radial cracks.

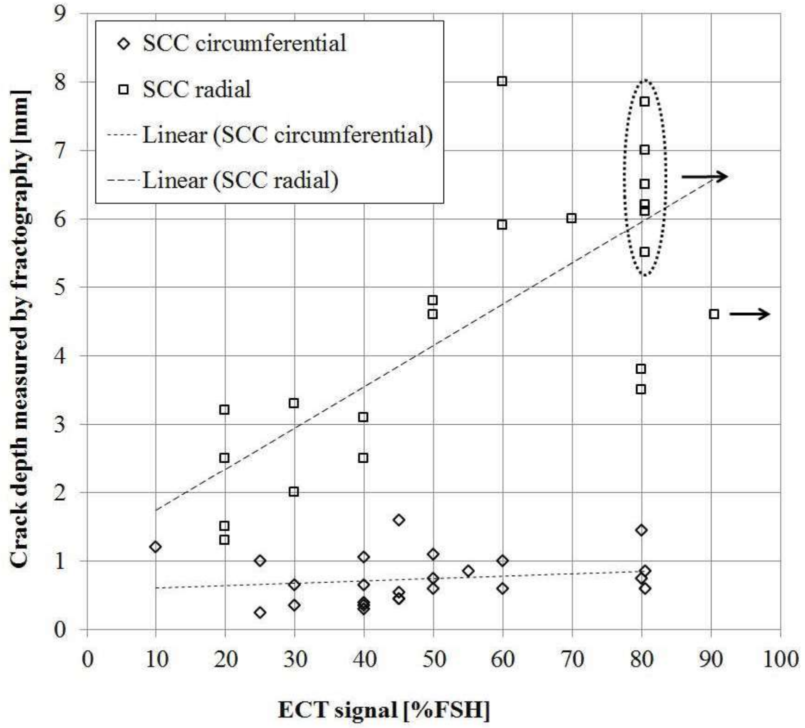

Figure 9 **Eddy current signal strength in % FSH (Full Screen Height) for SCC.**

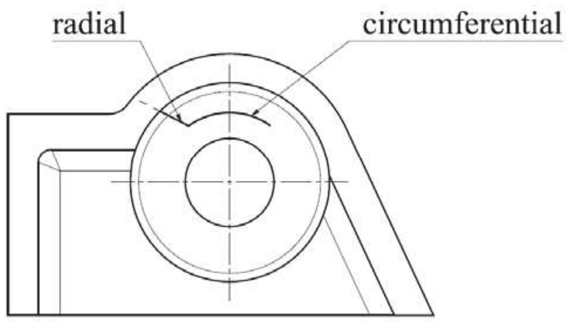

Figure 10 **Location of the circumferential and radial cracks in the longeron end pad.**

4.6 Component Engine Repairs

Michel Guillaume, Joel Kempter, Elias Pertschy, Lukas Müller, Daniel Epting

Zurich University of Applied Sciences / ZHAW

Tomas Valenta

SR Technics Ltd. Engine Repair & Overhaul

Jet engines are very complex systems with many structural parts. Some of them not only are affected by cyclic loading of mechanical stresses but also due to thermal stresses.

SR Technics is in the business of maintenance and overhaul of large commercial jet engines.

If the structural parts are out of tolerances, or structural damages are detected during overhaul, the parts will be scrapped. The costs for new parts mount up to several hundred thousand USD. Hence, SR Technics tasked the Zurich University of Applied Sciences to do an assessment for repair capabilities of fan frame shroud of the CFM56-7B (Boeing 737 NG, see Figure 11 below). The CFM56-7B has some changes compared to the CFM56-5B used on the Airbus 320. The fan frame shroud for CFM56-7B consists of three parts; the outer frame is made of 2219-T6 and the strut hub is Ti-6Al-4V. For the CFM56-5B the whole component is made of one part made of steel 17-4PH.

Figure 11 The fan frame shroud of the CFM56-7B engine

Several inspections on multiple CFM56-7B engines showed corrosion spots on the surface and in the holes on the fan frame shroud. Several factors were considered in order to figure out which kind of corrosion is damaging the structural part. Important was the analysis of the different materials involved in the assembly. The result confirmed the complex situation of fretting, pitting, and contact corrosion. The first conclusion of the OEM was that pitting corrosion is the only source of damage. Several grinding out processes depending on the depth of damage were analysed. For the holes, bushing repair solutions were studied and approved by detail stress and fatigue analysis. This repair (skin blending, with several options depending on the severity of the damage) can now be directly applied by the SR Technics engine shop.

In the meantime, the Outlet Guide Vane (OGV) Shroud of the same engine CFM56-7B (see Figure 12 below) was also affected by severe corrosion. On this component, mainly the surface was damaged by corrosion but the repair should fulfil the full functionality as prior to repair. The proposed solution was to remove the corrosion and use plasma-spraying technology to lay up the material. Important is the process specification to control residual stresses and the surface finishing to establish the original quality.

Very helpful was the use of modern simulation tools to optimize the processes and determine the stress on the component to ensure the airworthiness. Damages with a depth up to 1mm can now be repaired based on this investigation. SR Technics got the approval for this repair on the engine CFM56-7B.

