

Review of Aeronautical Fatigue Investigations in Germany during the Period April 2017 to March 2019

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Abstract

This review represents a compilation of abstracts on aeronautical fatigue investigations in Germany during the period April 2017 to March 2019 and is presented within the scope of the Meeting of the International Committee on Aeronautical Fatigue in Krakow, Poland, June 2nd-7th 2019.

The contribution of summaries by German aerospace manufacturers, governmental and private research institutes, universities as well as aerospace authorities was completely voluntary, and is acknowledged with sincere appreciation by the author of this review.

The same content will also be presented during the National German Review in front of the audience.

Enquiries concerning the contents should be addressed directly to the author of the corresponding summary.

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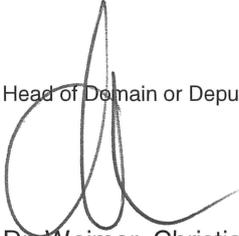
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1 Introduction

This review represents a compilation of abstracts on aeronautical fatigue investigations in Germany during the period from April 2017 to March 2019 and is presented within the scope of the Meeting of the International Committee on Aeronautical Fatigue in Krakow, Poland, June 2nd - 7th 2019.

The contribution of summaries by German aerospace manufacturers, governmental and private research institutes, universities as well as aerospace authorities (Table 1) was completely voluntary, and is acknowledged with sincere appreciation by the author of this review.

Enquiries concerning the contents should be addressed directly to the author of the corresponding summary.

Table 1: Mailing addresses of contributing companies and institutes

Abbrevia- tion	Details
AIRBUS	Airbus Operations GmbH; Kreetslag 10, D-21129 Hamburg, Germany, www.airbus.com
CRT	Airbus Central Research and Technology, Willy-Messerschmitt-Strasse, D-82024 Taufkirchen, Germany, www.airbus.com
HZG	Helmholtz-Zentrum Geesthacht; Max-Planck-Straße 1, D-21502 Geesthacht, Germany, www.hzg.de
IABG	Industrieanlagen-Betriebsgesellschaft mbH; PO-Box 1212, D-85503 Ottobrunn, Germany, www.iabg.de
IMA	IMA Materialforschung und Anwendungstechnik GmbH; Postfach 80 01 44, D-01101 Dresden, Germany, www.ima-dresden.de
LBF	Fraunhofer Institute for Structural Durability and System Reliability; Bartningstrasse 47, Darmstadt, Germany, www.lbf.fraunhofer.de
ISD	Leibniz Universität Hannover, Institut für Statik und Dynamik (Institute of Structural Analysis), Appelstraße 9A, D-30167 Hannover, Germany, http://www.isd.uni-hannover.de
PAG	Premium AEROTEC GmbH, Haunstetter Str. 225, 86179 Augsburg, Germany, www.premium-aerotec.com
PPI	Institute of Product and Process Innovation, Leuphana University of Lüneburg, D-21339 Lüneburg, Germany, www.leuphana.de/en/institutes/ppi.html

2 Full Scale Testing

2.1 PC24 – Major Tests

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The PC-24 is the latest and most advanced aircraft type of the Swiss-based company Pilatus Aircraft Ltd. As part of the type certification the PC-24 business jet aircraft had to fulfil the requirements of CS-23 Amendment 3, commuter category. Foreign concurrent validation of the EASA certification was performed together with the FAA (USA) and TCCA (Canada).

Full Scale Static Test (FSST)

A full scale static test (FSST) was performed which included a selection of critical load cases applied on the empennage, the wing and the fuselage. The FSST test-article was consisting of a complete PC-24 airframe in series aircraft configuration. The main goal of the FSST test campaign was to provide a final means of compliance to the CS23 requirements (CS23.305 & CS23.307 primarily), for demonstration of the structural integrity of the airframe by (amongst others):

- Demonstration of no permanent detrimental deformation of structure at limit load levels
- Demonstration of the capability to resist ultimate load levels for a period of at least 3 seconds
- Confirmation of stiffness and load distribution within the structure

The test campaign was performed and completed within the first half of 2017.

Full Scale Fatigue Tests (FSFT)

A full scale fatigue test (FSFT) and a separate full scale empennage fatigue test (FSEFT) was performed for compliance demonstration (Figure 1.) Both Tests were including a durability test phase where 2 x 30,000 flight hours (FH) were simulated plus a damage tolerance test phase with 1 x 30,000 FH. After completion of cyclic testing a residual strength test was carried out, followed by a tear down inspection (TDI). The aircraft design service goal (DSG) is 30,000 flights, corresponding to 30,000 landings or 30,000 FH. Especially to be mentioned is the test speed of at least 1,000 flights in 35 hours in full compliance with the test accuracy requirements. Both tests have entered fatigue cycling in early 2017 and were finished in June 2018.

Ground Load Calibration (GLC)

Part of the certification requirements was to carry out a Flight Loads Survey. To validate the flight load survey, a Ground Load Calibration test was performed (Figure 2). Loading was applied on the wing, empennage and engine beams in a series of tests, to assess the response of the structure and ultimately to derive load equations that can be used in conjunction with the flight data to associate known strains with aircraft bending moment, shear forces, and torsion during the flight load survey campaign.

The GLC test article was a PC-24 Prototype airframe, which was part of the flight test fleet. The test campaign was performed and completed in the first quarter of 2017.

Overview of test equipment

An overview about the used testing equipment is given below in Table 2.

Table 2: Equipment used for FSST, FSFT, FSEFT and GLC

	Actuators	Strain Gauges	Deflection transducers
FSST	33	Up to 800	Up to 33
FSFT	34	Up to 600	Up to 10
FSEFT	11	Up to 240	Up to 5
GLC	14 (on variable positions)	On-board systems used	Up to 16



Figure 1: PC-24 FSFT - Test Overview (with FSEFT and GLC in background)



Figure 2: PC-24 GLC - Test Overview

2.2 BD500-DADTT - Durability and Damage Tolerance Test of Bombardiers CSeries

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The DURABILITY AND DAMAGE TOLERANCE TEST (DADTT) for the type certification of Bombardiers CSeries (BD500) CS100 / CS300 aircraft family (now Airbus A220) started on schedule in August 2014 at IABG in Dresden and was accomplished in 2018 with the performance of the residual strength testing (RST). The number of cycles required for Entry-Into-Service was reached end of Nov 2014. Right after accomplishing RST the teardown inspection was started by IABG and it will be finished until mid of 2019.

A very fast fatigue test program was performed which was possible only because a preceding virtual test setup simulation has been performed by IABG to prepare and run such a sophisticated test fulfilling the customer test speed and accuracy expectations. The fast progress required also a very high availability of the test setup for a duration of more than 16,000 pure operating hours.

The testing was carried out on the CS100 model and obtained also sufficient data to enable successful certification of the CS300 aircraft. The Durability, Damage Tolerance and Residual Strength of the aircraft metallic structure were covered by the DADTT.

The metallic structure of the airframe was subjected to spectrum loading for a minimum of 3 times the Design Service Goal (DSG), i.e. 180,000 flight cycles, followed by a residual strength test and a teardown inspection.

The test specimen consists of

1. The pressurized fuselage extending from frame C277 to C1476
2. The right and left outer wing boxes attached to the centre wing box
3. Vertical stabilizer incl. rudder
4. Main and Nose Landing Gear (partly)
5. Dummy Structures
 - a. Horizontal Stabilizer
 - b. Main and Nose Landing Gear
 - c. Engine pylons
 - d. Flaps, ailerons, slats
 - e. Winglets

Typical repairs for metallic structures were introduced in order to validate the Structural Repair Manual. Artificial damages have been introduced into the structure at the beginning of the third DSG, after 120,000 flight cycles have been achieved.

In total the BD500 specimen was loaded by 124 hydraulic jacks and 7 restraints (Figure 3 and Figure 4). Measurement of strains and deflections with up to 1500 Strain Gauges and Displacement Transducers was required. The test airframe was subjected to a regime of fatigue loads, 24 hours per day, 7 days a week.

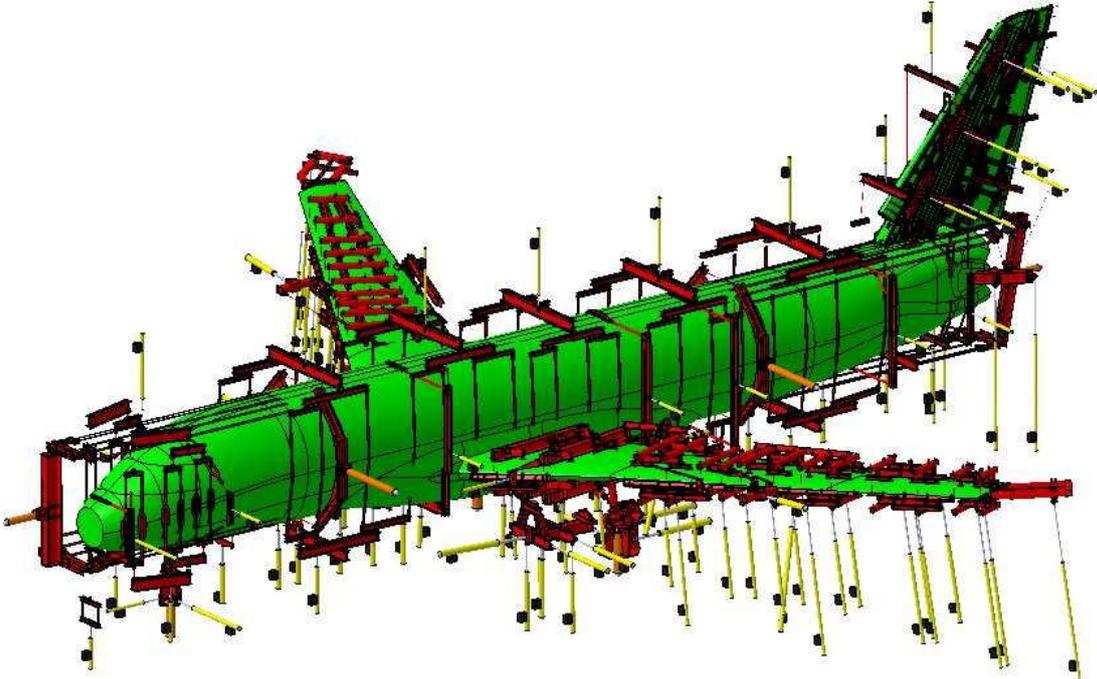


Figure 3: Overview of the load introduction tools for the BD500 Fatigue Test at IABG



Figure 4: The BD500 Fatigue Test Set-up at IABG in Dresden

2.3 Changing the Philosophy of Full-Scale-Fatigue-Tests derived from 50 Years of IABG Experience towards a Virtual Environment

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Over the last 50 years of full-scale fatigue testing, IABG has continuously developed new techniques in order to improve time, cost and quality for full-scale fatigue tests. Last year the performance of Airbus A350XWB EF2 full-scale fatigue test at IABG test lab in Erding has been completed faster than any other full-scale fatigue test in the past. Recent and ongoing testing activities on Airbus A220 (Bombardier C-Series) and Pilatus PC24 have confirmed the same speed-up by covering three life times within one single year of pure test performance.

