ASSESSING THE STRUCTURAL INTEGRITY OF POAF AIRCRAFT

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Abstract

This paper describes the methodology presently used on Portuguese Air Force aircrafts to assess fatigue life in order to maintain the aircrafts' structural integrity. This methodology uses experimental flight data collected by instrumentation systems, fitted in a discrete number of aircrafts. From the analysis of the recorded flight data, and other administrative data necessary to expand the results to a complete fleet, crack growth propagation models are developed. The resulting information is then used on the operational and logistical management of the complete fleet, i.e.: planning of the operational fleet usage; planning of non-destructive inspections and maintenance actions; management of all related logistical aspects. This methodology has been applied on VOUGHT A-7P CORSAIR II, LOCKHEED P-3P ORION, LOCKHEED C-130H HERCULES, SOCATA EPSILON and DASSAULT/DORNIER ALPHA-JET fleets. They will be exemplified by the DASSAULT/DORNIER ALPHA-JET application.

The Portuguese Flight Test Workgroup, being responsible for the development, implementation and operation of the instrumentation systems, has developed one cheap, modular and flexible Flight Load Data Recorder prototype, based in the application of the PC/104 standard. This system is also briefly presented.

1 Introduction

Safety and reliability of a structural component depend on factors that are initially associated to the design criteria and the quality requirements of the material and manufacturing processes. During its operational usage, safety and reliability of the components of civil or military aircrafts depend on prompt detection and immediate damage repair before critical dimensions, at which catastrophic failure occurs, are reached. In the late sixties and early seventies, structural failures in several aircrafts due to existing flaws, resulted in changes concerning the principles of design against structural fatigue, the damage tolerance analysis.

In this concept, it is assumed that fatigue cracks are already present in the structure at the initial construction of aircraft and the structure should be designed in such a way that these cracks cannot grow to a size that will lead to failure of the structure before it can be detected.

The damage tolerance design approach can be applied to the design of a new aircraft or by the reassessment of structural elements of existing aircrafts. The latter is common in the reassessment of the fatigue life of military aircrafts that were originally based on the Safe-life concept. The crack growth in a structural element depends primarily on the loads that are acting on the structural element [1]. Variations in the operational usage will, in general, result in different crack growth and therefore different times for the crack to grow to a specific size. In an aircraft fatigue management programme it is necessary to measure and record the data that allow the estimate of defects propagation over the life time of the aircraft, and taking into account the damage tolerance approach, based on the fracture mechanics, this corresponds to the applied stresses over a defect length [2].

The studies carried out in collaboration with Portuguese Air Force (PoAF), focused on VOUGHT A-7P CORSAIR II, LOCKHEED P-3P ORION, LOCKHEED C-130H HERCULES, SOCATA EPSILON and DASSAULT/DORNIER ALPHA-JET aircrafts. The aim of this study is to optimize the management of the fleet concerning the consumption of fatigue life and the logistics management and the actions of maintenance. In this paper only the analysis of DASSAULT/DORNIER ALPHA-JET aircraft will be described.

2 Flight Loads Monitoring

In the past and for a large period of time, the PoAF has performed the fatigue life assessment of its fleet by the usual methodology, i. e., by monitoring the load factor or the vertical acceleration in the centre of gravity.

To allow a more accurate evaluation of the aircraft structural integrity, one needs to know the loads applied on the various structural components, namely those defined as critical. This knowledge must be complemented with additional information allowing the characterisation of the mission type related with each flight and the establishment of a relationship between the applied loads and the flight manoeuvres. This methodology is generally named flight loads monitoring [3]. The equipment allowing the measurement, recording and analysis of the various monitored parameters are designated instrumentation systems.

2.1 Instrumentation Systems

The instrumentation systems used on the flight loads monitoring projects include airborne components and additional means used on ground. These systems allow simultaneous measurement of several parameters and therefore are also called multi-channel instrumentation systems. The number and characteristics of the several measured parameters depend on the particular objectives of each application project. Usually the measured parameters list includes:

- indicated or true airspeed;
- altitude;
- vertical acceleration on the centre of gravity or load factor;
- microstrains resulting from mechanical stresses applied on components and critical locations of the airframe, measured by full strain gauge bridges, in order to minimise the temperature effects and to achieve the best measurement sensibility [4].

