Review of Aeronautical Fatigue Investigations in Switzerland

April 2021 – April 2023

Swiss National Review

of the International Committee on Aeronautical Fatigue and Structural Integrity

Document: ZAV-KOR2023-001

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Summary

This document reviews the work that has been done in Switzerland in the field of aeronautical fatigue. Contributions to the document were made by Zurich University of Applied Sciences (ZHAW), Lucerne University of Applied Sciences and Arts (HSLU), and RUAG AG. This document represents a chapter of the ICAF National Reviews document that is published online on the ICAF website. The format of the review conforms to ICAF requirements.

Prepared and approved for the presentation for ICAF website www.icaf.aero

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1. Introduction

The present review gives a brief summary of the work performed in Switzerland in the field of aeronautical fatigue in the period from April 2021 to April 2023. Contributions were made by the following organisations:

- Zurich University of Applied Sciences (ZHAW); Centre for Aviation of the School of Engineering
- Lucerne University of Applied Sciences and Arts (HSLU); Department of Engineering and Architecture
- Gottier Engineering GmbH
- RUAG AG, Emmen; Business Area Air

The many interesting contributions are gratefully acknowledged, especially the effort of Xinying Liu, Viola Ferrari, Adrian Holtzhauer, Tobias Frischknecht, Cyrill Kalberer (ZHAW), Markus Gottier (Gottier Engineering), Dejan Romančuk (HSLU), and Ingrid Kongshavn (RUAG AG).

The financial support by the Swiss Federal Office of Civil Aviation (FOCA) is gratefully acknowledged for the activities of the Ageing Aircraft project. The Swiss Federal Office for Defence Procurement, armasuisse, is gratefully acknowledged for the supervision and funding of work carried out for military aircraft at RUAG AG, Emmen.

2. Swiss Aviation Activities

Kopter – a Leonardo Company

The Swiss Kopter Helicopter company became a Leonardo Company in April 2020. The original design of Kopter is modified to become a new generation of single turbine helicopter named AW09. This new helicopter will deliver the best performance in class, the largest cabin and cargo volumes, and outstanding modularity and modern electronics, see Figure 1 below. The engineering office of Kopter is in Wetzikon, the prototype assembly and flight testing take place in Mollis.

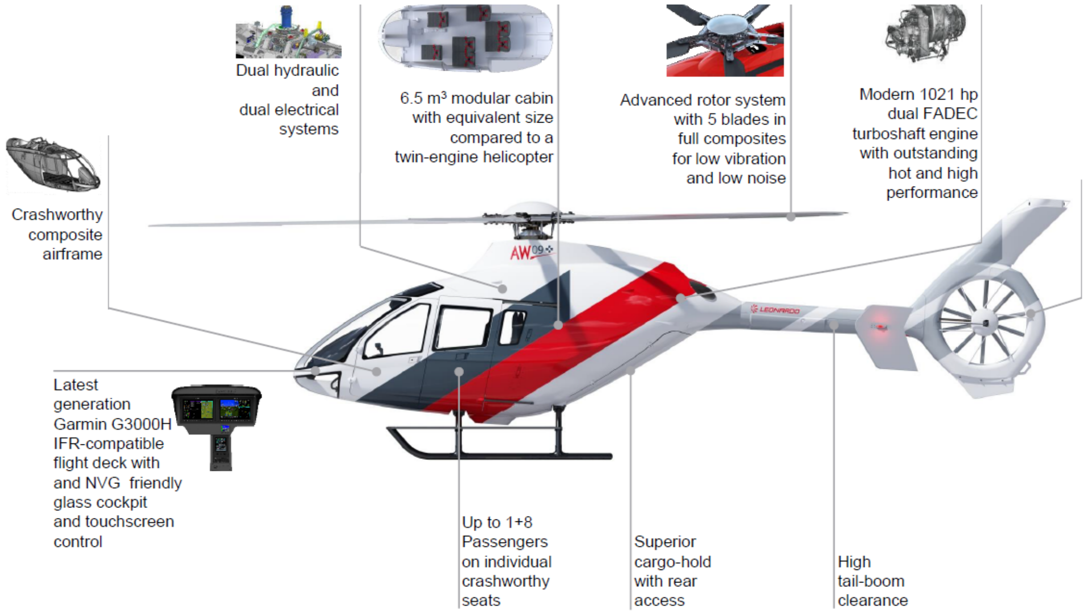

Figure 1: Kopter AW09

Kopter is working on an innovative hybrid electrical solution based on the AW09 for developing a very sustainable helicopter. The ultimate goal is to offer lower emissions, simplified emergency procedures with efficient operation and high safety standard. The new technologies are presented in Figure 2 below. These new technologies will provide new operational scenarios for operation in congested hostile areas, simplified training procedures, and improved public acceptance.

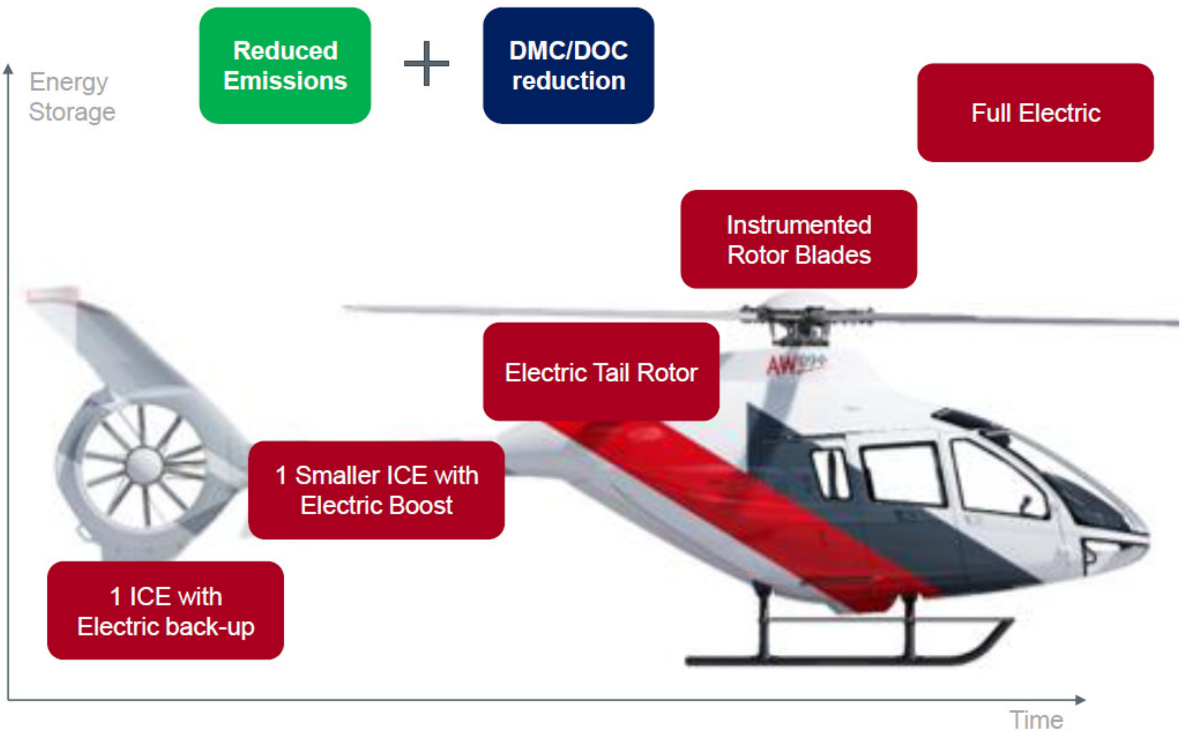

Figure 2: New technologies for the hybrid electrical solution

European Rotorcraft Forum at ZHAW

In September 2022, the European Rotorcraft was held in Switzerland at the Zurich University of Applied Sciences (ZHAW) for the first time. More than 200 attendees enjoyed a very interesting scientific program with the highlight of the landing of the Airbus EC635 in front of the main building of the University (see Figure 3).

Figure 3: Attendees of the European Rotorcraft Forum 2022

Pilatus Aircraft Ltd

The Covid crisis also affected the Swiss Aviation Industry. But Pilatus was very successful with their products PC-12 NGX and PC-24. In May 2023 already the 2'000th Pilatus PC-12 (see Figure 4) was delivered. For the PC-24 further improvements were done, as e.g. the spoilers (secondary structure) will now be manufactured from composite instead of metal. The big challenge is to find well qualified engineers for the future to develop new sustainable aviation products and services.

Figure 4: Pilatus PC-12 NGX

3. Ageing Airplanes (ZHAW)

Gottier Engineering GmbH, Markus Gottier; ZHAW, Michel Guillaume

Figure 5: Ageing airplanes Hawker Hunter (left), De Havilland Venom (centre), and De Havilland Vampire (right)

Hawker Hunter Mk.58 / T.Mk.68

The crack in the leading edge spigot of the Hawker Hunter presented in the ICAF review from 2021 is still under analysis (see Figure 6). The analysis initially focused on the loading of the component. For this purpose, a global finite element model (GFEM) of the wing and the centre part of the fuselage was created in Simcenter Femap (see Figure 7). The structure was largely simplified so that primarily shell and beam elements could be used. With this model, the interface loads could be determined using the linear solver of NASTRAN. Various wind tunnel data from the time of active military service were processed to be used as the load cases. This allowed a load case to be created at $n_z = 5.5$ which corresponds to the maximum load factor in civilian use.

