Prediction of Fatigue Crack Growth in Airframe Structures

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ABSTRACT

The paper describes the general proposal, function and performance of a prognostics system for fatigue crack growth in airframe structures. Prognostic capabilities are important to a superior Structural Health Monitoring System (SHM). An aim of the prognosis is estimation / prediction of a system / subsystem / structure remaining life, i.e. the time at which damage (crack, corrosion, wear, delamination, disbonding, etc.) will result in a failure of the considered element.

1. INTRODUCTION

Fatigue damage and its consequences are the most serious structural design and maintenance issues that have to be addressed. Several philosophies of how to decrease consequences of a fatigue hazard have been developed and applied. Serious aircraft accidents due to fatigue have contributed to this development and have started research efforts in this area. Two main philosophies for aircraft structure design are used nowadays:

Safe Life - an extremely low level of risk is accepted through a combination of testing and analysis that the part will ever form a detectable crack due to fatigue during the service life of the structure.

Damage Tolerance - structure has the ability to sustain defects safely until the defect is detected and repaired.

Structural Health Monitoring (SHM) - represents the next advanced step in structural damage monitoring and maintenance planning. An occurrence of structural damage is monitored by a sophisticated automated system. Its usage does not demand additional time from inspections and qualified personal. A significant part of a SHM system and its application in aerospace is prognostics. It shifts a structural maintenance program to an advanced level and brings significant benefits for an aircraft operator like efficient maintenance planning, effective aircraft usage, service cost decrease, safety increase, etc. Its application opens a new approach to aircraft design and brings a new advanced philosophy of structural lifetime estimation.

2. ENTIS PROJECT DESCRIPTION

This paper describes our approach to fatigue damage prognostics development. It is based on results from the ENTIS project supported by the Ministry of Industry and Trade of the Czech Republic. The main goal of this project is SHM system development, particularly its real application form, capabilities, conditions and issues definition. Experimental testing is a basic part of this project. Fatigue tests of aircraft structure parts have been done and structural damage has been monitored by an ultrasonic method. All structural specimens are parts of an L-410 UVP-E airplane (Figure 1), which is an all-metal high-wing monoplane powered by two turboprop engines. The airplane is certified in the commuter category in accordance with (FAR) PART 23 requirements. All structural specimens are considered critical parts of the aircraft structure in the sense of fatigue damage. The fatigue test arrangement is shown in Figure 2.

Figure 1. Aircraft Industries a. s. L-410 UVP-E airplane

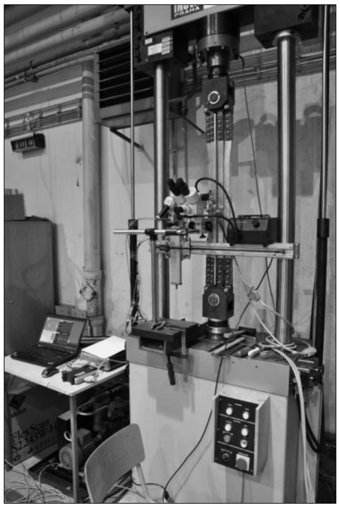

Figure 2. Wing Spar Fatigue Test

For the purpose of generation and registration of ultrasonic waves shear plate PZT actuators were used (Figure 3). Characteristics of the particular PZT actuators from Noliac used are shown in (Table 1). The actuators are characterized by their small dimensions, low weight, and low cost. This allows installation of large and non-expensive sensor arrays on an airframe with very small impact on the structure performance and aerodynamic properties. Moreover, the small dimensions of the PZT elements are suitable for integration of the sensor array on a flexible strip, which significantly accelerates the process of sensor array installation on the monitored structure.

