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STRAIN GAUGE CAPABIILITIES IN CRACK DETECTION

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ABSTRACT

Part of the structural inspections done to modern military aircraft can be technically replaced by automatic structural health monitoring systems (SHMS). One option for the system's sensor is traditional strain gauge, whose capabilities in structural damage detection were evaluated. A series of finite element analysis were done according to different damage cases in order to predict the output of strain gauge in crack affected area. Results from two fatigue tests have been used to test the strain gauge and prediction tool capabilities in real structural damage cases.

The experience gained showed noteworthy potential for the strain gauge based SHMS to be further developed into a flying prototype, as the formation and growth of fatigue cracks could be detected early enough in view of maintaining the flight safety. Biggest problems to be solved are in areas of damage size classification.

INTRODUCTION

The structural fatigue life of most military aircraft now in service is determined by a full-scale fatigue test. The test arrangement is a simplification of reality and the loads and test spectrum are based on calculations and hypotheses of future usage. If those are not accurately known during design phase, the operational use brings suprises such as cracking of the structure. Thus, the actual fatigue life of the structure can be considerably shorter than verified. This possibility forces to structural inspections, which has to start quite early in order to maintain flight safety. The numerous inspections and especially the disassembly and assembly of the surrounding structure to gain access can be very time and money consuming. If, at least, the most time consuming inspections could be automated, the cost saving potential would be significant. Because of multiple critical locations in the aircraft the sensors should be cheap; such as strain gauges.

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RESEARCH AREA AND SCOPE

This presentation describes the research done by Patria, Emmecon and Technical Research Centre of Finland (VTT) relating to a co-European RTP 3.20 AHMOS-project. The aim of the Finnish part of the AHMOS project was to develop methods and test the capabilities of strain gauge based structural health monitoring system in crack detection. The main interest was in typical riveted aircraft structure but a bonded repair patch and an integrally machined part were also studied. Different kind of cases were studied in order to gain an overall understanding of what can be reasonably detected. Emmecon and VTT developed a distributed microcontroller based SHMS concept for signal processing and data aquisition and Patria developed analysis tools for prediction of strain gauge behaviour in the damaged structure [1]. Two fatigue tests with typical aircraft structure were conducted for the system evaluation.

STRAIN GAUGE BASED SHMS

Strain gauges were selected as the sensors due to their proven performance in aircraft environment, VTT's experience in flight measurements and Patria's experience in subsequent analyses [2, 3]. Thus, the technology and practices of strain gauge measurements are mature and well tested. Specialised crack detection gauges are also developed but this research uses standard axial measuring strain gauges because the same measurement system can then monitor both crack occurence and aircraft usage. Strain gauges meet also the requirement of low-cost and provide potential for dual purpose usage by loads monitoring.

METHOD OF ANALYSIS

The development of computers has made it possible to construct global finite element (FE) models of the whole aircraft, and to introduce real aerodynamic and inertia loads. Into a global models detailed sub-models can be inserted, to which cracks of reasonable size can be modelled [4]. With the sub-model solutions one can predict the strain gauge behaviour according to different crack lengths. Figure 1 shows examples of a global model and a sub-model.

In the FE -analyses the global behaviour and the load paths of the structure were expected to be independent of crack existence and growth. This assumption bases on the residual strength requirements of the aircraft and it allows the usage of smaller sub-models. Of course in the FE-analysis a prior knowledge of probable crack positions must be available for the selection of critical locations which will be modelled and instrumented. This information normally exists after manufacturers full scale fatigue test and more information is gathered along aircraft type's usage.

A series of FE -analyses were done to estimate the crack detection range of strain gauges in undisturbed conditions. A flat plate with constant displacement boundary condition was analysed for different crack lengths (5, 10 and 20 mm) using centre and edge crack boundary conditions. Figure 2a shows dimensions and constant stress contours of one case.