With the advancement of virtualisation of the aircraft development and certification process, questions have to be raised as to how full-scale fatigue tests can be incorporated into these processes and how virtualisation can benefit from physical testing.

Virtualisation speeds up aircraft development to its next level. Virtual design rooms, linear numerical simulations with virtually unlimited number of elements and off-the-shelf software solutions from sketch to 3D printing – all these radical developments are changing time, cost and planning expectations. Does fatigue testing still fit into this context?

Nevertheless, the current EASA Part-21 AMCs still demand physical tests for new aircraft types. Drawing from the testing experience, failures are often first detected only during the full-scale fatigue tests. These aspects, in connection with the mental barrier to sacrifice the proven processes for the sake of economic gains, underline that full-scale fatigue tests are still state-of-the-art.

Considering the limits and the related efforts of a full-scale fatigue tests, virtualisation as a set of methodologies has increased safety probably more than pure fatigue testing and that both methods combined are today's standard in order to keep aircraft safety on its mandatory level. The purpose of full-scale fatigue tests are changing. The amount of available but unused information gathered during fatigue runs is still sizable, and changing the attitude towards fatigue tests could increase the return of investment.

This will lead to introducing new agile methods and processes into fatigue tests to allow the numerical methods to evolve continuously with the available experimental data. This could also result in a new philosophy of full-scale fatigue tests, moving from flight-by-flight testing towards purely artificially triggered load sequence in order to detect blind spots in the numerical analyses. This does require the certification philosophy to evolve hand-in-hand (Figure 5).



Figure 5: Full-Scale Fatigue Testing

3 Non-Destructive Testing and Structural Health Monitoring

3.1 Development and test of a load and damage health monitoring system and its implementation into a composite fuselage flight demonstrator

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Project Context and Objectives

In the frame of the platform GRA LWC (Green Regional Aircraft Low Weight Configuration) of the European Project “Clean Sky” Fraunhofer Institute for Structural Durability and System Reliability (LBF) has designed, developed, and implemented a load and damage monitoring system in a flight demonstrator of ATR. As flight demonstrator, the original metal panel of the fuselage structure was replaced by a CFRP structure by Leonardo (see Figure 6). The activities performed by Fraunhofer LBF within GRA LWC were supported by CSJU-GAM-GRA-2008-01 by the EC in the frame of JTI Clean Sky. In the preparation of the flight tests LBF qualified the system of fiber optic sensors for load monitoring and piezo sensors for damage monitoring on flat stiffened CFRP panels in a ground test campaign. The fatigue-proof and high tension capable optical fibers are advantageous compared to usual strain gauges in terms of reliability, sensitivity and material behavior in relation to the CFRP structure.



Figure 6: Flight test of the composite crown panel on an ATR 72-600 with a network of 94 sensors developed by Fraunhofer LBF. © ATR/Alenia Aermacchi(Leonardo)

In the flight demonstrator both structural monitoring technologies from Fraunhofer LBF were integrated. On the one hand a technology based on optical fiber Bragg grating (OFBG) consisted of a network of 40 sensors externally bonded on the panel for the

measurement of the longitudinal (16 strain sensors) and transversal strain (16 strain sensors) of the skin and stringer and eight fiber optic sensors were used for temperature compensation. With this technology stringer loads were monitored and in case of a debonding the difference in load distribution would be immediately detectable. The other technology was a network of 54 piezoelectric (PZT) transducers alternating as sensors and as emitters which were implemented with the aim to measure the structural behavior and based on this measurement, to detect impact damages within skin or stringers.

Static and fatigue test phases with and without impact damages were performed beforehand using four special flat CFRP panels for qualification of the load monitoring system for the flight test (see Figure 7). The optical fibers were embedded under the stringers through a co-boding process.

The static loading phase with limit load and ultimate load tests was performed to validate the FEM simulation as well as to verify that the panel and the SHM system can support the loads. In addition some panels were subjected to barely visible impact damage (BVDI) as well as visible impact damage (VDI). After performing the fatigue testing residual strength tests were realized to demonstrate the damage tolerance of the structure.

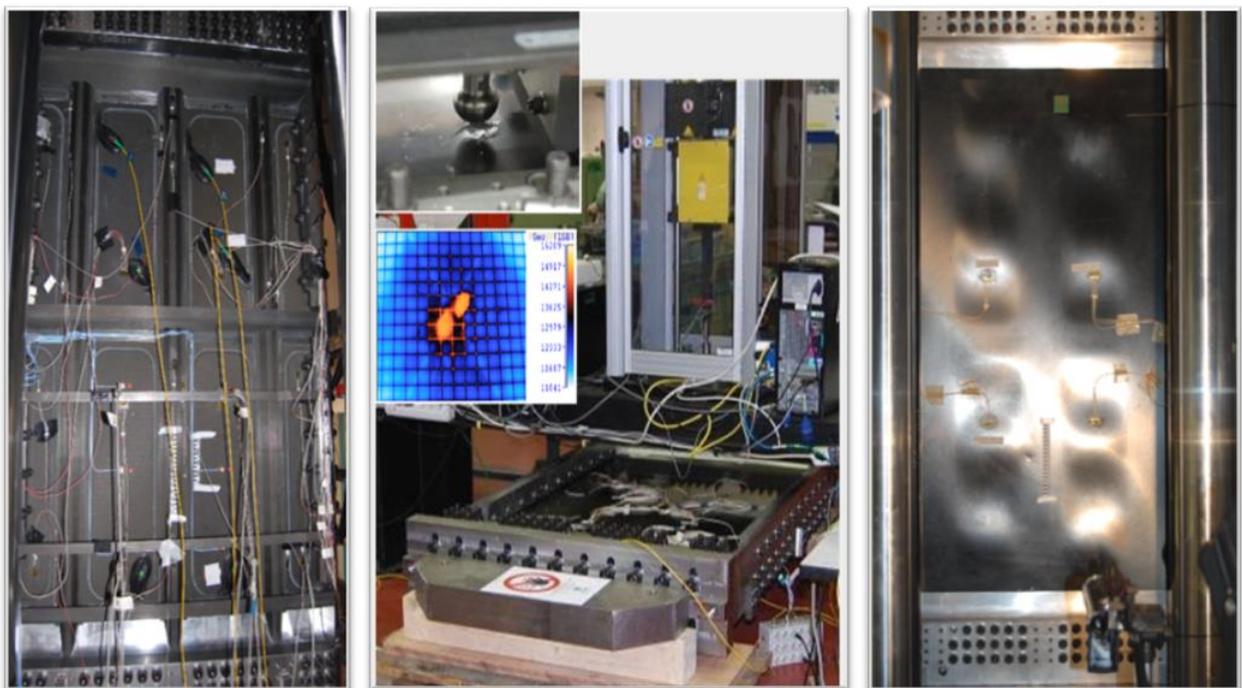


Figure 7: Left: Ground tests of the Load and Health Monitoring System integrated in a CFRP panel. Middle: Impact damage of a panel. Right: Fatigue and residual strength testing. ©LBF

Results and Conclusion

The developed and integrated Fraunhofer LBF SHM system for the non-damaged CFRP crown panel flight demonstrator have shown very good results for the monitoring of the structure during the flight.

With this activity, Fraunhofer LBF demonstrated the ability to monitor the behavior of a large composite airframe structure with a network of strain and piezoelectric sensors on

the ground and in flight. In the future, the capability to monitor structural components during flight and on ground will help to update load standards and to accordingly review the safety factors which are used for improved fatigue design of aircraft structures.

The optical fibers were able to monitor all the changes of strain levels inside the structure during the flight test. All the deformation of the substituted crown panel caused by the different flight maneuvers and flight phases were detectable and were visible by the strain signal recorded with the optical fiber network. A direct correlation between the strain signal and the recorded flight data was possible and reliable so that in reverse, the status of the plane during the flight would be derivable from the data of the optical fibers. All the applied loads during the flight were documented by the optical fibers so that a monitoring of the CFRP crown panel was reliably implemented.

For serial applications the glass fibers need to be protected against accidental damage. The capability to equip one single fiber with many sensor locations (multiplexing) and to integrate the small diameter fiber within the composite structure minimizes the weight in comparison with metallic sensors, gives new design space and offers protection against external influence on the fragile glass fiber. The ground test showed that a co-bonding process to integrate the optical fiber during the panel manufacturing underneath the stringers is possible without detectable structural effect.

The developed SHM technologies for detection of impact damages and for monitoring the behavior of the structure could make the regional aviation safer. Overall, the SHM system developed by Fraunhofer LBF is suitable for load and damage detection purpose and provides good results in terms of monitoring a large CFRP structure during ground and flight tests.

3.2 Evaluating the Influence of SHM on Damage Tolerant Aircraft Structures Considering Fatigue

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First patents concerning the integration of Structural Health Monitoring (SHM) capabilities into aircraft have been granted over 70 years ago [1]. Since then, numerous potential applications of SHM along the entire aircraft lifecycle have been identified in scientific literature. However, the application of this technology in commercial aviation is currently limited to field studies carried out primarily within aging fleets. Aside from recent advances in sensor technology, growing research effort has been committed to identify and quantify promising business cases for SHM [2]. Even though numerous individual applications have been investigated with regard to their lifecycle cost impact, current studies yield varying results. In order to facilitate the integration of SHM in commercial aviation we plan to put forward an integrated analysis framework for SHM technologies covering the entire aircraft lifecycle including design, operation and retirement. As part of this framework we introduce an evaluation approach expanding [3] to identify the impact of SHM on aircraft structural design considering structural weight, risk of structural failure and inspection efforts, as depicted in Figure 8. The proposed methodology is limited to damage tolerant structures prone to fatigue and considers only component sizing rather than the redesign

of structures. By using the established relation between structural dimensions, weight, inspection effort and probability of failure, the approach is calibrated with inspection intervals provided by the manufacturer as well as information from aircraft fatigue tests. Subsequently, the influence of aircraft Operations Monitoring (OM) and SHM can be analyzed using an ideal sensor system (no false alarms) as well as real sensor system (possible false alarms). Finally, a case study is presented demonstrating the suggested methodology utilizing a commercial narrow-body passenger aircraft. Employing the methodology within our integrated analysis framework a better understanding with regard to sensor performance requirements and the impact of SHM on operational aspects can be developed.

