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RESEARCH ON DURABILITY OF COMPOSITE MATERIALS USED IN REPAIRING AIRCRAFT COMPONENTS

BADANIE MATERIAŁÓW KOMPOZYTOWYCH WYKORZYSTANYCH DO NAPRAWY KONSTRUKCJI LOTNICZYCH

The paper presents methodology and results of research on durability of composite materials, which were applied to expedient repair of damaged aircraft components. Numerical calculation and experimental tests were conducted during research. The obtained results proved that failure of aircraft components is connected with local loss of stability in case of aircraft's skin and stiffness in case of girders and beams. The damaged components were repaired with the use of a metal insert and a composite patch. The use of a metal insert, which had the same stiffness as repaired elements, was a solution which allowed to restore local stiffness of repaired components. Composite patches which were formed with glass, carbon or aramid fabrics impregnated with epoxy resins, also created an adhesive bonds and joined all elements of repaired zone. The experimental tests proved that the executed repairs whose time was limited to 120 minutes improved stability, stiffness and fatigue life of the repaired components.

Keywords: *failures of aircraft, expedient repair, composite materials, composite patch, research on durability of composite materials.*

W artykule zaprezentowano metodologię i wyniki badań wytrzymałościowych materiałów kompozytowych, które zostały zastosowane do naprawy doraźnej elementów konstrukcji lotniczych. Podczas badania wykorzystano obliczenia numeryczne i badania eksperymentalne. Otrzymane rezultaty dowodzą, że uszkodzenie elementów konstrukcji lotniczych związane jest z lokalną utratą stateczności, w przypadku poszycia samolotów, oraz utratą sztywności, w przypadku belek i żeber. Uszkodzone elementy zostały naprawione z użyciem metalowej wkładki usztywniającej i łaty kompozytowej. Użycie wkładki o tej samej sztywności co naprawiany materiał pozwoliło odzyskać sztywność w trefie uszkodzenia. Łaty kompozytowe, które zostały uformowane z wielu warstw tkaniny (szklanej, węglowej i aramidowej) i nasączone żywicą epoksydową, pozwoliły na utworzenie złącza klejowego, w ramach którego scalono wszystkie elementy węzła naprawczego. Badania eksperymentalne dowiodły, że przeprowadzone naprawy, których czas był ograniczony do 120 minut, poprawiły stateczność, sztywność i trwałość zmęczeniową naprawionych komponentów.

Słowa kluczowe: *uszkodzenia samolotów wojskowych, naprawa doraźna, materiały kompozytowe, łata (nakładka) kompozytowa, badanie wytrzymałości materiałów kompozytowych.*

1. Introduction

Composites are often used in airplane structures because of their specific strength [5]. Therefore, the repairs with use of composite materials are an effective method of damages removal in case of aircraft 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, which are made of aluminum alloy [7]. The skins repairs include cracks of air components, dents and material losses of fuselage and wings skin [1, 3].

An assessment of repair 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 airplanes manufacturers create their 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 [2]. The experiences of some companies providing repair of airplane components show that the general principle of repair designing and its 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 [6, 9].

2. Methodology of research

The opportunities of repair of damaged components were assessed with use of chosen composite materials. The composite patches were shaped in order to substitute metal plates and riveted joints were replaced with adhesive joints. A repair of skin components was researched on the basis of "TS-8 Bies" aircraft and a repair of girders and beams was researched on the basis of "Su-22" aircraft.

The numerical calculations were used to assess the effect of defects in the research aircraft components. The numerical calculations were conducted with finite elements method (Nastran/Patran software environment). Hexagonal elements of HEX8 type were used to create numerical model. The loads were modeled with elements of RBE 3 type.

(*) Tekst artykułu w polskiej wersji językowej dostępny w elektronicznym wydaniu kwartalnika na stronie ***www.ein.org.pl***

The experimental research studies on composite materials used in repairing of aircraft skin, were conducted with use of a research stand. The stand was 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 (Fig. 1). The horizontal tail of “TS-8 Bies” aircraft 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 on the special frame.

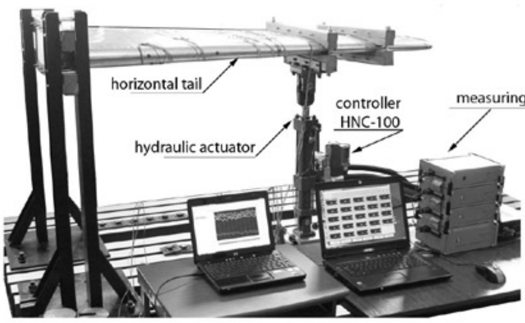

Fig. 1. The research stand with strain gauge bridge of National Instruments and ESAM Traveller Company

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 the Fig. 2.

