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Review of aeronautical fatigue investigations in France during the period May 2015- April 2017

TECHNICAL NOTE

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	NAME	Visa	Date	Title
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Abstract :

The present review, prepared for the purpose of the 35th ICAF conference to be held in Nagoya (Japan), on 5-6 June 2017, summarises works performed in France in the field of aeronautical fatigue, over the period May 2015-April 2017.

Topics are arranged from basic investigations up to full-scale fatigue tests.

References, when available, are mentioned at the end of each topic.

Correspondents who helped to collect the information needed for this review in their own organisations are :

- Bertrand Journet for Airbus Group Innovations
- Linden Harris and Peter Boesch for Airbus
- Vincent Bonnand for ONERA
- Etienne Deshaies for DGA Techniques aéronautiques.

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1. INTRODUCTION AND ACKNOWLEDGMENT

The present review, prepared for the purpose of the 35th ICAF conference to be held in Nagoya (Japan), on 5-6 June 2017, summarises works performed in France in the field of aeronautical fatigue and structure integrity, over the period May 2015-April 2017.

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- Etienne Deshaies for DGA Techniques aéronautiques.

They will be the right point of contact for any further information on the presented topics.

Many thanks to all of them for their contribution.

2. FATIGUE LIFE PREDICTION STUDIES AND FRACTURE MECHANICS

2.1. 3D CRACK PROPAGATION TESTING AND MODELING IN THICK ALUMINUM 2024 T351 (DGA TECHNIQUES AERONAUTIQUES)

DGA is continuously enhancing its capability to provide through-life support with the Air Forces for aircraft life extension and life management. Ageing fleet concerns and application of damage tolerance requires studying the propagation of elliptical cracks in thick structures. Those cracks have no direction of invariance and are 3D cracks. Lack of knowledge on those shapes causes usage of overly conservative propagation laws, and therefore to non-optimised inspection programs which increase the maintenance costs. Most of the times, as soon as the geometry run out the classical library, cracks are modelled by “through cracks” which is highly conservative, and critical crack size is underestimated due to a lack of precise residual strength criteria when the stress intensity factor is not uniform all along the crack front.

Two of the main needs to assess a damage tolerance approach, particularly for massive components such as spars, gear struts or thick lugs, are to develop a capability in 3D crack growth modelling and residual strength determination. Assessment of residual life based on crack growth is essential in the life management of critical components and require a high level of confidence. 3D cracking prediction is thus a major challenge to enhance the life of an ageing fleet at moderate costs.

There are various issues involved in 3D crack growth analyses. In general, the difficulties associated with 3D crack growth analysis in order to assess a damage tolerance approach are the following ones:

- Tests issues to follow the crack in the core of the material in order to calibrate and assess the good correlation between 3D crack growth computation and reality,

- Modeling issues concerning accurate calculation of stress intensity factors along a 3D crack front embedded in a component, especially at the edges of the structure.

- Determination of the crack growth rate in 3D space, with the choice of an accurate crack growth law.

- Predictions of the crack shape evolution during the propagation; the global behaviour cannot be determined from a simple point by point analysis.

- Use of an appropriate criterion to assess the residual stress of the cracked part.

The approach chosen for this project was a step by step method increasing the difficulties in terms of test follow-up and crack propagation modelling methodology.

First, we considered standard coupons (CT, CCT), which enables to obtain crack propagation laws. Different stress ratios were investigated, to be able to work with complex spectrum later on. It enables also to calibrate marking method and assess they have no influence on propagation.

Then technological coupons (notched corner crack bar, Figure 1) were used to put in place the different modeling strategy and tools. Results were carefully analysed in terms of crack propagation rate and shape prediction. Constant amplitude loading, at different stress levels but also complex spectrum were applied on the coupons. The influence of the initial shape was also investigated. All the results enable to select the most effective modelling strategy.

Finally a test was conducted on a landing gear part with a semi elliptical crack (Figure 2). To be able to follow the evolution of the crack front during the propagation test, ink injections and marking cycles have been used. A simplified but complex spectrum was applied, and inspections were put in place, as the part would have been analysed by damage tolerance analysis.

On residual strength, the transition between stable propagation and rupture were analysed carefully on different coupons, to compare different approach (from local to global) to estimate the residual strength. Recommendations were made from that.

The main objective was to investigate if the modelling predictions versus tests results are accurate and to quantify the delta in terms of crack shape and propagation time prediction in order to assess the reliability of each evaluated modeling method. Different modeling methods have been tested along the study with the Samcef finite element code from classic finite element method with Barsoum elements to Extended finite element method (XFEM). All of these modeling methods have been compared and the best compromise identified in terms of time consumed to model preparation/computations and accuracy of results is XFEM method.

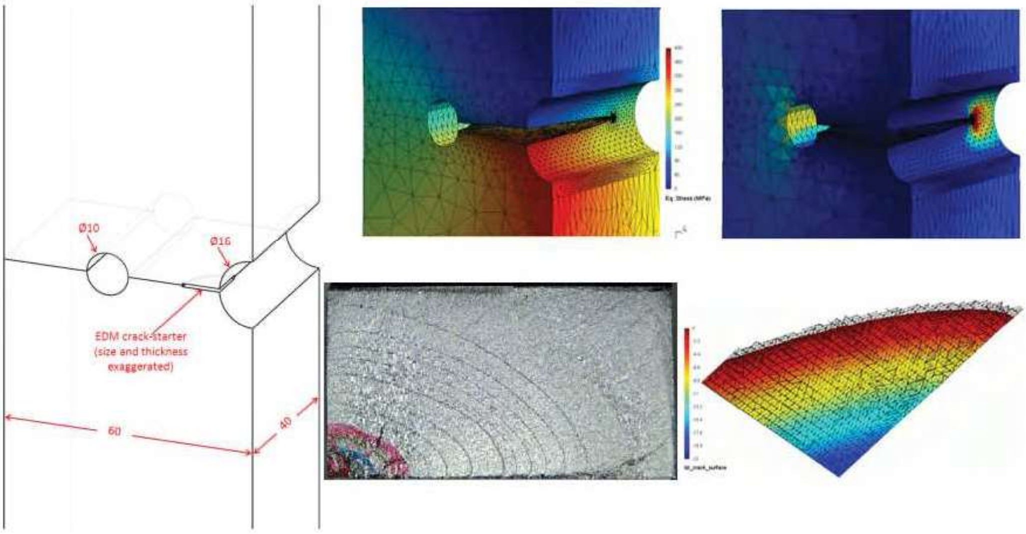

Figure 1: Notched corner crack bar: cracked surface of tested coupon and modeling

