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MINISTÈRE DES ARMÉES

Review of aeronautical fatigue investigations in France during the period May 2017- April 2019

TECHNICAL NOTE

19-DGATA-4-ST-2019

DIRECTION GENERALE DE L'ARMEMENT

DGA Techniques aéronautiques

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Abstract :

The present review, prepared for the purpose of the 36th ICAF conference to be held in Krakow (Poland), on 3-4 June 2019, summarises works performed in France in the field of aeronautical fatigue and structural integrity, over the period May 2017-April 2019.

Topics are arranged from basic investigations up to fleet fatigue monitoring.

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 :

- Manuel De-Araujo and Alain Santgerma for Airbus Commercial Aircraft
- Erembert Nizery for Constellium
- Myriam Kaminski for ONERA
- Olivier Gillet, Vincent Montlahuc, Anne-Cécile Marel, Cyril Pons and Bastien Bayart for DGA Aeronautical Systems.

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1. INTRODUCTION AND ACKNOWLEDGMENT

The present review, prepared for the purpose of the 36th ICAF conference to be held in Krakow (Poland), on 3-4 June 2019, summarises works performed in France in the field of aeronautical fatigue, over the period May 2017-April 2019.

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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. LIFETIME PREDICTION OF 3D WOVEN COMPOSITES (ONERA)

The use of composite material in both civil and military aeronautic applications keeps on increasing from the last decades, mainly due to its advantage in terms of ratio mass/stiffness/strengthening in comparison with standard solutions based on metallic material. 3D woven composite is a new kind of composite material, developed to obtain higher out-of-plane mechanical properties and interesting residual properties after impact. Safran Aircraft Engines has thus chosen such a material for the manufacturing of the fan blades of its new generation of civil engine. Moreover, with the continuous improvement of composite design methods during the last decades and the imperative of structural mass and consumption minimization for recent airliners, composite structures are subjected to loadings closer and closer to their static strength during longer periods. Consequently, a robust unified methodology to predict the real lifetime of industrial 3D woven components is expected by Safran Aircraft Engines. The static model named ODM-CMO [1] has been adapted to predict the lifetime of 3D woven composites. A significant effort has been made these last years [2] to build a full kinetic damage model defining multi mechanism kinetic damage evolution laws. This model allows to handle random complex fatigue loadings without using the notion of cycling, the damage law being written in a rate form $\dot{d}=\cdots$ ([2],[3]), and accounts for the observed mean stress effect [4]. It is now possible to track the evolution of the damage and the decrease of the Young modulus (Figure 1) to generate both the S-N curves (Figure 2) and Constant Life diagrams. A procedure of parameters identification has been proposed and validated with additional experimental tests carried out by Safran. Validation tests on the model have shown that measurement data and the simulation results are in good agreement. This model has been implemented into a FE code (ABAQUS) and applied on different structures with an increasing complexity in order to demonstrate its transferability to a design office in aeronautical industries Calculation strategies are also proposed in order to simulate large number of fatigue loading cycles of composite structures, particularly an innovative method of cycle jumps is proposed, relying closely on the model physics [3].

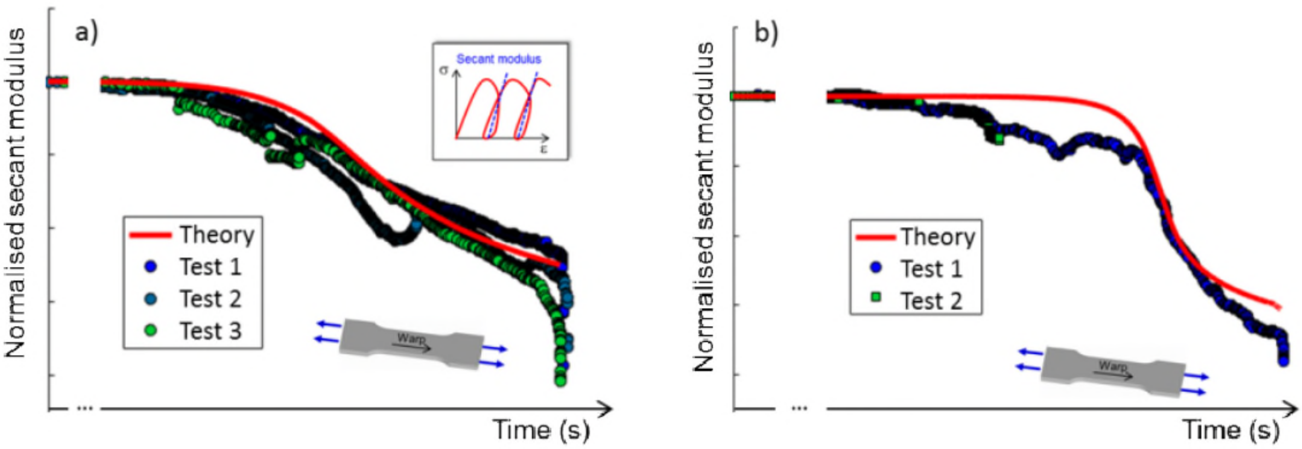


Figure 1. Comparisons between the experimental modulus and the estimations (identification) obtained with the model for fatigue tensile tests (a) at 60% and (b) at 40% σr in the warp direction, R close to 0 at f=5 Hz