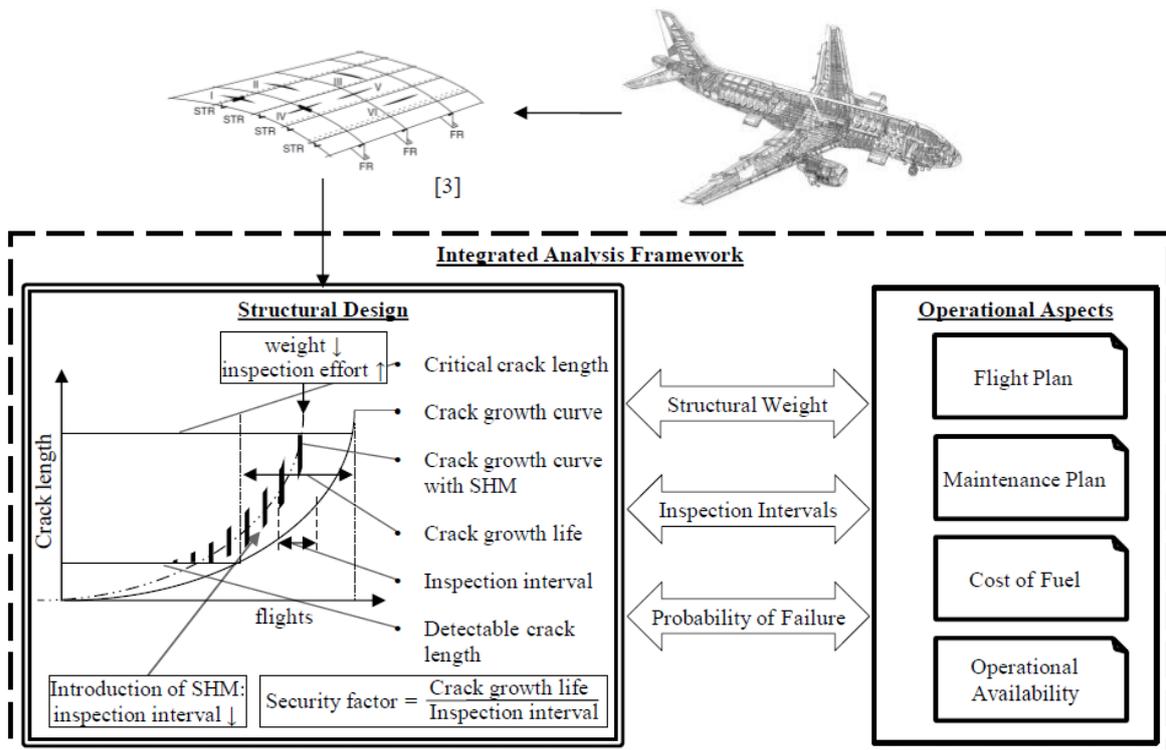


Figure 8: Schematic representation of design evaluation approach within integrated analysis framework.

Literature

- [1] W. C. Green, Pressure-detecting covering.
- [2] M. J. Bos, „Review of aeronautical fatigue and structural integrity investigations in the Netherlands during the period March 2015 - March 2017“ NLR-TP-2017-137, 2017.
- [3] H.-J. Schmidt und B. Schmidt-Brandecker, „Design benefits in aeronautics resulting from SHM“, Encyclopedia of Structural Health Monitoring, 2009.

4 Fatigue and Fracture of Fuselage Panels and Joints

4.1 Curved panel fatigue and damage tolerance testing

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For research projects of different customers IMA Dresden performed fatigue and damage tolerance tests of several curved fuselage panels. The tests conditions covered internal pressure loads with superimposed axial tension loads. All panels were loaded with complex flight-by-flight load spectra. Targets for the different panels were:

- Investigate design life of current and future fuselage designs
- Investigate crack growth and repair methods under fatigue loading

The panels under investigation had different design approaches:

1. "Classic" design with longitudinal and circumferential joints
2. Fiber metal laminate skin
3. Over wing door panel

For the damage tolerance tests panels contained artificial damages to determine crack growth properties. With multiple panels the characteristics for different kinds of damage were investigated. This included, e.g. saw cuts in various areas of interest or longitudinal cracks across broken frames. The test program was accompanied by static tests. In particular, after reaching an extension of a 2-bay crack, the residual strength properties under over-pressure conditions were determined.

The over wing door panel test is currently running (see Figure 9). This test contains a more complex loading program consisting of not only internal pressure and axial tension, but also bending, shear and active frame loading.

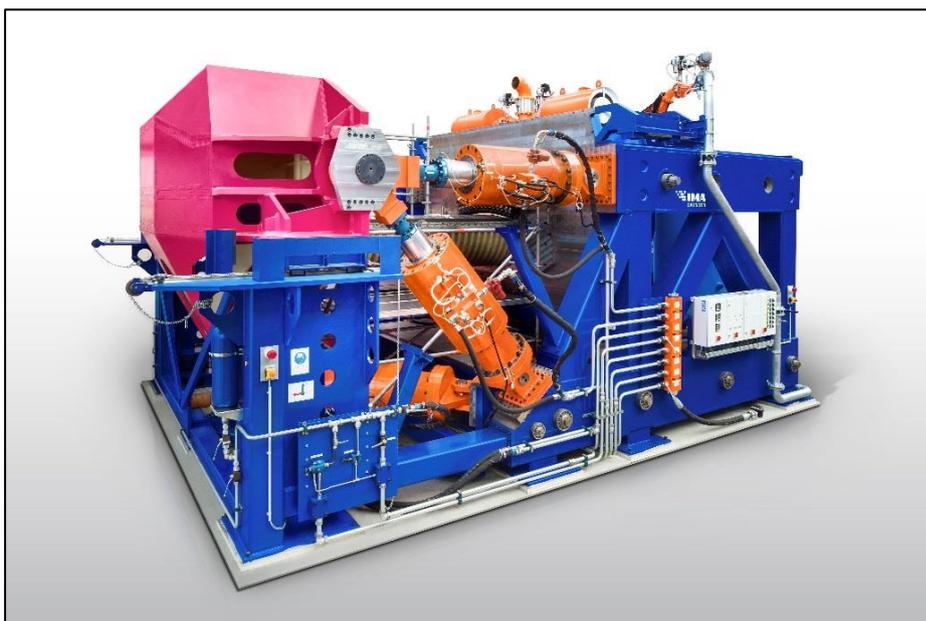


Figure 9: Panel Test Rig for over wing door panel test

5 Fatigue Life Assessment and Prediction

5.1 Crack propagation modelling and testing of a Compression Pad

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A Compression Pad of the A380 (see Figure 10) was inspected during tear down of the A380 inboard flap full-scale fatigue test after test completion. This inspection revealed a crack starting at the radius between shaft and disc part of the compression pad. At the end of the test the crack sustained several limit load cases, showing residual strength capability. This summary is based on the following presentation at the AA&S / PS&S 2018 conference: *Fatigue Life Determination of complex structure under complex loading - Case A380 Flap Track Compression Pad*.

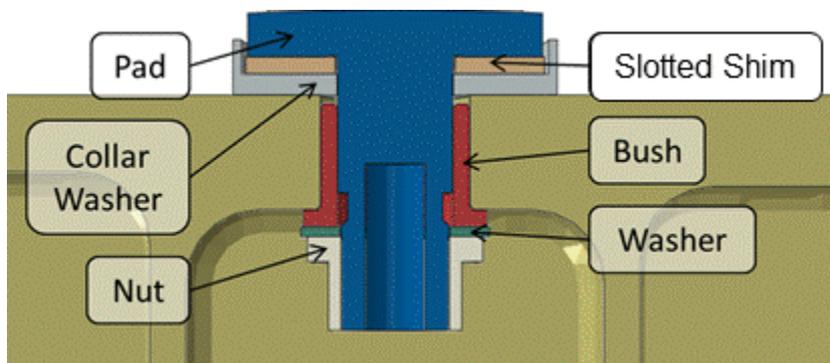


Figure 10: Detailed design and integration principle of the A380 Flap Track Compression Pad

It was decided to re-design the compression pads to reduce the local stress concentration, in order to avoid crack initiation. However, for not modified parts a further analysis was started, to better understand the crack growth behaviour. The standard crack propagation models cover the critical location, but the accuracy and detailed loading type can be further improved. Therefore it was decided to support the standard crack growth calculations with some component fatigue tests and detailed finite element modelling (DFEM) to reduce the conservatism in the fatigue assessment.

To get a more accurate loading on the part, interface load measurements from flight test campaign have been used. The component test loads are therefore based on flight test, including axial compression and friction. The fatigue test spectrum is simplified from measured spectrum for practicality, reducing the amount of load cases with a factor of approximately 15. The reduction was done by keeping the damage significant load cycles, rather than looking reducing based on load amplitude.

During the component fatigue test the stress level was monitored with strain gauges. Further non-destructive inspections were repeatedly performed to obtain information about the crack propagation. The results from the strain gauge measurement and the obtained crack length were used for the DFEM validation (Figure 11). The DFEM developed included non-linear elastic properties and interfaces modelled with contact elements. The local stress state provided by DFEM matched with strain gauge measurement

close to the radius. To support the standard crack propagation additional crack compounding factors were derived from the DFEM to model the cracks more accurately.

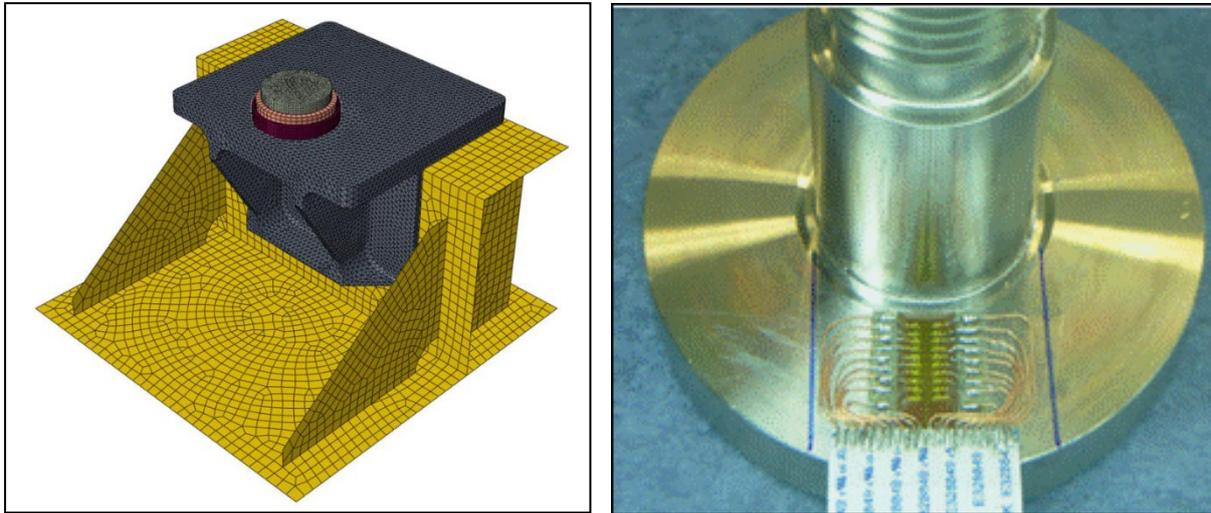


Figure 11: Left: DFEM of compression pad. Right: Compression pad equipped with strain gauges.