The identification of critical components and locations is based on data supplied by the airframe manufacturer, on historical records and on data supplied by others users of that kind of aircraft.

The set of airborne components is essentially formed by sensors, signal conditioning units and one Flight Loads Data Recorder (FLDR). The FLDR is the core of these instrumentation systems because is responsible for filtering, sampling, digital conversion and data recording, according to a compression algorithm previously defined and using its internal memory. Usually this compression algorithm is the so-called peaks-valleys sequential recording or master/slave recording algorithm [5]. With this algorithm, several processes are defined, each one related with a primary parameter and several secondary parameters. For each process, when occurs a local maximum (peak) or minimum (valley) on primary parameter, its value is recorded together with the values of all associated secondary parameters. Note that normally the various processes are recorded separately and the initial and final values are always recorded. Additionally all measured parameters values are recorded periodically, for instance with a period of 30 seconds, permitting the flight profile evaluation.

The instrumentation system is complemented with support means on ground, which are normally a portable computer and software. These means are used directly with the airborne FLDR for system configuration and data retrieval. With the data collected on flight, they allow the data processing, analysis and management of the resulting data base.

To allow the extrapolation of valid results for fatigue life assessment of all fleet's aircrafts, the flight recorded data are complemented with administrative data, which are usually for each monitored flight:

- mission type;
- used configuration;
- take-off and landing load or weight;
- recorded values on the airborne g-counter;
- administrative flight time.

The knowledge of these administrative data, for each aircraft of the fleet, and the knowledge of the fatigue life consumption for each mission type and for flight period, which are obtained from the in-flight recorded data, allows the individual assessment of each aircraft of the fleet.

To guarantee the continuous generation of valid results, a set of procedures were defined and applied for the operation of the instrumentation system, processing and presentation of the resulting data. This procedures list describes all actions to be executed by those involved on the project, in order to generate and maintain a valid flight data base, representing the aircraft usage by the PoAF.

2.2 Usage of the Flight Data

The usage of the flight data depends on the nature of each application project.

2.2.1 Parametric Equations

From the flight data we can evaluate experimental parametric equations as a relationship between the measured microstrains, in each critical location, and the corresponding load factor, or other measured parameters. Those experimental parametric equations are then compared with those supplied by the aircraft manufacturer, defined as reference parametric equations. Then they are used to extrapolate the stresses applied in the several aircraft structural components.

2.2.2 Load Spectrums

From the measured load factor values are also derived the load spectrum, which presents, for a defined usage period, the load factor occurrences. Normally several load spectrums are defined as resulting from the flight data and classified by mission type. These experimental load spectrums are also compared with those provided by the aircraft manufacturer and used as reference.

2.2.2 Fatigue Life Assessment

As will be stated, the knowledge of the parametric equations and load spectrum is extremely relevant for the fatigue life assessment. The fatigue life assessment of critical components is based on fatigue tests performed on laboratory, using specimens and load sequences representing the aircraft usage. These load sequences for test purposes are obtained from flight data, using the knowledge of the flight mission type and its incidence on the total of performed missions, witch is called mission mix.

The usage representative load sequences can be obtained directly from the load spectrum or by combination of the in-flight recorded sequences. Until now, in all studies performed load sequences generated from load spectrums were used [6, 7]. These load spectrums only show the number of occurrences, for a given usage period, for which the various load factor values were exceeded, so they don't contain information on applied loads variation which defines the fatigue damage. Therefore, on the representative load sequence generation, corresponding to the usage in a given period, we assume that for each occurrence of a given load factor value n_z, corresponding to a stress value $\sigma(n_z)$ defined by the valid parametric equations, results one complete cycle defined by the stress $\sigma(n_z)$ and by the stress corresponding to a unitary load factor $\sigma(1)$. A study [7] has demonstrated that this is a conservative and acceptable approach.