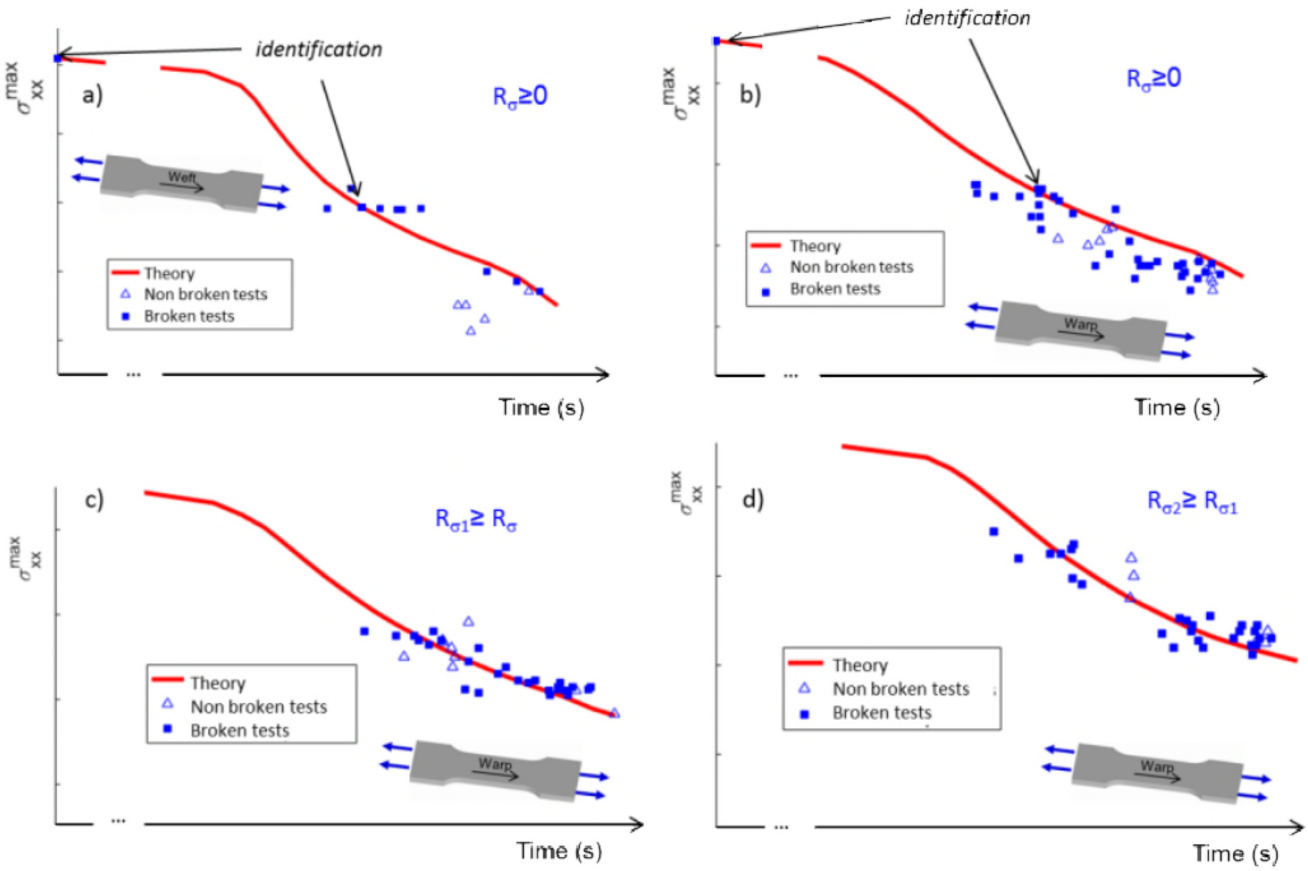


Figure 2. Comparisons between the predicted S-N curves with the model and the test results in (a) the weft direction with a stress ratio close to 0 at 5Hz and (b-d) in the warp direction for three increasing stress ratios (Rσ ≤ Rσ1 ≤ Rσ2) evolving between 0 and 1 at f=5Hz

[1] Rakotoarisoa C. Prévision de la durée de vie en fatigue des composites à matrice organique tissés interlock. Doctorate thesis of Université de Technologie de Compiègne, 2014.

[2] L. Angrand, Modèle d'endommagement incrémental en temps pour la prévision de la durée de vie des composites tissés 3D en fatigue cyclique et en fatigue aléatoire, Thèse de doctorat Mécanique des matériaux Paris Saclay, 2016.

[3] O. Sally, C. Julien, F. Laurin, R. Desmorat and F. Bouillon, "Calculations strategies to forecast lifetime of oxide/oxide composite structures under complex loadings", ECCM18 - 18th European Conference on Composite Materials, Athens, Greece, 24-28th June 2018

[4] R. Desmorat, L. Angrand, P. Gaborit, M. Kaminski, and C. Rakotoarisoa. On the introduction of a mean stress in kinetic damage evolution laws for fatigue. Int. J. Fatigue, 77:141-153, 2015.

2.2. A SPECIMEN TO EVALUATE SUSCEPTIBILITY OF ALUMINIUM ALLOYS TO L-S CRACK DEVIATION (CONSTELLIUM)

The current aircraft structures tend to increase the proportion of integral structure parts. One main advantage is the cost, through a reduction of the assembly complexity. The fatigue behaviour of integral structures is also improved due to a reduction in potential initiation sites at joints or rivets, and the stiffness is considered better. The design of new parts requires, however, an analysis of the damage tolerance behaviour (van der Veen et al., 2016). The present study focuses on the fatigue crack propagation of such structures. Complex parts are commonly machined in aluminium thick plates. Several studies demonstrate that cracks may deviate in unstiffened L-S crack configurations for standard alloys (Joyce et al., 2016; Sinclair and Gregson, 1997). T-stresses and mixed-mode loads are also affecting the crack deviation behaviour (Llopart et al., 2006). The L-S CT specimen is shown sufficient to differentiate some variations, e.g. variations between three positions through-thickness. However, the less deviating alloys are difficult to distinguish.

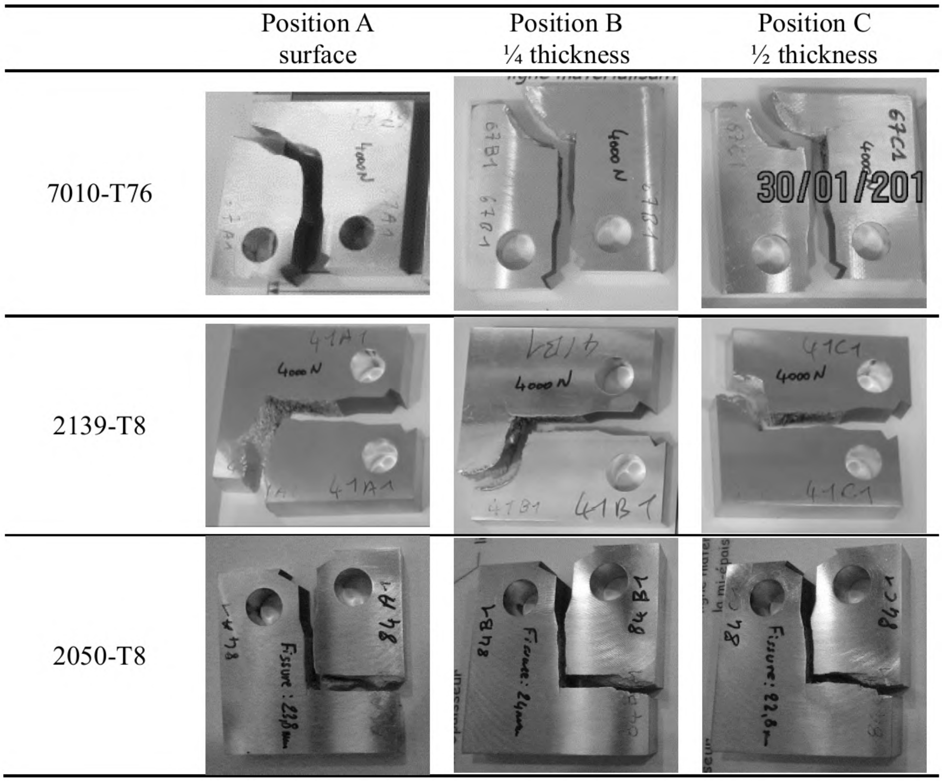


Figure 3. CT specimens in 7010, 2139 and 2050 after the fatigue test in the L-ST direction. The deviation is more pronounced for position A (near the surface of the plate), and for alloy 2050

A new asymmetric four-stiffener specimen (WEND) geometry have been proposed as an alternative test. Experimentally, in order to demonstrate the advantages of the WEND specimen, 7010-T76, 2139-T8 and 2050-T8 plates have been characterized. The same alloys are characterized using CT specimens. The WEND specimen is a lab-scale test closer to real structures than a CT test. In its first use, it allows to compare an alloy behaviour with a targeted lifetime vs crack path. In its second on-going use, it enables the comparison with Finite Element Modelling for a better crack path and lifetime prediction