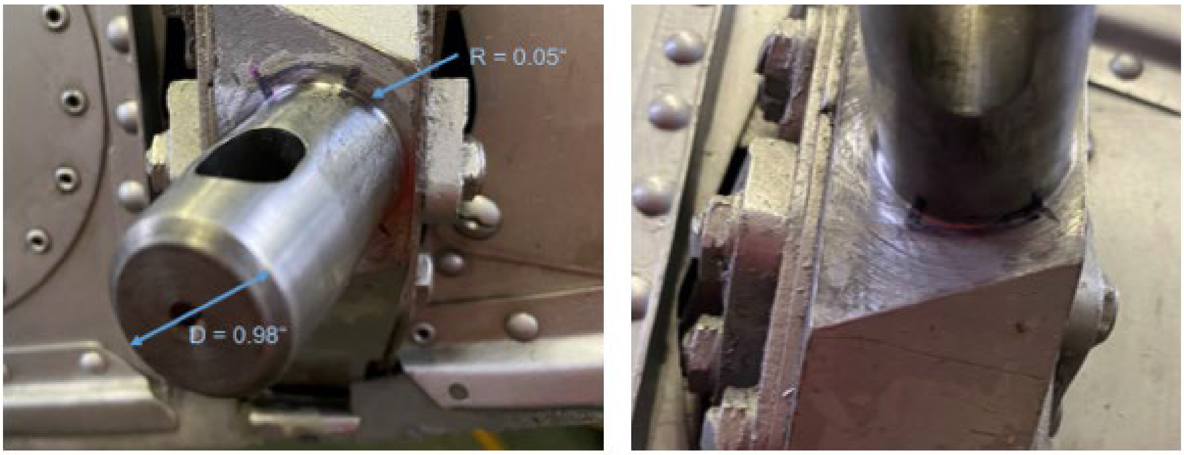

Figure 6: Leading edge spigot with the delineated cracks on the upper and lower side.

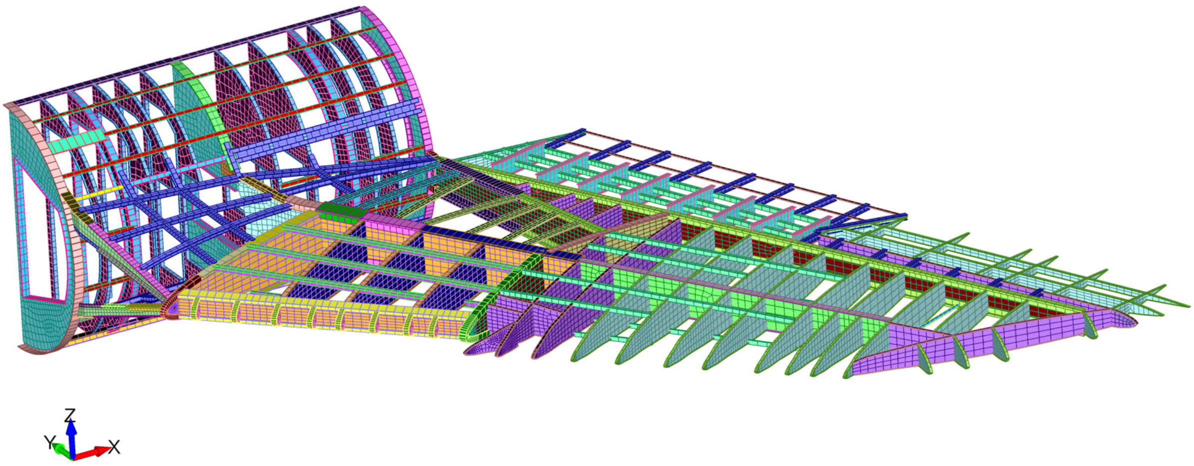

Figure 7: GFEM of the Hawker Hunter

The interface loads confirmed the previous conservative approach, which was based on the material data.

Subsequently, in order to further increase the degree of accuracy, the spigot and its socket were modelled in detail and integrated into the GFEM. The load transfer is established via a linear contact. The results revealed that the load is transferred further away from the spigot root than initially assumed. Thus, the maximum principal stress in the radius near the root is higher. The results of the 5.5g load case are shown in Figure 8.

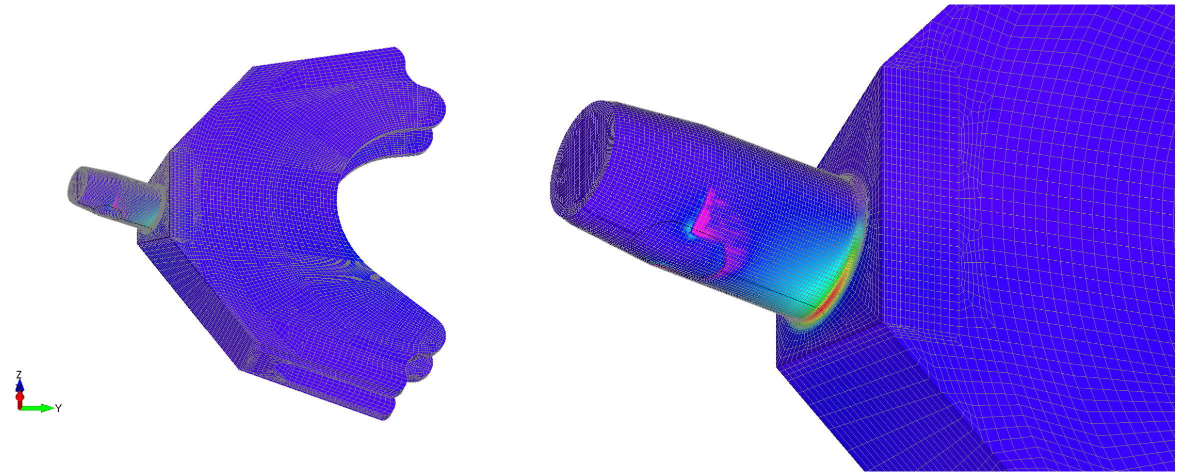

Figure 8: Max Prin Stress of the leading edge spigot (left) with the detailed critical location (right)

The resulting stress distribution is further used for a crack growth (CG) calculation using AFGROW in the form of the normalised stress correction factor. The spectrum is left unchanged: The FALSTAFF spectrum is scaled to the reduced civilian envelope, with the number of cycles doubled and the second part inverted to have a symmetrical spectrum (which is assumed to be present due to the fact that cracks both on the upper and lower side of the spigot were found). However, the evaluation with AF-GROW shows that the second inverted part has only a very small share in the damage contribution. It can also be assumed that the use of FALSTAFF leads to conservative results as the aircraft is now mostly used for passenger flights with comparatively few manoeuvres and few display flights only. The initial and critical crack sizes are currently under investigation. The total CG life of the component calculated with AFGROW is shown in Figure 9. It will be used subsequently to estimate the remaining fatigue life of the component. It can also be seen that the reduction of the maximum load factor from 7.5 to 5.5 markedly extends the fatigue life by a factor of approximately 3.

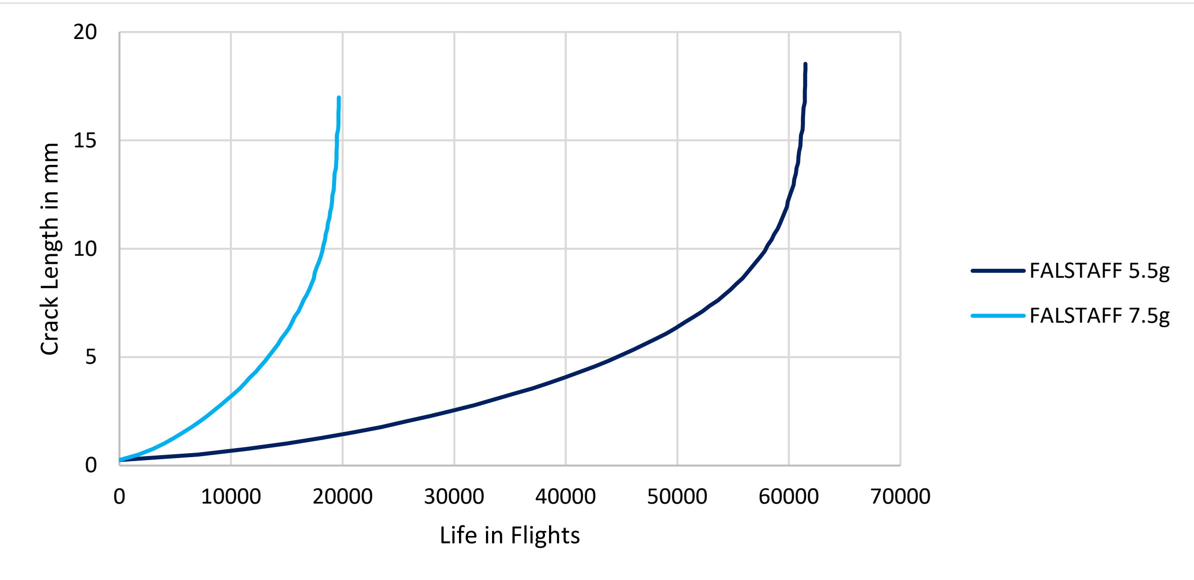

Figure 9: CG life of the leading edge spigot for both the civilian and military use

De Havilland Venom DH.112

It is planned to restore at least one Venom DH.112 to airworthy condition and operate it under civil registration, similar to the other Ageing Airplanes in Switzerland. The aircraft has achieved already close to three times the service life foreseen by De Havilland during its military service time. The life extension could be assured among other by means of a full-scale fatigue test (FSFT, see Figure 10). The recent work described hereinafter aims to determine the necessary steps from a structural point of view to assure the structural integrity in civilian use.

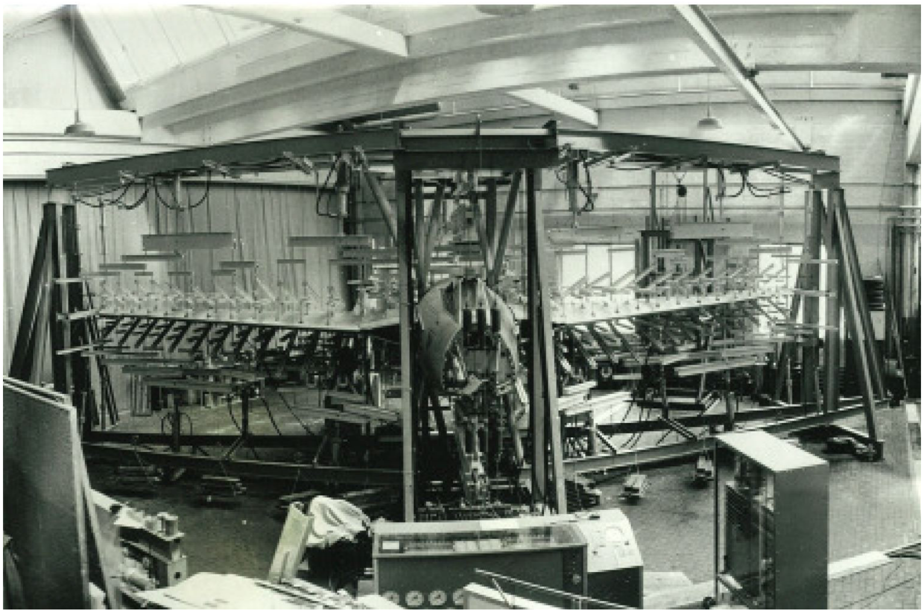

Figure 10: Setup of the Venom FSFT in F+W Emmen

Based on the reports when the aircraft was still in active military service, four critical locations could be identified that may be decisive for future life.

