THE USE OF COMPOSITE MATERIALS TO REPAIR OF AIRCRAFT SEMI-MONOCOQUE AIRFRAMES

M. Rośkowicz[1], T. Smal[2]

[1]Institute of Aircraft Technology, Military University of Technology, Warsaw, Sylwestra Kaliskiego Street 2, 00-908 Warsaw 49 Poland, e-mail: marek.roskowicz@wat.edu.pl, phone: +48 22 683 72 10
[2]Institute of Command, The Tadeusz Kosciuszko Military Academy of Land Forces, Czajkowskiego Street 109, 51-150 Wroclaw, Poland, e-mail: t.smal@wso.wroc.pl, phone: +48 71 7658 108

Keywords: repair, composites patch, semi-monocoque airframe, viscoelastic properties of composites

Abstract

The results of numerical calculation and experimental tests are presented, which proved that damaged skin of aircraft semi-monocoque airframe is connected with local loss of stability in the zone between stiffening components. The damaged skin area was repaired with a metal insert and a composite patch. The use of a metal insert; which has the same stiffness as a repaired plate, is a solution which allow to restore local stiffness of repaired component. Composite patches; which are impregnated with resins, create also an adhesive bond and join all elements of repair zone. They improve fatigue life of repaired component that was identified during experimental tests. The experimental research proved as well that viscoelastic properties of composite patches should be taken into account when monitoring their conditions with strain sensors.

1 Introduction

The repairs with use of modern composite materials are an effective method of damages removal in case of aircrafts components. The composite materials are bonded to the damaged structure with adhesives. The repairs executed with the use of composites are particularly accepted in relation to thin skin elements of fuselage and wing, which are made of aluminum alloy. The skin repairs include cracks of air components, dents and material losses of fuselage and wings skin [1, 2].

The assessment of fix feasibility is an important stage of preparation and execution of repair. There is a lack of standard guidelines which may be used to perform a repair. Individual planes manufacturers create own repair procedures for certain products. The procedures present a scope and technology of repairs without specific criteria. The assessment of repair feasibility requires defining the scope of repair and effect of damage to adjacent structure elements. The experiences of some companies conducting repair of plane components show that the general principle of repair designing and execution is reduction of a damaged component safety coefficient to a level not lower than 1.2. The assessment of safety coefficient changes can be provided with the use of modern numerical calculation systems.

The numerical calculations were used to assess the effect of defects in skin of thin structure to adjacent components. The experimental research studies were conducted with the use of testing machine; constructed especially for planned tests, in order to verify the conclusions of numerical calculations and assess the repair feasibility of damaged component with use of composites. Durability of repair zone were tested as well. The analysis of restrictions; connected with monitoring of repaired zone, were conducted with the use of fixed girder of Su-22 aircraft's tail plate.

2 Numerical calculations of damaged skin structure

The numerical calculations were conducted with the use of finite elements method (Nastran/Patran software environment). The model of "TS-8 Bies" aircraft horizontal tail was generated in order to execute numerical calculations. The horizontal tail was built with skin, stringers, ribs and two girders: front and rear. The geometric model of plane tail was generated (Fig. 1) with particular emphasis on accurate mapping of its individual components.

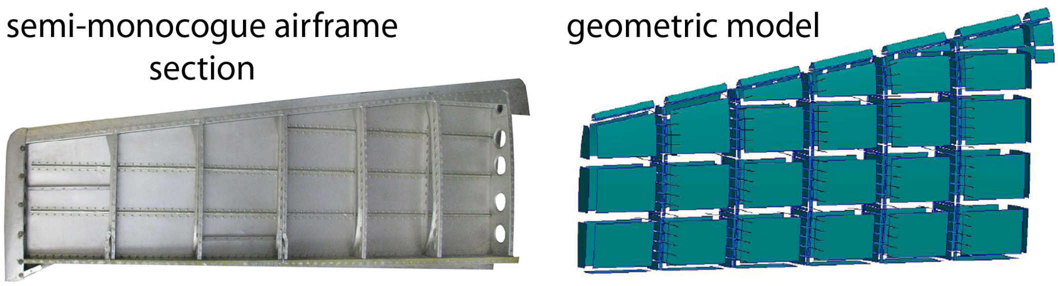

Fig. 1. The section of TS-8 Bies aircraft horizontal tail and its geometric model