A good alignment was achieved between crack growth prediction and crack growth results from the component test. Using advanced modelling techniques, supported by test evidence, provides opportunities to reduce maintenance cost impact and maintaining the high safety standard.

5.2 Machine Learning Application on Aircraft Fatigue Stress Predictions

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Machine learning applications are rapidly gaining momentum and already offer significant benefits in many areas of industry; from traffic predictions and flow of goods to fighting cancer, the world is turning increasingly to intelligent algorithms to overcome its most persistent challenges. With an ever-growing demand for air travel and an increasingly competitive market, it has now become a priority for commercial aircraft manufacturers to join the digital revolution.

The fundamental requirement for tailored aircraft maintenance solutions providing fully optimised scheduling without compromise to safety is knowledge of real-world aircraft usage. By harvesting fleet-wide aircraft usage parameters (e.g. ‘big data’ flight-by-flight recordings, meteorological conditions, etc.), and exploiting them with the application of validated machine learning algorithms, highly accurate predictions of the internal loading conditions of a structure become possible based on measured and recorded aircraft parameters (e.g. speed, altitude, flight-configurations, accelerations etc.). The major benefit of such data analytics is the possibility to recreate the real loading sequence as experienced by the structure. To fully validate any conclusions from machine learning, it must be recognised that other information will need to be considered in the context of the analysis, e.g. additional in-service data, full scale test results, theoretical analyses, etc.

The significance of the potential capabilities held by machine learning applications to predict aircraft structure internal load distribution is a major enabler for linking and comparison of practical aircraft usage against that of any average fleet or initial assumptions; the unlocking of such potential offers game-changing benefits in civil aviation.

The main input needed to make relevant and reliable predictions is a comprehensive dataset. Generally speaking, this and most other parts of the puzzle are already available; the main challenge for machine learning will be to integrate these parts harmoniously, obtaining new predictive capabilities by enhancing analytical power.

5.3 Probabilistic reliability assessment of a component in the presence of internal defects

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Metallic components and structures usually contain metallurgical defects as an inevitable result of any fusion welding or manufacturing process. Typical examples of defects are pores, non-metallic inclusions, cavities, lack of fusion or corrosion pits. Such discontinuities act as stress raisers (notches) and lead to premature crack formation in the early stages of fatigue life. Fatigue cracks are prone to initiate and grow from the small metallurgical defects not only on the surface but also in the interior of a component. Therefore, reliability and fatigue performance of metallic structures is determined by the distribution of defects contained in the most highly stressed volume. In this regard, fatigue life assessment of engineering structures in the presence of defects population has become of paramount concern from practical viewpoint. The need for accurate lifetime predictions is additionally supported by modern requirements on weight efficiency and minimizing fuel consumption, which should make the fatigue design less conservative without losses in the overall safety.

Effect of surface defects and roughness on the fatigue behavior has been extensively studied over the last 50 years. However, the nature of internal fatigue crack initiation and growth, the mechanisms of the fish-eye formation and the conditions responsible for the subsurface crack nucleation are not fully elucidated yet. This is related to a great complexity associated with the experimental determination of internal crack growth and to the fact that surface notches normally overshadow the internal defects in terms of their notch severity. Nevertheless, the topic of internal defects has recently become of great importance because some advanced manufacturing and material processing technologies have emerged over the last decades. A common attribute of these technologies is that the manufactured parts consistently show internal crack origins from various types of metallurgical defects in regions of high cycles (HCF) [1] and very high cycles (VHCF) [2]. Laser-assisted material processing techniques like laser beam welding (LBW) [3], laser metal deposition (LMD) [4] and selective laser melting (SLM) [5] are typical examples of the abovementioned technologies characterized by the inherent subsurface crack initiation. Parts produced by additive manufacturing possess some unique features with regard to the surface roughness, microstructure and defect distribution. Under these circumstances, both surface and internal defects may be responsible for fatigue failure. The crack origin position is always determined by a trade-off between the surface roughness

and the population of internal defects. However, if an additional post-processing technique is applied, e.g. laser surface remelting [6] or laser shock peening (LSP) [7], fatigue cracks inevitably originate from the subsurface discontinuities and, in some cases, even from microstructural inhomogeneities. Hence, the fatigue life of these components is entirely controlled by the growth of cracks initiated from internal defects, and the scatter in fatigue life is related to the scatter of defect size. Therefore, a quantitative characterization and probabilistic modelling of the fatigue life are needed in order to ensure the in-service security.

Traditional fatigue assessment of engineering structures with defects is based on the S-N curves with safety factors and is deterministic in nature. In this work, we propose a fatigue life assessment model for internally flawed materials based on the fracture mechanics approach which takes into account the size distribution of defects as well as their spatial distribution, influence of a vacuum-like environment at the crack tip of internal cracks, short crack behavior and some other relevant aspects. A methodology for simulating the scatter of fatigue life based on the statistical distributions of defect density and defect size is presented [1]. One of the major challenges in this regard is the definition of the most critical defect that initiates the worst-case fatigue crack. In contrast to conventional assessment procedures using the assumption that the largest defect initiates a dominant crack, in this work, two random variables – pore diameter and depth – are considered to identify the life-controlling defect. The evidence for this approach has been justified by experiments, i.e. very often, the crack-initiating pore is smaller than the largest pore in a specimen but it is located in a more critical location, e.g. closer to a free surface. Thus, the crack initiating capability of a defect relies on several factors, including its depth under the surface and also the local microstructure.

The global fatigue assessment model consists of several interdependent subroutines including the crack-growth module and the probabilistic module for generation of defects distribution in a specimen. The life-controlling defects are selected by the statistics of extremes for a given geometry, stress state and randomly distributed defects in a component. Crack initiation in the sense of nucleation of microstructurally short cracks is normally neglected. The most critical defect is identified by considering the shortest fatigue life among the defects. The applied crack-growth model predicts the fatigue life on the basis of the growth of a single crack from a given material defect. The model of physically short cracks based on the cyclic R-curve is used to quantify the accelerated growth of cracks in the near-threshold region. Finally, the probability of failure of a component is calculated as a product of statistical analysis of multiple random input data.

For the purpose of investigating the effect of defects on fatigue life, specimens have been manufactured from Ti-6Al-4V by LBW. A defocused laser subsequently treated the welded joints in order to smoothen the surface to promote internal crack initiation. The weld can be regarded as a single layer of material in an additively manufactured part. The overall process of additive manufacturing can be considered as a cumulative laser welding process, where the defects of each layer are integrated in a big population for a final part. From the practical viewpoint, however, it is easier and cheaper to produce a weld with relatively simple geometry, thus, excluding other factors. Moreover, the defect size and distribution is not affected significantly by additional thermal cycles. Another factor to be considered is the required amount of statistics that is needed for the model validation. In this regard, due to its simplicity, the LBW process does not impose any restrictions on

the number of specimens thus enabling to produce a large number of specimens to account for the large inherent scatter of fatigue testing. 2D maps (diameter-depth) of defects were measured by X-Ray analysis of the specimens. Fatigue testing was followed by thorough fractographical analysis of the broken specimens. The size- and spatial distributions of defects were subsequently considered in the model to predict the life-controlling defect in the specimens. Furthermore, LSP as a post-processing technique for the fatigue life extension is investigated.

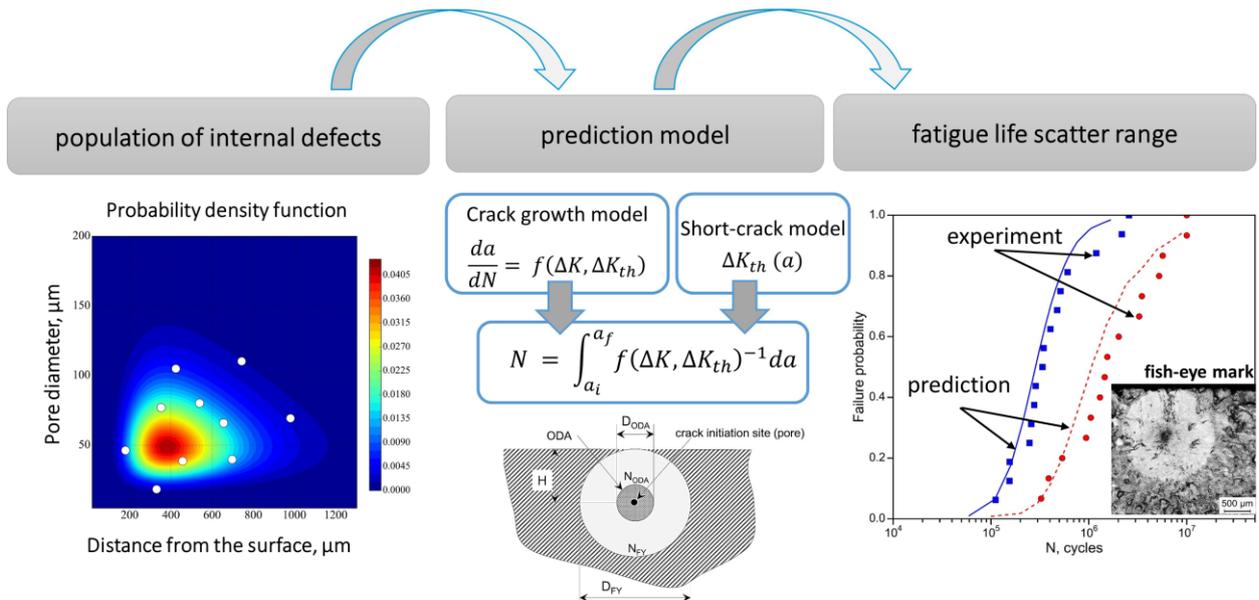


Figure 12: Schematic illustration of the probabilistic modelling for assessment of the lifetime scatter.