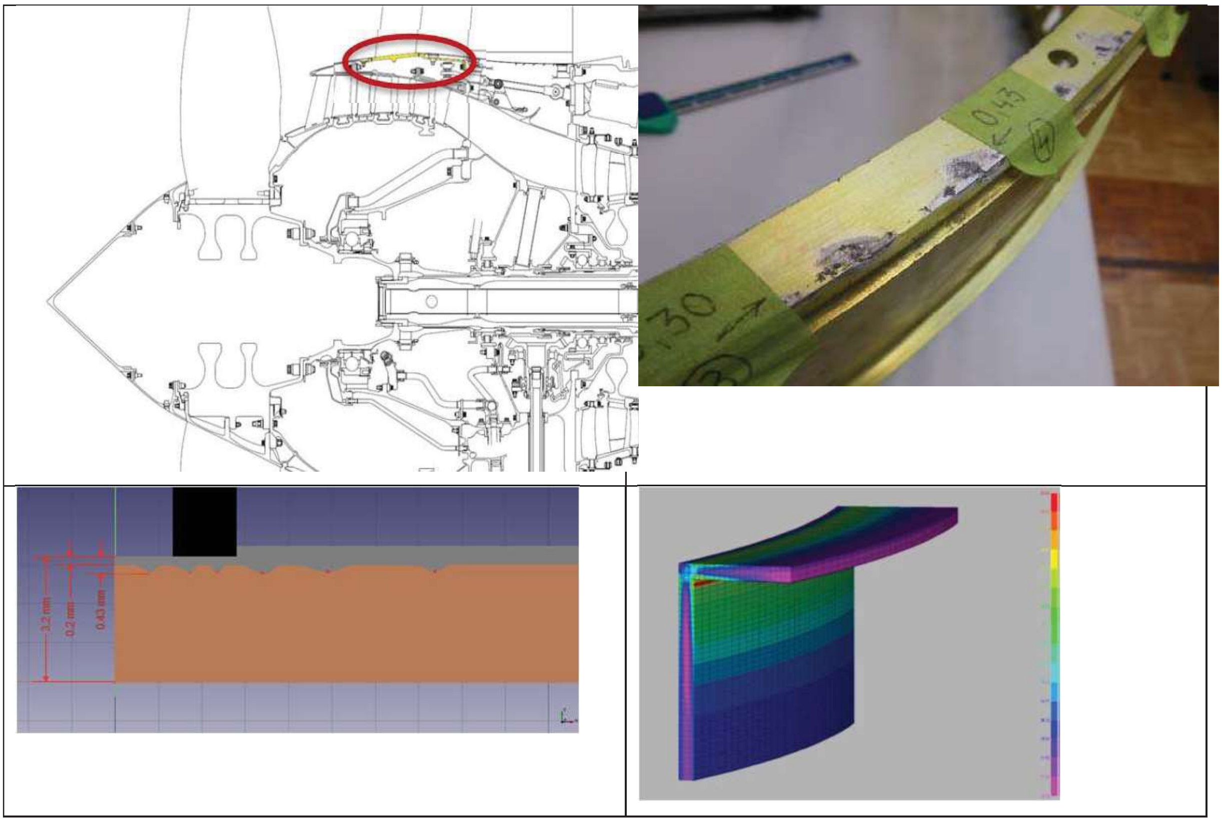

Figure 12 The Outlet Guide Vane (OGV) Shroud of CFM56-7B engine

4.7 Ageing Aircraft Activities for Annex II Airplanes

Michel Guillaume, David Frey, Fabio Mariani, Alexander Kuratle, Xinying Liu, Marina Vannelli

Zurich University of Applied Sciences / ZHAW

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Introduction

The Swiss Federal Office of Civil Aviation (FOCA) started an initiative to address the ageing aircraft where any OEM is anymore responsible for the aircraft design. These airplanes belong to the category Annex II. The following aircraft belong to this category:

- Pilatus P3
- DH112 Venom
- DH100 & 115 Vampire
- Hunter Mk58 and 68
- Mirage III DS
- C 3605
- Junkers Ju-52
- Super Constellation

These aircraft are still in operation in Switzerland by private holders. The regulations has changed in the last 80 years, see Figure 13 below.