On the basis of geometric model, numerical model was generated which included elements of SHELL type (skin, beams, girders' skin) and elements of BAR type (stringers and girder strips). There were boundary conditions (loads and mounts) assumed which correspond with the method of the tail mounting on the testing machine.
The model was loaded at the same place as the load element of the testing machine. It was assumed that the external force loading the tail would have value of 1000 N. The value was determined from the analysis, which described ultimate loads for this type of semi-monocoque airframe [3]. The loads were modeled with elements of RBE 3 type. The same material features were assumed for all finite elements of model, i.e. aluminum alloy 2024T4.
The stress and strain distribution was analyzed in the undamaged and damaged tail during numerical calculations. The damage was modeled in a shape of hole with a diameter of 100 mm, which was located in the middle of tail skin (Fig. 2).

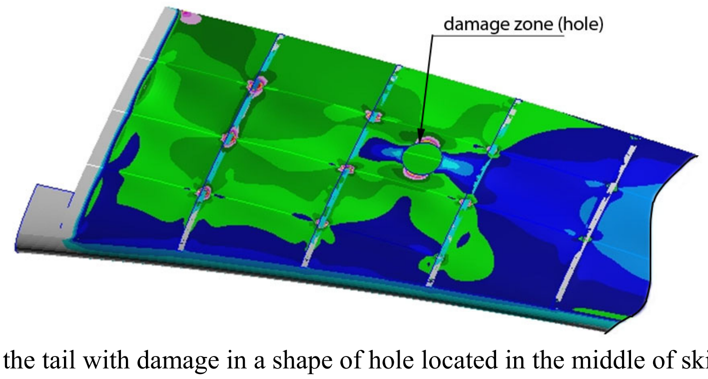

Fig. 2. The model of the tail with damage in a shape of hole located in the middle of skin between supporting beams of the generated airframe

The comparison of reduced stress distribution in the damaged and undamaged zone is presented in Fig. 3. In the area of damage an accumulation of stress was observed – more than twofold increase in stress in the direction parallel to the tail beam and decrease in stress in the direction parallel to the stringers. This kind of changes in a skin is specific in the case of local stability of skin loss (in the zone between the stringer and a beam of tail). On the basis of numerical calculations it was assumed that; due to damage, the changes in stress and strain are limited to skin zone bounded by ribs and stringers. Apart from these strengthen elements, the changes occur to the minimum extent. Simultaneously, the numerical calculations allowed to observe that the damage caused a greater effort of supporting elements, i.e. ribs and stringers directly adjacent to the damaged zone. At the same time, it was observed that the rivet connections between the supporting components and skin carried greater loads.

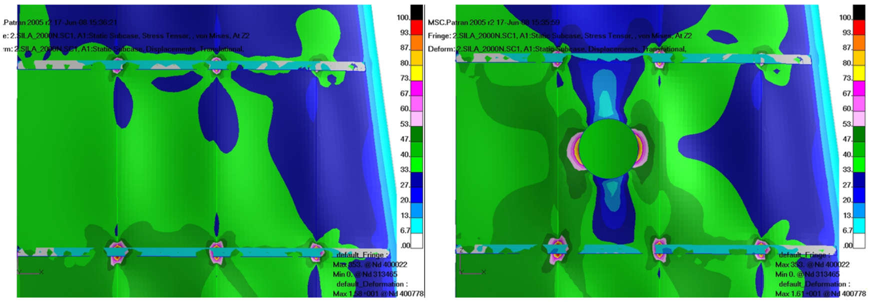

Fig. 3. The comparison of stress distribution in the tested tail: undamaged zone (on the left), damaged zone (on the right)

Taking into account the obtained results it seems that the main criteria of repair of thin semi-monocoque airframe; which does not include supporting components, are stiffness reconstruction of skin in a damaged zone and solution to maintain stability of repaired element.

