HEALTH MONITORING OF COMPOSITE REPAIRS OF AIRCRAFT STRUCTURES

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ABSTRACT

The work focuses on aircraft structures and their parts which are designed using the damage tolerance approach. The main goal is design, verification, monitoring and certification process of fatigue crack repairs of the selected critical places on the Czech light combat jet L-159A. The repairs were carried out using carbon/epoxy composite straps and patches and cold-setting glue SW 9323. A new type of repair was developed and tested. It applied a boron/epoxy composite and FM 73 glue system with the grit blast/silane surface preparation. The aim of the tests was to prove better characteristics of the repair with the use of boron double-sided patch in comparison to the carbon repair as identified by fatigue testing. Possibility of application of the health monitoring methods is discussed here, especially the acoustic emission method and application of strain gauges and optical fibers with Bragg grating for the detection of strains in monitored or repaired locations.

KEYWORDS

composites, repair, health monitoring, aircraft structures

INTRODUCTION

Aero Vodochody a.s., with more than 6500 manufactured military jet trainers, is the greatest producer in this category in the world. Aircraft types L-29, L-39, L-39MS a L-59 were delivered by Aero to more than 30 countries of the world. At the present time Aero offers the advanced light attack aircraft L159A and its training variant L159B which are used in the Air force of the Czech republic, as shown in Fig. 1.

During the development and certification process of the aircraft L159A, a full scale fatigue test of the airframe was conducted. As a result, real measured data for design of the repair of the typical L159A fuselage fatigue damage was available for the team of designers and analysts. This damage occurred during the full scale fatigue test of the whole aircraft and therefore the origin of the crack and its development is well known, see Fig. 2. During the test, the fatigue crack was repaired by the technology of bonded composite patches with a view to increase durability of the structure. This technology was used for the first time in 1995 by the full scale fatigue test of the aircraft L-39 MS.

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The airframe was loaded in the so-called "floating state" analogous to the same boundary conditions as during real operation. Data about deformation and stress in the damage area was observed during the test. This information enables the verification of theoretical and FEM global and local model analyses. The fatigue test did not prove growth of the crack with the applied repair until 4.2 lifetimes.

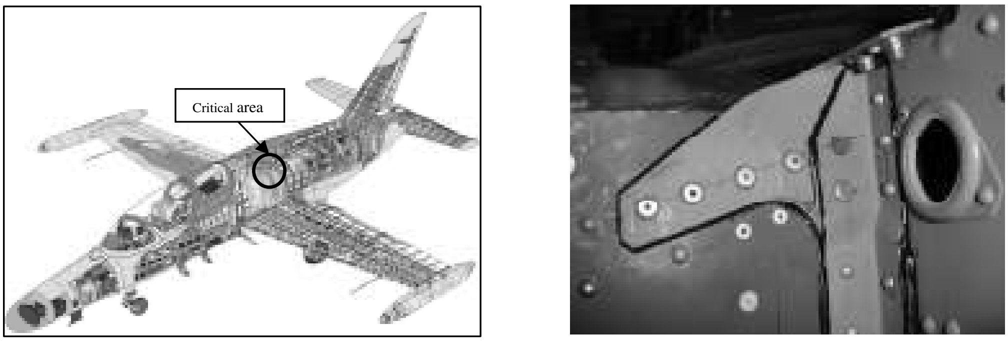


Fig. 1 & 2. Aircraft L159A and detail of the critical area at auxiliary stringer

REPAIRS OF AIRCRAFTS PARTS

Fatigue damage is a characteristic sign of aircraft operation. Early detection of damage and application of the suitable repair is able to lead to significant extension of fatigue life and cost cutting. Riveted metal patches are commonly used for these repairs. The patch transfers load in the damage area and reduces the crack growth rate by decreasing of the stress intensity factor at the crack tip. In some cases the crack growth rate can be stopped. The process of riveting however requires to drill holes for rivets which causes next attenuation of the structure and the holes can become new stress concentrators. In many cases this type of the repair is not applicable. The process of bonded composite shaped patches offers a suitable alternative to riveted repair. In this concept, composite patches are bonded to a cracked metal part instead of riveting [1]. A bonding provides many advantages over riveted repairs:

- more effective load transfer from cracked part to composite patch because of transferring load by whole bonding area instead of carrying by discrete points in case of riveted repair
- additional stress concentration does not occur due to absence of new drilled holes
- high durability of repairs in response to cyclic loading
- corrosion resistance
- stacking of orthotropic layers enables to design thin patches with high stiffness in loading direction of a repaired part
- possibility of integrating sensors in the patch for damage monitoring in the repaired part, glue joint or in the patch, [2]
- this technology enables the repair of complex shaped and hard to access parts.