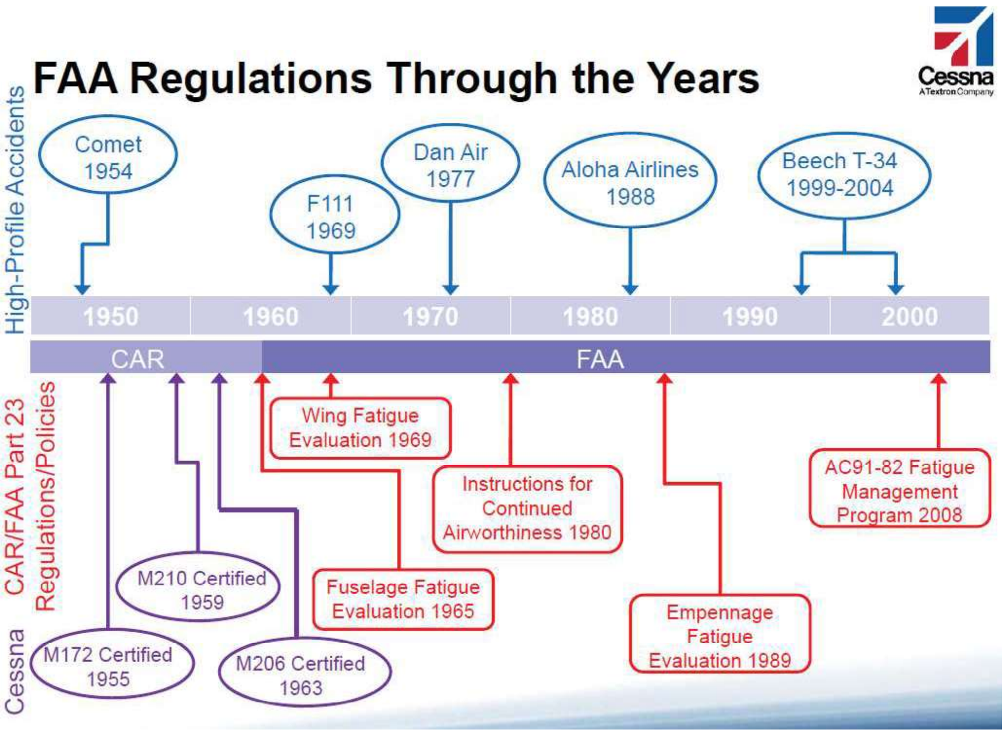


Figure 13 FAA regulations through the years by Cessna

The original design was primarily based on pure static considerations, no special fatigue investigations were done. Therefore, these airplanes are purely safe life airplanes.

For the Swiss military planes Pilatus P3, DH112 Venom and Mirage IIIS full scale fatigue tests were done at the Swiss Federal Aircraft Factory in Emmen. Mr Branger, the first Swiss National ICAF Delegate, was involved in all these tests.

The initiative of FOCA was motivated several incidence/accidents for example high time flying Cessna M400 series. FOCA contacted all the operators of Annex II airplanes and wanted to know the following information:

- Total number of flight hours and landings
- Total number of flight hours and landings for flight leader
- Number of pressure cycled for pressurized cabins
- Performed repairs and modifications
- Performed fatigue tests and results
- Incidences during service (Ad’s and SB’s)
- Fatigue critical structures: analysis, tests available?
- Current inspection program? Adequate and up to date?

The goal is to define special inspections, if life extension is necessary and based on the available data and results, to determine supplemental structural inspection documents to ensure the continuous airworthiness.

The information provided by the operators was far from complete. Especially engineering data was missing. The ex-military planes still were inspected according to the Military schedule and tasks. However, the ageing structure was not taken into account. The Pilatus P3 fleet leader reached 4’270 FH in 2012 but the original design service life was 2’500 FH whereas 5’000 FH was tested in a full-scale fatigue test in 1960.

The Centre for Aviation of the Zurich University was tasked to support FOCA and the operators to analyse the status of the Annex II fleets. Students and research assistants supported some of the work and a big help was the great experience of Markus Gottier in the field of fatigue design and modifications as well as repairs. The state of the activities is well documented.

Pilatus P3

The Pilatus P3 was designed as trainer aircraft and operated by the Swiss Air Force until 1990. The last aircraft was tested in the Swiss Federal Aircraft Factory in 1960 with a full-scale fatigue test using 19 hydraulic actuators. Only symmetric manoeuvers were tested based on a design spectrum with a block of 250 FH, the nz range was -3 g till +6 g. The service life was defined as 2’500 flight hours and 10’000 flights. No static test was performed, but after the first service life, a static load case of 7 g was applied successfully. The test was continued with the simulation of an additional service life of 2’500 FH. After that, a static test was done with an 8 g load and the right hand wing showed some cracks in the web of the spat at rib 12. The web was reinforced and the static test was successfully completed at 12.25 g. No major damage and catastrophic failure was reported.

For a first structural assessment, a special inspection program was set up. All available static and strength reports, as well as the reports from the full-scale fatigue test, were reviewed and served as basis to determine the inspection program on aircraft HB-RCH in Locarno. The partly dismantled aircraft was visited in February 2016. During this inspection, no cracks were discovered. But the main bolt of the main wing fuselage connection at fuselage 3 showed real corrosion and needed to be replaced by new manufactured bolt. For this, an AD was released by FOCA.

Another concern, was the installation of the bearings in the lugs. Especially the lug of the horizontal tail fairing in the aft fuselage showed special stress concentration, see Figure 15 on the next page.