3 The experimental verification

In order to verify the obtained results of numerical calculations, the testing stand was prepared (Fig. 4). The horizontal tail was mounted in the frame, which was bolted to the forceful floor. The loads were generated with hydraulic actuator controlled by HNC 100 controller of Rexroth Bosh company. The actuator with tail was also mounted with the special frame.

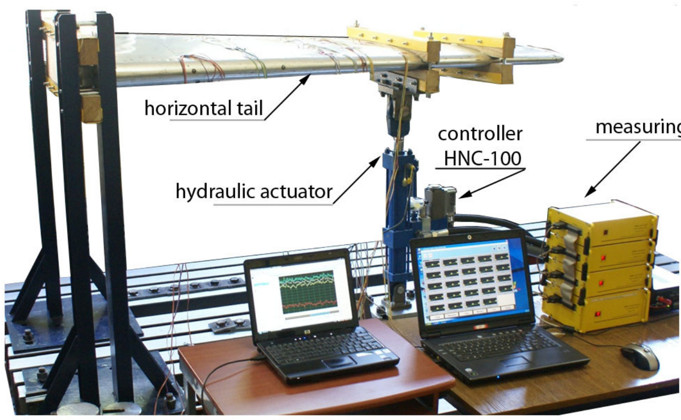

Fig. 4. The testing stand with strain gauge bridge of National Instruments and ESAM Traveller Company

The strain gauges were used to measure strain in tail skin. The strain gauges were arranged on the surface of skin according to the scheme presented in Fig. 5.

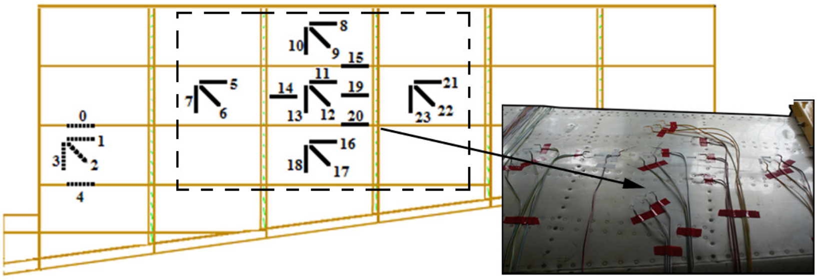

Fig. 5. The arrangement of strain gauges on the surface of tail skin

The research on strain in a skin; in statically loaded structure, was conducted before damage and after damage. The damage was in a shape of a hole with a diameter of 60 mm and was made by milling in the upper skin of tail according to the scheme presented in Fig. 6.

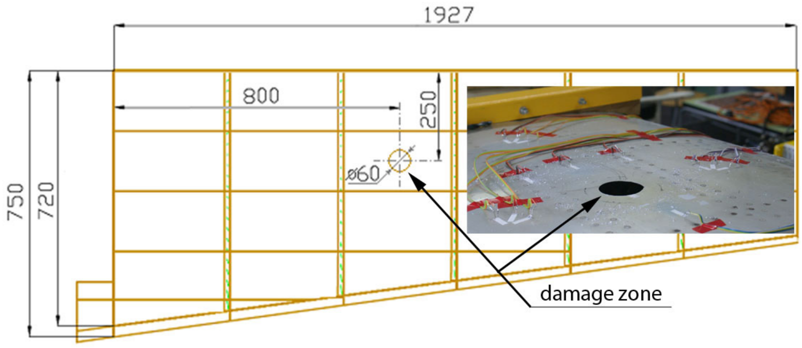

Fig. 6. The geometry of the damage and view of the hole made in the upper skin of tail

The horizontal tail was loaded with the force of 880 N that corresponded with a displacement of the actuator of 25 mm. The values of maximum main strain were determined on the basis of the strain gauges' rosettes placed in the subsequent construction zones. The values of strain were defined in chosen zones of undamaged and loaded construction (1. Case) and in damaged and loaded construction (2. Case). The results are presented in Fig. 7.