3 DASSAULT/DORNIER ALPHA JET Structural Integrity Project

As an application example, the DASSAULT/DORNIER ALPHA-JET Structural Integrity Project will be presented. The PoAF

DASSAULT/DORNIER ALPHA-JET are normally used on advanced training and on ground light attack, and are operated by the Flight Squadrom 103 – *Caracóis* and Flight Squadrom 301 – *Jaguares*, both located in Beja Air Force Base, in the south of Portugal. Figure 1 shows one of these aircrafts.

Fig. 1. PoAF DASSAULT/DORNIER ALPHA-JET

3.1 Instrumentation System

Figure 2 shows a block diagram of the instrumentation system [8]. Table 1 contains the measured parameters list, including its main characteristics.

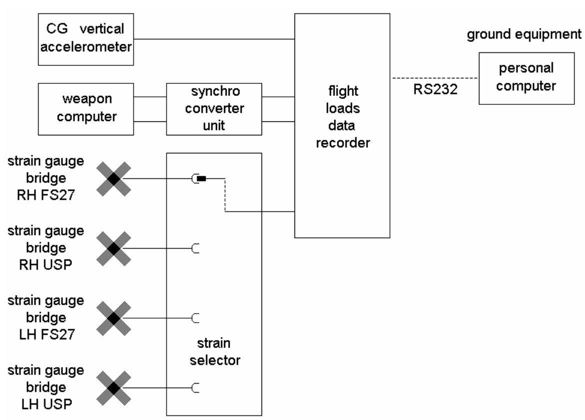

Fig. 2. Instrumentation system block diagram

parameter	signal source	Range
ind. airspeed	weapon	0 to 600 kts
altitude	computer	-500 to 50000 ft
vert. accelerat.	accelerometer	-4 to 9 g
microstrains (FS27 and USP)	4 full bridge strain gauges	-1000 to 4000 $\mu\varepsilon$
time	FLDR	1 sec. resolution

Tab. 1. Measured parameter list

As with the other instrumentation systems, the set of airborne components is essentially formed by sensors, signal conditioning units and one FLDR. According to the instrumentation system block diagram, the sensors installed on the DASSAULT/DORNIER ALPHA-JET are:

- one force balance linear accelerometer installed on main landing gear compartment and near the centre of gravity;
- four full strain gauges bridges, fitted in an installation module with protection against shocks, installed on location Upper Skin Panel (USP) and Fuselage Station 27 (FS27) of both wings. One of the installed strain gauge bridges is shown on figure 3.

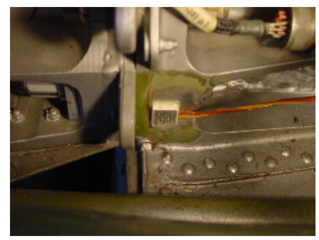

Fig. 3. FS27 full strain gauge bridge

Note that even if there are four full strain gauges bridges installed, the implemented instrumentation system only allows measurement of signal delivered from one of them. The selection of the active bridge is made by a connector's panel located near the FLDR.

The airspeed and altitude measurements don't require dedicated sensors, as the corresponding signals are obtained from the normal aircraft systems, namely from the weapon computer. These signals are in synchro format according to the standard ARINC 207.

For signal conditioning purposes, the filtering and amplification of the sensors' signals, including its excitation, are done internally on the FLDR. To allow the conversion of the synchro signals, (airspeed and altitude data), to voltages compatibles with the inputs of the FLDR a synchro conversion unit is used.

The used FLDR is a MOOG-ESPRIT ELAPS-IB and figure 4 shows a platform containing the FLDR, the synchro conversion unit and the active strain gauge bridge selection panel.

Fig. 4. Detail of the instrumentation system

On the DASSAULT/DORNIER ALPHA-JET FLDR there are five peaks-valleys sequential recording processes defined. In four of them, the primary parameter is always one of the four measured parameters and the secondary parameters are all others plus the time. The fifth process is named flight profile and implies the periodical (30 seconds) recording of all measured parameters plus time.