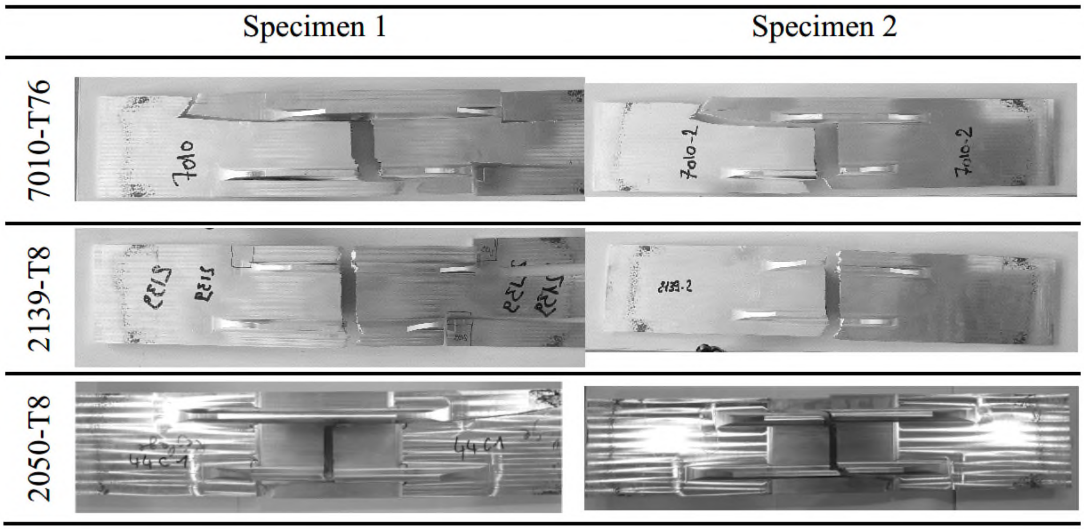


Figure 4. WEND specimens after the fatigue test in L-S direction.

Joyce, M.R., Starink, M.J., and Sinclair, I. (2016). Assessment of mixed mode loading on macroscopic fatigue crack paths in thick section Al-Cu-Li alloy plate. Mater. Des. 93, 379–387.

Sinclair, I., and Gregson, P.J. (1997). The effects of mixed mode loading on intergranular failure in AA7050-T7651. Mater. Sci. Forum 242, 175–180.

Llopart, L., Kurz, B., Wellhausen, C., Anglada, M., Drechsler, K., and Wolf, K. (2006). Investigation of fatigue crack growth and crack turning on integral stiffened structures under mode I loading. Eng. Fract. Mech. 73, 2139–2152.

2.3. RANDOM VIBRATIONS FATIGUE ANALYSIS AND TESTS (DGA AERONAUTICAL SYSTEMS)

This topic is an update of a study already presented in ICAF 34th conference French review.

DGA is in charge of qualification aspects and continued airworthiness for military aircrafts. In this context, DGA have to perform tests for the qualification of equipment in vibrations environment. But tests are expensive and can take a long time to do, relatively to operational issues. The objective of this study is not to replace all the tests by calculations – many data can only be obtained by test for now, damping for example – but to be able to anticipate a test in order to reduce hazard of non-predicted failures or to extend a structural qualification from a vibrations environment substantiated by test to another one lightly different. Moreover, some subjects are raised about crack initiations which cannot be explained by “classical” fatigue models, especially for rotorcrafts, because of the type of excitation: vibrations. So, with this study, DGA could also improve its knowledge about vibration fatigue in order to explain and to bring solutions to technical facts relative to this particular subject.

Indeed, vibrations environments are characterized by low stress levels, high frequencies (many fatigue cycles) and sometimes random spectra. These characteristics lead to some analysis difficulties: to get an initiation with low stress levels and to be able to count cycles for random spectra. The aim of this study was to improve DGA knowledge about random vibrations fatigue by evaluating some specific models from literature. They were assessed by two different approaches: numerical and experimental on coupons. The objective was to answer to some questions such as: is the model accurate? Is it conservative? How many parameters are needed? Another part of this study was about the ability of using this model for real structure.

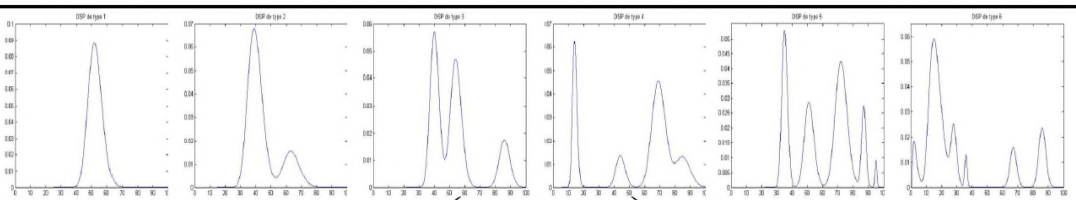
Damage calculations for random vibrations environment are based on “classical” damage calculations. The main difference and difficulty is the way to “count” cycles, because cycles don’t exist anymore. So, it is needed to estimate a probability density function of stress from the power spectral density of the excitation.

$$E[D] = \int \frac{S^b}{C} . E[P] . f_s(S)\, dS$$

- $f_s(S)$ represents the probability density function of having extracted from the signal an elementary cycle with an amplitude S.
- $E[P]$ is the average number of peaks of the signal per unit of time.
- $\frac{S^b}{C}$ represents the damage of an S amplitude cycle when the Basquin model is used to approximate the Wöhler‘s curve.

The different models from literature give different probability density functions. The first one, the Narrow band model, is purely theoretical and well none but, as its names suggests, it is not adapted to wide band signal. The other models are split in two types. The first ones are based on the Narrow band model: they add a corrective factor, which is often empirical, to correct and improve the Narrow band model. The second ones are direct models.

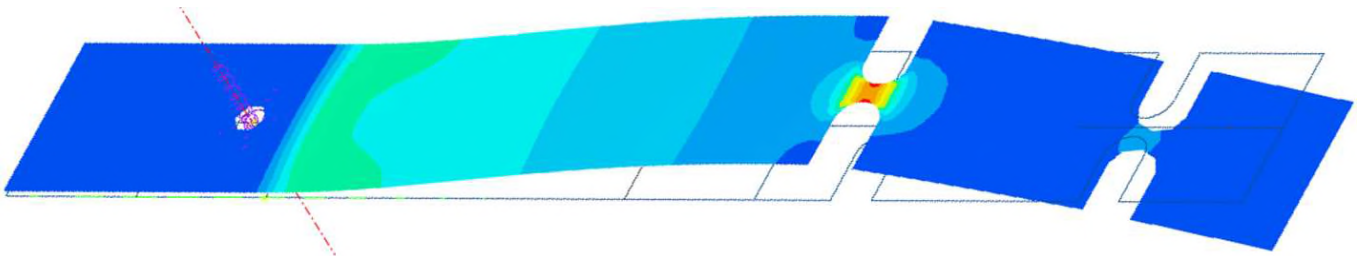
The numerical assessment of these models was made by comparison with “classical” damage calculations. More than 600 power spectral densities were generated at random on the [0 ; 100] Hz band.



$$S(f) = \sum_{i=1}^{n} A_i \times \exp\left(-\frac{(f - f_{ci})^2}{B_i} \right) \quad with \;\; n = 1..6$$

For each one, many corresponding temporal signal were also generated then the damage was calculated for each signal by “classical” method (Rainflow counting and Miner’s rule were used). The average of all these calculations (for each power spectral density) was the reference. The damage was also calculated directly from the power spectral densities with the different previous models. A comparison of the results led to assess their accuracy, scattering and conservatism.