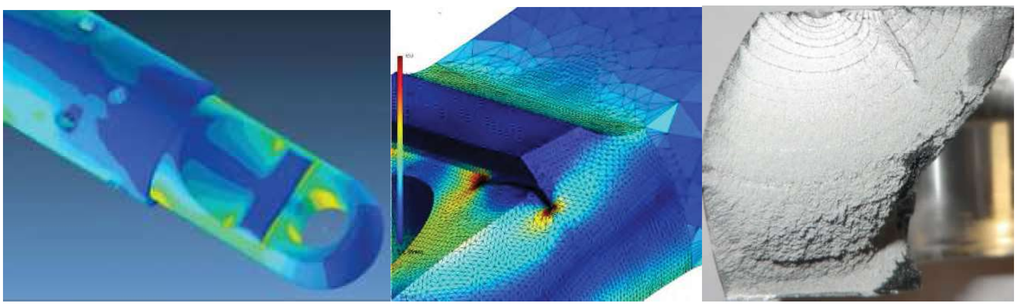

Figure 2: Landing gear part: cracked surface of tested coupon and XFEM modelling

2.2. FATIGUE CRACK GROWTH UNDER SPECTRUM LOADING AT TEMPERATURE AND AGEING EFFECT (AIRBUS GROUP INNOVATIONS)

A martensitic stainless steel 15-5PH has been evaluated in damage tolerance for pylon application within the frame of French public funded project PREVISIA(*). Since the application deals with long exposures (more than 10000 hours) to temperatures of 300 up to 400°C, the steel undergoes ageing through spinodal decomposition of the Fe-Cr matrix. This phase transformation bears some consequences on the fatigue crack growth properties. Fatigue crack growth rate equations have been formulated and validated to take into account the effect of temperature and ageing. These formulations have been coupled to PREFFAS model in order to predict fatigue crack growth under flight spectrum loading with the effect of temperature and ageing. Damage tolerance scenarios calculating the fatigue crack growth of a structural detail under pylon flight spectrum and taking into account the accumulated ageing during the flights at temperatures between 300 and 400°C have been computed. The results point out the effect of the flights temperature sequence order and the conservativeness of some scenarios which do not update the state of ageing of the alloy during the flight in their calculations. This type of computations may help manage a fleet of aircraft with respect to ageing and damage tolerance.

(*) the funding of the French National Research Agency is acknowledged. Partners in PREVISIA are: Airbus Group Innovations (coordinator), Airbus Commercial Airplanes, SIMaP Materials and Processing Science and Engineering department of the National Polytechnic Institute of Grenoble, CIRIMAT Materials Research and Engineering department of the National Polytechnic Institute of Toulouse, PPRIME Materials Physics and Mechanics department of ENSMA, LMT Mechanics and Technology laboratory of Ecole Normale Supérieure of Cachan, Aubert & Duval.

--- <u>Reminder on PREFFAS model</u>

Airbus Group Innovations has developed in the past a model to predict fatigue crack growth under spectrum loading, namely PREFFAS model, in the frame of linear elastic fracture mechanics. Effort to extend the model with new options has been continuous throughout the years. This model is used by Airbus Commercial Airplanes to determine the maintenance inspection intervals. This model was first developed in 1985 for cases of tensile loading on aluminum plates. It was further implemented on thin aluminum sheets, steels and titanium plates. Later on improvements dealt with the occurrence of compressive loading on aluminum plates. The model has been successfully evaluated on numerous types of flight spectra and related aircraft materials: inner wing, outer wing, pylon, fuselage of the Airbus aircraft family. Recent evaluations of the model have been dealing with the presence of residual stresses due to welding and with the exposure to temperature and ageing effects during fatigue cycling. This is the purpose of the paper to report on the second recent investigation dealing with temperature and ageing effect.

--- <u>Results details</u>

Fatigue crack growth test were run on aged and non-aged 15-5PH steel, at room temperature and at high temperatures (300-400°C). Ageing treatments were made at temperatures between 300 and 400°C for durations up to 15000 hours.

The testing results show that there is no influence of ageing on the fatigue growth rates at room temperature. At temperatures of 300-400°C, with 2 Hz cycling, the retardation effects are almost non existing, the fatigue crack growth rates increase with temperature and ageing severity.

The fatigue crack growth rates have been modelled with temperature and ageing treatment (use of results obtained by the academic partners at the metallurgical level for the spinodal decomposition). This formulation is used by PREFFAS model to make the predictions. This adapted version of PREFFAS has been validated on tests run under pylon spectrum, on aged/non-aged steel, at room temperature and high temperatures (300-400°C).

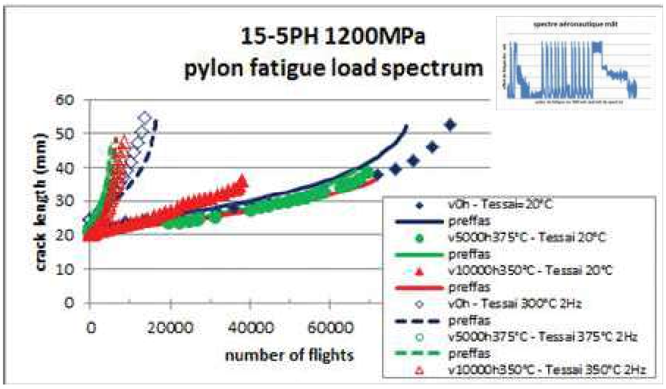

Figure 1 : validation of adapted PREFFAS model under pylon spectrum loading with temperature and ageing effect

Some prospective damage tolerance scenarios were computed with the adapted PREFFAS considering series of flights in which the temperature seen by the part can vary from flight to flight. Since spinodal decomposition is additive, it is then possible with the above modelling to estimate the cumulated ageing during the series of flights so that at each flight, the state of ageing can be known, and with the temperature of the concerned flight, the fatigue crack growth rates can then be estimated.

Damage tolerance scenarios for a service of 15,000 flights (15,000 flight hours) were calculated for a representative structural detail of a pylon panel. The damage scenario is that of an initial corner crack at a hole of a panel which is going to grow under the service fatigue loads and under the high service temperature while the material undergoes ageing.

Two types of scheme are computed:

a) Scheme where the material properties are fixed during the 15,000 flights (very conservative or unconservative).

b) Scheme where the material properties evolve with the ageing taking place during the 15,000 flights, ageing evolves with the flights (more realistic scheme).

The following 5 scenarios among the two types were calculated and plotted in the figure 2:

1. **flights at 375°C – aged15000h/375°C**

Scheme (a), very conservative scenario, all the flights are supposed to take place at a temperature of 375°C and the material property taken for the calculation over the 15000 flights is that of the material aged 15000h at 375°C. Failure takes place before completion of the 15000 flights.

2. **flights at 375°C – ageing with flights**

Scheme (b), less conservative scenario, all the flights are supposed to take place at a temperature of 375°C but the material property is updated every 1000 hours with regard to the cumulated ageing situation. Failure takes place before completion of the 15000 flights.

For the first 1000 flights, the material property taken for the calculation is that of the material aged 1000h at 375°C.

For the next 1000 flights, the material property taken for the calculation is that of the material aged 2000h at 375°C.

For the next 1000 flights, the material property taken for the calculation is that of the material aged 3000h at 375°C.

And so on… until the last 1000 flights where the material property is that of the material aged 15000h at 375°C.

Note that the calculation is a little bit conservative since the material is already supposed to be aged at the beginning. Improvement would be to update at each flight. It is thought that it would not make a big difference. The blocks of 1000 flights are small enough.

3. **flights by blocks successively at 375-350-320-300°C – ageing with flights**

Scheme (b), this is a less conservative scenario and a more realistic one. The flights are supposed to take place at different temperatures, first 5% of flights at 375°C, then 15% at 350°C, 30% at 320°C and the last 50% at 300°C. The update of the material properties with the cumulated ageing situation is made after each change of temperature.

For the first 750 flights (5% of 15000) at 375°C the material property taken for the calculation is that of the material aged 750h at 375°C.

For the next 2250 flights (15% of 15000) at 350°C, the material property taken for the calculation is that of the material aged 750h at 375°C + 2250h at 350°C.