The model development demonstrated that accurate lifetime assessment relies on careful consideration of the crack growth rates in vacuum and the short crack effect. The size distribution of the crack initiating defects differs from the overall defect population in a volume. On average, the life-controlling pores are slightly larger than the pores initially present before testing. Moreover, under reasonable assumptions, this shift can be predicted with good agreement. Random defect analysis can successfully predict the scatter of fatigue life and probability of failure, based on the given defect population (see Figure 12). A key role in the probabilistic modelling plays a criterion of a critical pore, i.e. a pore that initiates the dominant crack. An assumption based on the shortest fatigue life supported by the fractographical observations shows reasonable agreement with experiments.

References:

[1] Fomin F., Horstmann M., Huber N., Kashaev N. *Int. J. Fatigue* 116 (2018) 22–35.
 [2] Günther J., Krewerth D., Lippmann T., et al. *Int. J. Fatigue* 94 (2017) 236–245.
 [3] Fomin F. and Kashaev N. *Procedia Structural Integrity* 7 (2017) 415–422.
 [4] Shamsaei N., Simsiwong J. *Procedia Structural Integrity* 7 (2017) 3–10.
 [5] Beese A.M., Carroll B.E. *JOM-J. Min. Met. S.* 68 (2016) 724–734.
 [6] Siddique S., Imran M., Rauer M., Kaloudis M., Wycisk E., Emmelmann K., Walther F. *Mater. Design* 83 (2015) 661–669.
 [7] Maawad E., Sano Y., Wagner L., Brokmeier H.-G., Genzel Ch., *Mat. Sci. Eng. A* 536 (2012) 82–91.

5.4 An energy-based model for the fatigue analysis of fiber reinforced composite structures under load reversals

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The originally-developed FDM by [1,2] is based on classical laminate theory and has been implemented in the commercial finite element software ABAQUS as a user-defined material subroutine (UMAT) for layered shell elements. Based on the applied loads (e.g., tension, compression and shear) and the material orientations i (e.g., longitudinal and transverse to the fibre direction) strength and stiffness degradation factors (η_{R_i} and η_{E_i} , respectively) are introduced in order to describe the damage state ($\eta_{R_i} = \eta_{E_i} = 0$ completely damaged; $\eta_{R_i} = \eta_{E_i} = 1$ undamaged).

The FDM comprises of two main parts, a discontinuous and a continuous part. In the discontinuous part, the degradation is analysed under static loading using Puck's criterion [3] for static failure. In the continuous part, the degradation is calculated due to cyclic loading [2].

The main feature of the FDM is the energy-based hypothesis of Pfanner [4]. This hypothesis implies that the damage state, represented by strength and stiffness degradation, of a quasi-statically loaded material is comparable with that of a cyclically loaded material, if the magnitude of dissipated energy is equal. Furthermore, the model allows for a block-wise analysis of the load cycles employing a cycle jump technique in order to avoid the numerically-inefficient cycle-by-cycle analysis [1].

To use the FDM for the analysis of variable load amplitude cases, additional modeling elements are required. One of the main ingredients in the damage calculation in such cases is the S-N curves. Based on [5], Constant Fatigue Life Diagrams (CFL) were constructed and implemented in the FDM [6] in order to create different S-N curves. By using the CFL, the number of cycles to failure given by the S-N curves can be determined for any stress ratio, which is an important parameter for the damage evolution relationships contained in the FDM [2,6].

For a more realistic lifetime analysis, it is also necessary to consider passive damage effects resulting from variable amplitude loading. Accordingly, in the FDM, the computation of the tension and compression stiffness degradation factors is to be coupled. It is assumed that the compression-induced damage has an effect on subsequent tension-induced damage. This means that if compression damage is identified ($\eta_{E_i^c} < 1$), the tensile stiffness degradation factors ($\eta_{E_i^t} < 1$) will also evolve under compression loading. On the other hand, it is assumed that the tension-induced damage has no influence on subsequent compression-induced damage (Crack closure) [7].

In a further step towards a realistic fatigue analysis, the quantified effects of the manufacturing induced residual stresses are taken into account [8]. With a prior cooling simulation, the residual stress field is generated using the experimentally determined thermal expansion coefficients [9] before the damage analysis using the FDM. With the aid of the constructed CFL, the local change in the mean stress in the individual layers caused by

the residual stresses is considered. Figure 13 shows schematically the extended computation procedure.

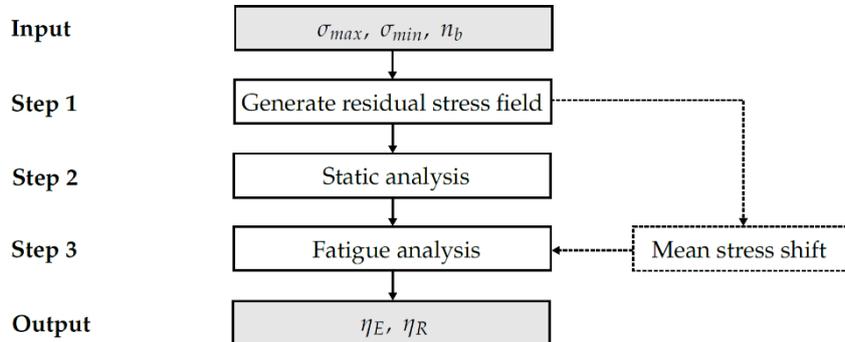


Figure 13: Extended simulation framework of the fatigue damage model

In the following, the results of a numerical study concerning the influence of load reversal on fatigue damage evolution using the extended simulation framework are presented for the case of cross-ply laminate. Herein, the influence of load reversals at variable amplitude loading (VA) is investigated. Figure 14(b) shows the simulation results of the damage evolution under VA. The results include comparisons of load sequence investigations of short ($n_b = 5 \cdot 10^3$) with long ($n_b = 5 \cdot 10^4$) load blocks, of tension-compression (TC) with compression-tension (CT) sequences and of VA loading with CA loading (cf. Figure 14(a)).

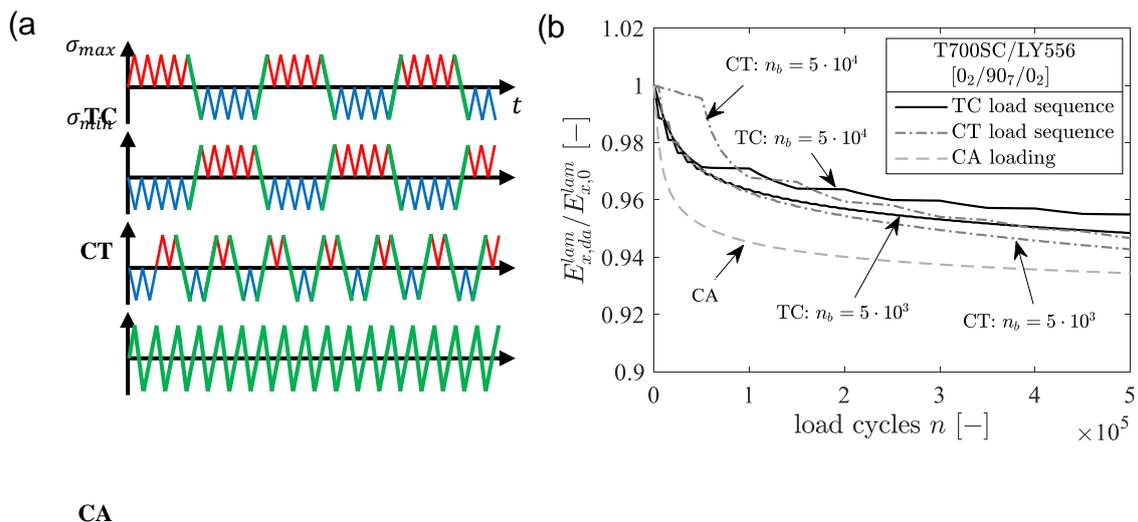


Figure 14: (a) Block loading scenarios with load reversals, (b) Block loading with different load block length and load sequences:

Tension-compression sequence (TC) with $n_b = \{5 \cdot 10^3, 5 \cdot 10^4\}$ and compression-tension sequence (CT) with $n_b = \{5 \cdot 10^3, 5 \cdot 10^4\}$, CA with $R_{glob}^{lam} = -3.26$ and $\sigma_{max}^{lam} = 105$ MPa.

When comparing short with long load blocks, the FDM calculates a marginally higher damage for short load blocks. This effect is visible in both TC and CT sequences. The deviation of the damage state in both cases (TC, CT) is less than one percent after $n = 5 \cdot 10^5$ load cycles. Comparing the damage evolution of TC with CT sequences, the FDM predicts a higher amount of damage with CT sequences. However, the deviation here is similarly low like in case of comparing short and long load blocks. If, on the other hand, the VA loads described are compared with the CA loading, a clear difference can be seen in the respective damage evolution. In the case of CA loading, the damage evolution is significantly higher in consequence of the higher load amplitude and the alternating load occurring over the entire load cycles.

To conclude, it was successfully shown that load sequence effects in the context of load reversals can be reproduced with the extended modelling approach. The various tension-compression load sequences examined showed a very similar damage evolution. In comparison to a constant tension-compression alternating load, the prediction with load sequences results in a significantly accelerated increase in damage.

References

- [1] Krüger, H. Ein physikalisch basiertes Ermüdungsschädigungsmodell zur Degradationsberechnung von Faser-Kunststoff-Verbunden [Ph.D. Thesis]: Leibniz-Universität Hannover. University of Hannover 2012.
- [2] Krüger, H.; Rolfes, R. A physically based fatigue damage model for fibre-reinforced plastics under plane loading. *International Journal of Fatigue* 2015, 70, 241–251.
- [3] Puck, A.; Schürmann, H. Failure analysis of FRP laminates by means of physically based phenomenological models. In *Failure Criteria in Fibre-Reinforced-Polymer Composites*; Elsevier, 2004; pp. 832–876.
- [4] Pfanner, D. Zur Degradation von Stahlbetonbauteilen unter Ermüdungsbeanspruchung; VDI-Verlag, 2003.
- [5] Kawai, M.; Itoh, N. A failure-mode based anisomorphic constant life diagram for a unidirectional carbon/epoxy laminate under off-axis fatigue loading at room temperature. *Journal of composite materials* 2014, 48, 571–592.
- [6] Brod, M.; Just, G.; Dean, A.; Jansen, E.; Koch, I.; Rolfes, R., & Gude, M. Numerical modelling and simulation of fatigue damage in carbon fibre reinforced plastics at different stress ratios. *Thin-Walled Structures* 2019, 139, 219-231.
- [7] Grosse, C.U. *Advances in construction materials*; Springer Science & Business Media, 2007.
- [8] Brod, M.; Just, G.; Jansen, E.; Koch, I.; Rolfes, R.; Gude, M. Simulation of the fatigue damage behavior of carbon composites under consideration of manufacturing induced residual stresses. *Proceedings of the 7th International Conference on the Fatigue of Composites*; 2018; p. 8.
- [9] Koch, I.; Just, G.; Tittmann, K.; Brod, M.; Jansen, E.; Gude, M.; Rolfes, R. Influence of stress ratio and manufacturing induced residual stresses to fatigue cracking of cfrp. *Proceedings of the 7th International Conference on the Fatigue of Composites*; 2018; p. 9.