As usual, the instrumentation system is complemented with support means on ground, which are one portable computer and software. The flight recorded data are complemented with administrative data, as mentioned above.

A procedures list was also defined and applied for the operation of the DASSAULT/DORNIER ALPHA-JET instrumentation system, data processing and results presentation. It describes all actions to be executed by those involved on the project, in order to produce and maintain a valid flight data base, representing the DASSAULT/DORNIER ALPHA-JET usage by the PoAF.

3.2 Data Processing and Analysis

As an example, figure 5 shows the time evolution of some of the measured parameters in one of the monitored flights.

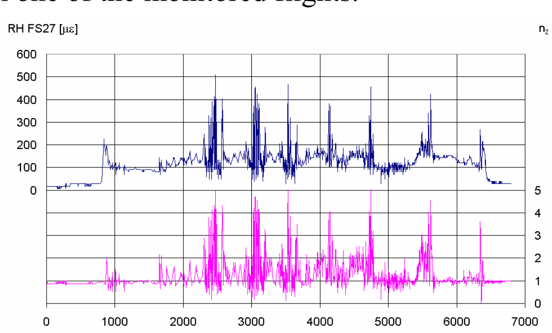

Fig. 5. Example of flight data (microstrains right FS27 – blue; load factor – pink)

As mentioned before, from these flight data we can evaluate the relationship between the measured microstrains in each critical location and the corresponding load factor, i. e., the experimental parametric equations. The experimental results obtained on right wing FS27 (figure 6) show linear correlation between stresses and load factor. The resulting experimental parametric equation relates the stress σ, in MPa, with the load factor n_z:

right wing FS27 $\quad \sigma = 47.8\, n_z \quad$ (1)

left wing FS27 $\quad \sigma = 39.1\, n_z \quad$ (2)

The reference parametric equation, supplied by the airframe manufacturer, for this critical location, is:

reference FS27 $\quad \sigma = 51.1\, n_z \quad$ (3)

Based on these results, figure 7 shows the experimental and reference relationships between stresses and load factor for critical location FS27.

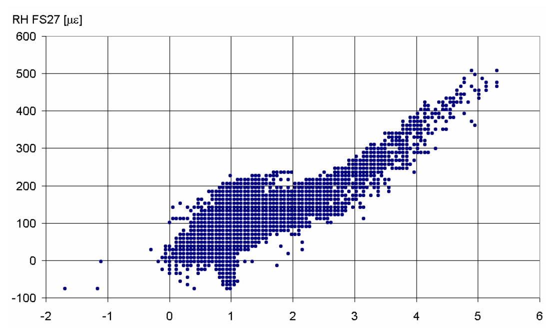

Fig. 6. Microstrains on right wing FS27 versus load factor

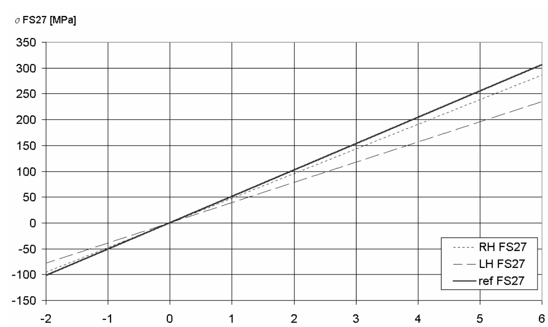

Fig. 7. FS27 parametric equations relationship

Note the good agreement obtained between the experimental results and those obtained from the reference parametric equation, supplied by the airframe manufacturer and used before on structural integrity analysis of these aircrafts. Figure 8 shows the results obtained for one of the USP critical locations. The non-linear correlation, observed in both wings, is the result of wing deflection and torsion. On the presented results, the large number of negative load factor occurrences on the data collection period resulted from a demonstration mission type.