Figure 2b shows predicted strain variation between uncracked and cracked structure. When a 10 percentage stress increase from virgin structure is selected as the alert level, then a centre crack should be detected from the distance of half crack length and an edge crack from the distance of crack length. In addition the stress increase seems to be independent of strain gauge location in a ± 45 ° sector from crack tip, which allows more tolerance for the installation procedures.

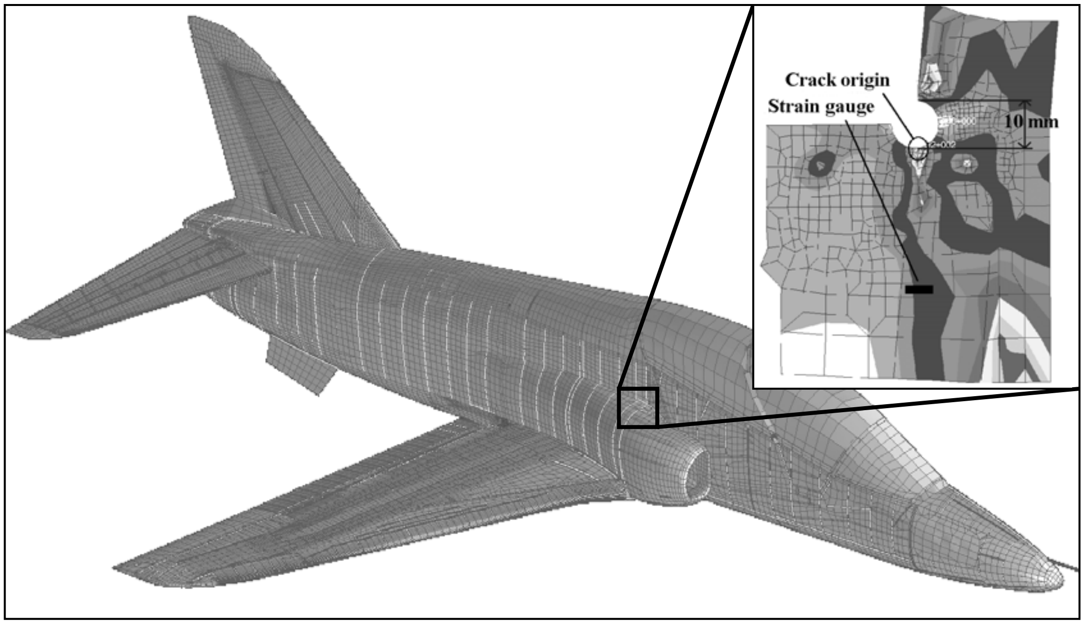

Figure 1. Example of a global FE -model with a sub-model showing stress contours.