6 Fatigue and Fracture of Metallic Components

6.1 Endurance Tests of Bearings for Airborne Applications

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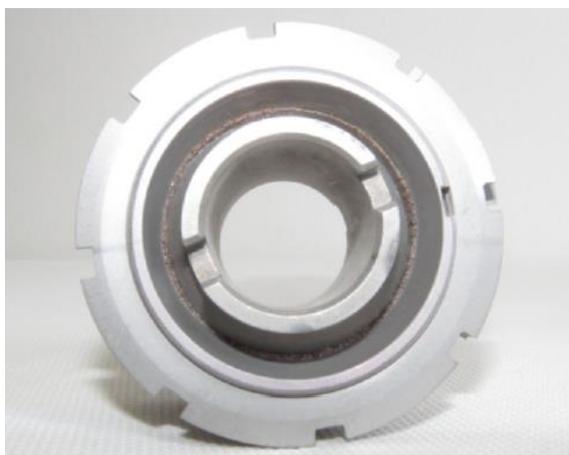
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Ball bearings are installed in various structures in aircrafts. That means, the bearings need to be qualified for airworthiness. This includes, depending on the installation area, several environmental tests, for instance according to RTCA-DO160. But also mechanical properties are assessed, for instance wear behavior and mechanical play. Of particular interest is also a low coefficient of friction within the bearing. This is needed to allow safe operation of the parts that are connected via the bearing. It also enables controlled loading conditions, e.g. it avoids bending in a rod that ideally would see just normal loads. IMA Dresden performed such tests in 2017 and 2018.

Depending on the actual application, the composition of test program differs. Usual components are:

- Endurance test, mainly to determine wear behavior
- Fatigue test
- Environmental tests such as
 - Temperature variation tests
 - Icing tests
 - Fluid susceptibility testing
 - Sand & Dust Testing
 - Salt Spray Test
 - Etc.

The influence of the environmental exposure to the bearing itself, e.g. corrosion effects, and on the performance, e.g. coefficient of friction, are determined during such kind of tests (see Figure 15).



a) bearing after sand test



b) bearing after salt spray test

Figure 15: Examples of bearings after different kinds of tests

6.2 Fatigue crack propagation influenced by laser shock peening introduced residual stress fields for thin aluminum specimens

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Laser Shock Peening (LSP) is a promising surface treatment, which allows the introduction of high compressive residual stresses in metallic components. LSP draws interest from the aerospace industry, where a damage tolerant design permits the crack initiation and propagation within components. High compressive residual stresses, as induced by LSP, are able to retard the fatigue crack growth and, therefore, to increase the fatigue life of the component. However, tensile residual stresses are always generated as well. These tensile residual stresses might accelerate the Fatigue Crack Propagation (FCP), which counteracts the fatigue life enhancement. Thus, the exact knowledge of the induced residual stress field and the fatigue crack retarding or accelerating mechanisms is necessary to apply LSP efficiently. In this work a step-wise simulation [1], predicting the FCP-rate, is utilized to study these fatigue crack retarding or accelerating mechanisms. The step-wise simulation, as shown in Figure 1, includes (i) a LSP-process simulation, (ii) a residual stress transfer between LSP-process model and FCP-model based on the Eigenstrain-method, (iii) calculation of the crack driving stress concentration factors and (iv) FCP-prediction using empirical FCP-laws, in particular the NASGRO-equation. Step (i) and (iii) are based on the finite element method in our work. The different steps are shortly described in the following:

- (i) LSP uses short-time laser pulses to vaporize material of the radiated surface. The vaporized material turns into plasma and expands rapidly. The heat expansion of the plasma causes a pressure at the surface of the solid material, which induces a mechanical shock wave traveling into the material. This shock wave deforms the material plastically. The gradient of the plastic deformations leads to elastic distortions of the material causing the residual stresses after the relaxation of the system. The LSP-process simulation takes the plasma pressure, acting at the material surface, as input and predicts the material response. The plasma pressure is adjusted for one-sided LSP using 5 J and 20 ns laser pulses with a square 3 x 3 mm² laser focus for AA2198-T3 and is applied to AA2198-T8 with a 3 x 3 mm² focus as well as AA2198-T3 and T8 for a 1 x 1 mm² square laser focus. The predicted residual stresses are experimentally validated [2]. In this work, a LSP-process simulation of a two-sided LSP-treatment is performed in AA2024-T3. The number of simulated laser pulse impacts is reduced based on a periodicity assumption of the plastic deformations. Hence, the aim of the LSP-process simulation is the prediction of the plastic deformations in a representative area.
- (ii) Predicted plastic deformations of the LSP-process simulation are transferred to the FCP-simulation using the Eigenstrain-method. Therefore, plastic strains of the LSP-process simulation are applied as thermal strains to the crack propagation model of the C(T)-specimen. The plastic deformations, predicted with the LSP-process simulation, were extended to a larger area based on the periodicity-assumption, already assumed in (i). Resulting residual stresses in the C(T)-specimen were validated with residual stress measurements using the incremental hole drilling method with electronic speckle pattern interferometry of peened specimens.

- (iii) Crack driving stress concentration factors are calculated using a C(T)-specimen model and assuming an interaction of residual and applied stresses. The calculation is based on the virtual crack closure technique, where the energy release rate after the crack extension by a finite length is linked to the stress concentration factor. The stress concentrations at the minimum and maximum applied load used to calculate the crack driving stress concentration factor range ΔK_{cd} and the crack driving stress concentration factor rate R_{cd} . ΔK_{cd} and R_{cd} are identified as main crack driving quantities, by applying ΔK_{cd} and R_{cd} to an unpeened specimen using specific external loadings. The FCP-rate of the unpeened specimen in this test set-up is in a good agreement with the experimentally measured FCP-rate of a peened specimen.
- (iv) The NASGRO-equation is used to predict the FCP-rate. While ΔK_{cd} and R_{cd} are the inputs of the NASGRO-equation, material parameters were identified by experiments of unpeened material.

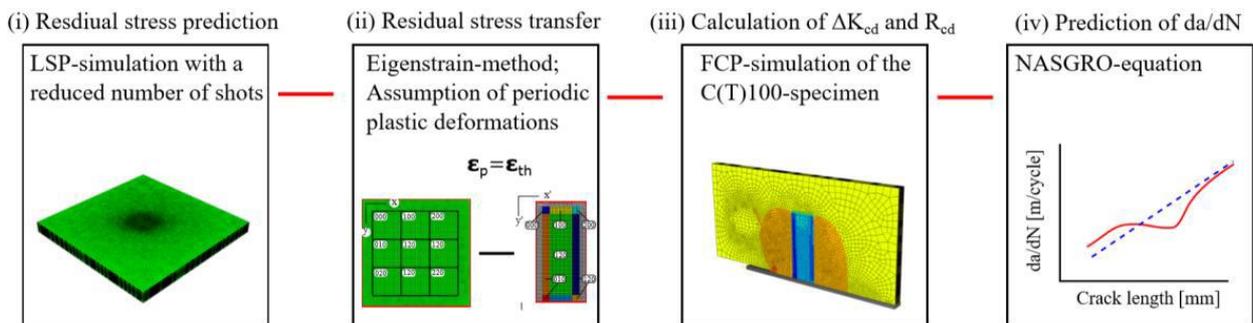


Figure 16: Multi-step simulation approach. The multi-step simulation takes the plasma-pressure as input and predicts the plastic deformations after the LSP-treatment (i). These plastic deformations are transferred to FCP-simulation based on the Eigenstrain-method (ii). The FCP-simulation calculates the crack driving stress concentration factors (iii), which are used as input for the NASGRO equation (iv) [1].

The experimentally validated step-wise simulation to predict the FCP-rate based on the LSP-induced surface pressure is used to study the crack retarding and accelerating mechanisms [3]. One of the most important mechanisms is crack closure, which occurs, if the external load is not able to open the crack completely due to compressive residual stresses. It is found, that the location of crack closure depends strongly on the position of the compressive residual stresses and does not have to be linked to the crack tip. This crack closure is interpreted as a change in geometry of the cracked C(T)-specimen, which strongly changes the stress distribution. Crack closure is able to reduce ΔK_{cd} which reduces the FCP-rate. The simulations and experiments showed that the fatigue crack retardation is dramatically reduced if no crack closure occurs. Furthermore, it is shown, that tensile residual stresses in front of the peened area are accelerating the fatigue crack while tensile residual stresses behind the fatigue crack do not lead to an acceleration of the crack propagation. Consequently, LSP should be applied as close as possible to areas where fatigue crack can be initiated, e.g. close to a weld.

References:

[1] S. Keller, M. Horstmann, N. Kashaev, B. Klusemann, “Experimentally validated multi-step simulation strategy to predict the fatigue crack propagation rate in residual stress fields after laser shock peening”, submitted (2018)

- [2] S. Keller, S. Chupakhin, P. Staron, E. Maawad, N. Kashaev, B. Klusemann, "Experimental and numerical investigation of residual stresses in laser shock peened AA2198", *Journal of Materials Processing Technology*. vol. 255, pp. 294-307 (2018)
- [3] S. Keller, M. Horstmann, V. Ventzke, N. Kashaev, B. Klusemann, "Experimental and numerical investigation of crack closure in residual stress fields generated by laser shock peening", submitted (2018)

6.3 Is the Civil Aerospace Industry ready to implement Laser Shock Peening into Maintenance Environment? Questions to be answered and minimum requirements from Aircraft Manufacturer's Perspective

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Laser Shock Peening (LSP) as surface technologies inducing residual stresses in metallic airframe is a well-established technology in aerospace industry [1], [2]. An overview of potential applications to ensure salvage for identified hot spots in terms of fatigue and crack growth performance in aircraft structures as well as to ensure the improvement of the economic and ecological impact of future aircraft structures by controlling the residual stresses has been reviewed in [3].