Sensitive choice of many parameters and strict observance of designed technological procedures are necessary for the repair to be effective. The critical phases for bonded composite repairs of metallic structures in light of their durability and efficiency are:

- selection of adhesive system and material for composite patch
- design of composite patch including environmental effects
- surface preparation and bonding process
- integration of the sensors in the patch or on the repaired part during the bonding process.

From the full scale tests results, it followed that the used technology of repairs has a significant effect on decreasing the crack growth rate in critical areas of the structures and thereby on increasing the airframe durability.

NEW TYPE OF REPAIRS - DEVELOPMENT AND TESTING

The next phase of our project examines a proposal of the certified procedure of fatigue damage repairs in light of contemporary knowledge, materials and technologies while including environmental effects. There is also is an effort to apply advanced monitoring technologies through smart structures. The fatigue damage of L159A longitudinal carrying system was chosen as a repair application demonstrator.

First type of repairs was carried out with technologies available at that time. For bonding of pre-cured carbon patches the adhesive SW 9323 A/B was used. The repair was carried out under laboratory conditions without any environmental effects. No sensors were applied for fatigue damage monitoring in this test. Successful application of the fatigue damage repair of the auxiliary stringer was verified during this test. The critical part is showed in Fig. 3. A double-sided carbon patch bonded with adhesive SW 9323 was used. But the test ran without the influence of temperature and moisture changes.

The aim of the new type repair development is to prove better characteristics of the repair with use of boron double-sided patch bonded by foil adhesive FM 73 in comparison to the repair of the first generation. The starting tests were conducted on simple specimens with a simulated crack and applied repair through constant value cyclic loading. Before application of the repair the tip of the crack was drilled and the hole was fastened by cold-working technology. The aluminum specimen geometry is drawn on the Fig. 4.

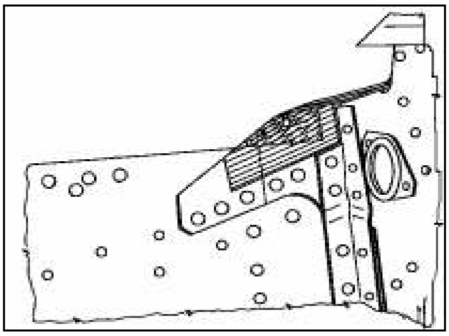
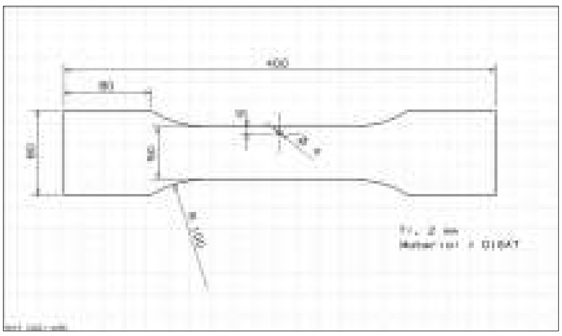



Fig. 3 & 4. Scheme of the composite repair and the aluminum specimen geometry

The foil adhesive FM 73 with cure temperature in the range of 80 -120 °C was chosen. Usage of foil adhesives enables to reach a consistent thickness of the bonded joint. The grit-blast/silane surface preparation or sol-gel technology creates a chemical interaction between the adhesive and aluminum-adherent and thus increases fatigue-durability of the repair.

FINITE ELEMENT ANALYSIS

The FE model of the whole aircraft (a global analysis model, fig 5) was used for the definition of technical specification of the full scale fatigue test. MSC Patran/Nastran were used for FEM analyses. The global model does not describe the damage area in sufficient details. Its purpose is to provide a correct layout of stiffness and deformation at the whole airframe and define boundary conditions for analysis of detailed parts. A local FE model of the critical area was created without and with applied repair, see Fig. 6. This local model is connected to global model by multi point constraints. The FE model was verified based on the static test results. It enables the design of the first generation of composite patch dimensions.