The aim of the second part of the study was to assess these models by confronting them to test results: some tests were made with a shaker on two types of coupons. To be able to assess properly the models, the coupons had to be “wide band”. Indeed, the standards for aircraft and rotorcraft, which were study in the frame of this work, present generally random vibrations on a wide band and some typical frequencies excitations corresponding to rotor or blades speed. In fact, a wide band response for “simple” coupons was quite difficult to obtain, that is why they were designed to obtain a transfer function with two natural mechanical frequencies. The first design of coupon was found in literature and has only one critical point. The second one was “home-made” by DGA: the design was optimized to obtain two different critical points, two resonance frequencies and a fixed predicted test duration.



First design of coupon

Second design of coupon

This optimization and the random excitation definition were made thanks to finite elements modelling (FEM) of the coupons and damage calculations with the different previous models. Finally, the first coupons tests enabled to adjust the complete model (FEM and damage calculation) and the second ones enabled to compare the models and to assess the possibility to make this type of damage prediction.

These two first parts of this study used a number of specimens found too limited to account for scatter and explore a wide range of spectra. So in order to get more accurate results, a huge test campaign has been launched and vibration tests on 100 classical aluminum 2024T3 coupons are about to start. There will be 50 coupons with the first design and 50 with the second. Some coupons will be use to define the material behavior in order to tune the numerical model and then the other will be used to assess the consistency between tests and calculation on random vibration fatigue. Many vibrational spectra will be assessed through these tests.

3. SIMULATION & TESTING

3.1. SMART SIMULATION AND TESTING (AIRBUS COMMERCIAL AIRCRAFT)

3.1.1. INTRODUCTION

In Airbus R&T campus located in Toulouse along with metallic investigations projects teams, a significant effort has been put on increasing detailed numerical simulations at different test pyramid scale levels aiming at spotting fatigue critical areas up to large scale components perimeter.

In parallel of increasing simulation accuracy for more detailed stress peak results, fatigue parametric model is also investigated to bridge for a better number of cycles for damage initiation and propagation assessment. Parameters such as loading frequency, temperature and humidity or again stress state and scale effect are evaluated in order to enrich the already existing parametric database on fatigue damage initiation.

Computational increasing capabilities are also used to investigate 3D crack growth simulation on complex geometries and load patterns enabling to predict with more accuracy crack paths and growth rates.

These streams of investigation are building the bricks of the Smart Simulation and Testing approach aiming at deploy large scale simulation platform able to assess critical areas early enough in time to influence definition and optimize out of cycle physical test components.

3.1.2. LARGE SCALE DETAILED SIMULATION & TEST

Large Scale detailed simulation models associating specific features for post processing in order to derive detailed stress peaks to be used for metallic fatigue evaluation have been developed on several platforms as shown here in here Figure 5 for a Fatigue R&T component cell dedicated for metallic technology demonstration. The predictions of the hotspots will then be correlated to test findings with the objective to measure the higher accuracy of the detailed simulation process approach.

Simulation features include refined parts mesh assembled by fasteners with contact idealization on selected areas together with enriched items information. Giving the complexity of the fatigue spectrum loading, detailed maximum fatigue damage angle with associated fatigue equivalent stress or load at R=0.1 can be mapped on all elements and fasteners.

Processing this detailed simulation output allowing a first qualitative screening of hotspots, structural configurations can be evaluated quantitatively associating a pre-computed structural catalogues database and critical areas can be selected for more detailed 3D local further refined DFEM evaluation

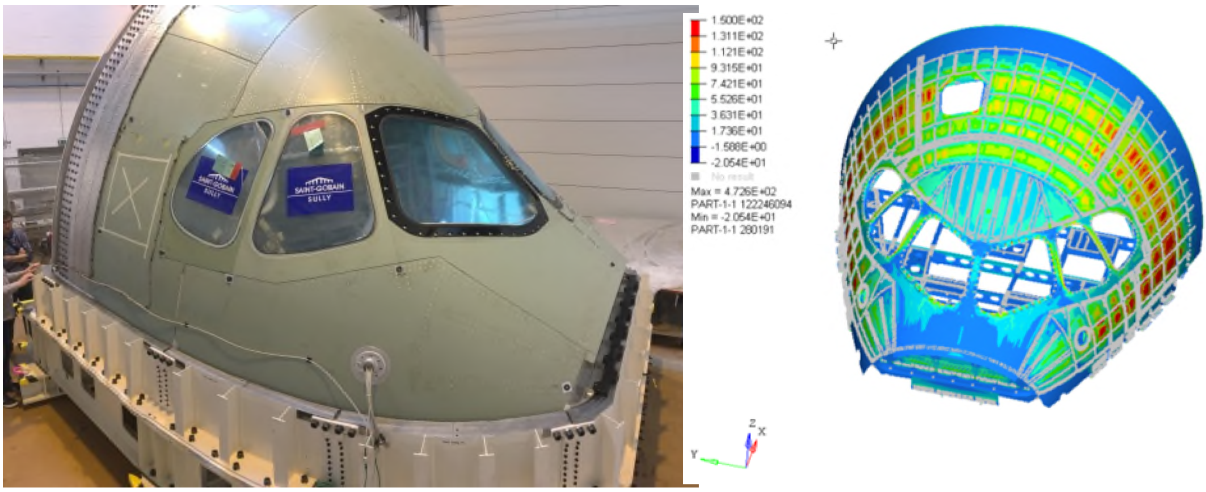


Figure 5. Cockpit R&T technology demonstrator test cell and its digital simulation twin

3.1.3. INVESTIGATING FATIGUE INITIATION MODEL

Fatigue initiation model today used in airbus is calibrated by several parameters, on top of stress peak, it accounts for material surface protection, mechanical treatment, structural configuration including assembly type parameters, material and scale effect.

In order to enrich this model, a PhD is investigating some additional parameters linked to cycling frequency, temperature and humidity. Also taking again advantage of detailed simulation some related parameters of influence have been investigated such as Scale effect. From a benchmark of approaches known in the literature the highly stress volume scale effect first introduced by R. Kuguel in 1961 as a function of 90% max peak stress volume as illustrated on Figure 6 was selected more promising.

Other related stress state influence parameters such as stress gradient and triaxiality are also looked into for more fatigue initiation parameters accuracy improvement.

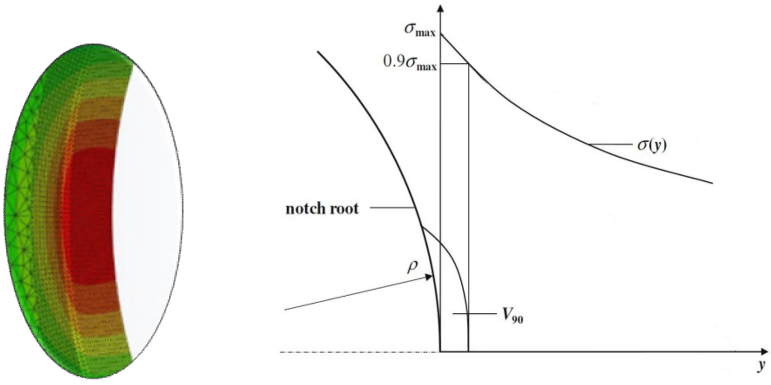


Figure 6. Highly stressed volume for Scale effect quantification

3.1.4. DETAILED 3D CRACK GROWTH SIMULATION

At the finer level of simulation sub-structure, detailed 3D Finite Element Models are used for crack propagation simulation. The specimen illustrated on Figure 7 has been manufactured from a plate where plastic deformation induced from manufacturing process was applied leading to residual stresses on the final manufactured part as an initial condition. Artificial damage and cyclic loading were applied. Initial stress conditions being validated by residual stress measurements and applied loading monitored with strain gauges led to successful crack growth simulation. Further simulations on large scale components with structural zooming for crack propagation are being tested.