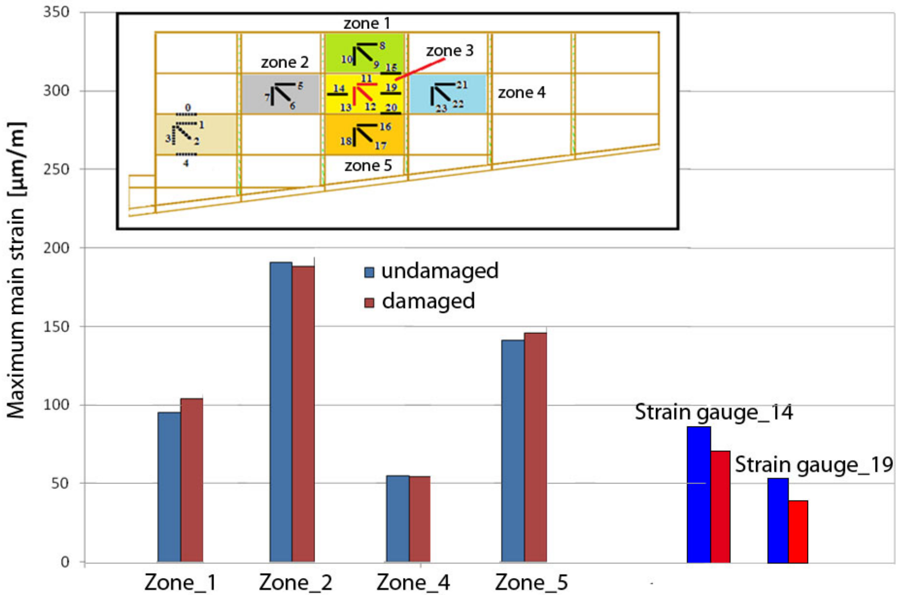

Fig. 7. Maximum main strain in the chosen zones of loaded construction

The results of performed tests showed that maximum main strain in the zones adjacent to damaged zone (outside the supporting elements) did not change significantly. A significant decrease in strain was registered in strain gauges no. 14 and 19 which were located in the damaged zone. The obtained results confirmed conclusions of the numerical calculations.

4 Research on construction repaired with composite materials

On the basis of the numerical calculations and the experimental research it was claimed that the main aim of damaged skin repair of thin components should be stiffness reconstruction of a part of skin in the damage zone (between supporting elements) and; therefore, solution to the issue of stability maintenance of the repaired element. With the use of results concerning repaired plates, which were loaded by shear [4] there was the method of tail skin repair proposed that included material loss supplementing by a disk-shaped insert of aluminum alloy 2024T4 and formation of one-sided strengthen composite patch performed with vacuum bag method. One of the essential conditions of the skin repair effectiveness is to ensure proper sensitivity of composite patch to deformation. Thus, the composite patch was made of 6 layers of glass fabric Synglass E81weighing 101 g/m^2 (Fig. 8).