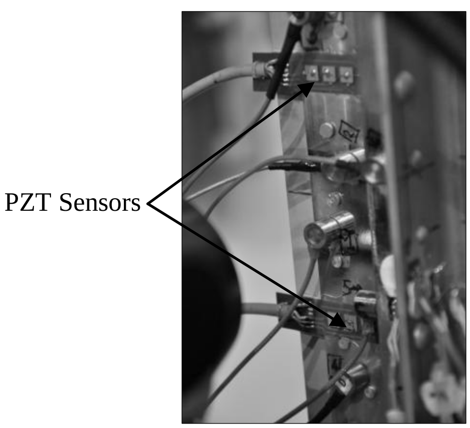

Figure 3. Wing Spar Fatigue Test

Type	Length [mm]	Width [mm]	Height [mm]	Maximum voltage [V]	Free stroke [μm]	Capacitance [pF]
CSAP02	5	5	0.5	+/-320	1.5	830

Table 1. PZT Actuator Parameters

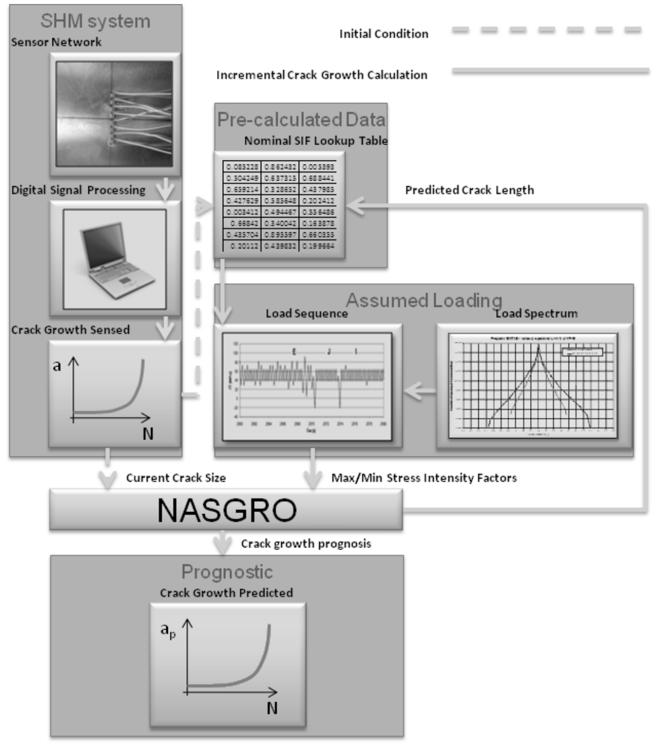

Figure 4. Block scheme of the crack growth monitoring system

The block scheme of the advanced signal processing for the automated monitoring of fatigue damage of the particular aircraft structural part is in Figure 4. A sparse sensor array enclosing the monitored area (Hot Spot) is used for collection of the data used for evaluation of the actual state of the structure. Therefore, the automated defect detection / sizing is based on evaluation of changes in direct signal paths, i.e. signals between individual pairs of PZT actuators where one PZT actuator works as source of the ultrasonic wave and the second one as sensor. First, Signal Difference Coefficients (SDCs) for individual paths are calculated evaluating differences between baseline and actual signals measured on the monitored structure. These SDCs represent a Damage Index (DI), which gives information on the extent of the structure damage. The defect occurrence is indicated and defect size is estimated using an Artificial Neural Network, which transforms a feature vector capturing significant features (DIs) of an identified defect to demanded parameters (defect occurrence, defect size), which are used as inputs of prognostics algorithm.

3. FATIGUE CRACK GROWTH PROGNOSTICS

The fatigue crack growth prognosis is used for RUL prediction. The RUL is defined by the crack length reaching its critical size limit. The concept of the crack growth prognostic algorithm is shown in (Figure 4). Input of the prognostic algorithm consists of the crack length observed during a particular inspection, i.e. measured by the SHM system, and the typical loading sequence. Several algorithms for the crack growth calculation can be found in literature (Beden & Abdullah & Ariffin, 2009). Those algorithms are based on various approaches: fracture mechanics & empirical models (Beden & Abdullah & Ariffin, 2009), or data based models (Forman, R. G. & Shivakumar, V & Cardinal J.W. & Wiliams, L.C. & McKeigham, P.C. 2005). Suitability of these approaches for a particular application depends on the actual type of loading, type of structure and other boundary conditions and available inputs for tuning of the crack growth model.

Our developed prognostic system uses the NASGRO equation as the crack growth model (*NASGRO Reference Manual, version 4.2*), Augustin (2009). This approach was selected for the following reasons: (1) This equation is widely used in aerospace, (2) All inputs required for the crack growth model related to duralumin aircraft structure are available in literature, (3) Influence of variable amplitude loading is accounted for in the NASGRO model, (4) This algorithm solves the crack growth in all three phases of the crack propagation (crack initiation, stable crack growth, unstable crack growth).