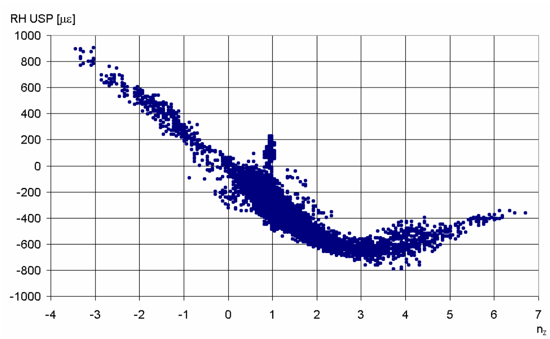

Fig. 8. Microstrains on right wing USP versus load factor

Figure 9 shows several load spectrums resulting from the flight data and classified by mission type. The reference load spectrums supplied by the airframe manufacturer are also presented. The presented results indicate that, in terms of fatigue life consumption, the usage severity depends on the mission type.

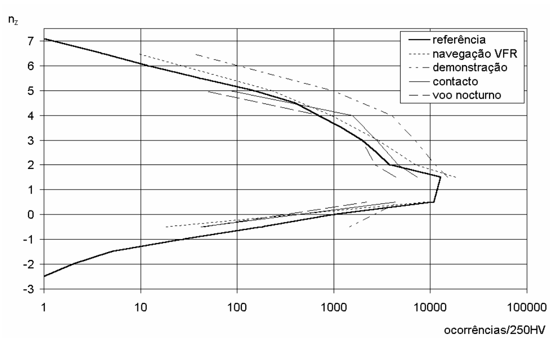

Fig. 9. Load spetrums for 250 FH and per mission type

The methodology and results from the DASSAULT/DORNIER ALPHA-JET fatigue life assessment of critical components is described in the next section.

4 DASSAULT/DORNIER ALPHA-JET Fatigue Life Assessment

As mentioned before, the fatigue life assessment of critical components is based on fatigue tests performed in laboratory, using specimens and load sequences representing the aircraft usage.

4.1 Material and Specimens

Fatigue crack growth tests under variable amplitude loading were carried out on aluminium alloy specimens, 100 mm width and 8 mm thickness, with a hole diameter of 20 mm containing two initial cracks used to initiate the fatigue cracks, as shown on figure 10. Precracking was carried out with the sequence loading used on the tests.

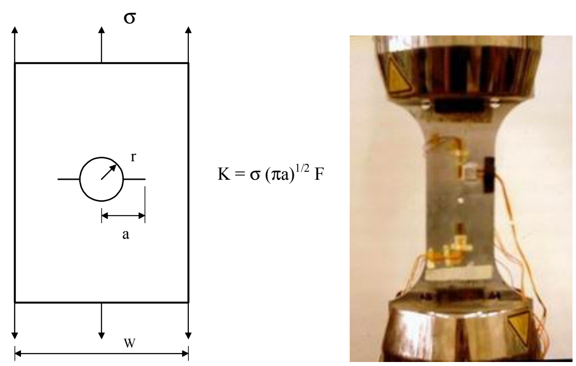

Fig. 10. Specimen configuration (left) and instrumented specimen mounted on a servo-hydraulic machine (right)

The aluminium alloy 2014-T651, used in the test program, is also present at the critical location in the aircraft. Its properties σ_{yield}=441MPa and σ_{ult}=490MPa were used and taken from [9].

All the specimens were instrumented with a full strain gauge bridge, as shown also on figure 10. The fatigue tests were carried out on a servo-hydraulic fatigue testing machine. The stress sequences were generated and controlled by a computer-based function generator. Crack growth was measured at regular intervals, from the specimen surface through two travelling microscopes at both sides of the specimen in order to monitor the crack length.

3.2 Load Sequences

The following stress sequences were used:

- PoAF based on g-counter readings and constituting the main raw data.
- PoAF2 a stress sequence derived from PoAF with the same rainflow content as

the previous spectrum but with a reduced number of points, as shown on figure 11.