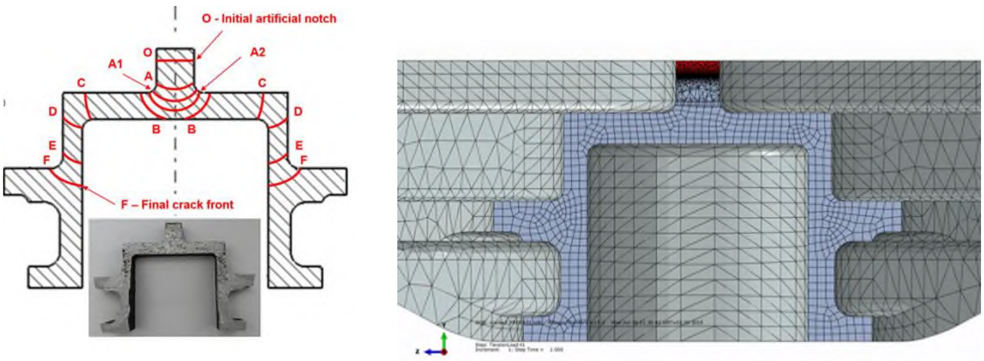


Figure 7. 3D crack growth Physical specimen simulation

3.1.5. OUTLOOKS

Know how capitalization on assessing hard structural nodes for Fatigue on metallic airframe structure from a global detailed simulation angle is expected to provide a finer and more exhaustive structural hot spot assessment. Starting earlier in the development with

Global detailed simulation will to enhance structural maturity definition and optimize physical tests lead time.

4. STRUCTURE FATIGUE TESTING

4.1. REAL TIME MONITORING OF A FULL-SCALE FATIGUE TEST (DGA AERONAUTICAL SYSTEMS)

The full-scale fatigue test is one of the major tests in the process of certification and qualification of a new aircraft type. Additional airframe fatigue tests may be also conducted in life extension programs. In both cases, setting-up the test facility can take up to one year of implementation, while the test itself can last for several years.

The experiment is based on a fatigue spectrum representative of the number of flight hours and the usage statistics expected during the service life of the airframe. The spectrum is applied by several actuators introducing a discretized load on the structure. By definition, the spectrum is divided in “units” that have to be executed multiple times.

During the test, three signals are constantly recorded: the command signal (CMD) set by the spectrum, the feedback signal (FDBK) corresponding to the real strength applied by the hydraulic jacks and the measured signal (MES) detected by strain gauges implemented close to the critical points. An algorithm was developed to process and compare those three signals. Thanks to an experimental matrix, the actuators involvement in the stress at a critical point and supposedly measured by a strain gauge is defined by a linear combination. Therefore, three damage values can be calculated from the CMD, the FDBK and the MES signals for each strain gauge at any time and more judiciously, after one unit has been entirely played. Most of the monitoring indicators lean on those values.

Indeed, the first goal of a real-time monitoring is to improve the knowledge and the compliance of the loading applied to the airframe. According to Basquin’s law of fatigue, a relevant indicator has been built from the calculated damage values induced by the CMD and the FDBK signals, showing the actual difference of severity in terms of stress observed at the critical points. When necessary, the adequate adjustment of one or several hydraulic jacks can be made during the test, mostly to prevent an overloading or an underloading of a critical component.

A second indicator was also created to ensure the global stability of the structural damage following each application of a unit. Until now, this indicator was only verifying the good reproducibility of a unit, by appreciating the difference in the damage values for all the critical points. For future tests, it will be improved and sharpened to guide and determine the visual inspection steps, which are currently set by the number of flight hours and not by the remaining potential of the components.

Ultimately with the real-time monitoring, the fatigue data can be processed and verified during the test and not exclusively when the test is over. The damage calculations produced throughout the tests can be compared to the simulated ones, enabling the deviations to be flagged and handled before the occurrence of any structural damages. Adjustments and modifications can be conducted at a suitable time and for instance, an irrelevant or malfunctioning strain gauge can be relocated or replaced to optimize the test.