For the next 4500 flights (30% of 15000) at 320°C, the material property taken for the calculation is that of the material aged 750h at 375°C + 2250h at 350°C + 4500h at 320°C.

For the last 7500 flights (50% of 15000) at 300°C, the material property taken for the calculation is that of the material aged 750h at 375°C + 2250h at 350°C + 4500h at 320°C + 7500h at 300°C.

Note that the calculation is a little bit conservative since the material is already supposed to be aged at the beginning. Improvement would be to update at each flight starting with the non-aged state.

4. **flights by blocks successively at 300-320-350-375°C – ageing with flights**

Scheme (b), same as case 3 but the sequence of temperature is reversed. The flights are supposed to take place at different temperatures, first 50% of flights at 300°C, then 30% at 320°C, 15% at 350°C and the last 5% at 375°C. The update of the material properties with the cumulated ageing situation is made after each change of temperature.

For the first 7500 flights (50% of 15000) at 300°C the material property taken for the calculation is that of the material aged 7500h at 300°C.

For the next 4500 flights (30% of 15000) at 320°C, the material property taken for the calculation is that of the material aged 7500h at 300°C + 4500h at 320°C.

For the next 2250 flights (15% of 15000) at 350°C, the material property taken for the calculation is that of the material aged 7500h at 300°C + 4500h at 320°C + 2250h at 350°C.

For the last 750 flights (5% of 15000) at 375°C, the material property taken for the calculation is that of the material aged 7500h at 300°C + 4500h at 320°C + 2250h at 350°C + 750h at 375°C.

Note that the calculation is a little bit conservative since the material is already supposed to be aged at the beginning. Improvement would be to update at each flight starting with the non-aged state.

5. **Scheme (a), flights at 300°C – aged15000h/300°C**

Very mild scenario, all the flights are supposed to take place at a temperature of 300°C and the material properties are those of the material aged 15000h at 300°C.

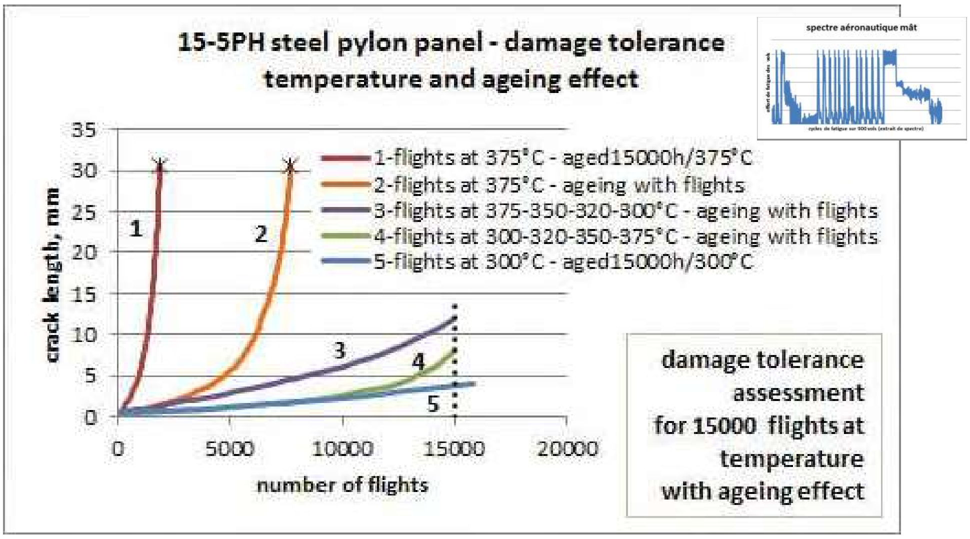

Figure 2 : comparison of the 5 scenarios

The figure shows that:

- Scenario 1 is very conservative. There is a need to take into account the microstructure evolution of the material during the High T flights.

- The damage tolerance capability is dramatically reduced with very high temperature flights and very long term ageing exposure (scenario 2).

- High temperature ageing reduces damage tolerance capability when it happens at the beginning of service (5% at 375°C makes the difference, scenario 3) or at the end of service (with 5% of the flights at 375°C, scenario 4).

- Scenario 5 is non conservative and unrealistic.

If the ageing of a fleet of aircrafts can be monitored, this type of calculation may help manage the ageing of the fleet in order to keep a good damage tolerance capability for the targeted commercial service duration.

2.3. IMPACT OF SHOT-PEENING ON THE FATIGUE LIFE OF A SINGLE CRYSTAL TURBINE BLADE (ONERA)

The fatigue analysis of blades in the hot stage of aeronautical gas turbines is extensively investigated to improve the engine reliability. Fatigue life assessment of these components submitted to fatigue thermomechanical loadings represents an important goal in the context of increased competition between engine manufacturers. Shot-peening is widely used on roots of high pressure turbine blade to postpone the crack initiation in stress concentration area. This pre-stressing introduces compressive residual stress and work hardening in a surface layer which will influence fatigue lifetime. However, due to the complex loading undergone by blades during service, the efficiency of the shot-peening treatment needs to be precisely evaluated.

The aim of the work realised in collaboration with SAFRAN was to propose a methodology which allows taking into account the impact of the mechanical state generated by shot-peening and its evolution on the fatigue behaviour of a single crystal nickel-based superalloy (AM1) used for high pressure turbine blade. The experimental work was firstly devoted to the determination of the initial mechanical state (residual stresses and work hardening) using X-ray diffraction on a single crystal material where the classical sin2ψ technique can't be applied. Residual stresses are then determined on shot-peened plane parallel samples or cylindrical specimens after shoot-peening, thermal and thermo-mechanical relaxation using the specific method developed by Ortner. A calibration method is also proposed in order to derive the work hardening from the peak broadening using tensile tests. For a shot-peened surface oriented along the [100] crystallographic direction, the measured profile exhibits a 160 mm-thick hardened layer, where the compressive in-plane stresses are up to 1000-1400 MPa. To quantify the effect of the crystalline anisotropy on the stress field induced by the shot-peening process, samples with surfaces oriented along the < 110 > crystallographic directions have been also analysed and reveal a 30% increase of compressive stresses for the component which is not crystallographically equivalent with respect to the [100] surface orientation. This experimental initial mechanical state is then introduced as an input in structure lifetime calculation. The well known method involving the direct introduction of eigenstrain profiles is used and improved to also introduce work hardening variables and the complete anisotropic mechanical state in all the integration points of the structure. The next step was to model, using here a crystal plasticity formalism, the evolution of these quantities under thermal and mechanical loads in accordance with the experimental data under an uniform temperature (650°C). It is important to notice at this stage that the constitutive model chosen to describe the behaviour of the material needs an accurate simulation of both the residual stress relaxation and the work hardening evolution during the cyclic loading. Finally the complete lifetime assessment carried out on notched shot-peened samples (representative of stress concentration on roots of turbine blades) a good agreement with measurements in the 635-800 MPa range of applied nominal stress regarding the number of cycles to crack initiation (figure 1) and the localisation (figure 2). The increase of the fatigue lifetime due to shot-peening is between a factor 4 and 10. For smaller applied stress, a deleterious effect of shot-peening is measured probably due to a higher sensitivity of the damage to the surface roughness of samples.