An example of LSP for fatigue critical components of military aircraft in maintenance environment in fixed established repair station is reported in [4], [5]. The application of LSP as retrofit solution for in service commercial aircraft is particular challenging and currently no applications are reported. Applying LSP as a structural modification in critical component of in service commercial aircraft implies treatment at the Maintenance Repair and Operations (MRO) all around the world during already scheduled maintenance to avoid Aircraft on Ground situation which can cost tens of thousands dollars a day. When a commercial aircraft is in a major overhaul maintenance, inspection programs, eventually repairs and structural modifications take place in a limited short time involving multi-skilled teams working at the same time around the aircraft making the accessibility of the structure not easy at all. Moreover it is a common understanding that the depth of compressive residual stress over 1 mm can be achieved only if high energy laser (i.e. large laser spot) is used [6]. The compressive shock wave generated by LSP has a lower attenuation travelling into the material if more planar shape as per the larger spot; while smaller spots are characterized by spherical shock waves with higher attenuation rate and shallower compressive stress.

LSP by high pulsed energy system and involving complex set up and tooling is not practical at all for in-service use (not compatible with airline maintenance constrains). To make LSP applicable at MRO and ensure reasonably simple setup and easy transportability to all around the world requires to develop a "portable" device (i.e. low energy laser). Fiber delivery system, in contrast to heavy, "rigid" and complex mirror systems, provide advantages in terms of accessibility of complex structures. On the other hand the limitation of the current capability to deliver short pulsed laser beam into silica core fiber is well known [7], [8]; 100mJ is the maximum energy possible to inject into silica core fiber (using 532nm laser with pulse width of 8nsec and fiber diameter smaller than 1mm). But this low

energy system that represent a solution for compact, light LSP device is said to be not capable to induce deep compressive stress.

The authors will demonstrate in this paper that low energy LSP system ($\leq 200\text{mJ}$, pulse width of $\leq 20\text{nsec}$) and associated small laser spot size (e.g. 0.7mm diameter) can determine high compressive stress in the near surface of 7175-T7351 material and compression depth above 1mm (Figure 17) if peening coverage is high enough. This residual stress profile is sufficient to extend the fatigue lives of critical components opening the door for development of portable LSP devices requiring low energy laser to be applied at MROs. The compressive stresses with a low energy LSP system is performed by scanning the surface to be treated with high percentage overlap (e.g. $> 60\%$ of the spot size) and advancing the line with similar overlap in the so called stepping direction. The residual stresses at near surface is influenced by the peening strategy, resulting in higher compressive residual stress component in the stepping direction compared to the scanning direction. This effect, only typical of low energy system and small spot size, tends to vanish with increased depth into the material. The investigations showed that even with at lower energy system (such as 532nm laser and 70mJ per pulse and shorter pulse width of 8nsec) the compressive residual stress can reach a depth in excess of 1mm into the material as shown in Figure 17. LSP with fiber delivery system provides interesting benefit in terms of accessibility to complex geometries and structures in particular in MRO environment, but the focus distance to the target when fibers are used is very limited (e.g. 20mm) becoming a challenge for treatment of complex 3-D shapes. The tolerances in positioning the focusing lens to the target application are very tight (e.g. $\pm 0.5\text{mm}$) resulting in possible change of laser spot size and shape at the target. It is important to quantify the range of variation of the spot size to determine the possible drop of compressive residual stresses because of reduction of laser irradiance (in GW/cm^2) in larger spots. Figure 17 shows the drop of compressive residual stress at near surface if the spot diameter increases from 0.6mm up to 1mm .

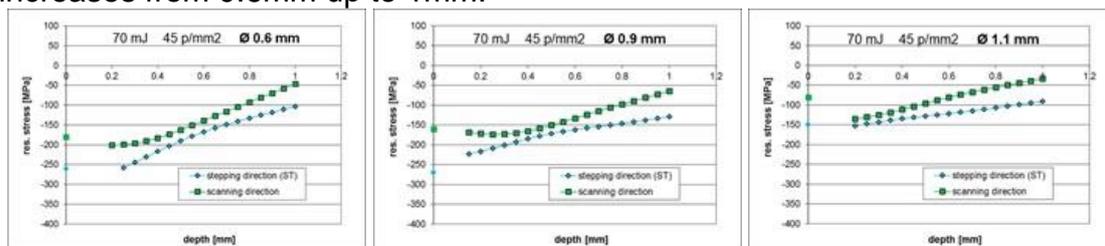


Figure 17: Residual Stress Profiles after LSP in 7175-T7351 material using 532 nm laser at 0.8 nsec pulse width and 45 pulses/mm² at increased spot size.

The capability to extend the fatigue lives of aircraft component subjected to LSP provided by low energy system have been verified testing under fatigue cycling load four point bending coupon (Figure 18). Figure 18 a) shows the fatigue lives up to final failure of four points bending coupons made of 7175-T351. Fatigue performance of bare material (no treatment, reference) has been compared with the one after shot peening process and low energy LSP treatment demonstrating a superior fatigue lives at all surface stress levels.

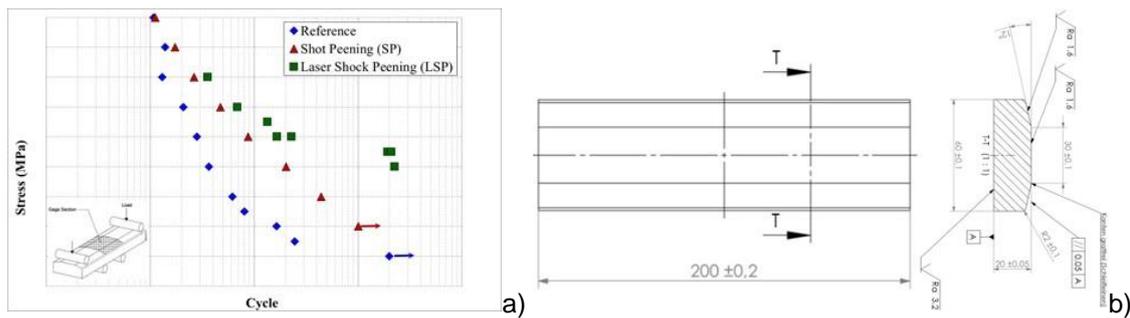


Figure 18: Fatigue test results with stress ratio ($\sigma_{min}/\sigma_{max}$) of 0.1 of four point coupons of 7175-T7351.

Finally, the authors review the minimum requirements of LSP portable device to ensure the compatibility with typical MRO operational environments. Robustness, repeatability and stability of the LSP process are key features to be considered during the development of portable system. In service, the system will operate all around the world in various environments showing the need for supply chain reactivity. The system should be “easy to handle”, providing appropriate training to Third Parties to operate and maintain it.

- [1] D. Furfari, “Laser Shock Peening to Repair, Design and Manufacture Current and Future Aircraft Structures by Residual Stress Engineering”, Advanced Materials Research Vols. 891-892 (2014) pp 992-1000, © (2014) Trans Tech Publications, Switzerland.
- [2] E. Hombergmeier, D. Furfari, N. Ohrloff, U. C. Heckenberger, V. Holzinger, “Enhanced Fatigue and Damage Tolerance of Aircraft Components by Introduction of Residual Stresses – A Comparison of Different Processes”, 27th ICAF Symposium – Jerusalem, 5 – 7 June 2013.
- [3] D. Furfari, N. Ohrloff, E. Hombergmeier, U. C. Heckenberger, V. Holzinger, “Laser Shock Peening as Surface Technology to extend Fatigue Life in Metallic Airframe Structures”, 35th ICAF Conference, Nagoya, Japan, 7-9 June 2017.
- [4] Lt. Ken MacGillivray, B. Dane, M. Osborne, R. Bair, W. Garcia, “F-22 Laser Shock Peening Depot Transition and Risk Reduction”, USAF ASIP Conf., 30/11–02/12, S. Antonio, TX, 2010.
- [5] LeAnn Polin, J. Bunch, P. Caruso, J. McClure, “Full Scale Component Tests to Validate the Effects of Laser Shock Peening”, ASIP 2011, Nov. 29-Dec. 1, San Antonio, TX.
- [6] A. K. Gujba, M. Medraj, “Laser Peening Process and Its Impact on Materials Properties in Comparison with Shot Peening and Ultrasonic Impact Peening”, Materials 2014, 7, 7925-7974.
- [7] Y. Sano, K. Akita, K. Masaki, Y. Ochi, I. Altenberger, B. Scholtes, “Laser Peening without Coating as a Surface Enhancement Technology”, JLMN-Journal of Laser Micro/Nanoengineering Vol. 1, No. 3, 2006.
- [8] T. Uehara, Y. Sano, I. Chida, M. Yoda, N. Mukai, H. Kato, “Laser Peening Systems for Preventive Maintenance against Stress Corrosion Cracking in Nuclear Power Reactors”, Proceedings of the 16th International Conference on Nuclear Engineering, ICON16, May 11-15, 2008, Orlando, Florida, USA.

6.4 Fatigue in Additive Manufactured aircraft: the long way to make it fly

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Manufacturing of metallic products is currently facing a historic revolution driven by a group of innovative technologies clustered under the name of Additive Manufacturing (AM) or, more colloquially, 3D-printing. As opposed to the conventional subtractive manufacturing methodologies, AM is able to create net-shaped products by means of the addition, layer by layer, of the needed material. The biggest benefit of the AM in respect

to the conventional manufacturing technologies is the substantial freedom the technology gives to the product designers in frame of the product development. Due to the relaxed geometrical constraints, complex parts characterized by optimized shapes can be realized (Figure 19). This can be translated in development of lighter and more cost-efficient parts and assemblies hitting two of the most important targets of the aeronautical industry.



Figure 19: Secondary Structure (Bracket) developed by Premium Aerotec GmbH. Conventional design (milling plus riveting) compared with topologically optimized design developed for the Additive Manufacturing (53% weight saving, 40% cost saving).

Despite the high investments, the AM still finds a limited applicability for the manufacturing of primary aircraft components (Table 3). This is also due the challenges relative to the fulfilment of the Fatigue and Damage Tolerance (F&DT) requirements, prescribed by the airworthiness authorities.

Table 3: Examples of primary structures as reported in the FAA Advisory Circular AC 25.571-1D of interest for the applicability of AM.

Wing and empennage	Control surfaces, slats, flaps, and their mechanical systems and attachments (hinges, tracks, and fittings)
	Integrally stiffened plates
	Primary fittings
Fuselage	Circumferential frames
	Door latches
	Window frames
Landing gear	Landing gear and their attachments

Engine mounts.	Engine mounts and their attachments
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The main steps of the AM process currently in-place for the production of aircraft secondary structures will be described and the limitations of this process in controlling the aforementioned characteristics of the produced AM parts will be then highlighted.