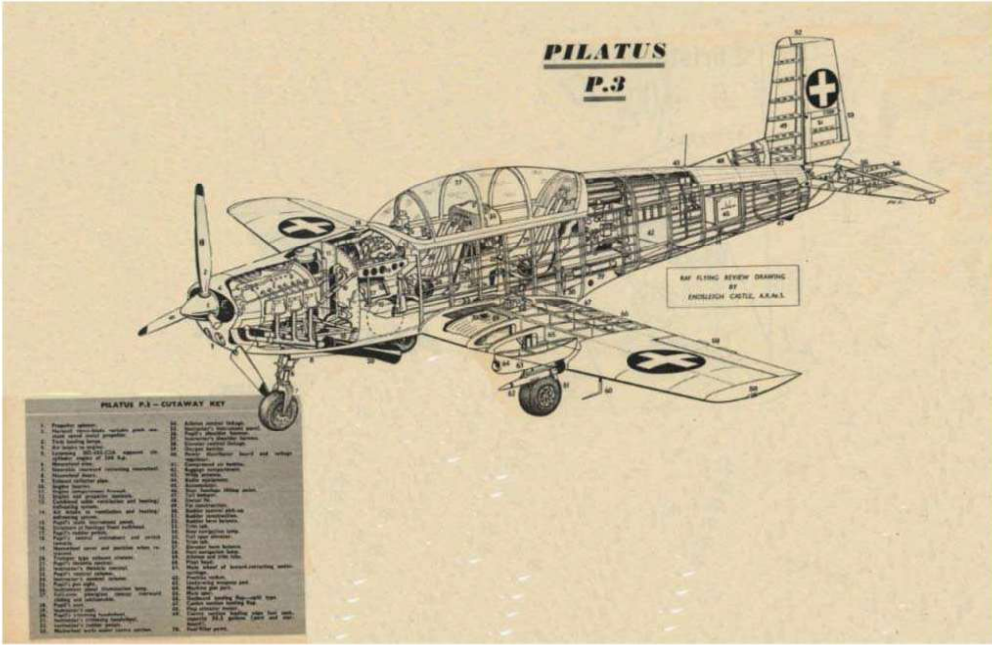


Figure 14 Structural layout of the Pilatus P3

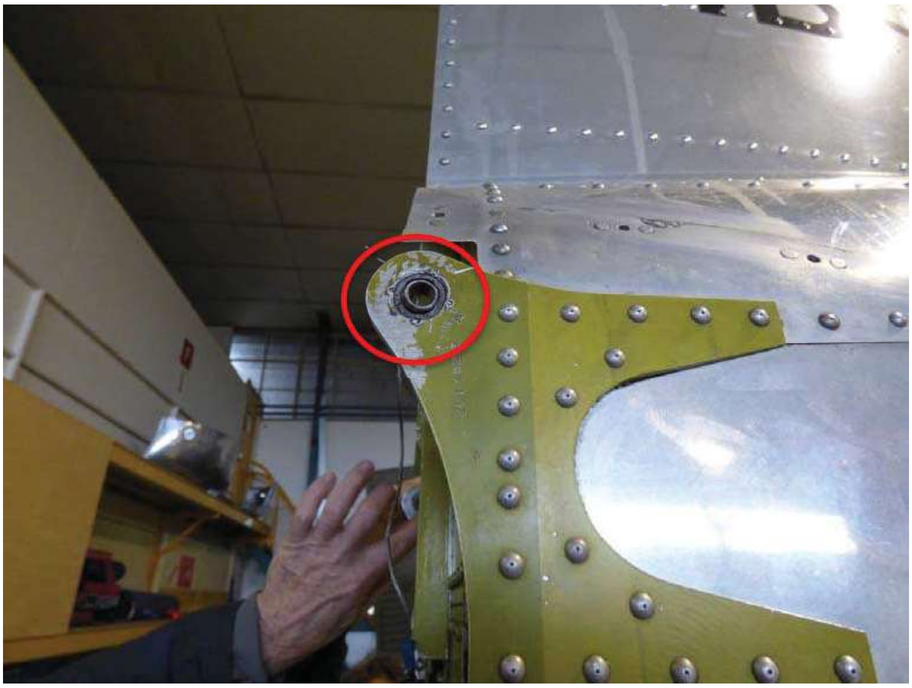


Figure 15 Figure of bearing for horizontal tail fairing on the fuselage

For this area, a structural analysis (see FEM model below) was done, followed by crack initiation and crack growth analysis. The line-by-line sequence of the P3 full-scale fatigue test program from 1960 was no more available. During the usage of the Swiss Air Force, only two aircraft were monitored by nz fatigue meters from Zambra & Negretti. Only exceedance plots from1960 till 1990 were available. For the fatigue calculations, the FALSTAFF spectrum was adopted to min nz of 3 g and max nz of 6 g. Based on special study, the spectrum was modified for the horizontal tail. The crack initiation life was much larger as the crack growth life; the total life showed a factor of more than seven times the service life of the original service life.

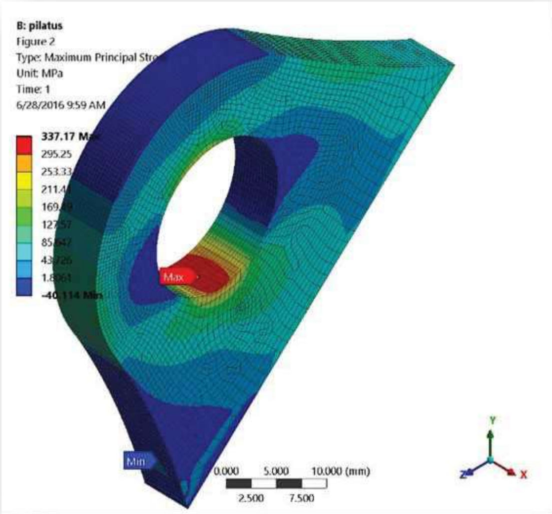


Figure 16 FEM model for the horizontal tail fairing to determine the principal stress for fatigue analysis