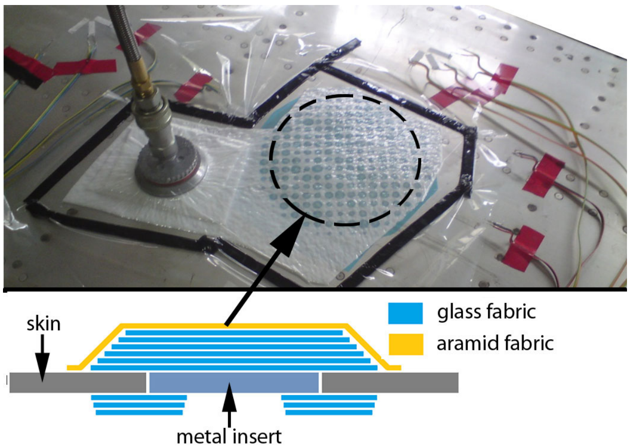

Fig. 8. The scheme of repaired zone

The layers of fabric in a disk-shape were arranged according to the scheme $(0/45/90)_2$ and in order to obtain a stepped effect the successive layers had a diameter of 5 mm less than each previous one. The first layer had a diameter of 100 mm. The last layer with a diameter of 110 mm was made of aramid fabric with a weight of 61 g/ m^2 in order to protect the composite patch against mechanical damage. The epoxy resin L418/H418 of German company MGS was used to lay-up the fabric glass. Because there was access to one side of the tail skin only, the problem of aluminum alloy insert holding up should be solved during composite patch formation. In order to achieve that the composite ring was formed on the inside of tail skin. The composite ring consisted of 3 layers of fabric glass and its outside layer had a diameter of 100 mm and an inside layer had a diameter of 40 mm. The applied ring made it also possible to seal a gap between the repaired skin and the insert which was necessary to perform the repair with vacuum bag method. In Fig. 9 there are the scheme of layers in the repaired zone and the view of repairing zone prepared to curing process presented.

The applied patch was cured at normal temperature for 24 hours and then heated at temperature of 80°C for 15 hours with heating blanket. The strain gauges no. 11, 12 and 13 were bonded on the composite patch in order to monitor conditions of the repaired zone (metal insert and composite patch) during durability tests (compare Fig. 8). The repaired structure were investigated during durability tests, in which the cyclic load was applied by shifting a hydraulic actuator with a frequency of 0,5 Hz. The shift of the hydraulic actuator was equal to 25 mm, as it was in static tests. 20 000 cycles were performed to analyze the strain changes in the repaired zoned, particularly in the composite patch (strain gauges no. 11, 12 and 13) and in an adjacent area to repaired zone. The changes of main maximum strain during durability tests are presented in Fig. 9. There are presented strain changes after 2 000, 5 000 and 20 000 load cycles.

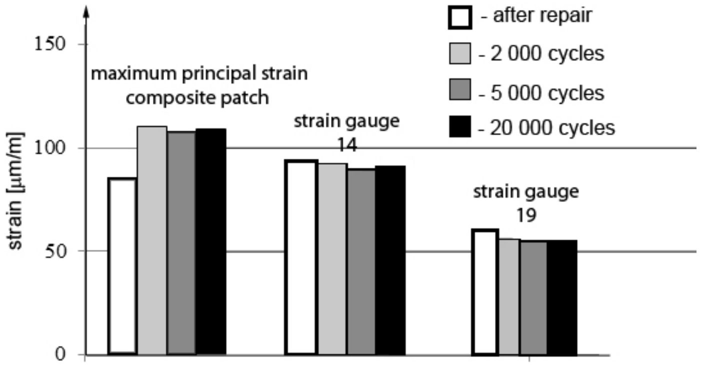

Fig. 9. Maximum main strain in the composite patch and in the adjacent area to repaired zone

It can be assumed; based on the research included in the paper [5], that repaired zone damage of this type (including disbanding of metal insert) causes significant change of maximum main strain in a composite patch and in a repaired skin. Since this kind of phenomenon did not occur during the conducted tests, it can be assumed that the repaired zone was not damaged again during durability research.

An interesting phenomenon of strain decrease in the composite patch in relation to time of measurement was noticed during the tests. The tested strains of the composite patch observed directly after durability tests were higher than strains measured a few hours after finishing tests. The same effects were not noticed in the strain gauges mounted on the metal components of structure. It seems that the observed phenomenon is connected to viscoelastic features of composite materials and it can be essential with regard to the process of composite patches monitoring. In order to confirm that the observed phenomenon exists, additional tests were performed using the same testing stand and repaired beam of Su-22 plane horizontal tail. Using this type of structure, higher values of strain can be obtained during durability tests without simultaneous failure of structure. The tail is made of aluminum alloy 2024 (tail skin) and aluminum alloy 2017 (tail strip). The shape and dimensions of the tested beam is presented in Fig. 10.