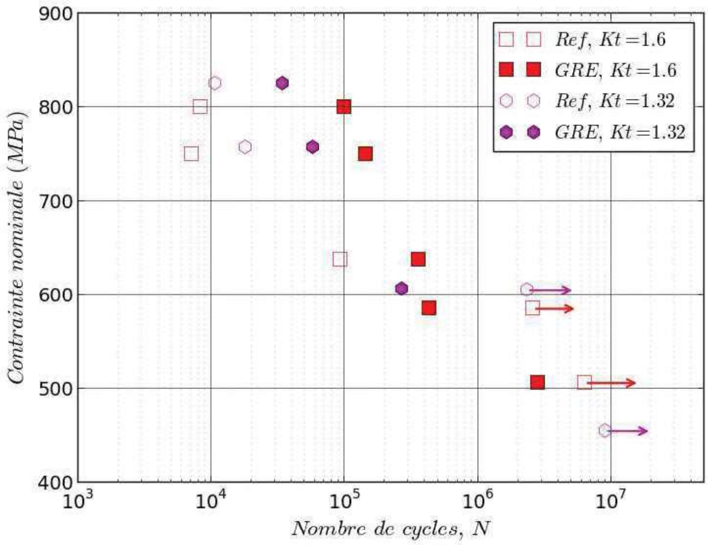

Figure 1: Fatigue curve of Kt=1.6 and Kt=1.32 fatigue test specimens made of AM1 single crystal superalloy (T=650°C, f=15Hz, Rσ=0).

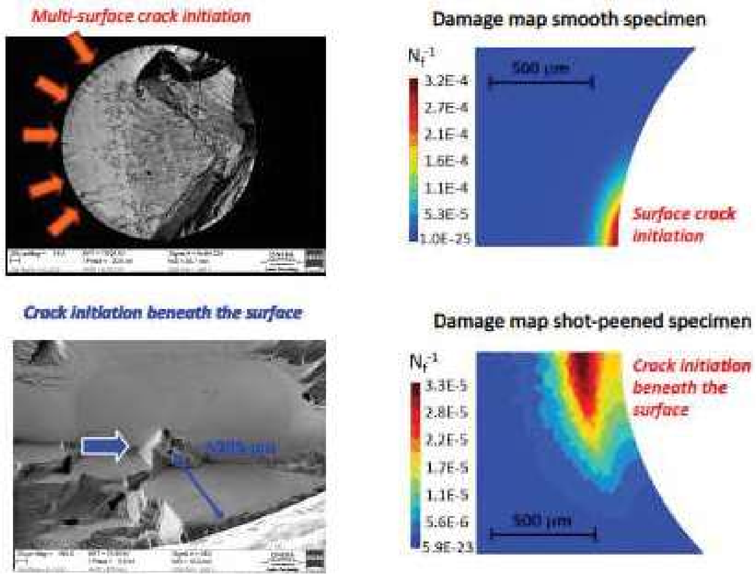

Figure 2: Right - fracture SEM micrographs in notched cylindrical fatigue test samples with a stress concentration factor Kt=1.6 subjected to an applied stress of 800 MPa at 650°C (f=15 Hz, Rσ=0). Left - local damage maps calculated.

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[2] MORANÇAIS, Amélie, FÈVRE, Mathieu, FRANÇOIS, Manuel, et al. Fatigue life of a shot-peened nickel-based single crystal superalloy: from measurements to modelling. In : 13th International Symposium on Superalloys (Superalloys 2016). 2016.

2.4. LIFE TIME PREDICTION OF FSW WELDED JOINTS UNDER UNIAXIAL AND MULTIAXIAL LOADS (ONERA)

Friction Stir Welding, otherwise known as FSW, was developed in 1991 by The Welding Institute. It is a solid state welding process which means that reached temperatures are lower than the melting temperature of the base material. This process without filler material enables the welding of non-weldable considered materials such as aluminium alloys of the 2XXX serie. Nowadays, FSW is more and more used in the aeronautical field as an alternative of the riveting process. When welding the precipitation hardened 2198-T8 aluminium alloy, an important hardness decrease is observed across the welded joint. The design of aeronautical structures welded by FSW needs, therefore, a good understanding and an accurate modelling of the impact of this hardness drop on the constitutive behaviour and the fatigue lifetime of the junction under uniaxial and multiaxial loads. Digital Image

Correlation (DIC) is used to measure local displacement fields inside the welded area, during specific mechanical tests. A constitutive model is thereafter identified and the simulated strain fields compared to those experimentally measured. An example of comparison between experimental results and the corresponding simulation is given in the figures 1 and 2 for a multi-axial loading. One fourth of the sample is simulated. The right localisation of the plastic deformation so as the reached values can be observed.

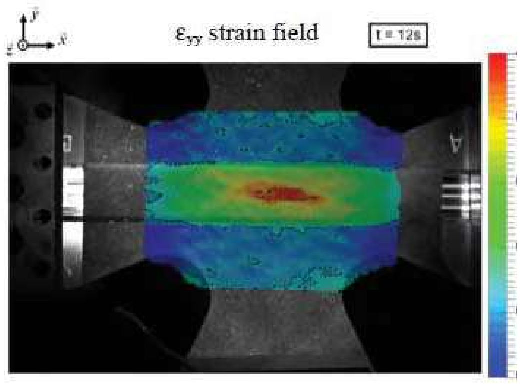

Fig 1: ε_{yy} strain field measured experimentally for $\sigma_{xx}=\sigma_{yy}=80MPa$. Isovalues at $\varepsilon_{yy}=0.1\%$ and $\varepsilon_{yy}=0.4\%$ are also added.

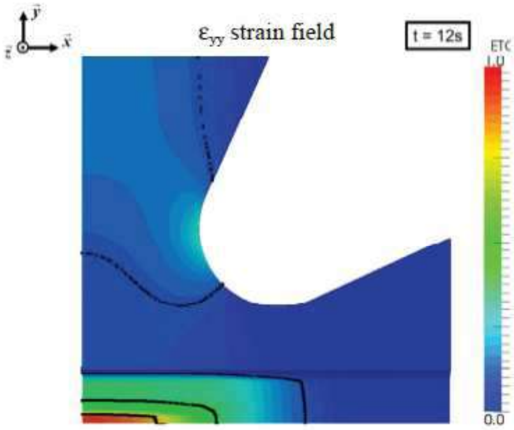

Fig 2: ε_{yy} strain field measured experimentally for $\sigma_{xx}=\sigma_{yy}=80MPa$. Isovalues at $\varepsilon_{yy}=0.1\%$, $\varepsilon_{yy}=0.4\%$ and $\varepsilon_{yy}=0,8\%$ are also added.

Then, once the constitutive model accurately validated, fatigue tests are performed in order to identify a fatigue model for lifetime predictions and rupture localisations. Figure 2, below, shows the comparison between the fatigue lifetimes experimentally obtained and the predicted ones, for both uniaxial and multi-axial tests and for several stress ratios. A quite good agreement is observed even if some tests are clearly too conservative. These results suggest that some improvements can be brought to both the constitutive and the damage models to better reproduce the experimental observations.