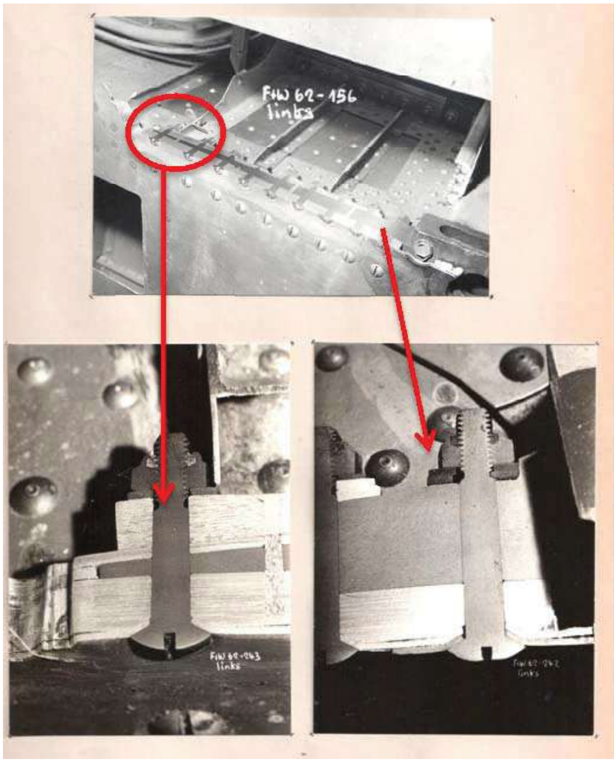


Figure 17 Fatigue critical area of lower spar flange close to rib 12, pictures from P3 test article

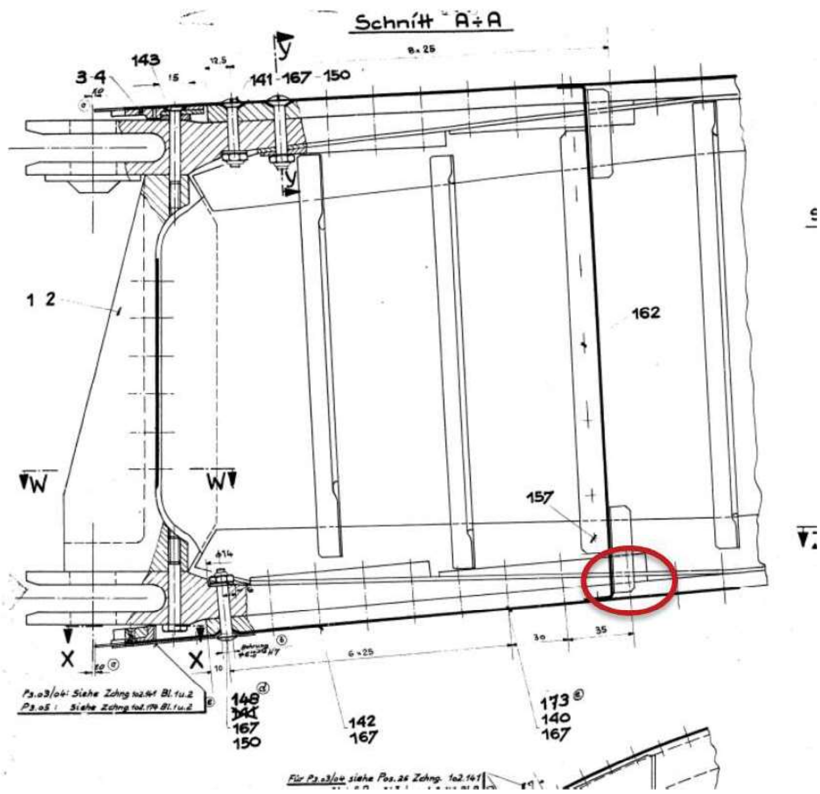


Figure 18 Critical location at main spar flange in the outer wing (red circle)

Another critical location is the lower spar flange in the area of rib 9 to rib 12 (intersection of inner and outer wing). In this area, the web section between rib 11 and 12 was cracked during the static test after the fatigue cycling of 2 service lives. In the final test report, this crack was mentioned to be a fatigue crack. A detailed analysis with loads and structural analysis could not confirm this fatigue damage. Our conclusion showed that this fracture is a pure static failure. However, this area showed high stresses. The crack initiation life showed one service life and the crack growth life was shorter than the service life. For these fatigue analysis the same adapted Nz FALSTAFF spectrum from the above study was used.

For the P3 full-scale fatigue test spectrum a flight takes 15 minutes. In the Swiss Air Force, the average flight took about 50 minutes. If we consider this fact, there is enough potential for further operation after inspection based on the damage tolerance studies to ensure the structural integrity.

When the study will be completed, all the results can be summarized and a special SSID can be developed. Nevertheless, for this final action a support from the P3 operators will be necessary.

Hunter Mk58/68

The Hunter aircraft was developed in the early fifties by the English Hawker Siddeley company. It was designed as a fighter and an air-to-ground attack aircraft.

In 1958 the Swiss airforce ordered 100 single-seat Hunters, consisting of 12 refurbished RAF F.6s, and 88 new-built F.58s. The Swiss Hunter fleet was then enlarged in 1972 and 1974 by the acqusition of two times 30 refurbished aircraft. 8 of the second set of 30 aircraft were double seaters or trainers. In 1994 the Hunter aircraft was completely withdrawn from the Swiss service. At this time the fleet had an average service life of about 1400 flight hours and the fleet leader (a trainer double seater) of about 1700 flight hours. It is to mention that the total life of the refurbished aircraft is not known, since the flight hours of the aircarft and the type of operation before the refurbishment programs are not known.