As a consequence of these limitations, not optimized secondary structures designed with extremely high safety factors to cover the “certainties of the uncertainties” are produced. Furthermore, constrained by strict F&DT requirements, the production and installation of aircraft primary structures using AM is currently not feasible.

In line with currently applicable Airworthiness regulations, a way-forward in order to overcome the current limitations of wider use of AM will be proposed, made of:

- Investigations aimed to get more complete understanding of the process, structure, roughness, F&DT performances relations;
- Development of AM products in a context of structural integrity.

According to authors’ perspective, following this strategy will support the development of more weight and cost effective AM secondary structures and start to move towards the vision of the development of AM primary structures.

6.5 A modeling approach for the fatigue behavior of laser drilled micro perforated structural panels

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Hybrid laminar flow control (HLFC) is a technology approach to reduce the drag of civil jet transport aircrafts. The aim of HLFC is to create a longer laminar flow along the airfoil on an aircraft. This is done by boundary layer suction through small holes in the skin panels. The holes are small (< 100 µm) and there are millions of them per square meter. The impact of these holes on the structural capability of the aircraft design is significant. The objective of this paper is to describe the best way how to model the influence of these micro drilled holes on the properties of the sheet material as an input for fatigue life calculations.

The main application of HLFC is on leading edges, e.g. wing or empennage. The extension over the chord of the micro drilling depends on the available design space for this feature and the aerodynamically efficiency of the solution. Underneath the micro drilled panels (see Figure 20) there is a controlled pressure level in order to get the desired suction of the external boundary layer. Via a smart design of this pressure chamber the structural function of the outer skin can be reduced from a primary structural element to a secondary level. Therefore, in this abstract the focus is on the fatigue properties and crack initiation of the design and not on the fracture mechanics of macro cracks.

The production process of micro drilled panels consists of several steps. An example of a description of the process can be found in [1]. To start a material sheet of desired thickness is selected, together with a hole diameter and pitch. For the laser drilling itself many parameters can be set, for example pulse energy level and amount of pulses per hole. With these parameters the geometry and quality of the hole will be defined, influencing: The entrance hole diameter, the exit hole diameter, the shape of the cone, the inner hole surface, the hole edges, the burrs. After the drilling process burrs will remain on the hole edges and some drops might have hit or stuck on the outer surface of the plate. To remove these burrs de-burring is necessary. De-burring can be done chemically or mechanically depending on the material used. This step in the production will alter the surface of the sheet.

Whereas the impact on static properties for the sheet is mainly determined by the remaining net section, the impact on fatigue properties will be much more complicated. The fatigue properties depend on the stress concentrations caused by the hole geometry. Moreover as fatigue is a material surface driven phenomenon [3] many of the production parameters influence the surface finish and thus the fatigue behavior of the material. Typically there are: the inner-hole surface quality, localized altered heat treatment around the holes, polishing or etching of the sheet surface. Therefore it is important to determine their influence on the fatigue life of the created parts.

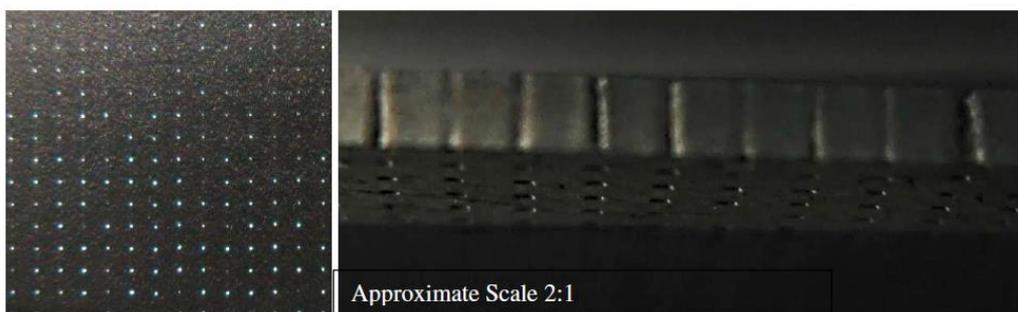


Figure 20: Top view and cross section of a laser micro drilled plate

The durability of the design is another important factor. As the HLFC is on the leading edge, impacts damage and erosion might occur. Impact resistance can be determined by the overall design; on the other hand the erosion resistance is a material property of the micro drilled sheet. This is essential as the holes need to remain open, so protective paints or coatings are not an option. The micro holes should not change significantly in size, otherwise the efficiency of the HLFC system will reduce. For designs with undrilled sheets, typically metals are better against erosion than composites. For micro drilled panels a comparison study was done [2]. In this study Titanium, Aluminium and CFRP were compared via a flight test performed with a SAAB 2000 aircraft. Several laser drilled panels were fitted on the wing leading edge. Over the time of 20 months of routine service these panels were monitored with regards to contamination and durability and the following results were obtained. Titanium is the most durable material with no evidence of degradation, the mean hole size appeared to be unchanged. The Aluminium panels displayed evidence of corrosion and micro-cracking. The hole size was significantly increased on

the outer surface. The composite material, flown only two months, had significant damages. Noticeable amounts of fibre were removed on the surface between holes. Based on the outcome of this study it was decided to perform the fatigue tests on titanium micro drilled sheets. For the fatigue tests, within the frame of the EU Clean Sky2 Project DELASTI [5], a test matrix, was setup to create a few full data sets to create SN curves for titanium CP (grade 2). Four SN curves were generated: two drilling patterns were selected ($0^\circ/-90^\circ$ and $-45^\circ/+45^\circ$) and tested in the two major material directions L and LT. The pitch and diameter was the same for all drilled specimens. The specimens have all been tested and the assessment of the test results is in progress. A comparison of the initial test results is shown in Figure 21. The slope of the four SN curves is similar for all for test variants. A first observation showed no significant difference between the 0° and 45° micro drilling pattern. This indicates that the distance between the holes is so big that there is no significant interaction between the separate holes for the fatigue initiation. This is supported by literature studies comparing pitch vs. diameter [4]. The rolling direction of the material resulted in a significant gap in the SN curves. With the loading in the LT direction giving somewhat better results. This offset is also seen in the undrilled sheets and not expected to be caused by the micro drilling. In a next step the test results are compared to SN curves of the base material to see if a stress concentration factor is sufficient to model these micro drilled holes. Also a factor to account for the high amount of holes is introduced. With this baseline some historical test results will be compared with the newly obtained data. Some further testing is planned.

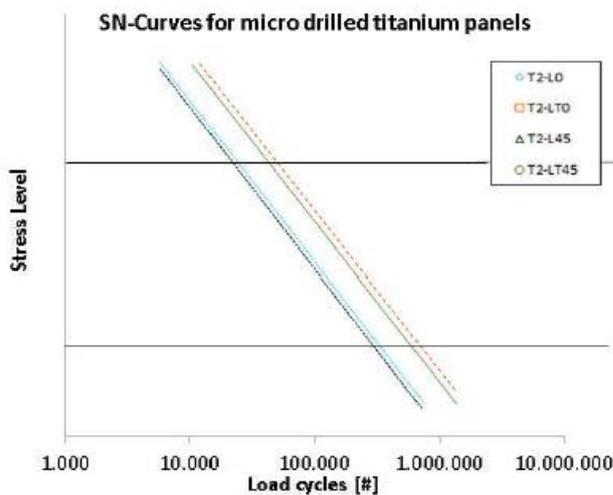


Figure 21: Initial test results from the test matrix.

References:

- Ref 1: High speed laser micro drilling for aerospace applications, A. Stephen, G. Schrauf, et al., New-Production Technologies in Aerospace Industry - 5th Machining Innovations Conference (2014)
- Ref 2: Durability of hybrid laminar flow control (HLFC) surfaces; T. Young, B. Mahony, et al.; Aerospace Science and Technology 7 (2003)
- Ref 3: Fatigue of Structures and Materials, J. Schijve, issue 2 (2009)
- Ref 4: Peterson’s Stress concentration Factors, W. Pilkey and D. Pilkey, issue 3 (2008)
- Ref 5 EU Clean Sky 2 Project “Development of advanced laser based technologies for the manufacturing of titanium HLFC structures/DELASTI” (grant agreement no: 687088).

7 Fatigue and Fracture of Composite Components

7.1 Fatigue and Damage Tolerance Test of a CFRP Part under Large Bending

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During a research project a spoiler was designed with a “flexible hinge” connection to the fixed wing box. Goal of the flex hinge was to connect the spoiler seamlessly to the wing box. The spoiler design was a combination of monolithic CFRP and Composite sandwich. Strain allowables are available for the base material to predict the damage tolerance behavior of the monolithic laminate. However, many strain allowables are obtained from tensile test. In this case the flex hinge is loaded mainly in bending, therefore it was decided to perform a set of small coupon tests under, for the flex hinge line typical, bending. This summary is based on the following reference: “Non-discrete spoiler with an adaptive shock control bump, ICAST2016: 27nd International Conference on Adaptive Structures and Technologies October 3-5th, 2016, Lake George, New York, USA J. Kirn, W. Machunze, M. Weber, F. Strachauer, Airbus Group Innovations (Airbus Defence and Space GmbH), Germany”.

A special test-bench (Figure 22) was designed to generate the typical bending curve and in which it was possible to test flexible hinge test specimens with repetitive loading. Two types of samples with different laminate lay-ups were manufactured using flight certified CFRP material. One was a standard quasi-isotropic, the other a triax lay-up. For the test two phases were defined: First a fatigue phase for the intact structure followed by a damage tolerance phase which includes impacts on the specimen. In this test campaign, the selected impacts were chosen such to simulate hail on ground at various energy levels. Each phase consists of minimum 20,000 cycles up to 100,000. Which is not necessarily covering a full aircraft life, but rather fulfils the specific requirements inside the research project. One cycle is a complete deflection from retracted to fully deployed (air break out) and back. Therefore the system is displacement controlled rather than force controlled.



Figure 22: Test bench and schematic principle of bending test

After the first test phase each sample was subject to an ultrasound inspection to see if any damage had occurred. In the second test phase the damage of the “hail” impact was assessed before and after the cyclic testing. It could be seen, that for all but one specimen the damage did not increase during the cycling. Only on sample FH2-3 with a 20 J impact a growth in damage was detected. This was the specimen with the highest energy level and the largest detectable damage.

Concluding it can be said, that a new test setup was successfully build to test the repeated bending of the specimens. The initial design for the flexible hinge is robust and damage tolerant without considering environmental conditions.