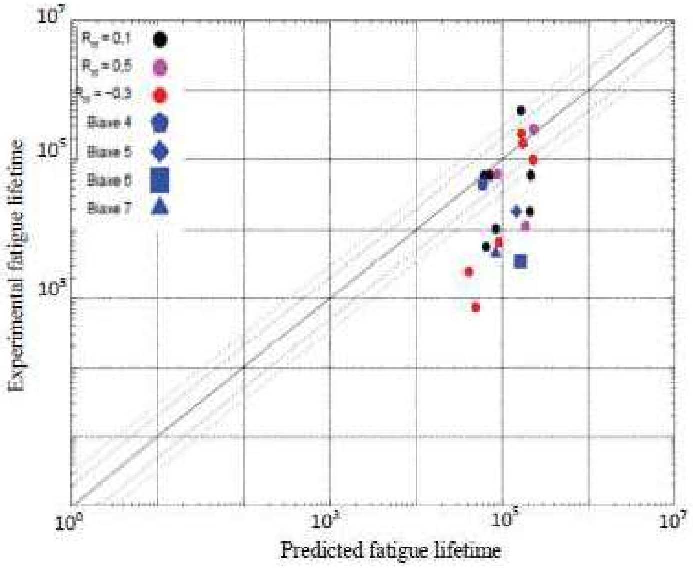

Fig 3: Comparison between the fatigue lifetimes obtained experimentally and the predicted ones. Blue symbols refer to multi-axial tests. Circles refer to uniaxial tests performed with several stress ratios.

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2.5. EXPERIMENTAL AND NUMERICAL INVESTIGATIONS OF THE INFLUENCE OF MULTIPLE SITE DAMAGE ON THE FATIGUE LIFE OF CORRODED ALUMINUM STRUCTURES (DGA TECHNIQUES AERONAUTIQUES)

DGA is in charge of qualification and continued airworthiness for military aircrafts. In this context, DGA have performed tests and numerical studies for the safety assessment of corroded aircraft structures.

Structural damages in corroded aircraft are very often characterized by the simultaneous presence of cracks initiated at the bottom of corrosion pits, which can be referred to as multiple site damage phenomenon (MSD), figure 1.

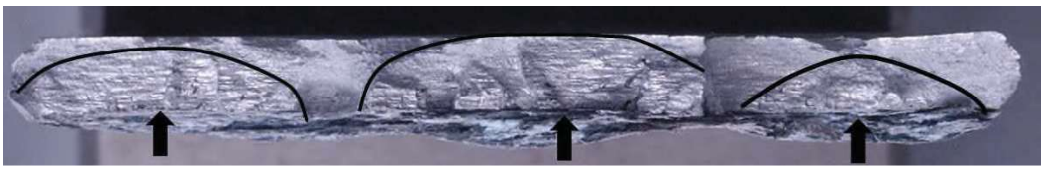


Figure 1: multiple site crack propagation at the bottom of corrosion pits

In the 90's, MSD in longitudinal lap joint panels have been studied extensively, but, so far, MSD initiated on corrosion is far less documented. Also DGA led experimental and numerical studies on the influence of MSD on the total fatigue life of corroded aluminum structures.

Furthermore, the random distribution of corrosion pits and the complexity to model MSD crack growth led DGA to investigate the Equivalent Initial Flaw size method (EIFS) to predict the fatigue life of corroded aluminum structures.

This paper summarizes the results obtained during several campaigns of tests and the results of modeling exercises.

First of all, the fatigue tests and fractographic analyses carried on sea air corroded specimens showed, as expected, an important decrease of the total fatigue life compared to non-corroded specimens, even for slightly corroded specimens, figure 2. The total fatigue life includes a crack initiation part at the bottom of corrosion pits, a multiple site crack propagation part with possible interacting effects depending on the pits density and their depths, and the MSD residual strength.

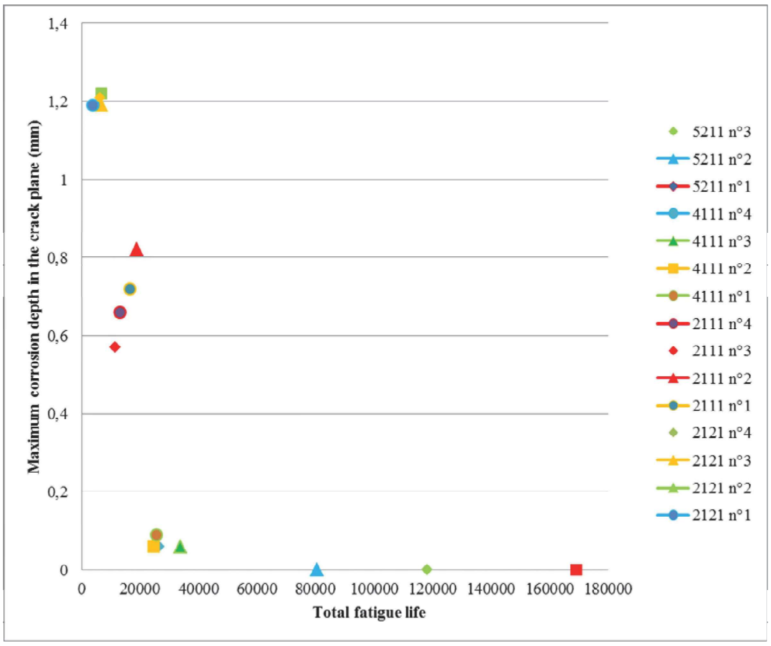


Figure 2: fatigue life to rupture vs corrosion pit depth of 2024T3 aluminum specimens

The reduction of the total fatigue life is mainly attributed to the sharp geometry at the bottom of pits which emphasizes the mechanism of the crack initiation, and led us to model corrosion pits by semi-elliptic cracks. The interaction effects on crack propagation needed more investigations and the influence of MSD residual strength on the fatigue life had been, as far, neglected.

Fatigue tests and modeling exercises have been performed on aluminum 2024-T3 specimens with multiple coplanar and no-coplanar semi-elliptic sharp notches (pits modeling), to investigate the influence of MSD on the fatigue life, figures 3 & 4. The results showed an important decrease of the total fatigue life compared to a single-site semi-elliptic cracked specimen (1.5 to 2.5 times), which shows the importance of the MSD phenomenon and the necessity to use an appropriate fatigue and fracture mechanics model. According to the numerical analyses, the interaction phenomenon between adjacent sites is the main cause of this shorter multiple site crack propagation life.

Figure 3: 3 coplanar crack sites configuration – confrontation FEM model/multiple cracks propagation fatigue test

Figure 4: 3x2 no-coplanar crack sites configuration - multiple cracks propagation fatigue test

Finally, an EIFS method has been applied on the sea air corroded specimens fatigue tests. The EIFS calculated for different 2024-T3 corroded specimens are consistent with the pits depth at the cracks sites, although they have a larger size, and seem to be independent of the loading.

The main lessons we learned from this experience are the danger to use a single site model to predict the total fatigue life of MSD, an improvement of the understanding of the influent parameters implied in MSD initiated on corrosion pits which outline the complexity to model accurately these MSD phenomenon, and a degree of confidence in the EIFS method to predict the total fatigue life of a corroded aluminum, even if further testing are required.

3. IN SERVICE OCCURRENCE ASSESSMENT

3.1. C-135 FUEL TRANSFER PIPE LOADS MONITORING, TESTS AND SIMULATIONS (DGA TECHNIQUES AERONAUTIQUES)

DGA is in charge of airworthiness issues for military aircrafts, including the fuel tanker Boeing C-135. In this frame, a study was launched to explain leaks occurring during fuel transfer in the aircraft wing aerial refueling pods located on the wing tips.