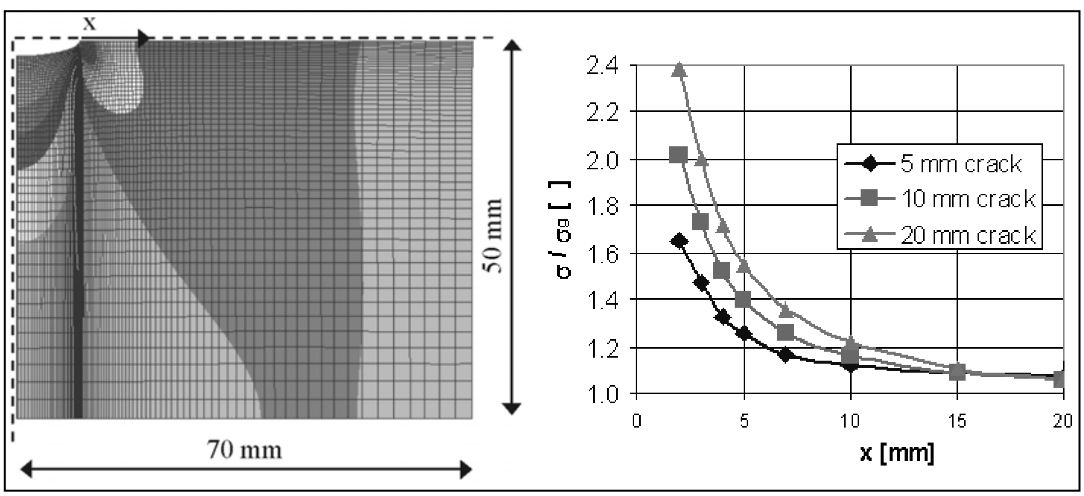

Figure 2a and 2b. Dimensions of analyzed plate and predicted strain gauge behaviour near crack tip. Typical vertical stress contours shown over FE mesh.

TEST-CASES

The capabilities of FE -analyses to predict strain gauge behaviour in real structure near the crack tip as well as the functionality of microcontroller based data aquisition and post processing system were tested in a fatigue test of a scrapped Finnish Air Force Hawk centre fuselage. The centre fuselage forms the fuel tank, and the pressure loads in the tank determine mainly the fatigue life of the structure. In the fatigue test the tank was filled with water, and constant amplitude pressure cycles were introduced by pressure air (Figure 3) [5]. This kind of test permitted the development of real fatigue cracks in real aircraft structure.

Figure 4 shows another component level fatigue test done for lower wing skin details [6]. The test loading was axial spectrum load and cracks initiated on the upper and lower surfaces of a hole drilled through the stringer. Strain gauges were installed to the upper surface of the stringer and to the lower surface of the wing skin. This test demonstrated strain gauge's crack detection capabilities in a thick aluminium piece, where critical crack size is short. The spectrum load introduced new requirements for damage detection logic.

Figure 3. Centre fuselage fatigue test arrangement.

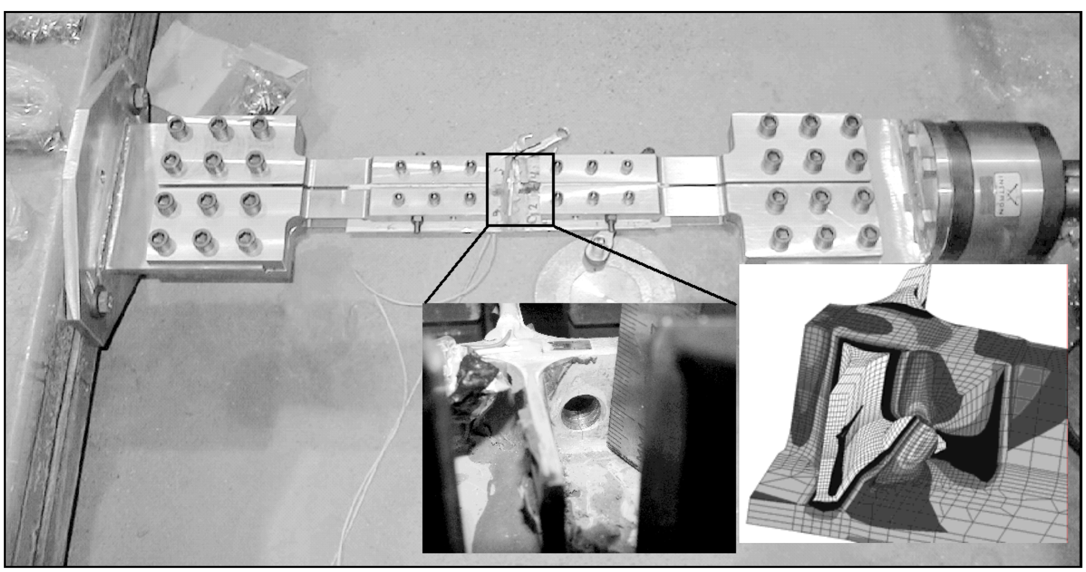

Figure 4. Lower wing skin test arrangement and part of the sub-model. Upper strain gauge can be seen on the top surface of the stringer.