During the 36 year period of Swiss Hunter era the aircraft was modified and upgraded several times, mainly to gaurantee to operate the aircraft in the 2 roles with new weapon systems.

In 1994 a few of this retired Hunter aircraft were purchased be a private organisation. Therefore, a new structural maintenance concept was required to be established. The most fatigue critical structural locations had to be defined and afterwards a fatigue investigation was necessary at each of these locations.

A first in depth investigation for the Hunter fuselage was carried out in a master thesis. It revealed that the upper two fastener locations of the forward to mid-section fuselage joint figure below was very critical and a detailed fatigue investigation was deemed to be necessary.

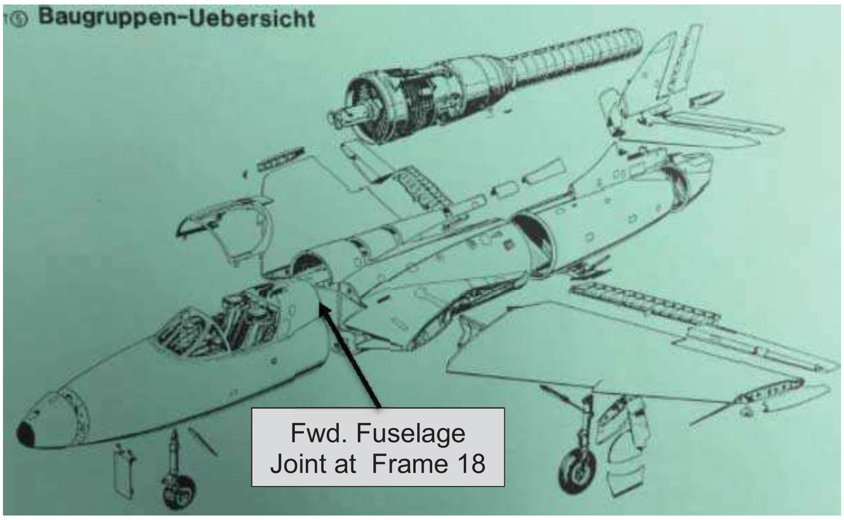


Figure 19 Wing- and Fuselage Joints of a Trainer Aircraft

In order to determine the crack growth lives at all critical locations at this joint (see Figure 20 below), several steps had to be performed.

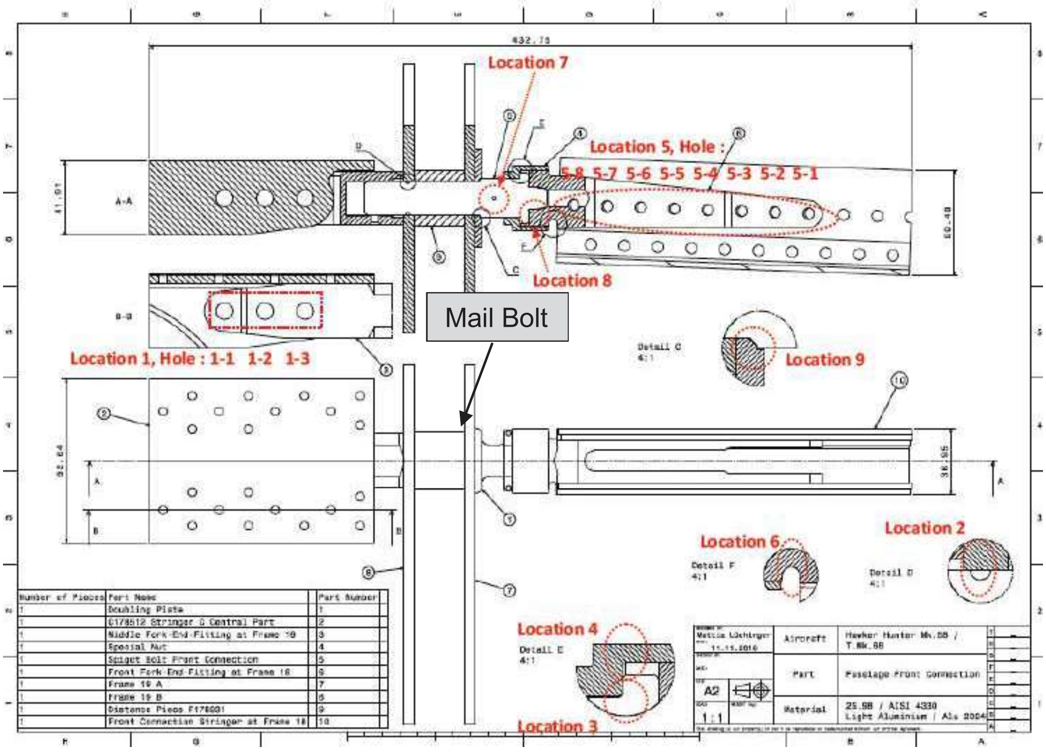


Figure 20 Critical Locations in Upper Fwd. Joint at Frame 18

At first a loading model was defined. Then the stresses at the critical locations were determined by means of a detailed finite element model with contact elements and the resulting bending moments and shear forces of this loading model. Due to the fact, that no information of the pretension force of the upper main bolt was available, three different pretension forces (10%, 30% and 50% of F_{ty}) were assumed, to investigate the impact of the pretension force on the stresses at the different critical locations. It revealed that only one location of the main bolt (i.e. location 2) is strongly affected by the pretension of the main bolt.