The main spar has a kink at the position of rib 2. In addition, the spar cap is interrupted at this point to facilitate manufacture by folding. This disrupts the load path of the bending moment and leads to a stress concentration in the web. Cracks have already been found at this point in the FSFT (see Figure 11). It is decided that this location and also possible modifications should be investigated further.

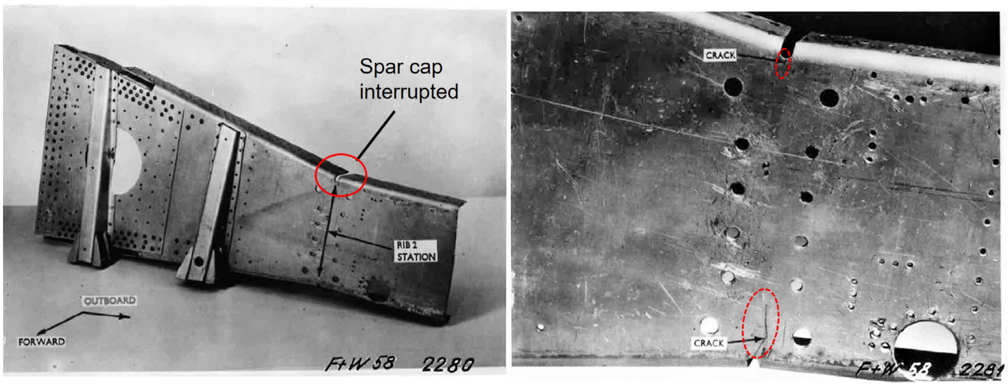

Figure 11: Interrupted spar cap in the main spar

Cracks were found regularly in the wheel wall and partially also in its planking during military operation. These parts are easily accessible for inspection and some reinforcements to the planking have been made. However, the importance of the planking as a load path needs to be analysed.
In the region of the wing fuselage connection joints, various critical locations were identified as well. Some of them have been mitigated with modifications. The bottom wing root fitting is under tensile load when exposed to the lift and cracked at the boreholes (see Figure 12). It must be assumed that a failure of that part would lead to a fatal accident, and it thus needs further investigation.

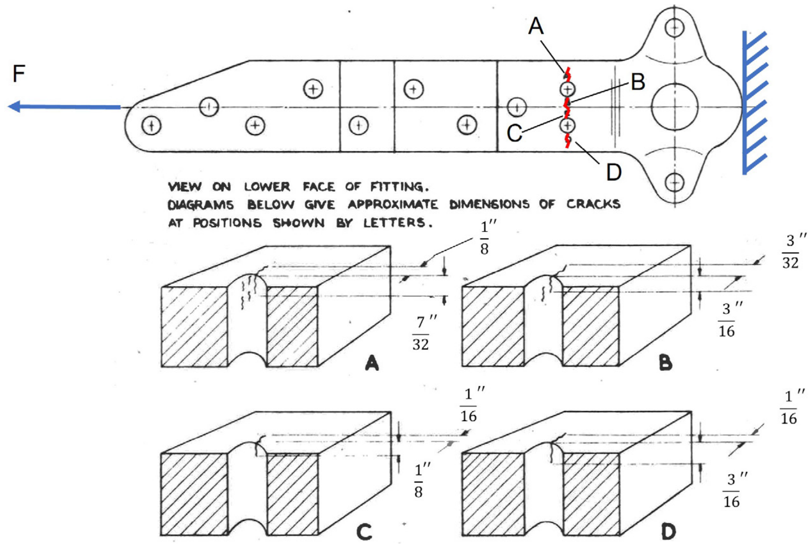

Figure 12: Possible cracks indicated in the bottom wing root fitting

The FSFT also revealed cracks in the planking of the right wing at rib 7, even after a reinforcement was installed. The cracks were only found using an x-ray inspection under load. This location itself is not considered as fracture critical. Nevertheless, the consequences of a failure or a combination of failures needs to be assessed.
The first step for the estimation of the future fatigue life is driven by the intended operation. The aircraft will be mostly used for air shows and the associated ferry flights. Only a limited number of flight hours per year is foreseen. Furthermore, the maximum allowed load factor is reduced to $n_z = 5.5$. A flight envelope was thus created which was used to determine the most critical load cases. The air loads were subsequently calculated for a load case with $n_z = 6.67$ at $Ma = 0.6$, since validation data from the FSFT and from wind tunnel tests was available. From the different tools evaluated, Flow5 showed the best agreement as it can be seen in Figure 13. A reasonable correlation between the forces determined with Flow5, the cylinder forces from the FSFT and the wind tunnel data can be observed. Using these loads, the bending and torsion moments of the wing can be calculated. The future work consists of developing an appropriate model to determine the loads at the four aforementioned critical locations. The spectrum will be a FALSTAFF spectrum scaled to the actual envelope, as only limited data of the actual spectrum is available. Only then, the criticality can be assessed using CG calculations.

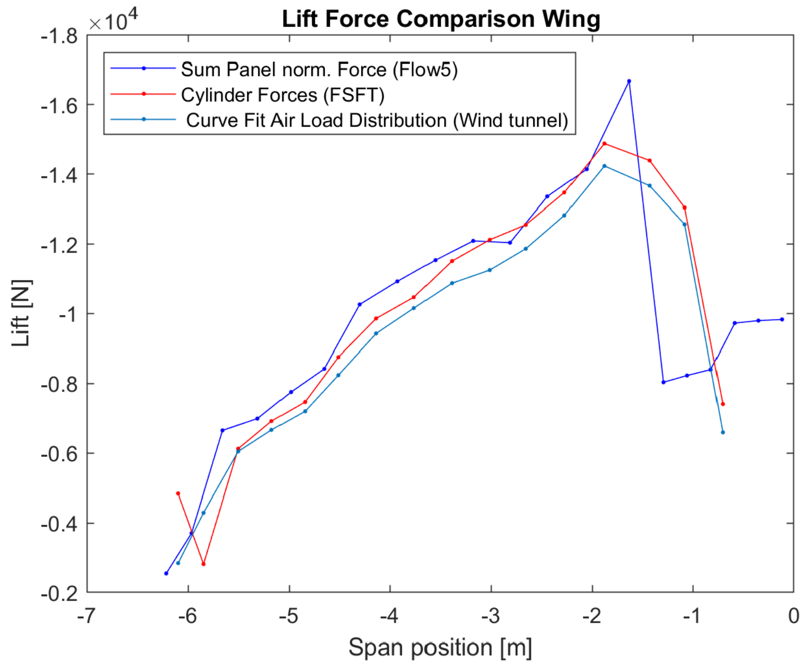

Figure 13: Comparison of lift forces from panel code, FSFT, and wind tunnel data

De Havilland Vampire DH.110 / DH.115

The investigations regarding the Vampire have only recently started. The aim of these investigations is similar to that of the other two aircraft already described, which is to ensure the structural integrity of the aircraft and that it can be safely operated beyond the safe life provided by the manufacturer. To this end, the remaining documents were reviewed first and a list of critical parts was compiled. A total of 20 components were identified that must be regularly inspected for fatigue. Of these, four are classified as fracture critical traceable (see Figure 14) and seven as maintenance critical. Cracks were found in the fracture critical traceable components during the FSFT. They are located at the interface between wing and fuselage as well as between fuselage and engine. The inspection interval is 600 FH, for the remaining components it is mostly 50 or 100 FH. These intervals were defined in the time when the aircraft was still in service in the Swiss Air Force.

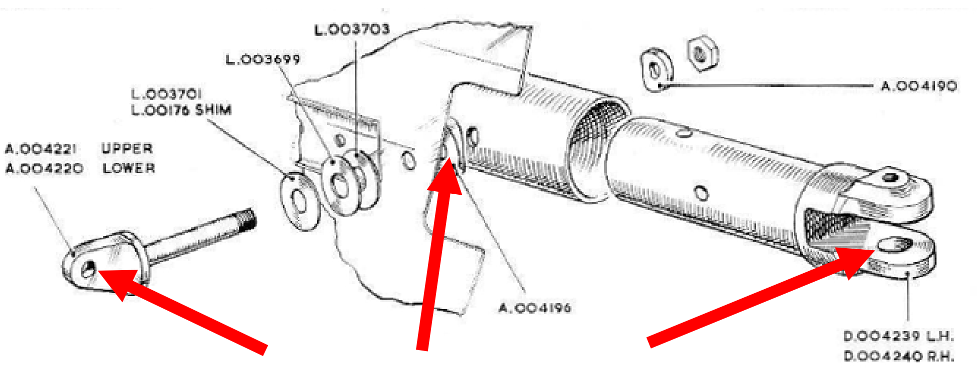

Figure 14: Three fracture critical traceable locations on the engine (left) and wing (right) attachment fitting

The loads were calculated as external loads using simple panel methods (XFLR5 and OpenVSP). The basis for this was the flight envelope of the Vampire adapted to CS23 and to the intended civil usage. As the civilian use will consist of both scenic flights with few manoeuvres and flights with aerobatics, the resulting spectrum for the analysis will be adapted accordingly. It will essentially consist of two FALSTAFF sequences, one of which is adapted to the scenic flight envelope in the load factor range of 1-2g and the second to the aerobatic envelope with load factors between 1-4g.
A beam model is initially used to represent the wing of the aircraft. The static loads can be validated with data available from De Havilland.