Crack in Fuselage Frame

In the centre fuselage fatigue test a crack was detected in a fuselage frame web (Figure 5a). When detected it was about 10 mm abobe the rivet seen in Figure 5a. After the detection a strain gauge was installed below the rivet and the crack growth was monitored. The result of the measurement is shown in Figure 5b, where crack growth can be clearly seen. During the constant period the crack was stopped at the rivet hole. When the crack started to grow again the test had to be stopped for repairs.

When detected, the crack was already longer than allowed for operational aircraft. The effort in FE-prediction analysis was put in shorter cracks, which would be more interesting in real aircraft SHMS. In Figure 6 the predicted stress changes according to different crack lengths and sensor positions are shown. If a 7 % increase is selected as the alarm level a 9 mm crack should be detected at the distance where a strain gauge can be installed in real aircraft. The critical crack length in the area is over 22 mm, which means that the sensitivity of strain gauges is in this case good enough and strain gauges could be used for SHMS purposes [7].

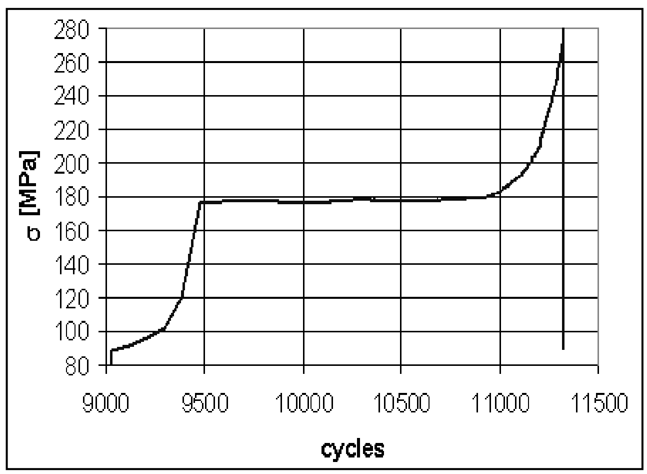

Figure 5a and 5b. Detected crack in the fuselage frame and measuremt results of the strain which is installed below the rivet. The crack was stopped in the rivet hole between cycles 9 500 and 11 000.

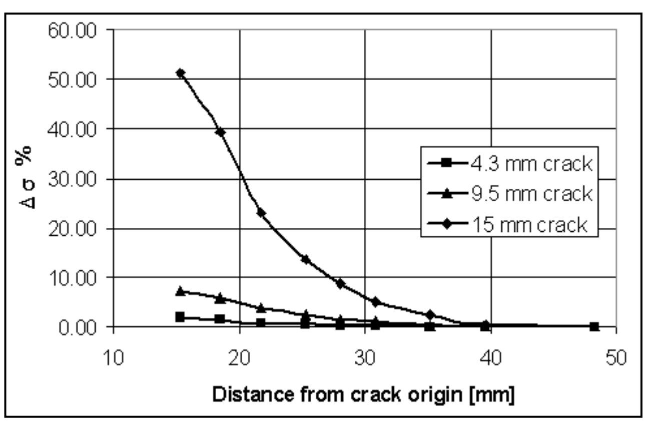

Figure 6. Predicted stress change as a function of sensor location and crack length.

Multi-Site Damage (MSD)

A more difficult damage mode to detect, the MSD in a rivet row, was expected to occur in the fuel tank side skin. The structure is a stiffened sheet to which normal pressure causes bending and membrane stresses. The main problem is shorter critical crack length due to interaction between the cracks origin from adjacent rivet holes. In the test bench the rivet row was intrumented with three strain gauges: one on the rivet row, and the two other 16 mm above and below the row.