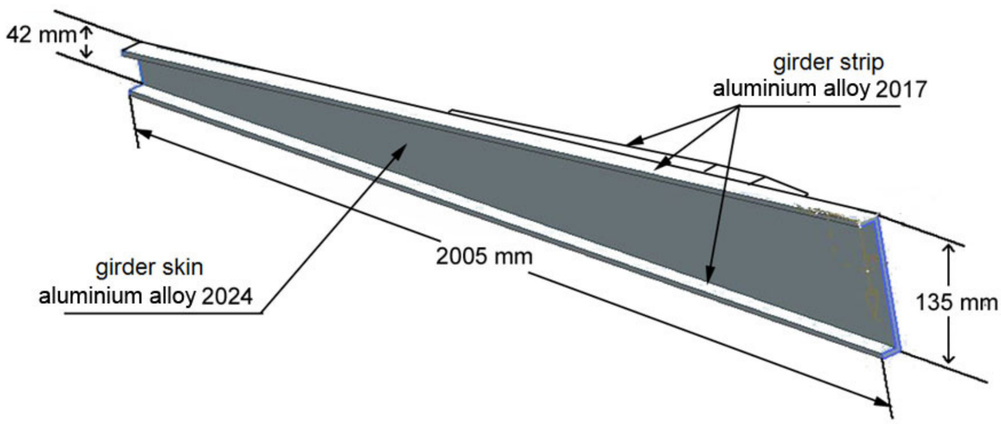

Fig. 10. The shape and geometric dimensions of the tested beam

The extraction of material in the upper strip of beam (Fig. 11) was repaired with metal insert and composite patches.

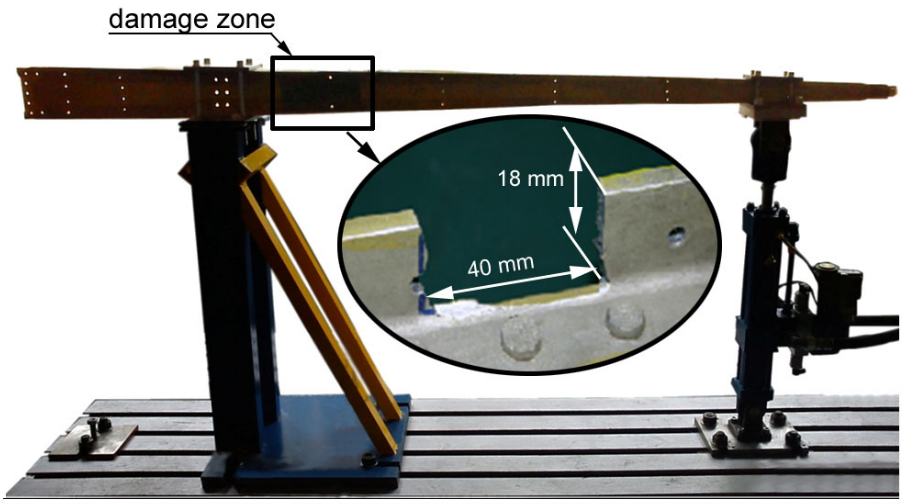

Fig. 11. The beam mounted in the testing stand and the view of damaged zone

To perform repair there were a fabric glass, a carbon fabric and a carbon-aramid fabric with a weight of 160 g/m^2 and Epidian 57/Z1 epoxy adhesive used. The repair was performed according to the scheme presented in Fig. 12a. The two strain gauges were mounted in the area of repair on the composite elements above the edges of metal insert – Fig. 12b.