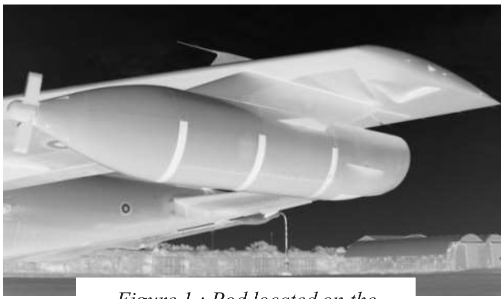


Figure 1 : Pod located on the

During an in-flight refueling, the receiver connects to the drogue, which trails from the tanker via a flexible hose, through a "probe," a rigid arm placed on the receiver aircraft's nose or fuselage. Once in position, the fuel circulates from tanker to receiver aircraft.

The study consists in measuring loads applied on a Y-pipe located in the pod where the leaks appear and explaining to what extent cracks can happen in service.

Figure 2 : Pipe in the air refueling system

To achieve this goal a 6 months in service flight campaign was led with an instrumented Y-pipe in order to better understand the fuel transfer process and the functioning of the probe-and-drogue air-to-air refueling system. Strain gauges were bonded on the pipe, accelerometers and thermal sensors were installed in the pod, and a data bus reader was connected to the fuel transfer control unit. All signals where recorded through an acquisition unit.

Loads applied were measured with the strain gauges. Thermal stress corrections on recorded data had to be done in order to take into account the environment in which the recording is made.

Experimental flights with defined manoeuvers were first performed to record data in specific phases of flight. Then, the test cell was returned to the French forces to be used as usually. A database of air-to-air refueling signals was created from the sensors data. This database was then analyzed to identify specific actions or events to evaluate the resulting pipe damages.

At the same time, a hydraulic test rig was set up and numerical simulations were carried out. The tests provide information about the burst static load and a number of fatigue cycles to initiate cracks in the pipe. The simulations were used to identify critical points in the pipe and to evaluate the fatigue life. Experimental and numerical simulations were complementary. Results were compared and correlated.

Finally, some specific and potentially damaging events were identified during the fuel transfer. The measured flight loads applied in the test rig led to the apparition of leaks at nearly the same number of cycles as in real use. Besides, cracks do not seem to be caused by an out of limits static load. Numerical simulations were well correlated with the test rig tests. Improvement axes have been defined, related to the pipe structure and manufacturing and also to the fuel transfer process.

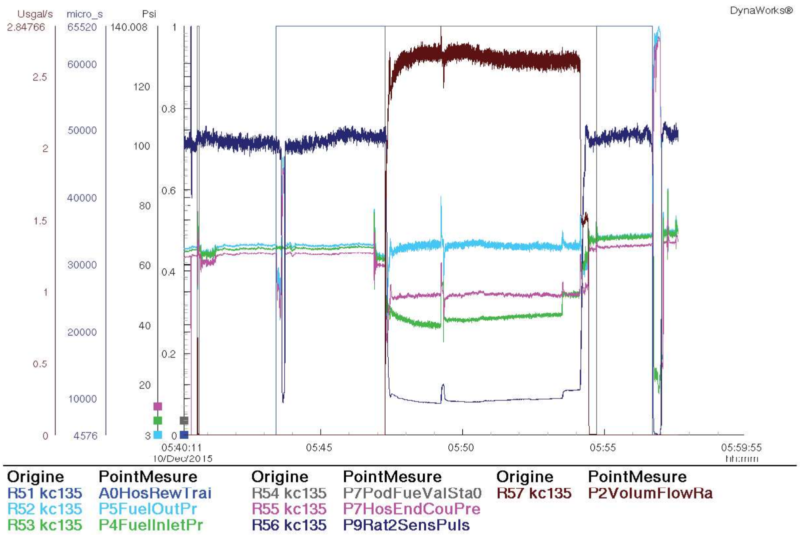


Figure 3 : Fuel transfer record

4. OPERATIONAL LOAD MONITORING, FLEET MONITORING & MANAGEMENT

4.1. OPERATIONAL LOADS MONITORING PROGRAM ON WATER BOMBER CANADAIR CL-415 (DGA TECHNIQUES AERONAUTIQUES)

DGA is in charge of airworthiness issues for Military and State aircrafts, including the water bomber Canadair CL-415 fleet used for wildfire fighting. Due to its specific missions, sometimes implying severe usage conditions - as scooping on heavy swell – the aircraft structure is submitted to high-level solicitations and technical events were reported for different areas of the airframe. Therefore, in a purpose of improving flights safety and maintenance schedule, an operational loads monitoring (OLM) program has been conducted on the fleet since 2010.

The two main goals of the OLM program have been, through a better understanding of flight and ground loads, to be able to give fleet management advices to the French Government civil defense agency in charge of the Canadair CL-415 fleet exploitation and to improve future technical events treatment by DGA. In order to do so, two aircrafts were fully equipped with strain gauges, accelerometers and data acquisition systems. Sensors were installed both on areas where technical events occurred (such as main landing gear and bottom hull) and on main structural parts (wing spare, wing root, fuselage). Measurements were recorded for a specific flight test campaign and for normal activity missions during two years.

For each instrumented structural part, the same general approach has been followed for data exploitation:

- ❖ Part modeling for critical areas identification and link between local stresses and strain gauges measurements;
- ❖ Identification of damaging flight phases (or configurations);
- ❖ Definition of fleet management indexes (static indexes for high-level solicitation detection with preventive inspection recommendation and/or damage indexes for fatigue damage assessment);
- ❖ Calculation of these indexes based only on data from the flight data recorder (FDR), which is inboard all the CL-415 aircrafts.

The last step is crucial in the purpose to give to the French Government civil defense agency a fleet management tool for uninstrumented aircrafts. Indeed, even if two aircrafts were equipped with sensors for the OLM program, this instrumentation was not intended to be widespread and only FDR records will be available for the fleet management. In order to compute indexes from FDR measurements, two different strategies have been followed according to the structural part:

- ❖ Physical identification: the objective was to determine the link between FDR measurements and the damage process, and to correlate with strain gauges measurements;

- ❖ Using of neural network. A neural network is a computational approach used, generally speaking, to identify hidden laws between inputs and outputs. In the OLM program, inputs were pre-selectioned FDR extractions and outputs were strain gauges measurements. Laws identification is made by feeding the algorithm with several flights data, both for inputs and outputs. Then, laws validation can be done on remaining flights. If a good correlation is found between strain gauges measurements and their reconstitution using FDR extractions on these remaining flights, laws are validated and indexes calculation is thus possible for uninstrumented aircrafts.

With the conclusions issued from the OLM program, the French Government civil defense agency will be able to order preventive inspections on structural parts hardly solicited during a flight – this evaluation being made in a rational way using static indexes. The OLM program will also enable a homogeneous fatigue damage evolution on all CL-415 aircrafts, using damage indexes. These both fleet management improvements will lead to a flights safety improvement.