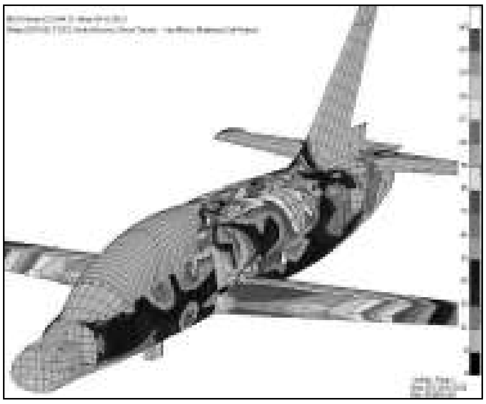
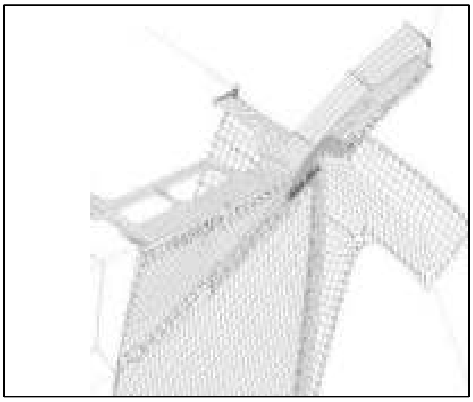



Fig. 5 & 6. A global FE model and FE submodel of the critical area at auxiliary stringer

The analyses of various patch shapes both on test specimens and on local model of airframe and the damage at the CTU in Prague (Abaqus 6.6 code) were performed.

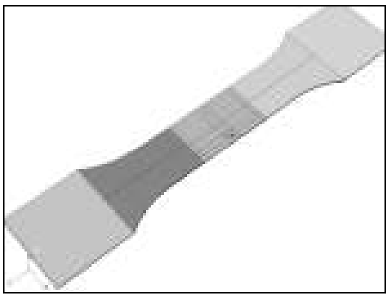
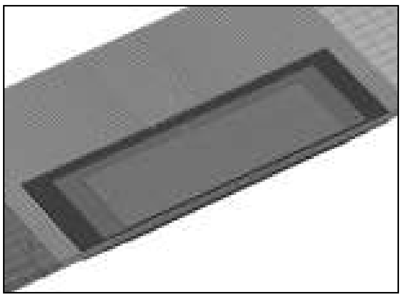



Fig. 7 & 8. The aluminum specimen geometry depicting the asymmetrically cracked/drilled gauge region and the geometry/orientation of the applied composite patch in the CTU FE model.

The model of specimen [6] compares two specific patch fibre and adhesive system combinations, namely, carbon fibre/adhesive SW 9323 and boron fibre/adhesive FM 73. The geometry of the aluminum specimen in question is depicted in Fig. 5 while Fig. 6 shows the geometry and orientation of the applied composite patches. The geometry of the tri-layer composite patch has been modeled as three distinct parts, geometrically tied together. The geometry of the adhesive layer was set to 0.2 mm. Isotropic strain hardening was implemented into the material model for the 2024-T3 aluminum, since the examined load case would approach yield and neglecting its inclusion would invariably lead to erroneous results. The patches were bonded onto the specimen employing an adhesive layer composed of cohesive elements whose elastic response has been defined by a precise, uncoupled, tractionseparation law. In order to examine the possibility of failure within the cohesive interface, a quadratic nominal strain damage initiation criterion was enforced along with an exponentially softening damage evolution law, incorporating element deletion upon complete degradation of material properties at all integration points within the cohesive elements. Furthermore, the composite patch materials also include a 4 parameter Hashin based, damage initiation criterion with a linear softening damage evolution law, uncoupling the contribution of developed shear stresses to failure in net tension. The inclusion of element deletion upon complete material degradation compliments the aforementioned failure model implemented for the cohesive interface layer, thus enabling the analyst to completely model failure in any constituent desired.