The SHM system is focused on the so called Principle Structure Element (PSE). PSE's are those elements of a primary structure which contribute significantly to carrying flight, ground, and pressurization loads, and whose failure could result in catastrophic failure of the airplane. Sensors are installed on hot-spots in order to provide information on the actual status of the structure integrity, (i.e. actual length of the fatigue crack). Each hot-spot is treated separately, (i.e. prognosis of the crack growth for particular hot-spot is done without accounting for the effect of presence of other cracks out of the hot-spot). However, each hot-spot may contain multiple cracks.

3.1. Input Crack Length

The crack length observed during a particular inspection is used as an initial crack length for the prognostic algorithm. Evaluation of the crack length is done using a feature vector, which consists of DIs. The vector is applied on an input of an Artificial Neural Network as was described above. The prognostic algorithm propagates the initial crack length into the future using a typical loading sequence. Thus, the prognosis is done for each inspection, i.e. for each crack length observed.

3.2. Typical Loading Sequence and Flight Loading Spectrum

The type of the wing flange specimen loading is the same as the real loading on a real aircraft wing flange. It is a combination of a bending moment and an axis force.

The loading sequence represents a series of loading cycles (Figure 5) affecting the structure. Duration of a loading cycle is constant for the whole sequence, i.e. 1/3s. Each cycle is described by its maximal and minimal stress levels [σ_{min}, σ_{max}]. The maximal and minimal stress levels are expressed as multiples of a nominal stress σ_0. Thus, we have a pair of numbers {n_{min}, n_{max}} so called load factors for a single loading cycle.

A typical flight spectrum for a particular aircraft (Figure 6) is used in order to define a typical flight loading sequence. The typical flight spectrum was defined according to FAA AC 23-13A. The total loading spectrum during the flight composes two loading spectra: wind gust and maneuvers.
The loading sequence can be derived from the typical flight spectrum in various ways. Most often the block loading sequence or random loading sequence is used:

<u>Block loading sequence</u> – loading cycles of the same amplitude are organized in blocks, which consist of a number of loading cycles.

<u>Random loading sequence</u> – loading cycles, which have various amplitudes, are randomly organized in the loading sequence (Figure 7).