- PoAFO a stress sequence containing all load cycles of PoAF but without the load cycles where $\sigma_{max}<\sigma_{op30}$, where σ_{op30} is the opening stress level of the cycle where the maximum stress level is reached, calculated at regular intervals. The basis of the above consideration is the Crack Severity Index (CSI) concept [10], developed at the Dutch National Aerospace Laboratory (NLR), which has been proved to be applicable to assess the damage tolerance of fighter aircraft.

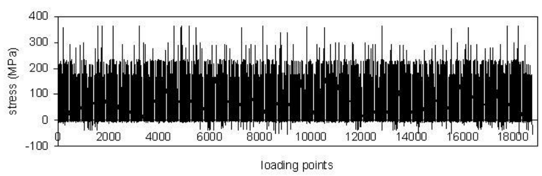

Fig. 11. Load sequence PoAF2

The stress level distribution within each load sequence, respectively PoAF, PoAF2 and PoAFO are shown on table 2. It shows that the missing points in the stress sequences PoAF2 and PoAFO are mainly on the low level stress values.

stress level (MPa)	number of turning points		
	PoAF	PoAF0	PoAF2
X > 200	700	700	700
100 < X < 200	2341	2341	2341
50 < X < 100	7195	7435	6297
0 < X < 50	9508	3346	5562
X < 0	4211	2423	3905
missing points		7710	5150
total	23955	16245	18805

Tab. 2. Stress level distribution for each spectra

4.3 Experimental Results of Crack Length

Testing variable amplitude loading in servo-hydraulic machines, is a delicate task, since in general the machine hardware and software only controls constant amplitude loading. Therefore, a careful control of the PID parameters must be achieved in order to obtain the assigned loading to the specimen. For this purpose, the load cell signal was acquired during part of the flight sequence and compared with the assigned signal and a monitoring of the maximum and minimum peak loads for each flight load sequence was registered.

Figure 12 shows a fairly good similarity between two specimens tested with the same load file. So it seems that the reproducibility of experimentation is quite good.

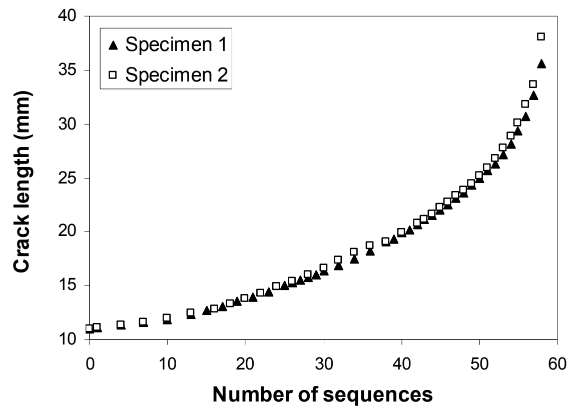

Fig. 12. Experimental results of crack growth

4.4 Simulation of Crack Growth

The crack growth per cycle depends on the stress distribution around the crack tip, which is characterized by the stress intensity factor.

For the specimen with two cracks emanating from a circular hole of radius r in a finite plate under tension σ, the following stress intensity factor equation was used [11]:

$$K = F\left(\frac{a}{W}, \frac{r}{W}\right)\sigma\sqrt{\pi a} \tag{4}$$

where a is the crack length and W the specimen width. The correction factor F that has been used on the following calculations, as shown on figure 13, has been interpolated from the previous values presented in literature [12].

The crack growth under service loads can be determined by summation of the crack growth caused by each individual load excursion. It is usual to correlate the crack growth rate (da/dN) with the stress intensity factor range ($\Delta K=K_{max}-K_{min}$) of a load variation. The crack growth rate depends also on stress ratio ($R=\sigma_{min}/\sigma_{max}$), therefore empirical crack growth relations were established that account for the effect of *R*. There are simple crack growth laws that can only be used for constant amplitude loading, and other crack growth laws that take into account the variation of the stress ratio *R*. These crack growth relations can be divided into those that describe the relation empirically, such as the Forman law, and laws that are based on the idea of crack closure, such as Elber [13] and NASGRO law [9]. It has been also noted that overloads and under loads can affect the crack growth and several empirical models have been proposed to take into account these effects on a variable amplitude crack growth test, such as Wheeler [1, 2] retardation factors. These models are based in the crack tip plasticity but need extensive testing in order to determine adjusting coefficients to take into account the retardation or acceleration of crack growth due to respectively overloads and under loads.