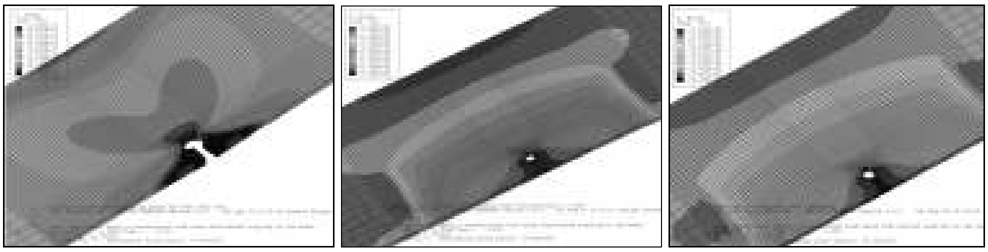


Fig. 9,10 & 11. Mises stress distribution in the aluminum, without and with carbon or boron patch respectively. Note the effect of compliance discontinuity.

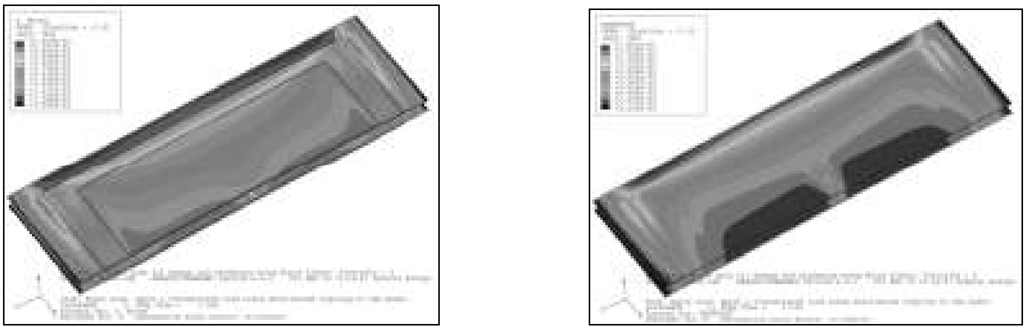


Fig. 12 & 13. Mises stress distribution for all the patch layers in the boron case and fibre tension failure criterion in the large patch layer

The distribution comparison of illustrate (see Fig. 9,10 & 11) that the stress near the cracked/drilled hole region of the aluminum substrate have been dramatically reduced even when compared to the carbon patch case. An effect due to the compliance discontinuity generates high stresses at the ends of the adhesive and in fact satisfies the failure criterion at the regions located most longitudinally distal from the cracked/drilled hole region, see Fig. 12 and 13. It will be verified experimentally.

Figure 14& 15show the distribution of stress-distance profiles from various nodal paths in the aluminum substrate. It clearly indicates the superiority of the applied boron/FM 73 system when also considering the minimization of peak stress in the aluminum substrate discussed previously. The boron patches equally or out-perform the carbon patches utilizing a nominal thickness of half magnitude; this is a critical consequence in the aerospace sector where reduction in weight is paramount.

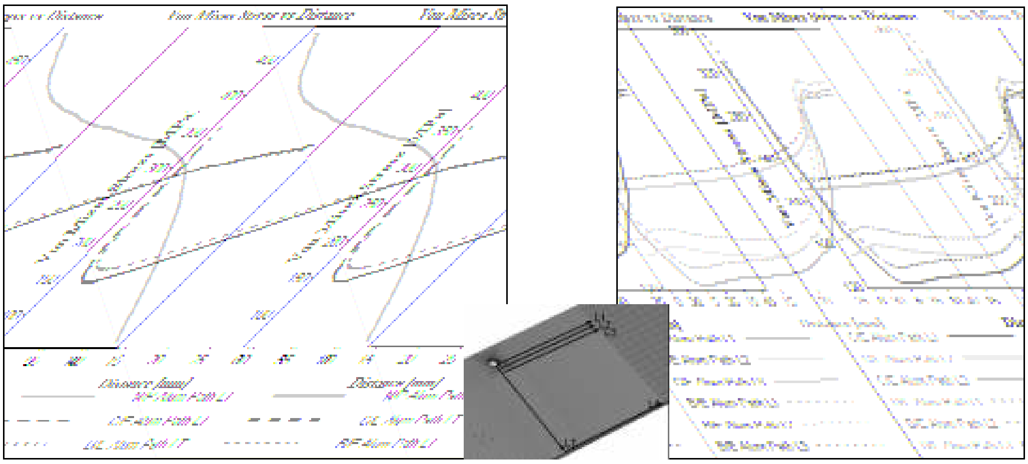


Fig. 14 & 15 Comparison of the Mises stress-distance profiles in aluminum parts for the carbon/SW9323 and boron/FM 73 system for various nodal paths.
(LT =transversely – on the left, L = longitudinally – on the right)