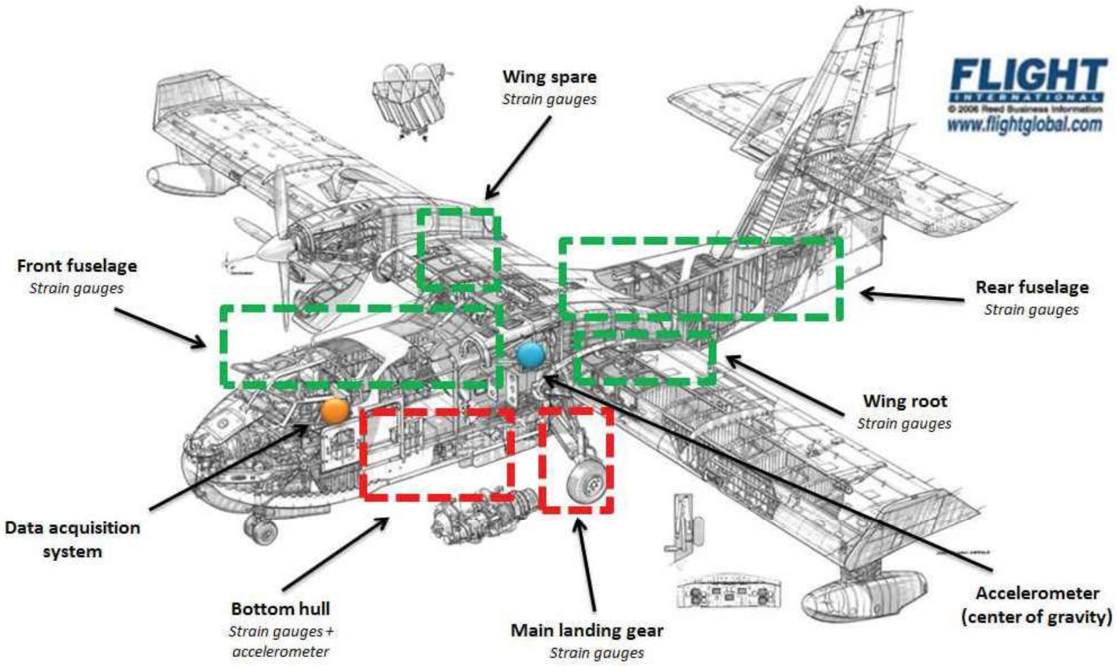


Figure 1: Instrumentation of Canadair CL-415

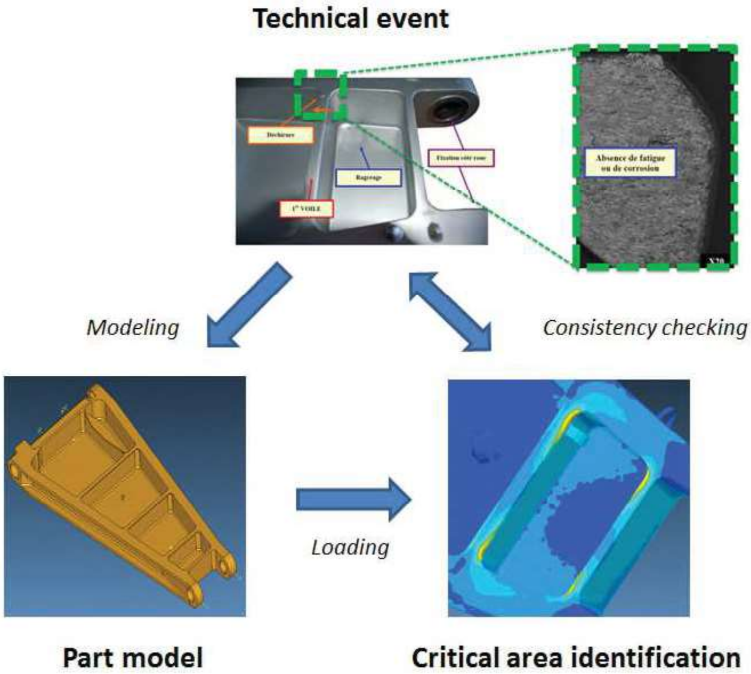


Figure 2: Modeling and critical area identification for lower member (main landing gear)

5. FATIGUE TESTING

5.1. CHALLENGES TO REPLACE FULL SCALE AIRCRAFT FATIGUE TESTING BY PREDICTIVE ANALYSIS (AIRBUS)

This topic is detailed in a paper presented in the 29th ICAF Symposium.

Objective

The objective is to have the capability to predict the actual behaviour of Aircraft Structure under applied loads up to failure for the purpose of replacing or reducing structure tests. (Ref. [1])

With this top level objective comes benefits for early optimization, reduced product development time, and increased product maturity.

Results

With today's capability in advanced non-linear analysis, prediction of test behavior has been demonstrated for many Aircraft Programs and structural materials and failure conditions. The paper will present some of these success examples at aircraft Full Scale level;

1. The accuracy of the A320 Wing Test failure prediction.

2. A400M Wing test risk mitigation activity.

3. A350 Full Scale Aircraft Predictive Test model containing more than 20M nodes and run in an Abaqus Non-linear model.

Nevertheless there exist elements which would render such an analysis unsuitable as a means of compliance for certification;

1. Metallic Fatigue – an easy process and capability to identify in a reliable way any fatigue sensitive hot spot

2. Damage prediction – reliable methods to determine and predict any composite and metallic damage growth in order to establish conservative inspection thresholds

3. Failure criteria for structure considering known failure mechanisms and the ability to accurately predict failure mechanisms for new materials and modes.

<u>Innovation Steps still needed</u>

There exists a need to build confidence in the advanced analysis modelling and simulation activity not only through verification and validation process but also by the use of management of uncertainty

Work is in progress to demonstrate that the important building blocks of the analysis are verified against empirical theories or through the use or experimentation to ensure that the underpinning methods and their results are reliable. The use of DofE can be used to fine tune testing objectives to be more in line with PVT parameter requirements, sensitivity studies, and to establish the zone of relevance rather than to establish material allowable of prove failure modes.

Similarly use cases are being identified to validate the full scale predictive test models are capturing well the science of the materials and structure when extrapolated to a whole aircraft experiment. It is important for the analyst to understand and quantify the uncertainty of analysis, test results, and process which needs to be used in a fully predictive environment

During this activity, THE most important aspect is to increase the confidence PVT with Structures analysts, Experts, Managers, Certification Authorities, and the final Customers being Airlines and fare paying public.

The use of Smarter tests providing the right data earlier in the development process is a key enabler to employ the use of advanced data analysis tools in the future.

But by far the biggest challenge is the need to establish a new generation of Computer which is needed to turn around the analysis of huge non-linear or constant re-meshing FEM in short periods suitable for structure optimization and fatigue and damage tolerance product maturity. Currently PVT is selected as a disruptive game changer for the Quantum Computer development..

Finally the development of the correct Skills and resources to be able to build models, analyze them, and derive the correct results in a repeatable manner will need considerable investment over the next 10 years in order to be ready for the next development product.

Conclusions

The manufacture of Aircraft is built around well-defined and proven processes and validated by a strict certification process. The safety of the product is paramount and both the airlines and fare paying passengers need to feel confidence in its operation.

REFERENCES

[1] Airbus forum of Predictive virtual Testing. L Harris

5.2. A350 XWB MAJOR FATIGUE TESTS (AIRBUS AND DGA TECHNIQUES AERONAUTIQUES)

The A350 XWB is subject to full airframe fatigue testing as part of its certification requirements. As for previous aircraft families Airbus decided to test the entire primary structure in a series of multiple test specimens.

One A350-900 airframe was therefore split into 3 parts, 3 test installations were created and 3 large tests were performed on 3 different sites.

The front fuselage part is called EF1. It consists of the cockpit and the forward fuselage with the installation of a nose landing gear plus the gear door loading mechanism. It was located in Toulouse, the test preparation and the test performance were done by DGA-TA.