4. Structural Risk Assessment with SMART|DT (RUAG AG / ZHAW)

RUAG AG, Viola Ferrari, Michea Ferrari; NRC (Canada), Min Liao; ZHAW, Michel Guillaume

Structural risk assessments are fundamental to support the aircraft structural integrity (ASI) where a pure analytical “lifing” approach cannot show full compliance with design requirements and mitigation actions as e.g. a safety by inspection needs to be defined. With increased computational power, probabilistic risk assessments (PRA) can be used instead of deterministic models. The main benefit of a PRA is that it takes uncertainty into account and leads to less conservative estimates. However, it still meets safety requirements. This practical application should assess the use of SMART|DT for RUAG AG. SMART|DT is a tool for probabilistic structural risk assessments based on damage tolerance for the general aviation industry. The goal is to apply it to military systems as well. A case study on the F/A-18 is conducted and the results are compared with those of the tools from NRC and USAF.
The master curve methodology is used to predict deterministic crack growth from a single deterministic fracture mechanics model. The master curve is then shifted and scaled by using different values for the fracture toughness and crack size. Further input data is needed for the material properties (incl. scatter), the initial crack size distribution through the concept of the equivalent pre-crack size (EPS), and the applied stress. A sensitivity study showed that the accuracy of the spectrum and the EPS are most relevant for the PRA.
A case study on the inner wing trailing edge flap knife edge (see Figure 15) is used to compare the results with other tools. Cracks were found during inspections in the Swiss fleet and they origin from the severity of the Swiss usage spectrum. ProDTA (from NRC) and PROF3.2 (from USAF) with different equations to calculate the probability of failure (POF) are used as a reference. Figure 16 shows the results of the different tools. The results from SMART|DT are comparable to the reference using the Lincoln equation and conservative compared to the tools using the Freudenthal equation. The results using the Freudenthal equation delivers better results when to start the inspection as a comparison with the in-service findings showed.

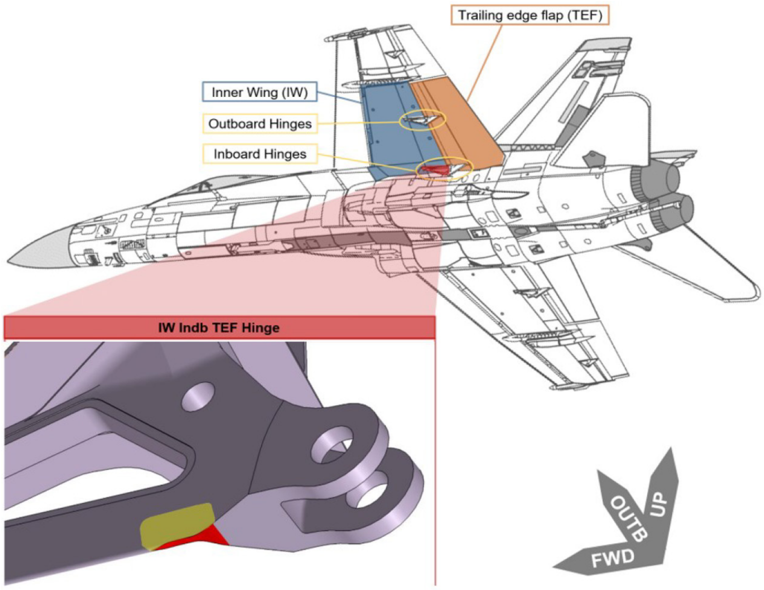

Figure 15: Overview of the knife edge location on the F/A-18

SMART|DT shows to be a conservative tool. However, extensive data sets are needed for the EPS and also for the probability of detection (POD) of different cracks which in turn also depends on numerous parameters such as the material, the method, or environmental effects. The tool could therefore only assist in the decision-making process for fleet safety measures at the moment. Nevertheless, data acquisition strategies, possibly as an international collaboration, could arise from this project.

Further details on the PRA using SMART|DT are given in the paper *Ferrari, V., Liao, M., Ferrari, M., Guillaume, M. (2023) Practical Application of Structural Risk Assessment with SMART|DT, Proc. of the 31st ICAF Symposium.*

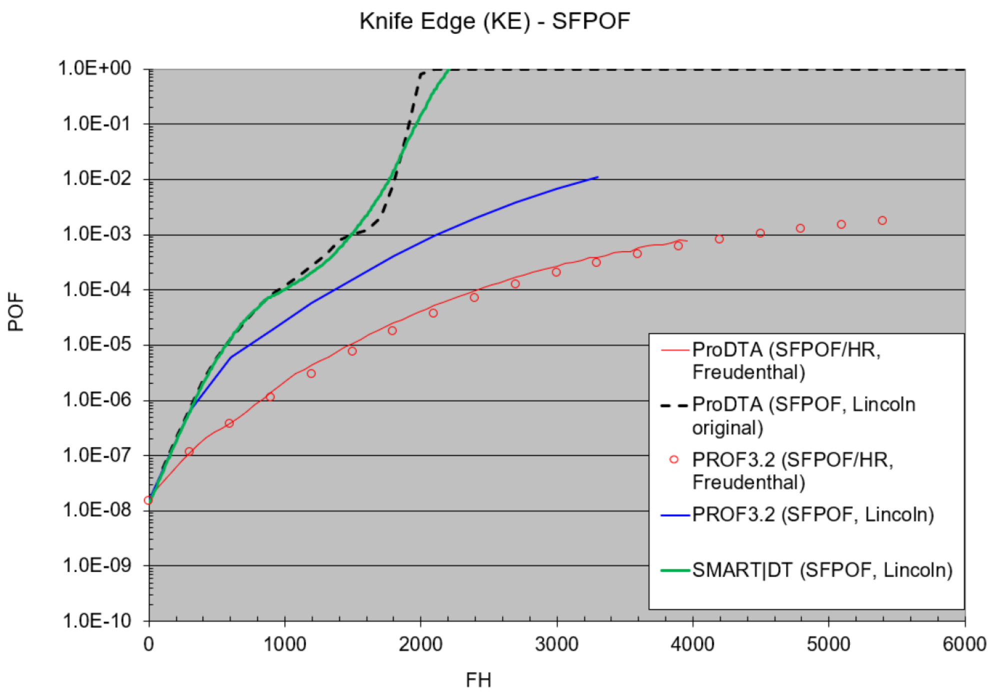

Figure 16: Single flight probability of failure (SFPOF) results from different tools

5. Activities Hochschule Luzern (HSLU)

Lucerne University of Applied Sciences and Arts, Dejan Romančuk

Activities in aeronautical structural integrity at the Lucerne University of Applied Sciences and Arts (Hochschule Luzern - HSLU) are carried out at the Institute of Mechanical Engineering and Energy Technology (IME) within the Competence Center of Mechanical Systems (CCMS). Progress has been made in Structural Health Monitoring (SHM) and in the field of metallic Additive Manufacturing (AM).
Investigations into the feasibility of a metallic AM structural integrity concept for load-carrying, safety-relevant components have been performed. First steps have been initiated, in collaboration with Pilatus Aircraft and supporting activities (AM printing, CT scanning and post-processing), for a research project of the Swiss Federal Laboratories for Materials Science and Technology (EMPA) to assess the role of a lack-of-fusion defect in the fatigue performance of AM titanium parts.
For the SHM activities, an outline of a project, which is being carried out in collaboration with RUAG, is provided in the following:

Structural Health Monitoring – POD Study

HSLU (A. Gut, D. Romančuk) in collaboration with RUAG (H. Eijsermans, M. Lüchinger, M. Ferrari) performed investigations into Structrual Health Monitoring (SHM) sensor systems. The goal was to assess damage sensing systems for the application in difficult to access, fracture critical and durability critical locations on the F/A-18 platform. As a goal, the fleet availability shall be increased and the down-time reduced. This shall lead to an overall maintenance cost reduction with the requirement of an equivalent or improved reliability compared to the existing NDI requirements. The key elements of the successful implementation of a sensor framework in a SHM system for an aircraft application is the determination of a probability of detection (POD) curve and its validation against the location specific POD and durability requirements.
A so called comparative vacuum monitoring (CVM) sensing system provided by SMSystems is used for selected aluminum 7050-T7451 structural locations. The CVM principle relies on placing a sensor onto the surface where damage is expected to occur. The sensor contains fine channels in parallel which are open to the surface. Once the sensor has been installed on the surface, the channels form closed "galleries" to which a vacuum can be applied. The sensor is connected to a vacuum source with an accurate flow- and pressure meter (e.g. the PM200 monitoring device). If a crack develops, a leakage path will form and the vacuum level will be reduced in the sensor manifold.
In a lab test program, the CVM system has been evaluated for two "hot-spot" locations. The lab testing included familiarisation testing to determine the detection threshold and POD screening tests for preliminary POD curve derivation. The specimens used in the lab were designed to closely represent the

locations design (thickness, stress concentration and surface treatment) and loading condition with a corresponding stress distribution and crack growth scenario. The specimen design process using FEM analysis is illustrated in Figure 17. A very similar process is used for a second potential SHM location, which is shown in Figure 18. The specific specimen design allows cracks to naturally grow at the desired location.