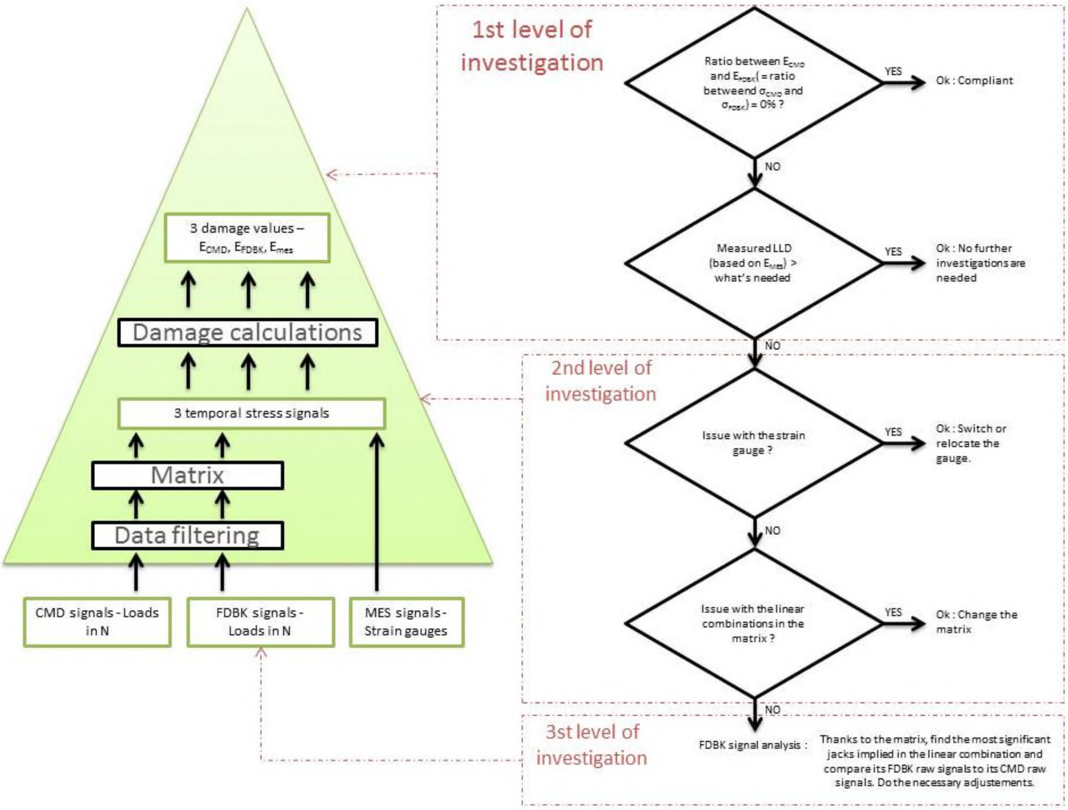


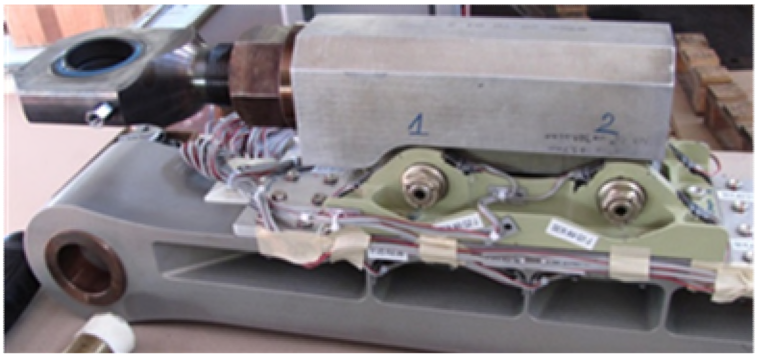
Figure 8 : Real time test monitoring principle

4.2. DETAILS FATIGUE TESTS FOR RAFALE AIRCRAFT (DASSAULT AVIATION)

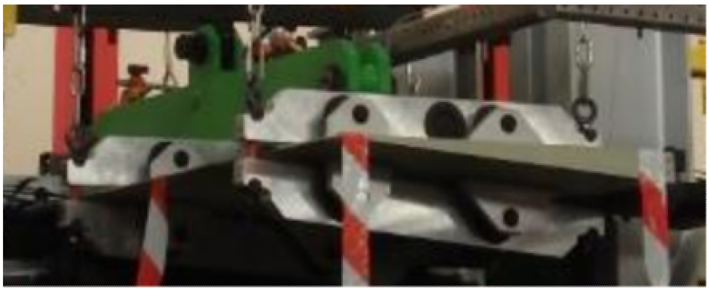
Detail tests have been carried out in the DGA Aeronautical Systems testing facilities to extend significantly the RAFALE Service Life under complex spectrum, including crack initiation and growth phases (on artificial damages), and finally, residual strength tests. The tests support calculations on specific locations of the RAFALE structure :

• Wing Rib 1 (Wing secondary attachment + Inboard elevon and its actuator attachments):

• Outboard elevon actuator attachment (Wing side):



• Outboard elevon main bearing & actuator attachment (Elevon side):



5. FLEET FATIGUE MONITORING

5.1. RELIABILITY APPROACH APPLIED ON FATIGUE SAFETY FACTORS FOR FLEET MONITORING (DGA AERONAUTICAL SYSTEMS)

Structures are often sized by a determinist approach. For a given load and material strength, this approach determines the ability of the airframe to sustain this load. Nevertheless, physical randomness is always present in every mechanical environment. Indeed, military aircraft loads (wind gusts, missions, manoeuvres), geometry design and material properties introduce some degree of randomness. These ones are considered into the aviation regulation at different levels such as the use of material design values (99%/90% probability with 95% confidence) or scatter factor. For instance, AIR 2004/E standard, which is one of the regulations for French military aircraft, specified the use of a scatter factor for fatigue structure demonstration.

These scatter factors lead to prevent the dispersion of each constitutive part of the system. They usually come from statistical data, with no clear other justification than the positive feedback on former aircraft. These factors emanate from a strict regulatory environment which can be awkward. Especially, if the impact of a modified scatter factor on the occurrence probability of an adverse event cannot be evaluated.

The goal of the reliability approach is to carry out a quantitative study about the structural failure based on random phenomenon. Then the failure probability of the airframe related to a specific scatter factor is obtained by this method. This method is currently implemented at DGA in order to quantify aircraft safety level.

According to AIR 2004/E standard, the scatter factor is equal to 5 for all critical parts which cannot be inspected on a non-monitored aircraft. Thus, the certified lifetime comes from the result of the full-scale fatigue test divided by 5. The standard says that in other cases, the scatter factor can be reduced according to qualified authorities.

Most French military aircraft are equipped with load monitoring systems (flight parameter recorder, and g-counters). These systems give direct or indirect access to the in-service loads. In that case, French military authorities allow reducing the scatter factor from 5 to 3 with a safety level at least preserved. A monitoring leads also to reduce this factor in other countries such as 3.33 in UK. However, this reduced scatter factor is used on calculation for every monitoring system like either flight parameter recorder or simple g-counters (vertical g-level detection). One may wonder to what extent the safety level is modified by the monitoring system.

A preliminary study has been conducted concerning the safety level induced by different monitoring systems. As the fatigue damage computation is different, a comparative analysis was made. More specifically, several z-acceleration spectra coming from a g-level statistical survey were implemented. All the damage values were compared thereafter. The key point was to build a random spectrum using stochastic modelling. The aim was to estimate the difference of safety level between the g-counter method and the temporal one.

Then, the main study has been conducted concerning the safety level of aircraft without monitoring system compared with monitored ones. To carry out this study, a reliability analysis has been completed considering material and load scattering. The first point consists in determining the probability that the real fatigue damage exceeds the calculated damage with or without monitoring system. Indeed, for multiple cases, several fatigue damages are computed from the input data as “triplet”: (spectrum, scatter factor (k), and material Wöhler curve). For instance, fatigue damage is calculated with the following input data:

1- (g-counter spectrum, k=3, deterministic Wöhler curve) for a monitored aircraft,

2- (specification spectrum, k=5, deterministic Wöhler curve) for an aircraft without monitored system,

3- (g-counter spectrum, k= 3, stochastic Wöhler curve) for a ‘real’ fatigue damage.

Military aircraft have a very large amount of different missions, leading to many different load spectra. Thus, these computations have been done for several random load spectra.

The second point of the reliability study is to find the adequate scatter factor to obtain the same safety level. To carry out this point, a parametrical analysis has been done in order to find the scatter factor that leads to the same probability between monitored/no-monitored aircraft.

In the future, investigations will be carried out to extend the conclusions with other methods to obtain random acceleration spectrum with the cumulative number of occurrence vs g-level.

5.2. FATIGUE CRACK GROWTH APPROACH FOR FLEET MONITORING (DGA AERONAUTICAL SYSTEMS)

Aircraft, and especially military ones, are subjected to unpredictable variable amplitude loadings. Fighter aircraft structures are essentially sized by operational loads that vary widely according to the missions flown by the aircraft (combat, cruising, training, airshow display). For safety reasons, it is necessary to be able to estimate each aircraft's actual fatigue "consumption" relative to its potential. For this purpose, most French military aircraft are equipped with load monitoring systems (flight parameter recorder, g-counters and others acquisition systems). These systems give direct or indirect access to in-service loads.

The data are regularly processed for each aircraft throughout its lifetime. The cumulative fatigue damage is calculated at various points of the structure pointed out as being critical as a result of the full-scale fatigue test. This information enables the Armed Forces to optimize the fleet management in terms of structures potential. Also, for fleet life-extension purposes and usage predictions, the use of more precise damage and crack growth prediction models is a major concern.