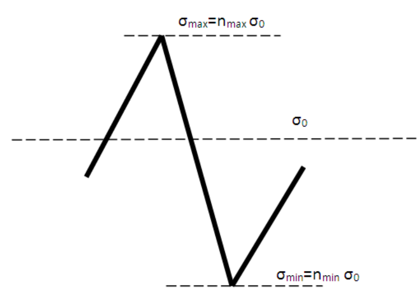

Figure 5. Typical Loading Cycle

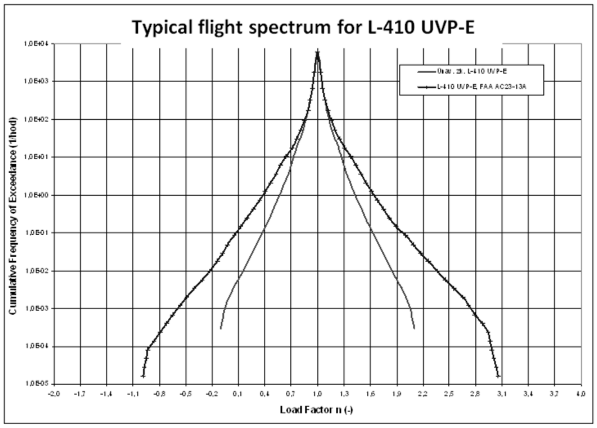

Figure 6. Typical Flight Spectrum for L-410 UVP-E

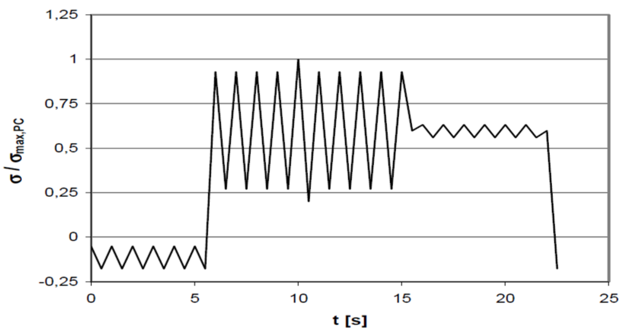

Figure 7. Random Loading Sequence for One Flight

3.3. Stress Intensity Factor

The calculation of the stress intensity factor (SIF) at the crack tip for a nominal load is based on a finite element analysis (FEA), which requires knowledge of the structure and crack geometry. The FEA provides stress energy release rates for the tip from where stress intensity factor $K_{\sigma 0}(N)$ can be calculated. The FEA analysis is time consuming and too complex to include in an online system. Online FEA calculation of the stress intensity factor $K_{\sigma 0}(N)$ is replaced by a lookup table for our purpose.

3.4. Crack Growth Equation

Calculation of the crack increment for a particular load cycle is done using the NASGRO equation of fracture mechanics Eq. (1).

$$\frac{da}{dN} = C\left[\left(\frac{1-f}{1-R}\right)\Delta K\right]^n \frac{\left(1-\frac{\Delta K_{th}}{\Delta K}\right)^p}{\left(1-\frac{K_{\max}}{K_c}\right)^q} \qquad (1)$$

where C, f, ΔK_{th}, K_{max}, K_c, p and q are model parameters given by the structure material and geometry.

4. Prognostic Algorithm Performance

Two sets of experimental data from laboratory fatigue tests of wing flanges (Figure 2) were used as inputs for the prognosis algorithm performance assessment. In both tests, a two-tip crack 2 x 1.27 mm long was initiated at the rivet hole. One tip pointed to edge of the flange – external crack, and the other one to the flange axis of symmetry - internal crack. The intention of the experiment was to 1) evaluate performance of the crack measurement system and 2) obtain fatigue crack growth data. Ideally the only two major cracks near the measurement site would have been those initiated intentionally. This was the case for the first set of experimental data. However, during the second experiment, several other cracks formed within the test article, contaminating the experimental data. Nonetheless, the performance of the fatigue crack prognosis model is compared with both sets of experimental data.

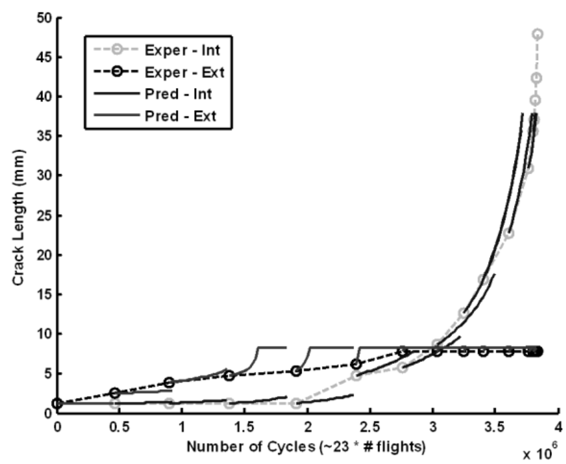

Figure 8. Fatigue Crack Prognosis with First Set of Experimental Data

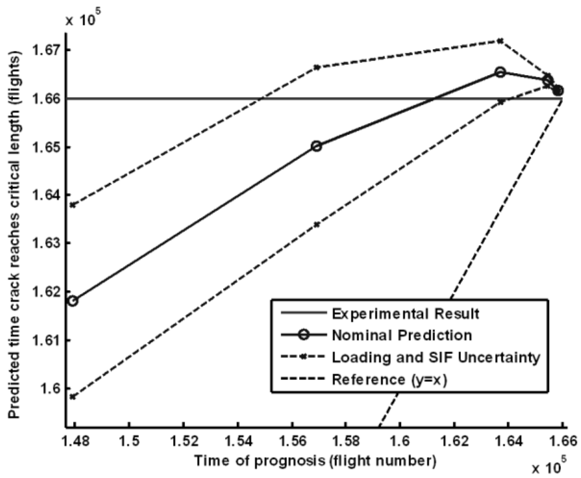

Figure 9. Predicted Times to Internal Crack Length of 38 mm (first experimental data)

Figure 8 and Figure 9 show the outcome of the fatigue crack growth prognostic algorithm applied to the first set of experimental data. In Figure 8, the dashed data lines represent the crack lengths measured by fractography at specific times during the test for the internal and external cracks. The solid lines indicate the predicted crack growths where the crack growth prognosis was initiated at each fractography measurement of the crack length. Each prognosis proceeds no farther than 20,000 flights into the future (where each flight consists of approximately 23 loading cycles).