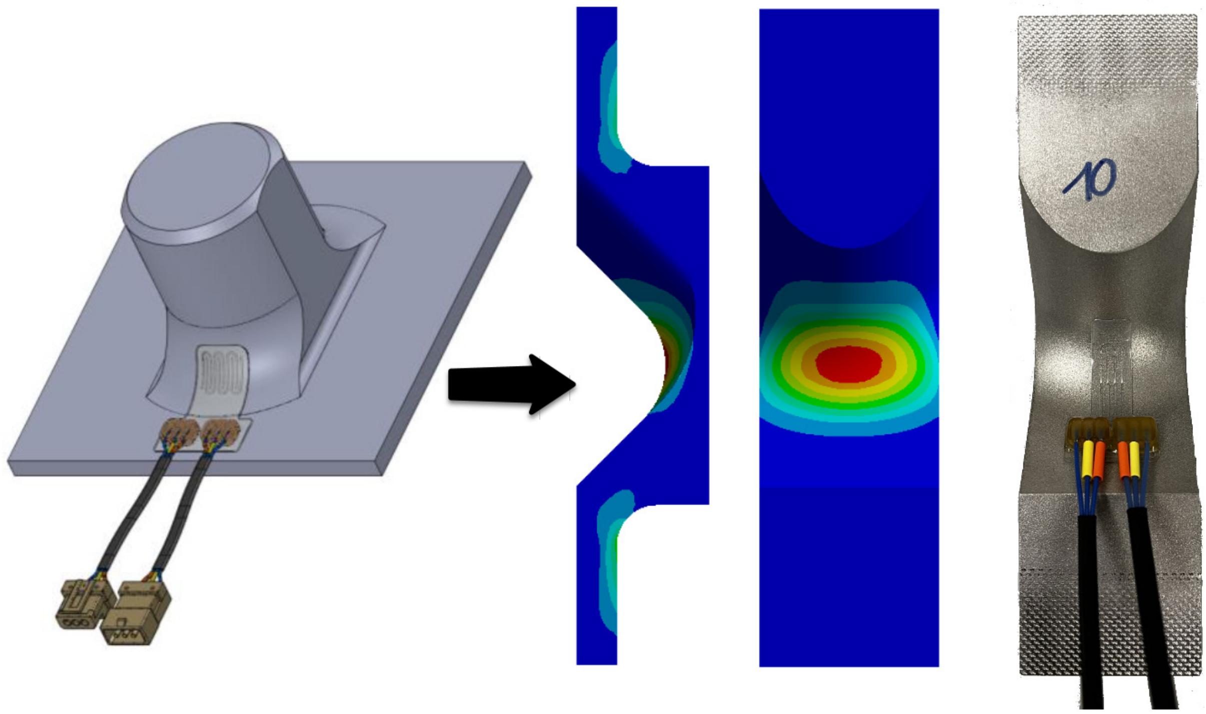

Figure 17: First potential SHM location on the aircraft (left) is translated into an axially loaded lab specimen (bare, shot peened surface)

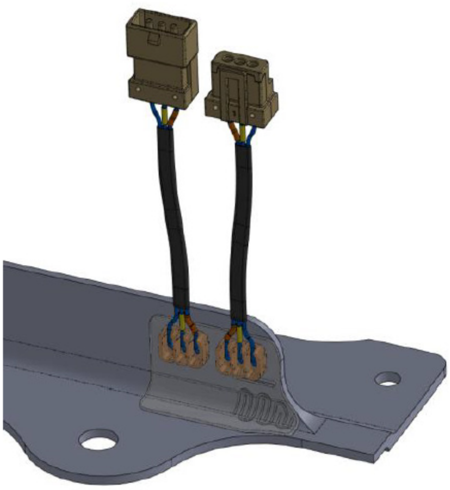

Figure 18: Second potential SHM location (primered surface)

The POD testing procedure requires fatigue cycling and continuous crack monitoring. For the continuous crack monitoring, a separate lab device was required which allowed for an early crack detection on specimens during fatigue cycling. For the inspections with the PM200 device, the specimen was periodically removed from the test bench and inspections were performed until a crack detection signal above the inspection threshold was provided. The sensor was then removed and the crack length at detection was determined under a microscope.

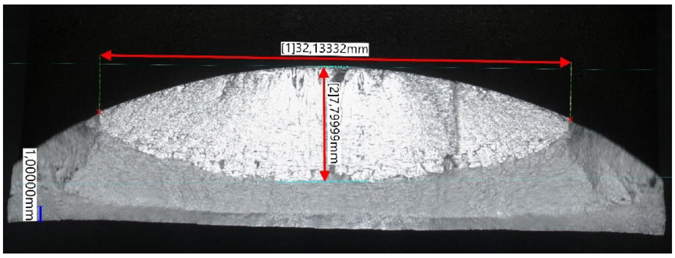

Figure 19: Fracture surface of a sample

POD curve derivation and the calculation of the $a_{90/95}$ value was performed with 15 specimens and the one-sided tolerance interval (OSTI) method. The concept of the OSTI POD is, that there exists a normal distribution describing the crack lengths at which detection by the SHM system is first made. The POD for a given crack length is then the proportion of cracks that have a length less than the given length, i.e. the POD curve is the cumulative probability density function (CDF) of the normal distribution describing the distribution of crack lengths at first detection.

The 95% confidence interval and the value of $a_{90/95}$ specifically is then the one-sided tolerance interval on the POD curve as shown in Figure 20 (a). A statistical significance test that tests the hypothesis that the underlying population of a sample is normally distributed is required to pass to justify the validity of the OSTI POD method for this application. The Log-Normal Q-Q plot, indicating the normal distribution of the crack lengths at first detection, is shown in Figure 20 (b).

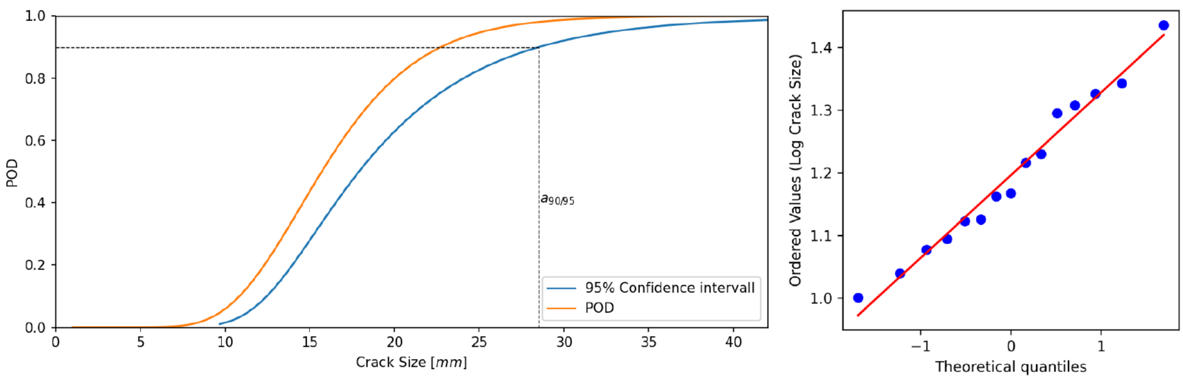

Figure 20: (a) POD curve with confidence interval and (b) Log-Normal Q-Q Plot for Crack length at detection of the first location

Conclusions from POD testing performed at HSLU:

- The SMSystems CVM technology provides consistent and reliable results and is an easy to use and mature system for aircraft applications. Depending on the specific POD requirement of a location, the CVM technology may be a sensible SHM solution. Great importance must be given to the review and selection of optimally suited locations for the SHM CVM system.

- The $a_{90/95}$ values for the selected locations were higher than initially expected, based on supplier data for sheet material. This is likely due to the shot peened surface and the thicker, machined part (compared to sheet). More testing with altered parameters is required.
- The CVM technologies POD is strongly dependent on the specimen geometry (thickness, radii) and surface condition (bare, primed, shot peened). POD results cannot be easily transferred between different locations. Most locations selected for monitoring will require separate POD testing.
- The number of specimens for POD testing should be higher. A number of around 20 specimens is proposed instead of 15 when using the OSTI method.
- A reduction of the detection threshold is possible to increase the systems sensitivity and to slightly lower the $a_{90/95}$ value. However, an increased probability for false positives is to be expected.

6. The Swiss F/A-18 Structural Fatigue Management Program (RUAG AG)

RUAG AG, Etienne Girard

It is well known that the Swiss F/A-18 fleet usage is severe and to a certain extent this was already foreseen and considered during the development of the aircraft. With the continual aging of the fleet and additional service life extension goal, ensuring the structural integrity of the Swiss F/A-18 while maintaining the same level of fleet readiness is a major challenge for the program.
In fact, the fatigue damages discovered in the Swiss fleet tend to indicate that, for some area of the aircraft, the fatigue life consumption is already very advanced and that the initial Design Certification Basis of the aircraft structure is already exceeded or will be exceeded before the Out of Service Date (OSD). This reality raised some concern within the Swiss F/A-18 community of the expected challenges that would impact fleet availability and would need to be addressed by the maintenance and engineering community.
While some of the fatigue damages can be managed by the means of recurring inspections and repairs, refurbishment programs or part replacements, some other locations are not suitable for such a structural integrity approach due to various reasons (poor accessibility, high maintenance efforts and/or costs). If no actions are taken, the later locations will form a major burden for the program and may drastically impact aircraft readiness and availability.

A robust and strict fleet fatigue management can help achieve the service life goal while lowering the risk of unplanned and prolonged downtime due to repairs, inspections, modifications or replacements. Several other F/A-18 operators have already successfully put in place such a program and have shown that it is a very effective way to control the fatigue accumulation of the airframe. An active aircraft fatigue management contributes to improve the continued safety of flight and operational readiness requirements.

In short, aircraft fatigue management is the process of monitoring and controlling fleet usage and fatigue damage accumulation for the purpose of maximizing its economic life without compromising its operational effectiveness and integrity.

Through awareness and several discussions, the need for a robust fatigue management plan became evident. In mid-2019, a collaborative effort began between the various F/A-18 stakeholders. RUAG led this effort with the goal to define and roll out a Structural Fatigue Life Management Plan (SFLMP) for Swiss F/A-18 fighter jet. The SFLMP was officially rolled out at the start of 2022. This was a joint effort between the Swiss F/A-18 Operational, Engineering and Maintenance communities which each their own role and objectives.

Organisation	Operation	Maintenance	Engineering
Objective/Role	Successfully complete the assigned missions! →Readiness →Availability →Affordability	Perform the planned maintenance to ensure fleet availability, deal with un-planned events! → Predictability	Perform predictions and define actions to ensure the safety and ensure long-term fleet availability
Metrics	▪ Nbr. of aircrafts ▪ FH Budget	▪ TAT, MTBF ▪ Man Hours	▪ SLL, Fatigue accumulation (FLE)

Figure 21

Since its inception, several effective actions have been defined and implemented successfully. This was possible by defining measures in all three important areas of fatigue management: (1) fatigue awareness, (2) usage characterization and (3) guidance and control. Fatigue awareness was achieved through briefings, training and various fleet management documents. Usage characterization was achieved through the gathering and processing of aircraft usage data in a timely and accurate manner. And guidance and control was achieved by modifying the mission profile, prioritizing aircraft usage and imposing flight restrictions on severe damaging missions.