A detailed FE-model was constructed and strain gauge behaviour for different cracks was analyzed. The summary of the analysis is shown in Figure 7. The decreasing stress in upper and lower strain gauges is due to their location outside the $\pm 45\ °$ sector from crack tip. In the fatigue test the area was inspected when the MSD cracks were about 6 mm long. Due to instrumentation problems only lower strain gauge results were available. Taking into account the rivet influence a difference of -34 % in strain level was predicted for the strain gauge. In the test a difference of -12 % was measured leaving large error between predicted and measured stresses.

The analyzed case, which includes geometrical nonlinearity, is difficult for FE -analysis and probably needs more detailed mesh than that used. In this case the allowed crack length is about 4 mm so the strain gauges detected the cracks early enough to maintain flight safety. But the crack size could not be predicted.

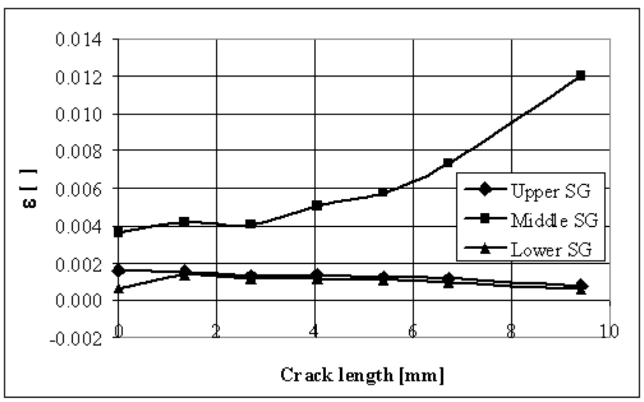

Figure 7. Predicted strain gauge response according to different crack lengths. Strain gauges located 10.5 mm from crack origin on and 16 mm over and below the rivet row.

De-Bonding of a Repair Patch

In the centre fuselage fatigue test unexpected MSD occurred in the fuel tank outer skin: About 10 cm long crack in the rivet row of a frame to skin attachment. The damage was repaired with a bonded 0.8 mm steel patch. A strain gauge was installed on the patch in order to monitor the expected de-bonding. A series of nonlinear 2D FE -analyses with different de-bonding lengths were conducted. Predicted strain gauge behaviour is shown in table I and measured stress history in Figure 8.

Based on FE results the de-bonding should be detectable with a strain gauge. If a typical de-bonding size of one inch is considered, the measured stress should increase 9 %. The constructed FE -model seems to predict initial stress very well: Predicted stress –220 MPa and measured stress –215 MPa. At the time, when de-lamination's semi-width was 15 mm, the FE -model predicted a stress of -208MPa. The nearest measurement in fatigue test is the 21570 cycle with the stress –129 MPa differing considerably from FE prediction and leading to conclusion that the 2D analysis used is not working properly. In this kind of a case with a curved skin and large geometrical displacements a detailed 3D FE -model is mandatory for damage size classification. A component level fatigue test can be a more cost effective solution.

TABLE I PREDICTED STRAIN GAUGE STRESSE ACCORDING TO DIFFERENT DE-BONDING LENGTHS

Half de-lamination [mm]	Stress (p=0.82 bar) [MPa]	Detached elements
2.4	-233.0	5
5.1	-223.8	10
7.9	-216.5	15
11.1	-211.5	20
14.4	-208.5	25
18.1	-212.8	30
22.1	Patch buckled	35