Four uncertainties are considered in our work: Loading uncertainty, SIF uncertainty, Prognostics model uncertainty and Measured crack length uncertainty.

Figure 9 shows the predicted times the crack will reach the critical length. The solid curve represents the actual time predicted by the algorithm. The dashed curve includes an offset of this prediction accounting for minimum uncertainties due to loading and SIF uncertainties:

<u>Loading uncertainty</u> – *One of sources of uncertainty in crack growth prognosis is the assumed loading sequence (random loading sequence drawn from the typical loading spectrum for this type of aircraft). For several initial crack lengths, several different crack prognoses were run, each using a random ordering of the selected loading sequence. The upper end of the 95% confidence interval on the standard deviation of the times from these prognoses was then calculated.*

<u>SIF uncertainty</u> – *A key parameter in fatigue crack growth models is the crack Stress Intensity Factor (SIF). For purposes of this project, SIFs are evaluated using a Boundary Element Analysis software package. Crack geometries are entered into the software, and post-processing yields an estimate of the stress intensity factor. The time to setup and execute this stress analysis is too long to perform for each step of the fatigue crack prognosis. Therefore, a lookup table has been generated for selected crack lengths (minimum, medium, and maximum). The lookup table has been generated under the assumption that a single stress may be used to determine a baseline SIF, and this baseline can then be scaled by a load factor to calculate an actual SIF as a function of crack length and loading condition. The SIF for particular crack length is then calculated using the values tabulated in the look-up table and an interpolation technique, which is the source of the uncertainty in the SIF estimation.*

Prognostics model and measured crack length uncertainties are not described in this paper.

The solid horizontal line indicates the time to reach this crack length as indicated by the experimental data. In the figure it can be seen that as the time until the internal crack reaches the critical length gets smaller, the identified uncertainties do not account for the error between the prediction and the experimental results.

Figure 10 shows the prognostic algorithm applied to the second set of experimental data. As mentioned before, the second set of experimental data was contaminated by additional unintended cracks propagating during the test. Predicted times to reach the critical crack length are presented in Figure 11.

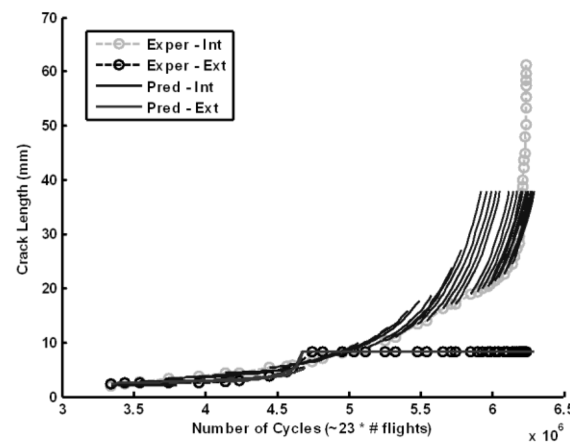

Figure 10. Fatigue Crack Prognosis with Second Set of Experimental Data

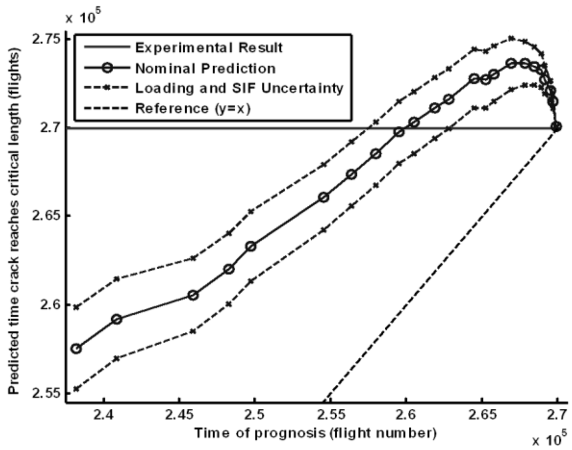

Figure 11. Predicted Times to Internal Crack Length of 38 mm (Second Experimental Data)