Since the very first measures were rolled out (mid-2019), the impact of the program has been shown to be very effective at reducing the fatigue consumption on the overall fleet. A 35% drop in fatigue consumption rate has been recorded. The overall cumulative lifetime fatigue rate has dropped by 4.6% as well. And finally, the projected fatigue usage at the Design Service Goal (DSG) based on the last 3 years rate shows a potential overall reduction of 12.6%.

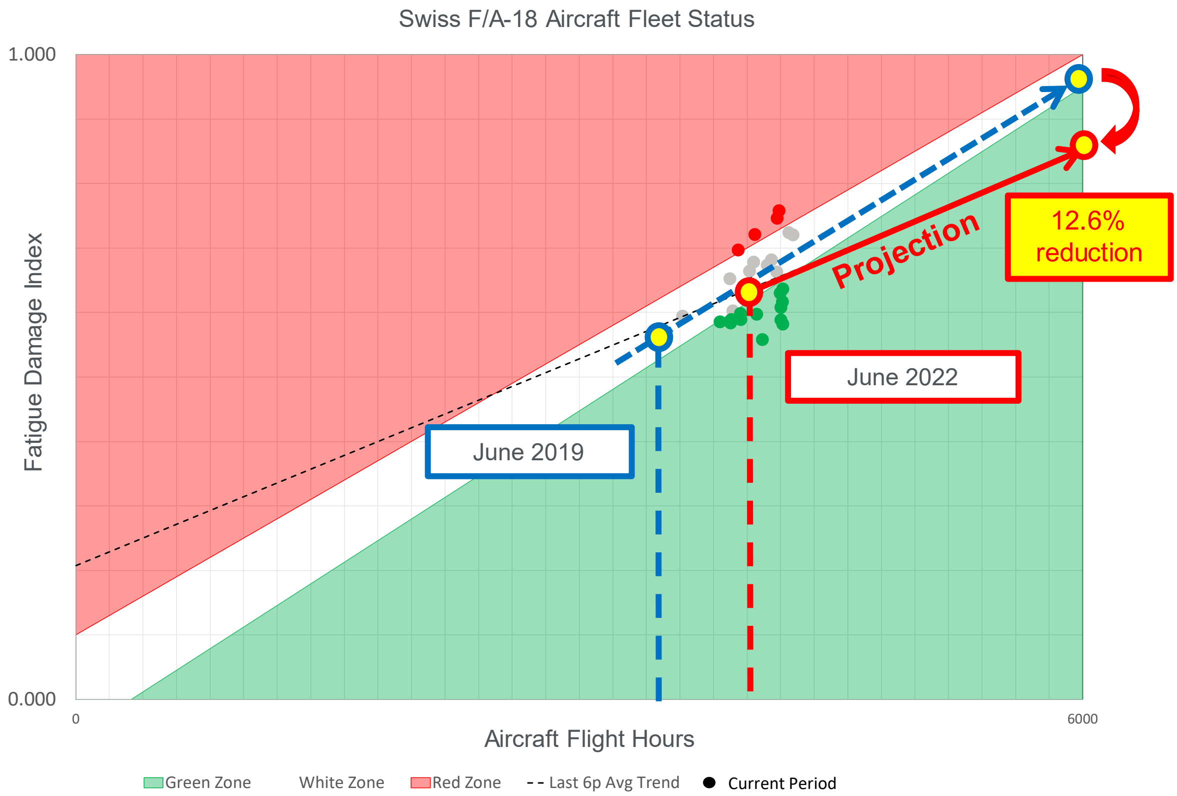

Figure 22

In conclusion the SFLMP is a living program. The SFLMP community meets twice a year to evaluate the effectiveness of each measure in place and discusses about adjustments needed to continually optimize the fatigue usage of the fleet. The program has demonstrated its ability to monitor and control the fleet aircraft's future usage.

To a large degree, the success of the Swiss F/A-18 fleet fatigue management program is a result of the close collaboration of the Operational, Engineering and Maintenance communities and their desire and ability to implement and monitor effectively fatigue damage control measures.

7. F/A-18 C/D Structural Refurbishment Program (SRP) (RUAG AG)

RUAG AG, Hakon Eijsermans

Despite the fact that the Swiss F/A-18 structure received design changes by the OEM, at the end of the full-scale fatigue test several structural deficiencies were identified. The primary structural deficiencies have been treated within the first F/A-18 Structural Refurbishment Program (SRP).

Furthermore, since the development of the SRP1, additional structural deficiencies have been discovered by other F/A-18 users, which are also relevant for the Swiss F/A-18 fleet. In order to assure the A/C structural integrity, measures on the A/C fleet have to be taken. Preventive modifications will reduce the need of unplanned repairs or new periodic inspections. For some critical locations, periodic inspections and repetitive repairs are not an alternative. This is due to accessibility constraints, which require a significant disassembling work to assure the required access. In order to optimize the A/C fleet availability and to maintain the fleet as economically as possible, the required preventive modifications and the one time deep inspections shall be combined in the Structural Refurbishment Program 2 (SRP2).

Due to reasons of criticality and accessibility, all SRP2 locations are categorized into different packages. To form the different packages, all locations are grouped in A/C zones such as forward fuselage, center fuselage, aft fuselage, inner wing, outer wing and landing gear to share synergies during preparation/post and testing work. These synergies include accessibility, component removal, riggings and/or functional checks.

The total SRP2 package is split in two, with the highest priority locations to be implemented on the fleet first and the remaining locations secondly. The high priority package is often referred to as package red and will be completed in 2023. In the meantime, the implementation of the second package ("package yellow") has started and will be completed several years from now.

To support the implementation phase, the engineering department is constantly in contact with the shop floor to assess the bottlenecks together. This has led not only to many improvements to the designs of the modifications, but also to many standard repair solutions that can be executed on the shop floor directly without involvement of the design department. These efforts have led to a reduction of the aircraft turnaround time.

For each aircraft, the SRP2 is a single event. Therefore, it is important to know if further actions are required after SRP2 till the end of life. The modifications are specifically developed to last, but SRP2 consists of many inspections as well. For these locations, fatigue and damage tolerance analyses are performed to evaluate if further inspections are needed, and if so, how often.

8. F/A-18 Centre Barrel Test – FTS3 (RUAG AG)

RUAG AG, Ricardo Filipe do Rosario, Marina Fernandez Nuñez

The Titanium Centre Barrel Test is a collaboration effort between the Australian Government, represented by Australian Department of Defence and the Swiss Federal Council, represented by the Federal Department of Defense, Civil Protection and Sports. The Steering Committee for this joint project is composed of armasuisse (ar, Switzerland) and the Defence Science and Technology Group (DSTG, Australia). The project is supported by RUAG AG (Switzerland) and the Royal Melbourne Institute of Technology (RMIT, Australia).

The objective of the collaboration is to obtain fatigue life data necessary to optimise and extend the life of the Swiss F/A-18C/D fleet. The Swiss center barrel if made of a recrystallization annealed Ti6Al4V. This alloy was specifically chosen to optimize the fatigue properties of the structurally significant wing carry through bulkheads, in expectation of a severe Swiss usage spectrum. The goal is to test the bulkheads with a 20% increase in Swiss service life and to obtain detailed evidence of sufficient fatigue resistance.

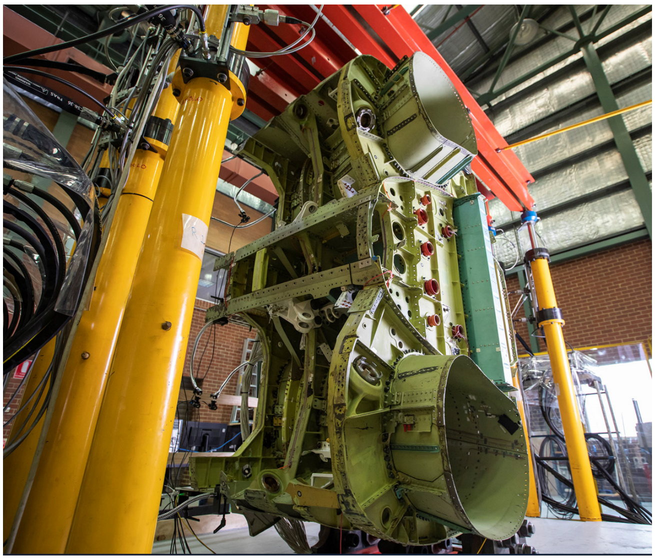

Figure 23: FTS3 – F/A-18 Centre Barrel Test Rig

9. F/A-18 Trailing Edge Flap Component Test – FTS4 (RUAG AG)

RUAG AG, Ricardo Filipe do Rosario, Marina Fernandez Nuñez

The Trailing Edge Flap Component Test is a durability and residual strength test, with the primary objective to validate the repair of the inboard hinge under the Swiss baseline fatigue spectrum. A secondary objective of the test is to collect additional data on fatigue damage in the remaining structure. The project was requested by the government of Switzerland, represented by armasuisse (Swiss Federal Council and the Federal Department of Defense, Civil Protection and Sports). RUAG AG (Switzerland) leads and technically supports the test which is performed by the National Research Council (NRC, Canada).

Figure 24: FTS4 – F/A-18 Trailing Edge Flap Component Test Rig

10. Coupon Truncation Studies (RUAG AG)

RUAG AG, Ingrid Kongshavn, Larissa Sorensen, Beat Schmid

To reduce the cost and time of performing full scale, component and coupon test programs, the number of cycles in a test spectrum needs to be kept to a minimum, without altering the effects of the spectrum on the fatigue life. This is especially important for spectra containing sequences of dynamic loading, such as buffet, as the raw spectra can easily contain over a million cycles. There are several methods of reducing the number of load lines, such as by capping, filtering or dead-banding spectra, for example. It is, however, essential to evaluate the effects of this truncation on the fatigue behaviour. While analytical predictions are the most efficient way to predict the effects of truncation, predictive results may vary considerably based on the material data and the damage models chosen for the analysis. With this in mind, a study was performed to test the effects of truncating two spectra with and without dynamic loads. The goal was to better understand the effects of truncation during all the phases of crack growth and to have test data available for comparison to analytical results. The study was led by RUAG AG in Switzerland with the support of RMIT University in Australia.