In a second step the stress sequences at each of the critical locations had be defined. The basis for these spectra was the standard fighter spectrum FALSTAFF with a bilinear nz vs. FALSTAFF-level relationship. For that following conditions had to be fulfilled: 1 (FALSTAFF-level) is assigned to nz(min) = -3.5g, 7.5269 is assigned to nz(min) = 0 g and 32 is assigned to nz(max) = 7.5 g.

The crack growth analyses were then performed with the AFGROW computer program. In general simplified fracture mechanical models were used, leading to rather conservative crack growth lives. The initial crack size was assumed as 1.27 mm (0.05”) and the critical crack size was determined by means of material thoughness and a 1.2 times increased reference stress.

The results of these study showed that the variation of the crack growth life at the different critical locations were very big; i.e. between 2’000 flights and 400’000 flights. This leads to the conclusion, that more in-depth crack growth analyses supported by crack initiation analyses are necessary, in order to finalize a new structural maintenance concept.

At this point it is also to mention that this aircraft was designed as a safe life aircraft and thus, crack growth analyses alone for the used material would lead to too short inspection intervals. This is the reason, why both crack initiation and crack growth lives must be considered in the new concept.

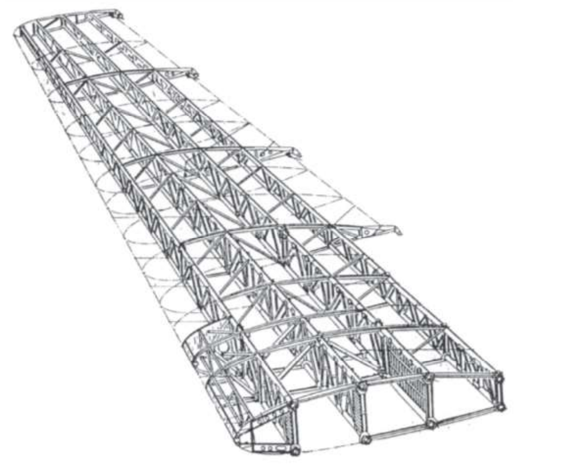
Junkers Ju-52 from JU-AIR

The Swiss Air Force operated from 1939 till 1990 three original Junkers Ju-52 with BMW engines. Since 1991 the three Ju-52 are operated by JU-AIR, a private organization in Dübendorf near Zurich. From April till October JU-AIR is pretty busy with passenger flights. Most flights are fully booked with 17 passengers. The Ju-52 makes about 250 FH per year per airplane. The maintenance reports show, that apart from accidental damages no repairs had to be performed.

Figure 21 The three Swiss Junkers Ju-52 in flight

The German Lufhansa Stiftung also operates a Ju-52, called D-AQUI. This airplane has a very interesting history regarding usage. The D-AQUI has much more total flight hours than the Swiss Ju-52; more than 25’000 FH. The Swiss fleet leader reached in 2016 a total of 10’893 flight hours.

The German D-AQUI showed first cracks in spar at 12’949 flight hours and 25’495 flight cycles in 2003. The wing structure has 4 spars with tube dimensions at upper and lower side, see Figure 22 below.



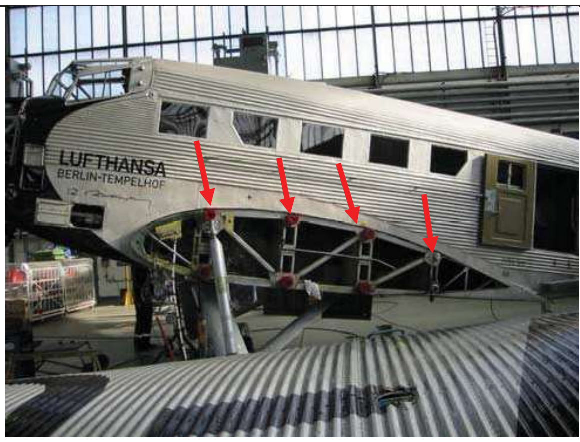


Figure 22 Wing structure of Ju-52

The wing structure consists of truss and beams with four spars. This means the wing has a fail-safe capability. The German have done several studies at the Hamburg University of Applied Sciences to understand the loads, structure, spectrum, and fatigue life of spars. Even strain gauge measurements were done, to get more information on the stress, based on manoeuver and gust loads. The conclusion was that ground-air-ground cycles have a major impact on fatigue life of the wing spars.

The Swiss FOCA started in 2015 reviewing all the available documents from the JU-AIR, including the information from the Lufthansa Stiftung about the D-AQUI. The goal was to assess the continuous airworthiness of the Swiss Ju-52 regarding ageing aircraft.