5. CONCLUSIONS

The prognostic system allows the possibility of fatigue damage growth prediction and mitigates an issue of a corrective maintenance planning. Our prognostic system design is based on the traditional method (NASGRO equation) used for crack propagation modeling for damage tolerance relating analyses. Prognosis of simultaneous propagation of multiple cracks is possible. In this case, multi-dimensional lookup tables for SIF estimation have to be used in order to account for interaction between individual cracks. A connection of the SHM and Prognostics system brings a novel advanced capability of an interactive fleet management and prognostics regarding fatigue damage growth. It shows a new dimension of maintenance planning and organization of all maintenance tasks are done in real time that is estimated by aircraft operators regarding their requirements and minimal costs.

The accuracy of the crack growth prediction is influenced by several parameter uncertainties. This was demonstrated by the prognostics applied on two fatigue test results. Those parameters uncertainties (loading, SIFs, crack size estimation, etc.) have to be considered. The prognostics output could be influenced by addition boundary conditions (additional cracks – flange fatigue test 2). In this case prognostics results do not follow the real crack propagation curve exactly. A solution is monitoring of changes of boundary conditions and adjustment of prognostics input parameters with regard to these changes.

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NOMENCLATURE

Da	Crack size increment
DI	Damage Index
f	Opening Function
FEA	Finite Element Analyses
K_c	Fracture Toughness
$K_{\sigma 0}$	Stress intensity factor
σ_{min}	Minimal stress level
σ_{max}	Maximal stress level
n_{min}	Cycle minimal load factor
n_{min}	Cycle maximal load factor
PSE	Principle Structure Element
PZT	Lead Zirconate Titanate
RUL	Remaining Usage Life
SDC	Signal Difference Coefficient
SHM	Structure Health Monitoring
SIF	Stress Intensity Factor

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BIOGRAPHIES

Jindrich Finda (March 28th, 1980) earned his Master of Science in Aircraft Design from Brno University of Technology, Faculty of Mechanical Engineering, Institute of Aerospace Engineering in 2003 and his PhD. in Methods for Determination of Maintenance Cycles and Procedures for Airplanes/ Airplane Assemblies from Institute of Aerospace Engineering in 2009. Jindrich Finda works as a Scientist II R&D. His work is aimed on the SHM system development, (developing algorithms for advanced ultrasonic signal / image processing, and algorithms for automated defect detection, localization, size evaluation and prognosis of the defect growth, SHM integration into aircraft maintenance plan).

Andrew Vechart earned his Master of Science in Computation for Design and Optimization from Massachusetts Institute of Technology in 2011 and his Bachelor of Science in Mechanical Engineering and Physics from the University of Wisconsin – Milwaukee in 2009. He has been a R&D scientist with Honeywell focusing on vehicle health management projects since 2011. At Honeywell, he performed prognostic algorithm development for a structural health monitoring (SHM) application. He is also program manager and project engineer for a Honeywell program to update existing and design new firmware and software for an FPGA application. His interests include structural health management, embedded systems, and scientific computing.

Radek Hedl (November 19th, 1973) earned his Master of Science in Cybernetics, Automation and Measurement from Department of Biomedical Engineering, Faculty of Electrical Engineering and Computer Science, Brno University of Technology and his PhD. in Cybernetics and Computer Science from Department of Biomedical Engineering, Faculty of Electrical Engineering and Computer Science, Brno University of Technology. Radek Hedl works as a Sr. R&D Scientist in the Mechanical & Simulation Technologies group. In particular, he is a leader of CBM/SHM sub-group. His responsibilities include development of the resource and technology capabilities of the CBM/SHM group in Brno, involvement in definition of long term strategy technology roadmaps in the CBM/SHM area, and leading R&D projects. His R&D activities include developing algorithms for advanced ultrasonic signal / image processing, and algorithms for automated defect detection, localization, size evaluation and prognosis of the defect growth.