Truncation of a spectrum with low dynamics

A wing root bending moment spectrum containing low dynamics was tested as part of a coupon program to support a full-scale test of a Swiss center barrel. The centre barrel is made primarily of a forged recrystalization annealed Ti6Al4V. Filtering the spectrum with a 9% max/min rise/fall filter predicted no change in the analytical crack initiation (CI) life (<0.25 mm) and a 0.5% change in crack growth (CG) life (0.25 mm to failure). This reduced the number of load lines from over 1 million, to approximately 30 000. To verify the analytical predictions, high Kt single edge notched coupons were tested with a truncated and an untruncated spectrum in the 'L-T' orientation at two load levels. The coupons were lightly etched to reflect aircraft production processes and to reduce scatter.

A fully reversed spectrum was used, and marker bands were added to support the evaluation of the crack growth rates, from small cracks until final failure. Due to the microstructure of the RA titanium material, creating reliably visible markers is significantly more challenging than in other materials such as aluminum. A detailed description of the marker development is presented in the ICAF 2023 paper: *Kongshavn, I., Barter, S.A., Sorensen, L. (2023) Measuring Small Fatigue Crack Growth with the Aid of Marker Bands in Recrystallized Annealed Ti6Al4V, Proc. of the 31st ICAF Symposium.*

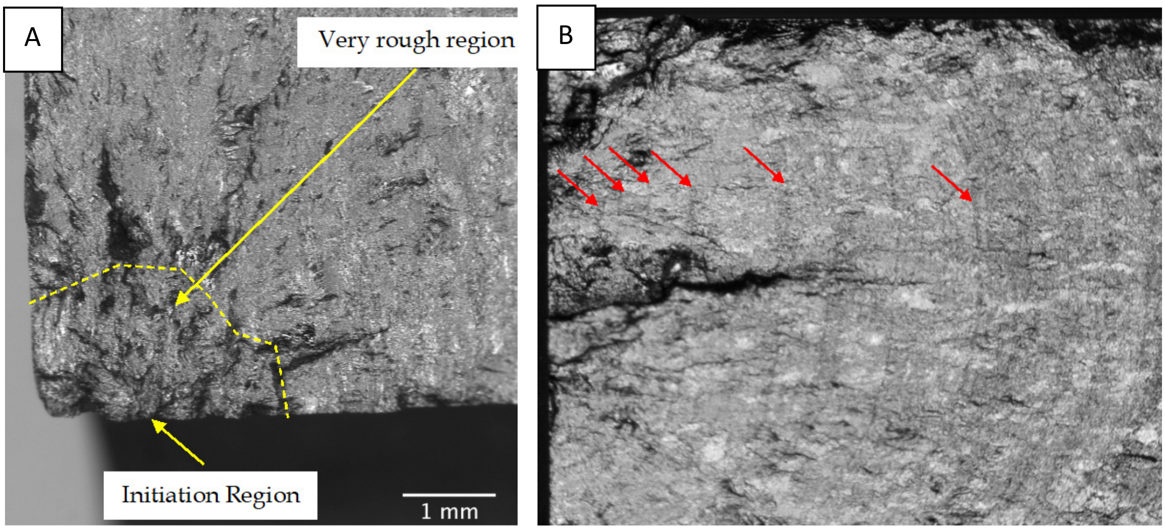

Figure 25: Forged RA Ti6Al4V SENT coupons – A: Example of a fracture surface near the origin in RA Ti6Al4V and B: of some of the marker bands visible in the optical microscope.

By measuring the repeats of the marker bands, a crack growth curve was measured for each coupon using post-test quantitative fractography. Although was a slight difference in the log-average CI lives measured between the truncated and untruncated spectra, based on the total lives to failure, the spectrum severity difference was marginal, which agrees well with the analytical prediction.

A second study was performed with the same truncated spectrum on 7050-T7451 low Kt coupons. Again, there was little effect of truncation on the fatigue life.

Truncation of a spectrum with high dynamics

The effects of truncating spectra containing significant dynamic loading was tested on low Kt coupons made of aluminum alloy 7050-T7451 thick plate (L-T orientation). To increase the probability of crack initiation, thus reducing test scatter, the length of the coupon in the test area was optimized and the coupons lightly etched with a bath simulating aircraft production processes. Two load levels were tested.

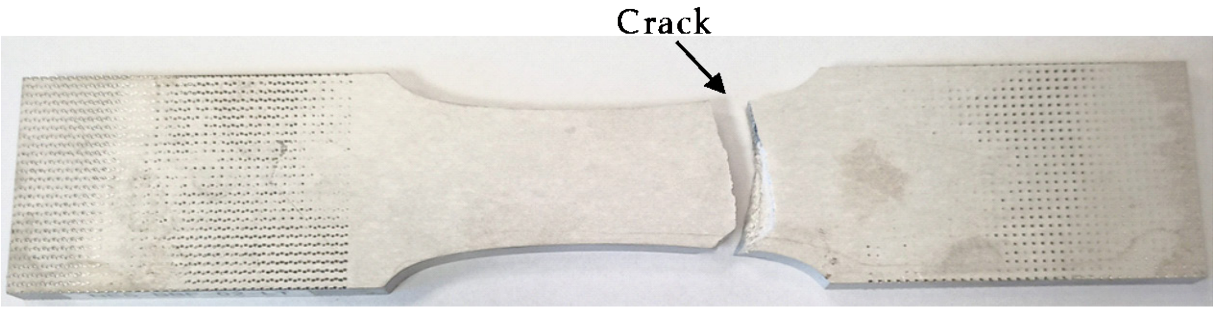

Figure 26: Etched low Kt coupon made of AA 7050-T7451 thick plate.

A typical fighter aircraft asymmetric spectrum containing buffet sequences was chosen, and filtered by 6.8% using a max/min rise/fall filter. This reduced the number of turning points from over 2 million, to approximately 200 000. Two versions of the spectrum were tested, one fully reversed to represent fillets and radii on the aircraft and one containing tension loads only, to represent lug locations. Two types of constant amplitude bar-coded marker bands were added at the end of each spectrum block, to assist in measuring crack progression using post-test quantitative fractography.

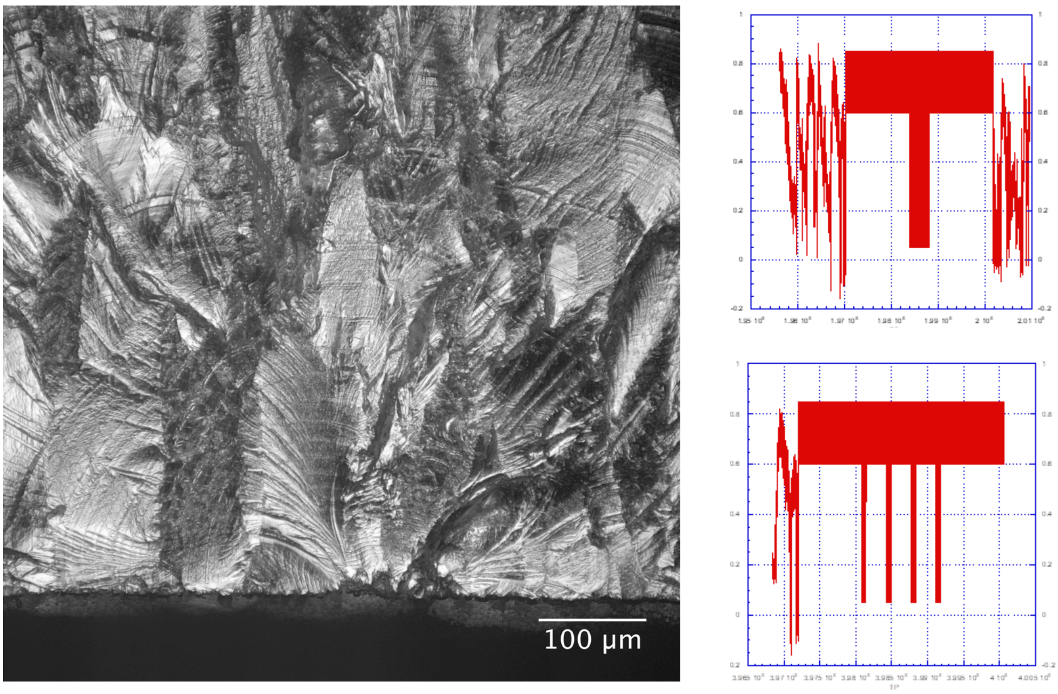

Figure 27: Some examples of marker bands (thin white lines) near the origin in a test specimen containing truncated and untruncated spectra. The constant amplitude bar-coded markers are shown.

With the aid of the marker bands, it was possible to distinguish and measure the different effects of truncation on the growth rate when the crack was small (CI life), versus at larger depths (CG life). Analytically, a different effect on truncation was predicted for the CI region than for the CG region of crack growth. The test results showed overall a smaller effect on truncation than that predicted analytically.

11. Additive Manufacturing (RUAG AG)

RUAG AG, Luca Barloggio, Beat Bachmann, Stefan Bräutigam, Esmeralda van der Kuip

RUAG AG and the former divisions such as RUAG Space have experience with the production of non-airworthy, additively manufactured parts. They are made of plastics and metals, they are used for wind tunnel models, replacements parts for land systems, ground support equipment, prototypes and tools.

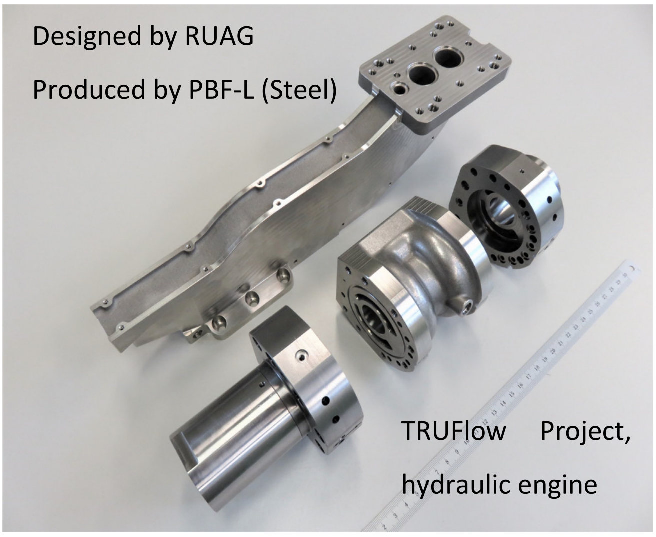

Figure 28

But what about airworthy parts? Several OEMs and military fleets operators are taking advantage of AM technologies to produce “flying” parts with unprecedented complexity while saving weight, time and costs. However, due to the novelty of the technology in aviation, the certification of AM-parts is a challenging task. The industry and the regulators (EASA and FAA above all) are working on the standardization of this technology and on guidance material for AM-part certification.