The current inspections are done on a condition-based procedure. During winter season more heavily inspections were done where also parts had to be removed to get access to the structural details. The FOCA assessment comes to the following conclusions for the ageing aircraft action on the Ju-52:

- The current inspection plan must be updated, critical crack length for each structure has to be added and inspection capability for the crack size below the critical length must be investigated and documented.
- For the repair on the wing spar of the airplane HB-HOP (date of repair 1966) no structural analysis is available, this has to be done and approved by FOCA.
- Detailed structural and fatigue analysis for the engine mounts, control surface attachments including horizontal and vertical tail, must be performed to ensure continuous airworthiness.
- Assessment of the service life for the Ju-52, original design by Junkers, was 8’000 flight hours.
- Damage tolerance analysis for critical components and determination of inspection intervals and methods. For this study also material investigations must be considered, original material for spar tube is Dural 44, which seems to be equivalent to alu 2024-T3 based on chemical decomposition.
- The spectrum for the Swiss Ju-52 must be developed, analysed and compared to the usage of the D-AQUI. Comparison with MINITWIST spectrum would be recommended.

- The horizontal tail spindle is a single load carrying structural member, this is a safe life component and therefore detail analysis is strongly needed to ensure the continuous airworthiness of Ju-52.
- A special Supplemental Structural Inspection Document SSID must be developed and approved by FOCA to ensure the continuous airworthiness of the next coming years.

In the summer of 2016 a first Bachelor thesis was done to develop a first master event spectrum for the Swiss Ju-52 usage. This study showed that gust loads are important and will influence the fatigue life, furthermore the weight and especially the fuel level in the tanks have an impact. The Swiss Ju-52 fly longer and with a different fuel sequence of the tanks. The Swiss Ju-52 have a total of 7 tanks whereas the D-AQUI only has 6 tanks. The Swiss Ju-52 mostly fly with 17 passengers, except of pilot training flights. In spring 2017 a new study was launched to measure N_z spectrum and sequence for the whole season with MEMS loggers from MSR in Switzerland.

At the inspection of HB-HOT a crack in the lower spar No. 1 on drain hole was observed in the fuselage wing area. The spar will be replaced by a new one. This offers us to instrument the spar No. 2 and No.3 on the wing section with strain gauges and to do flight measurements to better understand the loads and correlate them with Nz recording. The goal is to operate the airplane by mid of May.

An initial study with first Bachelor thesis was also done in summer 2016 for the horizontal tail spindle, see Figure 23 below. This part is made of special alloy (steel and bronze) and only static strength data is available from old data sheets of 1930 (Flieg Norm). Even durability life data was not known. The safety and hazard analysis based on current standard CS23.1309 showed that this component is very critical if structural failure will happen. Therefore, further investigation and instrumentation with strain gauges was recommended.

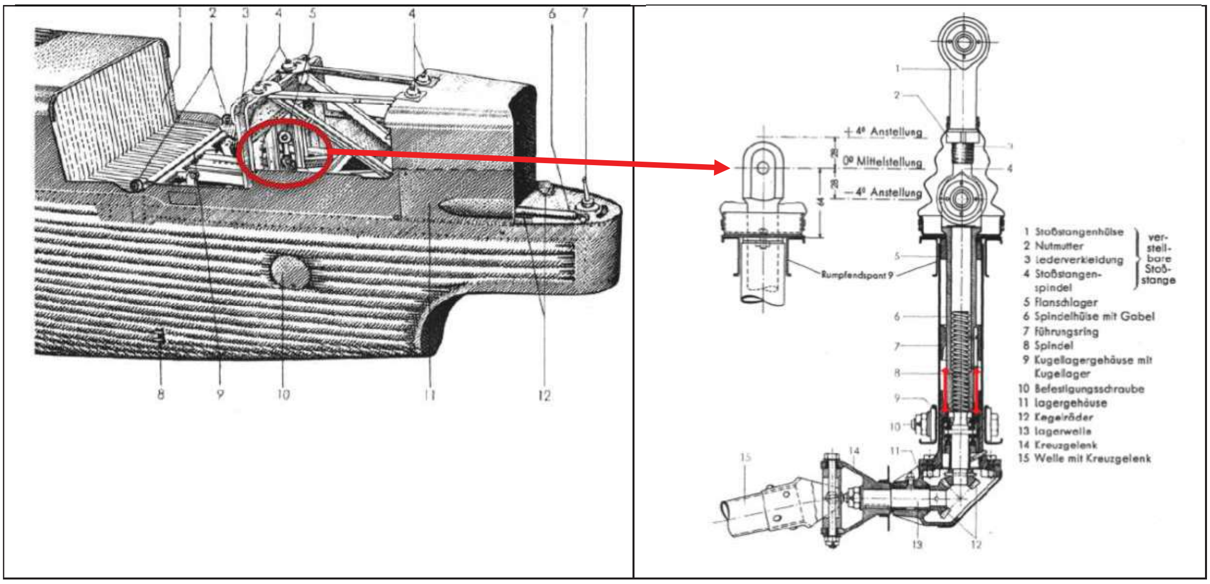


Figure 23 Horizontal tail spindle in aft fuselage and detail construction of the spindle

The investigation of Ju-52 will take at least one to two years to get the Supplemental Structural Inspection Document SSID to ensure continuous airworthiness of Ju-52 in Switzerland.

The challenge for these ageing Annex II airplanes is not only the knowledge of structural and fatigue engineering, it is also the financial budget to do all the analysis and to operate these planes on safe manner.