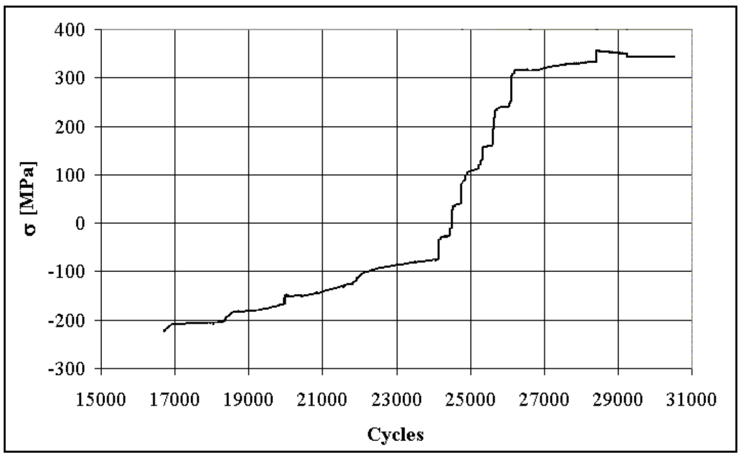

Figure 8. Measured stress history of the strain gauge.

Drain Hole Crack

In the wing detail fatigue test an integrally machined lower wing skin detail was tested (Fig. 4) [6]. In the test a strain gauge was installed directly below the crack location. Because of spectrum load, the strain gauge stress was compared to the load actuator force in damage analysis. In real aircraft installation the cylinder force could be replaced by a strain gauge installed near critical location but outside crack path. To avoid false alarms a suitable trigger level should be set and analysis of crack existence be done starting from medium stress level. The measured results are shown in Figure 9 where the highest value equals to crack length of 4 mm. Distance from strain gauge to crack origin was 7 mm. As can be seen the crack was well detectable.

A solid FE -model was constructed (Fig. 4) in order to predict strain gauge behaviour with different crack lengths. The results in different flight conditions were within 10 percent compared to flight test results for virgin structure [8]. Due to limitations in computer power the crack couldn’t anyway be modelled precise enough for proper predictions. Also in this kind of a case the most cost efficient solution for strain gauge behaviour in crack affected area is still a component level fatigue test.

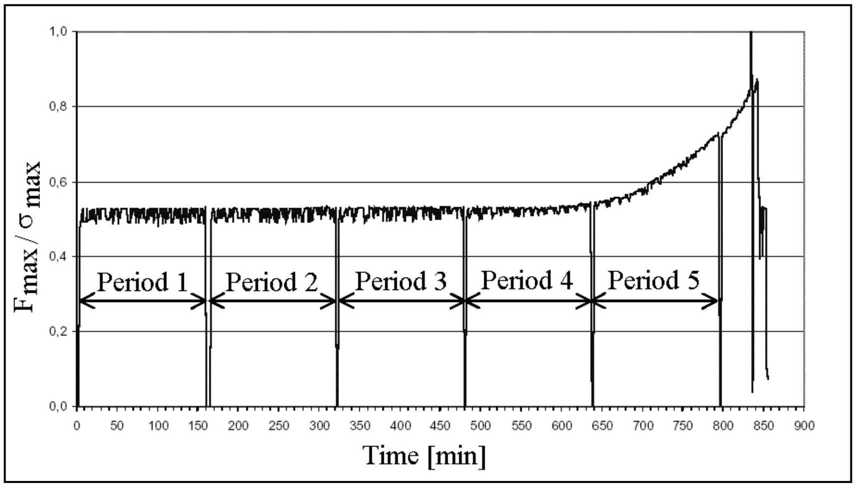

Figure 9 One minute maximum applied force divided by maximum measured strain gauge stress under spectrum load. Periods are between inspections when the test was stopped.

RESULTS AND THEIR EVALUATION

Based on test experience with representative aircraft structures it is possible with strain gauges to detect cracks early enough to maintain flight safety. The measured difference in stress level due to damage growth is clearly visible (Figs. 5b, 8,9). The FE -models can predict if the strain gauge can detect crack, but damage size evaluation requires in most cases tests. The other option could be extremely detailed FE -models but it leads to cost increase compared to small component level tests. The necessary equipment for instrumentation is relatively low cost providing possiblities to lower the cost of operation with aging aircraft types.

In the future more systematic work is required for damage size classification. Especially if component level tests for the specific installation are expensive to arrange. Basically all necessary elements for application prototype are available but no operational application is known to exist.

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