RUAG AG identified potential benefits for MRO activity and defined the goal of certifying the first additively manufactured part on the F/A-18, following the philosophy shown below and the EASA certification memorandum CM-S-008/3 – Additive Manufacturing.

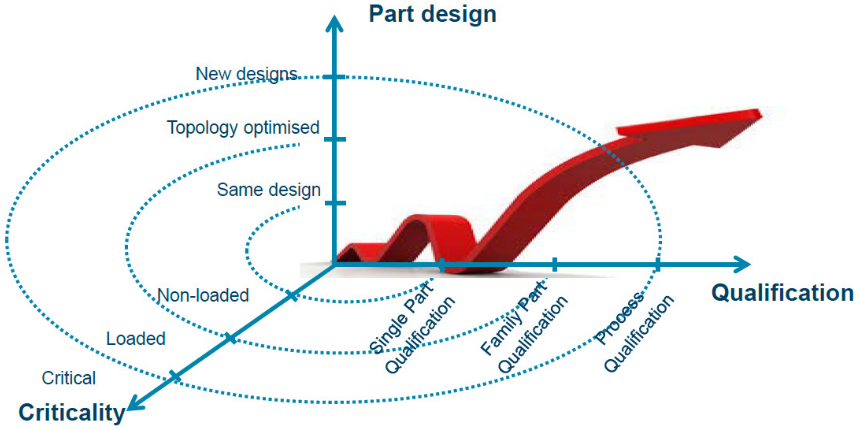

Figure 29: Certification of AM parts. Source: Structures qualification and certification approach supporting AM introduction for system parts, AIRBUS, EASA-FAA Workshop Presentation, 2019.

The main objective of the project is to gain AM know-how by producing the first "flying" component on the F/A-18 structure. Involvement of the Swiss AM suppliers SAUBER and Georg Fischer were selected in the know-how build-up, as they are excellent candidates in terms of materials availability and production for aviation. Special attention will be paid to qualification and certification of the AM process. The certification process is also applicable to other AM technologies. The candidate part is a hinge of a door below the V-Stab. The AM-part will be produced using powder bed fusion laser based technology; aluminum and titanium alloys are the primary target materials.

12. Evaluation of Risk for Stress Corrosion Cracking using FMEA (RUAG AG)

RUAG AG, Andreas Uebersax, Raphael Zehnder, Daniel Rölli, Sandro Christen

Introduction

Stress Corrosion Cracking (SCC) is a known threat to aircraft structural integrity, in particular for aging aircraft such as the F-5E/F Tiger. Due to the various parameters of influence, an accurate prediction of SCC is mostly unfeasible. Thus, airworthiness is often guaranteed by general inspection requirements without any specific guidance towards specific parts or locations.

In the presented approach a Failure Mode and Effects Analysis (FMEA) is used: possible hazard scenarios are compiled and ranked, based on their severity and probability of occurrence. This procedure allows for the identification of the most critical locations and provides the basis for the definition of mitigation actions.

General Procedure

In a first step, all primary structural parts made of the SCC prone aluminium 7075-T6x of the F-5E/F aircraft are identified. In a pre-evaluation, the amount of parts is narrowed down by sorting out parts that are either irrelevant for the Swiss configuration or that are only marginally susceptible to SCC. In a third step, the actual FMEA is run for the remaining parts. The plausibility of the rating is checked by comparing the processed probabilities of parts with an actual SCC damage history. It shows that their probability attributed by the evaluation process is ranking within the two highest categories. Therefore, the probability ranking is considered credible.

Definition of the FMEA matrix

The FMEA matrix is built up with four columns. All entries from each column are combined with the entries in all other columns as generally shown in Table 1. Assuming i, k, m and n entries in these columns, the resulting FMEA matrix theoretically features i·k·m·n rows of different combinations, hereafter referred to as scenarios. However, the amount of rows can usually be reduced by sorting out irrelevant scenarios.

It shall be recalled that the failure mode is defined as “part failure”, thus the probability and severity evaluations are made with respect to that incident, not for SCC formation. Therefore, the only potential failure mode in column 2 is “failure”. The considered potential effects in column 3 are:

- loss of structural integrity
- loss of redundancy
- loss of cabin pressure
- loss of attached parts
- jammed landing gear

There are more possible effects, but the above listed effects were estimated the most important for the present investigation.

Table 1: FMEA matrix build-up methodology

Column 1	Column 2	Column 3	Column 4
Part	**Potential failure mode**	**Potential effect of failure**	**Potential cause of failure**
Part 1	Failure mode 1	Effect 1	Cause 1
			⋮
			Cause n
		⋮	⋮
		Effect m	Cause 1
			⋮
			Cause n
	⋮	⋮	⋮
	Failure mode k	Effect 1	Cause 1
			⋮
			Cause n
		⋮	⋮
		Effect m	Cause 1
			⋮
			Cause n
⋮	⋮	⋮	⋮
Part i	Failure mode 1	Effect 1	Cause 1
			⋮
			Cause n
		⋮	⋮
		Effect m	Cause 1
			⋮
			Cause n
	⋮	⋮	⋮
	Failure mode k	Effect 1	Cause 1
			⋮
			Cause n
		⋮	⋮
		Effect m	Cause 1
			⋮
			Cause n

Finally, the considered potential causes of failure are:

- sustained loads during ground time
- stresses due to interference fit fasteners
- stresses due to mounting constraints

Determination of probability and risk

To define the probability of each scenario, the sustained load is roughly determined in a first step. The level of the sustained loads, i.e. loads due to fuselage bending during aircraft ground time, loads due to interference fit fasteners or loads due to mounting constraints, is assumed to be the key driver for the formation of SCC.

Table 2: Initial Probability Level

Parameter	Description	Initial Probability Level
Level of sustaining load/stress	Low (or compressive)	0
	Medium	1
	High	2

Other parameters of influence are accounted for by a Level Correction Factor. This factor is derived by summing up the individual corrective contributions according to Table 3.

Table 3: Probability level corrections

Parameter	Description	Correction
Extrusion	Non-extruded part	0
	Principal grain direction in parallel to sustaining loads	+1
	Principal grain direction normal to sustaining loads	+2
Environment	Standard atmosphere	0
	Particularly corrosive environment	+1
Detectability during regular maintenance	Difficult visibility	0
	Good visibility	-1

The Level Correction Factor is added to the Initial Probability Level of Table 2. This results in a Corrected Level of Sustained Load which is translated in a probability level according Table 4.

Table 4: Translation of Corrected Levels of Sustained Loads to Standard Probability Levels

Corrected Level of Sustained Load	4	3	2	1	0	-1
Standard Probability Level	A	B	C	D	E	F

Table 5: Hazard risk evaluation matrix

	Probability				
Severity	Frequent A	Probable B	Occasional C	Remote D	Improbable E
Catastrophic I	1	2	4	6	9
Critical II	3	5	8	11	14
Marginal III	7	10	13	15	16
Negligible IV	12	17	18	19	20

Risk level
Negligible
Low
Medium
Serious
High

The probability level is then combined with the severity in a standard hazard risk evaluation matrix, as shown in Table 5. In total, 108 scenarios for 34 parts are evaluated with this method. An example is shown in Table 6.

Table 6: Example of the evaluation of a part (lower longeron F.S.90 to F.S.137.5) with its final Hazard Risk Index (HRI) of 11 regarding SCC

ID	A/C Config	Part ID	*P/N*	Part Name	Potential Failure Mode	Potential Effect of Failure	Severity	Comments to Severity Determination	Potential Cause of Failure	Probability	Comments on Probability Determination	CAT	HRI
FMEA Matrix ID	**A/C configurations:** *F-5E* *F-5F* *F-5E/F*	**Part ID**	**Part number of the component**	**Part name of the considered component**	**Potential Failure Modes are:** *failure/damage*	**Potential Effects of Failure are:** Loss of structural integrity Loss of redundancy Loss of cabin pressure Loss of attached parts Jammed landing gear	**Severity levels are:** *1 (I): CATASTROPHIC* *2 (II): CRITICAL* *3 (III): MARGINAL* *4 (IV): NEGLIGIBLE*	Justification of the severity rating	**Potential Causes of Failure are:** Static loading during ground time Stresses due to Hi-Lok fastener Stresses due to mounting constraints	**Probability levels are:** *A: FREQUENT* *B: PROBABLE* *C: OCCASIONAL* *D: REMOTE* *E: IMPROBABLE* *F: eliminated (not considered)*	Justification of the probability rating	**Risk Categories are:** *1A, 1B, 1C, 1D, 1E, (1F)*	**HRI value:** Values from 1 to 20
4	F-5E	P02	6-10782	Longeron, Lower, F.S. 90 to F.S. 137.5 LH	failure/damage	Loss of structural integrity	2	A failure of the part is seen to be critical for the structural integrity of the A/C	Static loading during ground time	D	**GENERAL PART DESCRIPTION:** ▪ Longeron is part of the primary fuselage structure (torque box) ▪ Scallops could lead to notching effect **LEVEL OF SUSTAINING LOAD:** ▪ **Load level:** low (0) **LEVEL CORRECTION FACTOR:** ▪ **Extrusion:** yes, principal grain direction parallel to main loading direction (1) ▪ **Environment:** standard atmosphere (0) ▪ **Detectability:** Upper flange visible through access door, limited visibility of lower flanges (0)	2D	✔ 11

Results

Out of the 34 parts, six parts are identified with a significant risk for a SCC failure, whereas two of them have a known SCC damage history. Future detailed analysis and inspections of these parts will reveal the accuracy of this simple predictive method.