Military aircraft were historically designed with a safe life philosophy, that is to say designed to demonstrate their entire lifetime with no damage. However, due to fatigue life extensions and their associated tests, cracks may be detected. The initiation models are yet insufficient to cover the entire lifetime. In order to maintain the operational capability as well as the design safety margins, the technical authority introduced the damage tolerance philosophy: cracks are allowed below a given size provided propagation process is well known (slow, stable, predictable). This change of philosophy opens prospects in terms of crack propagation control. Indeed, new and more effective monitoring methods are developed.

Systems currently in operation on the French Air Force Mirage 2000D and Rafale enable the acquisition of large number of parameters and thus advanced monitoring methods. The use of this information gives access to continuous parameters such as g-accelerations, roll, pitch ... This data associated with crack growth models such as PREFFAS or ONERA could provide information about the propagation phase if any life extension of these Aircraft requires crack monitoring. The airframe potential consumption during the operational life can thus be estimated by combining initiation and propagation models.

Studies are conducted concerning the use of both initiation and propagation models to control the airframe potential consumption. For the aircraft equipped with g-counters with no temporal data, only initiation models can be used to control the crack propagation. The prediction qualities of this method are analyzed.

Initiation models enable to control the processing damage. In the case of aircraft with temporal flight parameters recorders, the initiation model is based on a unit damage matrix. In practice, the monitoring systems data are treated to know the real load sequence applied to the critical structural points. This load sequence is then used to perform a rainflow counting. The result gives a matrix that has to be multiplied by the unit damage matrix to determine the damage. Thus, the cumulative damage law is linear. As a consequence, the damage law is linear too.

In the case of aircraft equipped with g-counters (with no temporal data), the cumulative damage law is linear too but it is necessary to build an acceleration usage spectrum with the cumulative number of occurrences vs g-level, then to extrapolate and to discretize it. The discretization gives elementary damage by referencing to Wöhler curves or abacus. The total damage is then obtained by cumulating all elementary damages multiplied by their number of occurrences.

Furthermore, propagation models are investigated for the fleet providing temporal data. At this stage, experiments were carried out on CT samples and are ongoing on a more realistic geometry close to a spar one.

In the future, investigations will be carried out to extend the conclusions made on aircraft equipped with flight parameter recorder to aircraft only equipped with g-counters. It means adding a system to build spectrum using the g-counters output. This point needs further research in order to take load history into account in the propagation process.

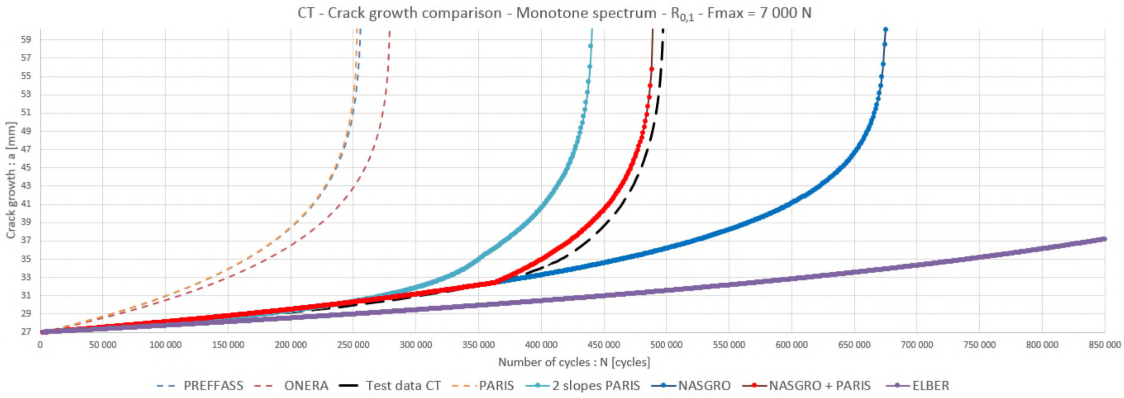


Figure 9 : Crack growth correlations on CT coupons

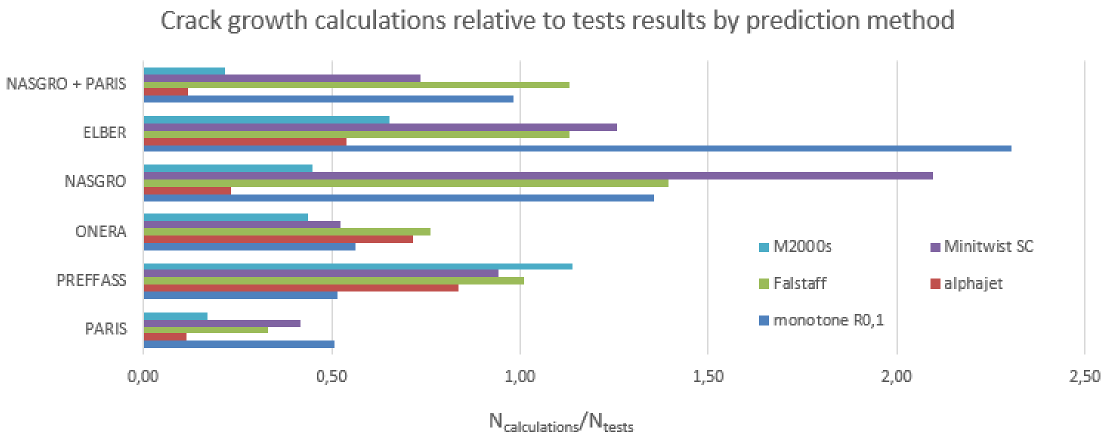


Figure 10 : Crack growth calculations relative to test results by prediction method

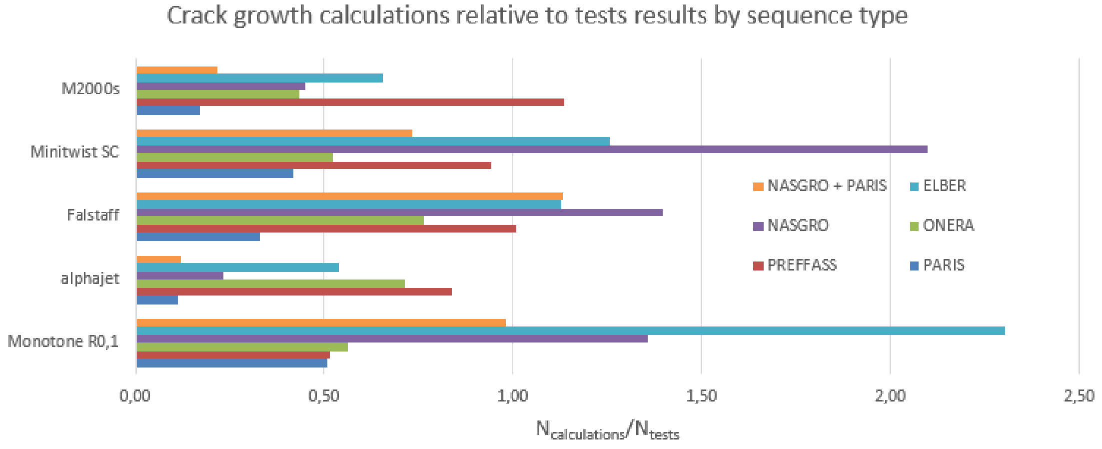


Figure 11 : Crack growth calculations relative to test results by sequence type

5.3. FATIGUE SPECTRUM : SEVERITY COMPARISON FOR G-COUNTER FLEETS (DGA AERONAUTICAL SYSTEMS)

DGA Aeronautical System studies the load factors from g-counter fleets thanks to reports completed for each flight. The totals of exceedances are then clustered by missions, versions and aircraft configurations.