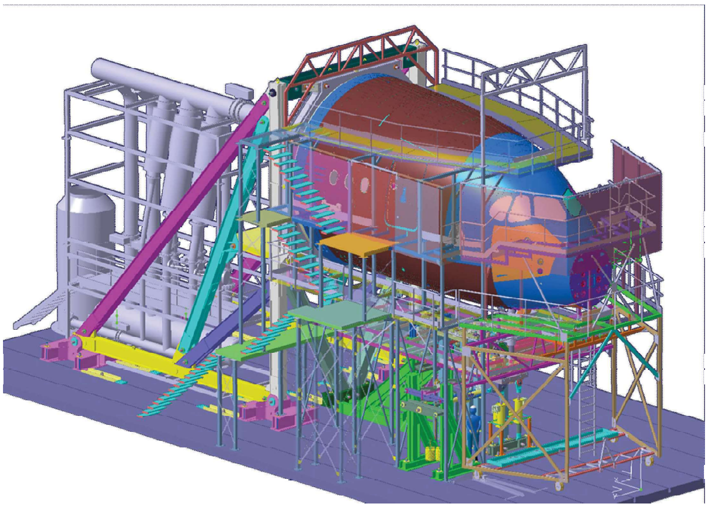
The center fuselage and the wings with the main landing gears, called the EF2, is the largest specimen of the trio. The installation is the most complex with regards to the load introduction and load support points; the test performance is slower and subject to higher level and severity of damages. The EF2 test is installed in Erding, close to Munich in Germany. It is contracted to the IABG mbH, a test institute with the experience of Airbus aircraft large fatigue tests.

The aft fuselage, called EF3, consists of the rear pressurized and non-pressurized parts of the fuselage, complete with the pressure bulkhead, the cargo doors and a part of the vertical tail plane. The horizontal tail plane is represented by a load introduction dummy. The EF3 test was performed in the in-house Airbus test facility in Hamburg.

All three major fatigue tests performed simulated flights up to 3 times the Design Service Goal of the A350 XWB. The EF1 has been transferred to tear down, the EF3 is being prepared for the transport to the tear down and the EF2 is subject to the final Residual Strength Test campaign (status end 2016).

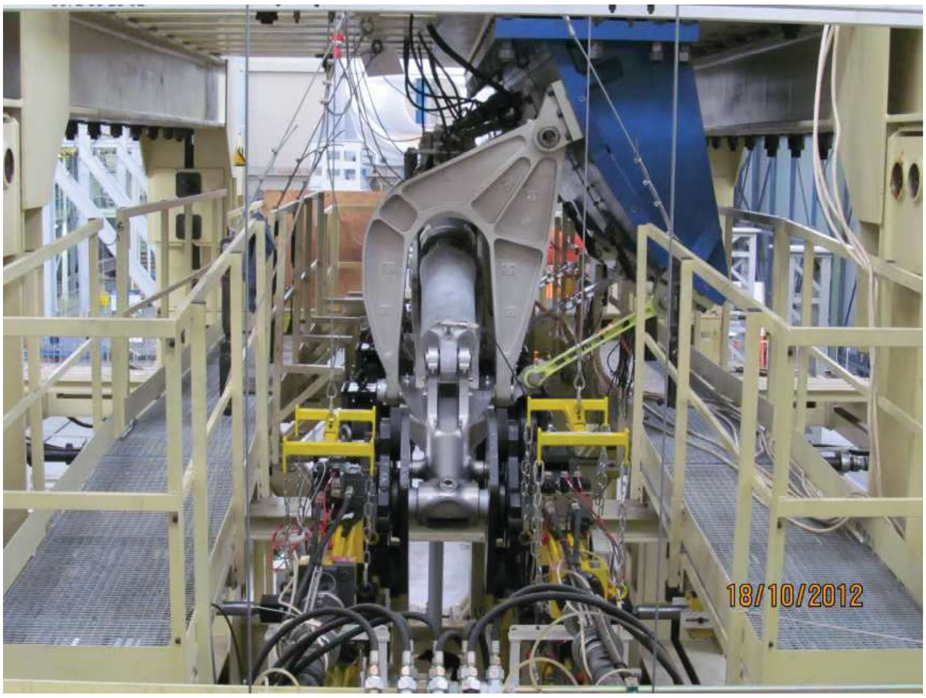
The A350 XWB aircraft concept introduces new technologies and employs a high percentage of composite materials in its construction. The results of the three tests were consistent and provided excellent validation of the design both to Airbus and its customers. We have been able to demonstrate the high maturity in the aircraft design at a very early stage of its development.

In the presentation to the ICAF 2017 given in the 29th ICAF Symposium the results of the tests which were performed in a record time will be shared. It will be also presented the test specimen and the measurement results after the equivalent of 72 years in service life.

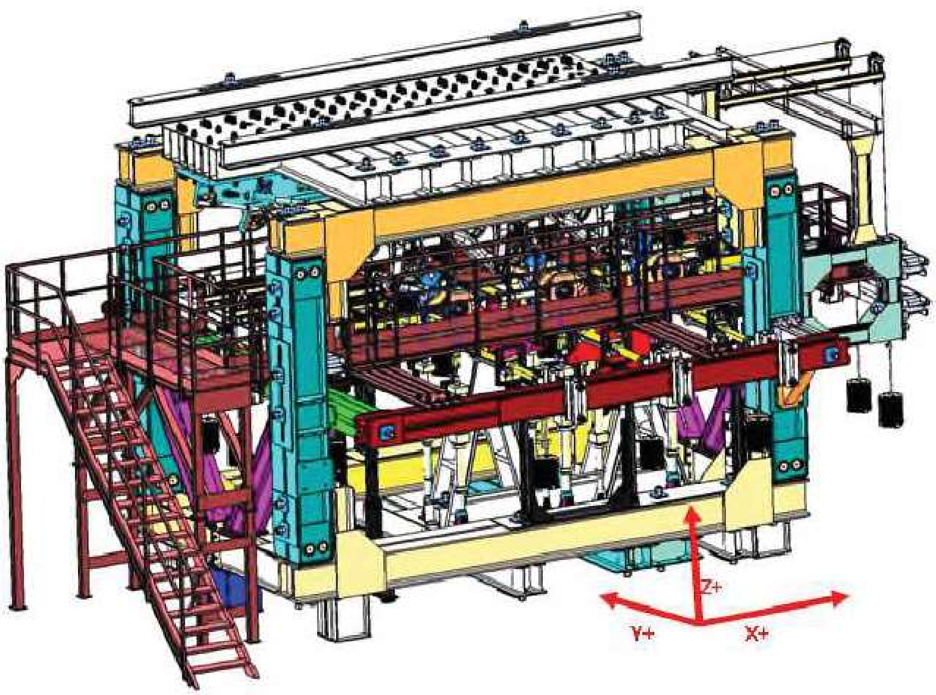


5.3. A400M MAIN LANDING GEAR FATIGUE TEST (SAFRAN LANDING SYTEMS AND DGA TECHNIQUES AERONAUTIQUES)

DGA TA has now completed the performance of the A400M Main Landing Gear Fatigue Test on a specimen representative of the series production.



The aim is to simulate 50,000 cycles (SF) to validate 10,000 Flight Cycles (FC) which is the design goal. Load spectrum includes the application of combination of flights grouped in blocks of 200 flights. Load spectrum at the end is in line with the typical aircraft usage (mission mix) defined by a breakdown between four basic missions. Typical ground loads, kneeling actuator loads and extraction retraction loads are simulated.



The loading is achieved using twenty four independent load channels (eight per leg), each fitted with spherical bearings at both ends. All load transducers are located as near as possible of the load introduction point to have a load value recorded by the transducer very close to the real load introduced into the specimen.

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