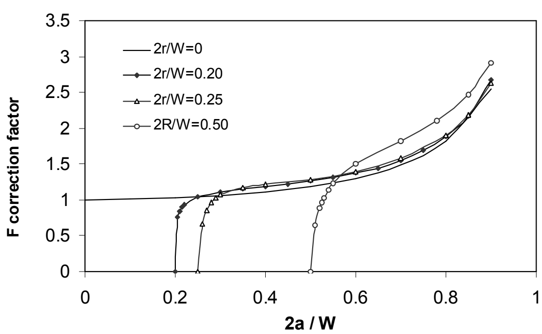

Fig. 13. Correction factor for stress intensity factor

Elber [13] introduced the concept of crack closure, which occurs as a consequence of crack tip plasticity. The effective stress intensity factor range ($\Delta K_{eff}=K_{max}-K_{op}$) was introduced and relates to the stress range that contributes to crack growth. Other mechanisms have been proposed to explain crack closure but will not be addressed in this study [3].

One of the equations that relate crack growth rate data with the effective stress intensity factor range is the NASGRO 2.0 equation [9]:

$$\frac{da}{dN} = C_n \frac{(1-f)^n}{(1-R)^n} \Delta K^n \frac{\left(1-\frac{\Delta K_{th}}{\Delta K}\right)^p}{\left(1-\frac{\Delta K}{(1-R)K_c}\right)^q} \tag{5}$$

where $f=(K_{op}/K_{max})$ is the crack opening function, K_c is the critical stress intensity factor and ΔK_{th} is the threshold stress intensity factor range which is approximated by a function of the stress ratio and of the threshold stress intensity factor range at $R=0$ (ΔK_0), by the equation (6):

$$\Delta K_{th} = \Delta K_0 \left(\frac{4}{\pi}\tan^{-1}(1-R)\right)\left(\frac{a}{a+a_0}\right)^{1/2} \tag{6}$$

The empirically derived constants, *C*, *n*, *p* and *q* were obtained from the NASGRO user's manual [9] and are shown on table 3.

C_n	n	K_c	p	q	ΔK_0	α
0.185e-10	2.8	973	0.5	1.0	94	1.5

Tab. 3. NASA/FLAGRO constants, SI units

The NASGRO law is based on the concept of crack closure and several models have been proposed to determine the stress opening level within a cycle.

In this study the crack opening function, *f*, for plasticity induced crack closure was defined by Newman [14, 15], and is given by the equations (7) and (8):

$$f = \frac{K_{op}}{K_{max}} = \begin{cases} A_0 + A_1 R + A_2 R^2 + A_3 R^3 & R \geq 0 \\ A_0 + A_1 R & -2 \leq R < 0 \end{cases} \tag{7}$$

where:

$$\begin{aligned} A_0 &= \left(0.825 - 0.34\alpha + 0.05\alpha^2\right)\left(\cos\frac{\pi\sigma_{max}}{2\sigma_0}\right)^{1/\alpha} \\ A_1 &= \left(0.415 - 0.071\alpha\right)\left(\frac{\sigma_{max}}{\sigma_0}\right) \\ A_2 &= 1 - A_0 - A_1 - A_3 \\ A_3 &= 2A_0 + A_1 - 1 \end{aligned} \tag{8}$$

σ_{max} and *R* are respectively the maximum stress and the stress ratio for each load cycle, σ_0 is the flow stress taken as the mean value between

σ_{yield} and σ_{ult} and α is the plane stress/plane strain constraint factor.

The retardation model is based on a cycle by cycle analysis of crack growth and on the determination of the actual part of the cycle that causes crack propagation, through the effective stress intensity factor. The possible retardation of a load excursion is accounted for as long as the affected zone of subsequent loads has not grown through the peak-load affected zone. A more detailed description of the model is given in reference [10].