VERSION : XXX - Unit : XXX

	LOAD FACTORS								
MISSION	-1.5	0	3	4.5	5.5	6.5	7.5	8.5	Flight Duration (h)
A1	0	176	2060	126	0	0	0	0	39,8
A2	0	0	6909	2000	0	0	0	0	5,5
A3	0	0	1000	0	0	0	0	0	1
B1	0	43	1783	43	0	0	0	0	23
B2	0	3846	12308	1538	0	0	0	0	2,6
C1	0	0	971	194	194	0	0	0	10,3

Figure 12 : Example of clustering from a g-counter report.

To draw a logarithmic fatigue spectrum, the calculated exceedances are then revised to a ratio of exceedances for 1 000 flight hours.

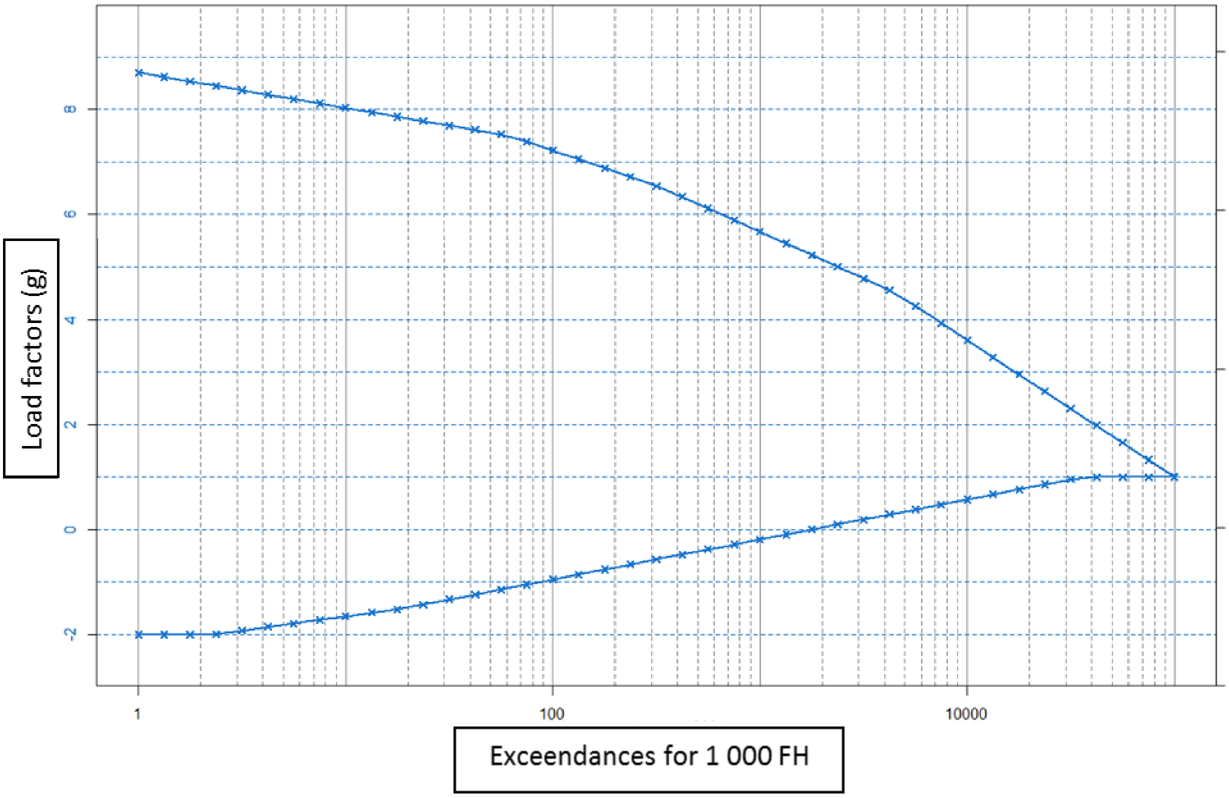


Figure 13 : Example of logarithmic fatigue spectrum drawn from g-counter reports

At this point, it was possible to graphically compare two spectra, from two successive years for example, but only when all the points of one spectrum were strictly above or below the points of the other one. In the case of crossing curves, the damage calculations were needed to conclude.

That is why a study was conducted to define the impact of shapes within a spectrum and the great difference of damage that can be induced by a slight variation of the curve.

The analysis showed that parts of the spectrum were more sensitive for the damage calculation. “Damaging fields” were considered to narrow down the investigation on those areas and to include the variations due to missions, configurations and areas of the aircraft concerned by the damaging calculation.

These “damaging fields” are plots of values on the Nz-exceedances graph which quantify the added damage (in %) according to the area where the spectrum curve lies.

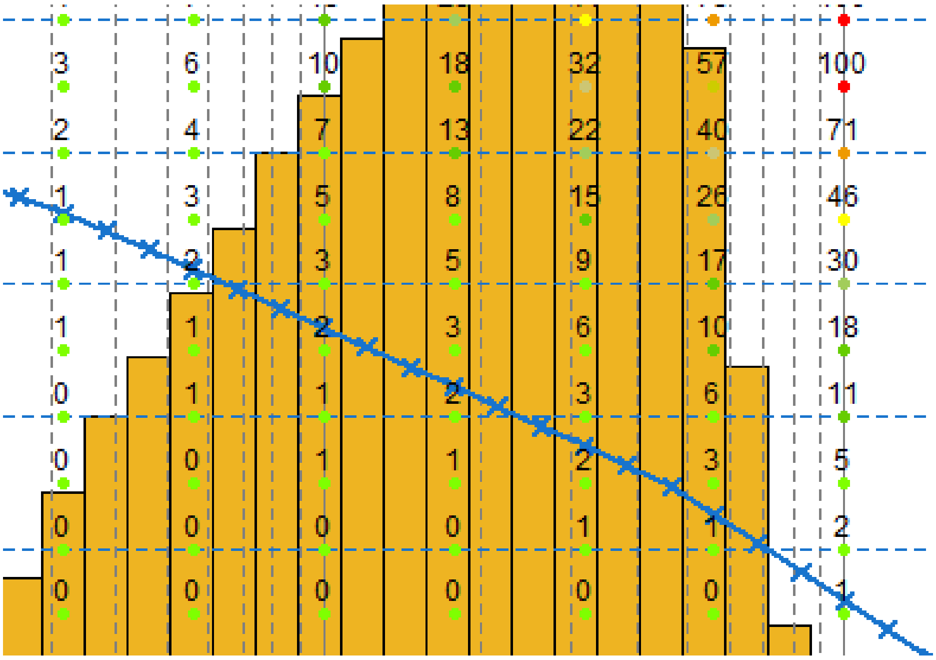


Figure 14 : Example of damaging field (zoomed in the most damaging area of the spectrum) used to underline the effect of the shape

At the end, a computing tool was finalized, providing a severity ratio between two selected spectra, building its calculation directly on the g-counter reports.

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