STATIC AND DYNAMIC TESTS OF GLUED JOINTS

Static and dynamic tests of different types of glued joints were carried out at the Czech Technical University [3], in Aeronautical Research and Test Institute and in Aero Vodochody. The main goal of these test were to recognize material parameters of the glue, static and fatigue characteristics of different types of joints and combinations of glued materials.

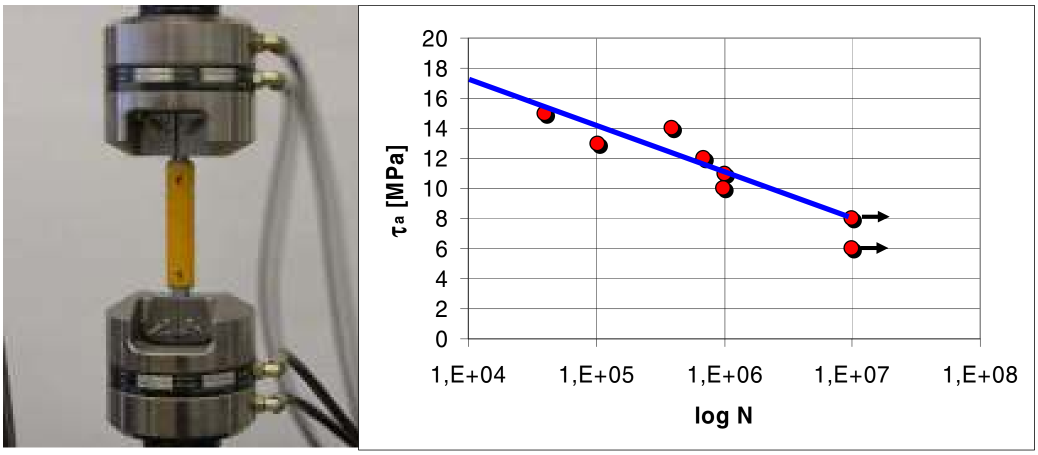


Fig. 16 & 17 Fatigue testing of glued joint of G/E composite tubes and S-N curve of the joint

Examples of the fatigue tests in tension and torsion are presented on the Fig. 16. Dynamic tests were carried out on testing system MTS 858.2 at the department of Mechanics of the CTU in Prague. The testing system makes possible apply the combination of the dynamic loading with axial load and torque. The evaluation of behavior of specimens and the fatigue life (total number of cycles) was determined after destruction of specimen. The Stress-life curve of the glass/epoxy joint with using of the Spabond 345 glue is presented on the Fig. 17.

STRAIN, FRACTURE AND HEALTH MONITORING METHODS

To validate FEM results and obtain information about local strains of basic materials or composite patches different experimental methods of strain analysis were done in collaboration of the CTU in Prague and Aeronautical Research and Test Institute in Prague. For example:

- photostress analysis with using of polariscope LF/Z - 1
- photogrammetry analysis with using of ARAMIS equipment
- thermoelastic analysis (SPATE)
- clasical strain gauge analysis
- optical fibers with Bragg gratings (FBG sensors)

Comparison of the stress concentration around the hole in axial loaded composite plate is demonstrated on the Fig. 18.