4.5 Comparison Between Predicted and Experimental Crack Growth

The crack growth obtained for the three spectra load can then be analyzed and compared with the predicted crack growth. From the three very close curves, presented on figure 14 and concerning PoAF stress sequence, we can conclude that:

- position of the stress variations within the stress sequence has a very limited influence (PoAF2 creation), since the two load spectra with the same rainflow content gave similar crack growth. The two curves are almost superposed, as shown in figure 16 with a small difference of about 5% on the total number of flight hours.
- stress levels where the maximum stress is below the opening load have minor influence on crack growth, therefore the opening stress used in the CSI-calculation is a valid concept. The duration of a test with an omission sequence is obviously shorter than the original file with the same final results so this conclusion would be important.

Again, all curves of the predicted crack growth are very close with no visible difference and reveal that the crack growth lives are exactly equal (4900 flight hours) for every PoAF load sequences used on this study. The conclusion about the severity of the sequence can now be analyzed and compared with other sequences for different aircraft usage.

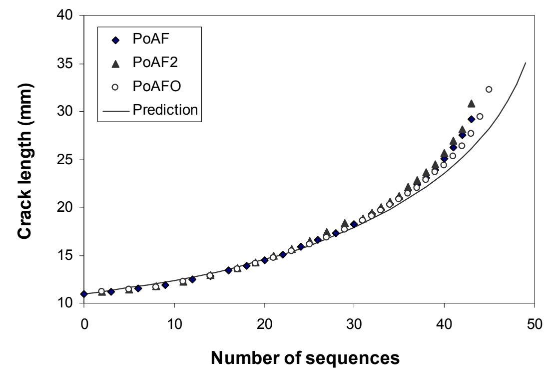

Fig. 14. Predicted and experimental crack growth for PoAF, PoAF2 and PoAFO sequences

5 PC/104 Flight Loads Data Recorder

The Portuguese Flight Test Workgroup (*Grupo de Trabalho de Ensaios em Voo* – GTEV), being responsible for the development, implementation and operation of the instrumentation systems, has developed recently one cheap, modular and flexible FLDR prototype, based in the application of the PC/104 standard [16].

In the development of the PC/104 FLDR prototype several available PC/104 modules were used:

- one EUROTECH PC/104 CPU module, PENTIUM 233Mhz processor, with a Disk-on-Chip memory module;
- two EUROTECH PC/104 data acquisition modules, each one with eight analog inputs and four analog outputs;
- one EUROTECH PC/104 28Vdc power supply module;
- two dual full strain gauge bridge amplifier.

The system was encapsulated using a EUROTECH DURACHASSIS NEMA and was programmed and configured in a DOS environment. The installed connectors and pin layout are similar to those used in the MOOG-ESPRIT ELAPS-IB, installed in the instrumented PoAF SOCATA EPSILON aircraft [17]. The main idea is the use of this aircraft in flight testing of the FLDR prototype, without any special modification. These flight tests are delayed and will start in the summer of 2006.

The PC/104 FLDR prototype has limited data acquisition capacity, due to the utilisation of the available modules. Software has been developed allowing data recording in its memory using two different algorithms: periodical recording of all measured data; sequential peaks-valleys data recording, permitting the full definition of the recorded processes. For the moment the recorded data are retrieved with a PC serial connection.
Presently, environmental tests according to the MIL-STD-810F standard [18] are being performed, in order to access the compatibility of the FLDR prototype with the airborne environment. These tests are also necessary in order to develop an eventual certification plan.

6 Conclusions

The studies on aircraft structural integrity carried out with the cooperation of Portuguese Air Force have shown:

- correction of the methodology used for the assessment of the structural integrity of the aircrafts;
- the methodology allowed monitoring of fatigue life of each aircraft using damage tolerance approach;
- a optimization of maintenance procedures and rational use of aircraft fleet.

The development of a cheap and modular PC/104 FLDR prototype was carried out, the first tests are being conducted and soon it will be used in flight tests, using a PoAF SOCATA EPSILON aircraft.

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