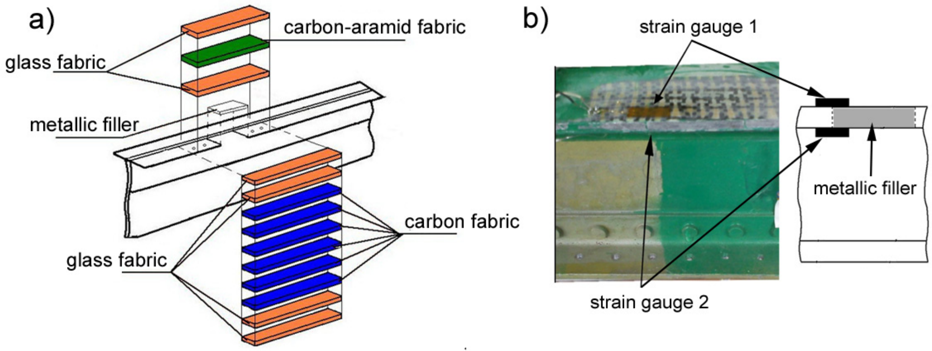

Fig. 12. The repair scheme of the damage beam – a) and the place where strain gauges were mounted – b)

The beam were subjected to cycling bending by shifting a hydraulic actuator down at the distance of 20 mm (it was achieved by load of 890 N). 15 000 cycles were performed during test in three stages of 5 000 cycles. The tested beam was off-load after each stage and strains in the composite patches were checked. The subsequent stages were executed at intervals. The first measurement of strain was performed directly after first stage (5 000 cycles) and the further measurements were conducted after 12 and 48 hours of interval. The results of test are presented in Fig. 13. The measured strains were returning to initial level with increasing interval between performance of measurement and completion of durability test. It seems to be connected with viscoelastic features of the applied composite patches.

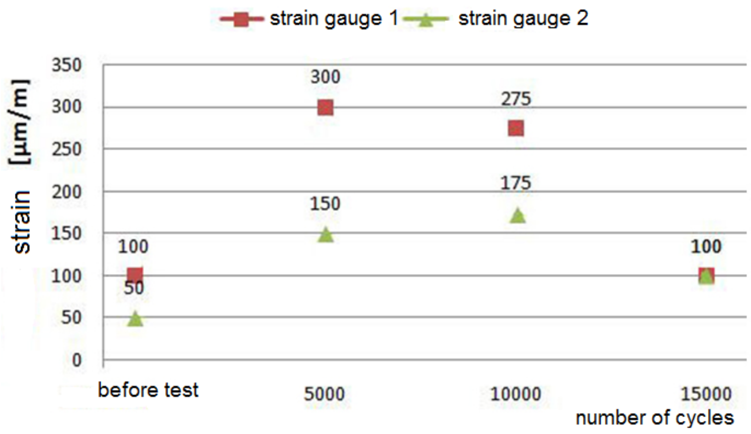

Fig. 13. The values of strain in repaired zone after 3x5 000 cycles of load (displacement 20 mm caused by force of 890 N)

4 Conclusion

Taking into account the performed studies, following conclusions can be drawn:

- The skin damage of a semi-monocoque structure causes a local change (increase) in strain mainly in the damage zone, which is limited to adjacent ribs and stringers.
- A stiffness reconstruction of damaged panel should be the main criterion in designing of semi-monocoque structure skin repair in order to protect it against local loss of stability.
- The use of composite materials and metal inserts is an effective technology, which can be used to perform repair of skin of thin aircrafts components. The use of insert which has the same stiffness as a repaired element is a solution that allow to restore local stiffness of repaired panel. The composite patches, which are filled with epoxy resins, join all elements of repaired zone and increase its fatigue life.
- One of solution that can be used to composite patches monitoring is strain sensors employment. When applying this kind of sensors, a certain limitation should be taken into account which involves viscoelastic features of composite materials.

References

[1] Jones R., Chiu W.K., Smith R., Airworthiness of composite repairs: Failure mechanisms. *Engineering Failure Analysis,* 2, pp.117-128 (1995).
[2] Baker A., Rose F., Jones R. (eds). *Advances in the bonded composite repair of metallic aircraft structure*. Elsevier Science Ltd, Oxford (2002).
[3] Cichosz E. *External load of aircraft*. WAT, Warsaw (1968).
[4] Rośkowicz M. Stability of composite repair plate. *Bulletin WAT*, 4 (648), pp. 257-272 (2007).
[5] Kijewski P., Rośkowicz M. Diagnose of composite patch. *Technology and Assembly Automation,* 2, pp. 49-55 (2011).