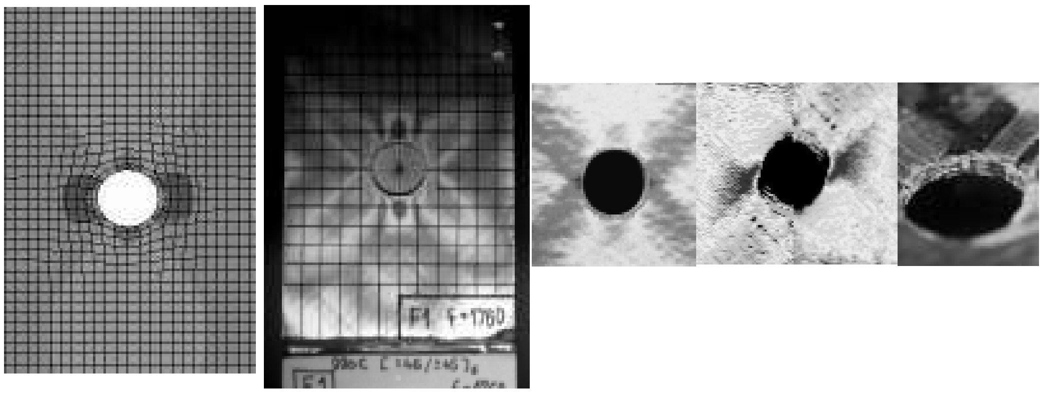


Fig. 18 Comparison of the stress concentration around the hole in axial loaded composite plate: FEM, photostress, ARAMIS, SPATE, strain gauge

Getting experiences lead to choosing of the followed methods, which were further developed and applied for fracture and health monitoring of composite joint and composite repairs:

- strain gauge data monitoring
- fibre FBG sensors application
- acoustic emission monitoring

Example of acoustic signal measurement of bonded joint is demonstrated on the Fig 19. Amplitude of the acoustic signal as well as summation of the acoustic events with comparison of loading force are depicted here. It could be see a changing of the slope on the acoustic hits curve and a growing of acoustic signal amplitude before breaking.

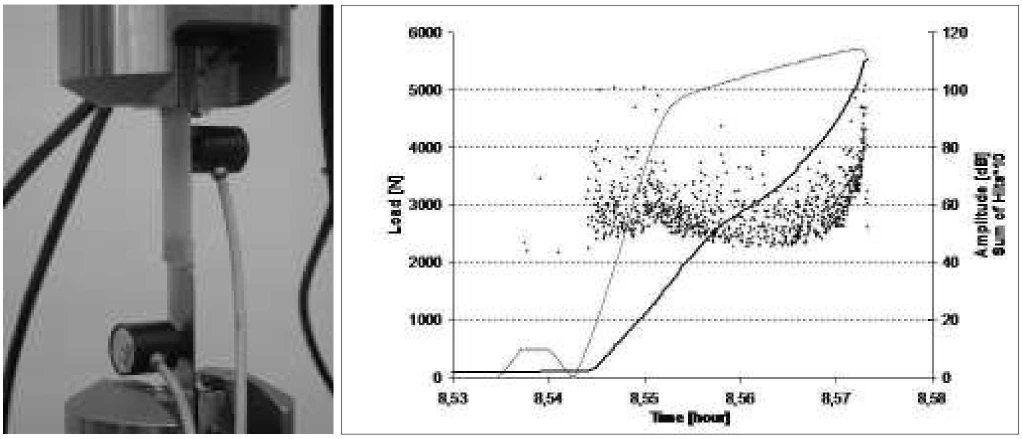


Fig. 19 Acoustic emission monitoring of bonded joint

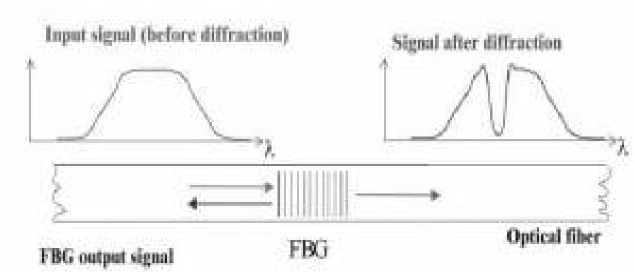
FIBER BRAGG GRATING SENSORS

Optical fibres are often used by a health structure monitoring. They have a number of advantages over conventional electrical strain gauges. The most significant advantages are that:

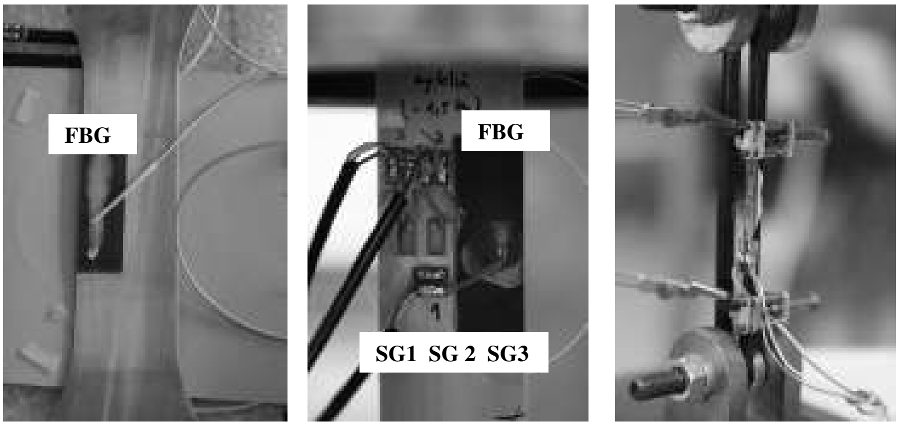
- they are immune to electromagnetic fields
- they have the ability to take many measurement points along a single fibre - greatly improving the ease at which sensors can be multiplexed
- they can be embedded within or bonded to structures without the risk of de-bonding during operation.

Several different optical sensing techniques have found their way into the market place but fibre Bragg gratings (FBGs) are commercially one of the most successful. Fiber gratings are made by laterally exposing the core of a single-mode fiber to a periodic pattern of intense ultraviolet light. The exposure produces a permanent increase in the refractive index of the fiber's core. This index modulation is called a grating. At each periodic refraction change a small amount of light is reflected. All the reflected light signals combine coherently to one large reflection at a particular wavelength when the grating period is approximately half the input light's wavelength. This is referred to as the Bragg condition, and the wavelength at which this reflection occurs is called the Bragg wavelength. Light signals at wavelengths other than the Bragg wavelength, which are not phase matched, are essentially transparent. This principle is shown in Figure 20. Therefore, light propagates through the grating with negligible attenuation or signal variation. Only those wavelengths that satisfy the Bragg condition are affected and strongly back-reflected. The ability to accurately preset and maintain the grating wavelength is a fundamental feature and advantage of fiber Bragg gratings.

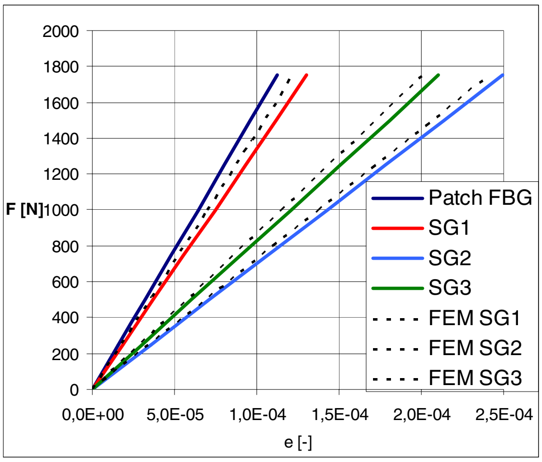
One of the first our commercial application of fibre Bragg grating was in the monitoring of strains in composite structures [4]. The fibre sensors are laid on or between adjacent layers of fibre prior to resin bonding. Otherwise fibres could be placed directly into the fibre tows by application of filament winding technology. Such "smart structure" can then be used to monitor strains (loads) or damage during the lifetime of the structure. The photos of the CTU experiments on the Fig. 21 and 22 show the strain gauges and fibre sensors being laid on the carbon composite patch and in the composite loop connection



Figures 20 Working principle of FBG [6]



Figures 21 & 22 Application of the strain gauge (SG) and fibre Bragg gratin sensors (FBG)



Figures 21 Comparison of the FEM, Strain gauge and FBG sensors strain signal measurement in different places on the carbon patch

CONCLUSION AND FUTURE WORK

Performed full scale testing of airplane showed feasibility of carbon/epoxy composite repairs of fatigue cracks on an airframe critical place. Results from performed FEM analyses of stresses and damages on the specimens with patches confirmed convenience of boron/epoxy repairs that was designed and tested for repairing of the auxiliary stringer L159 airframe. Various methods of stress strain analysis and damage monitoring of the structural part and composite patches will be tested and compared.

Optical fiber sensors, acoustic emissions method and strain gauges monitoring are now in progress by the fatigue tests of above described specimens. The aim of such work will be a continual monitoring of the cracks stability and a bearing capacity of the composite repair. It should be included an assessment of the safe damage zone in the glue layer interface.

ACKNOWLEDGMENTS

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