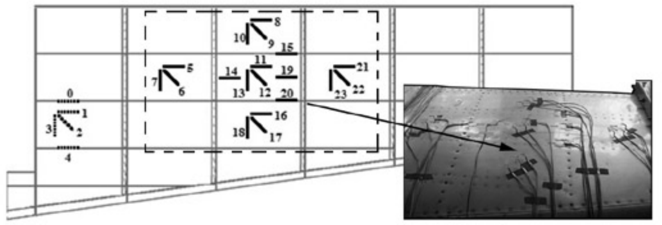

Fig. 2. The arrangement of strain gauges on the surface of tail skin

The effectiveness of girder repair was checked by a comparative analysis of stiffness of girder which was undamaged, damaged and repaired. The stiffness was determined during bending tests. The fixing method of the tested girder and the used research stand is presented in the Fig. 3. The distance between supports was 717 mm and the girder was loaded between supports with use of loading stanchion. A change in strain of the loaded girder was determined in function of the force. A strain sensor was mounted under flange of the girder. On the basis of the obtained results it was possible to assess an effectiveness of applied repairs.

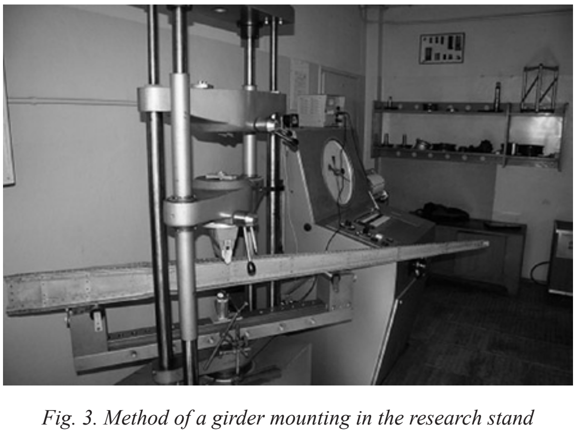

Fig. 3. Method of a girder mounting in the research stand

3. Research on composite materials used in repairing aircraft skin

3.1. Numerical calculations of damaged skin structure

A 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. 4) with particular emphasis on accurate mapping of its individual components.

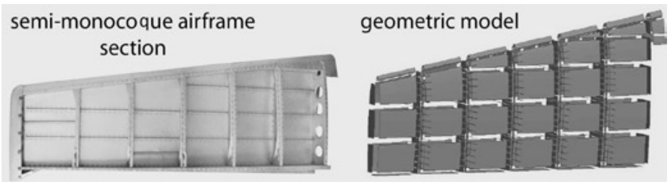

Fig. 4. The section of “TS-8 Bies” aircraft horizontal tail and its geometric model

On the basis of the geometric model, numerical model was generated which included elements of SHELL type (skin, beams, girders’ skin) and elements of BAR type (stringers and girder’s flange). There were boundary conditions (loads and mounts) assumed which correspond with the method of the tail mounting on the testing machine.

The numerical model was loaded in the same way as the real component in the experimental research. It was assumed that the external force would have value of 1000 N when loading the tail. The value was determined in the analysis, which described ultimate loads for this type of semi-monocoque airframe [7]. The same material features were assumed for all finite elements of model, i.e. duralumin 2024T4. The stress and strain distribution was analyzed in the undamaged and damaged tail during numerical calculations. The damage was modeled in the shape of a hole with a diameter of 100 mm, which was located in the middle of tail skin (Fig. 5).

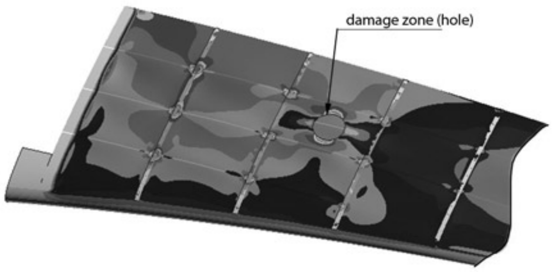

Fig. 5. The model of a tail with damage in the shape of a hole located in the middle of the skin between supporting girders of the generated airframe

The comparison of reduced stress distribution in the damaged and undamaged zone is presented in the Fig. 6. In the area of damage accumulation of stress was observed. More than twofold increase in stress was noticed in the direction parallel to the tail girder and decrease in stress was noticed in the direction parallel to the stringers. This kind of changes in the skin is specific in case of loss of local stability of the skin (in the zone between stringers and girders). On the basis of numerical calculations it was assumed that due to the damage, the changes in stress and strain are limited to skin zone bounded by ribs and stringers. The changes occur to the minimum extent outside these strengthening elements. Simultaneously, the numerical calculations prove 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 joints between the supporting components and the skin carried greater loads.

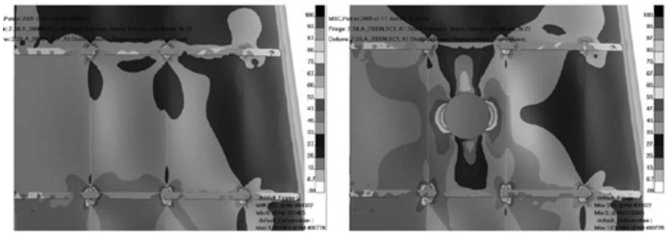

Fig. 6. The comparison of stress distribution in the tested tail: undamaged zone (on the left), damaged zone (on the right)

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

3.2. The experimental verification

The experimental tests were conducted in order to verify the obtained results of numerical calculations. The research on strain in the skin; in statically loaded structure was conducted before damage and after damage. The damage was in the 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 the Fig. 7.

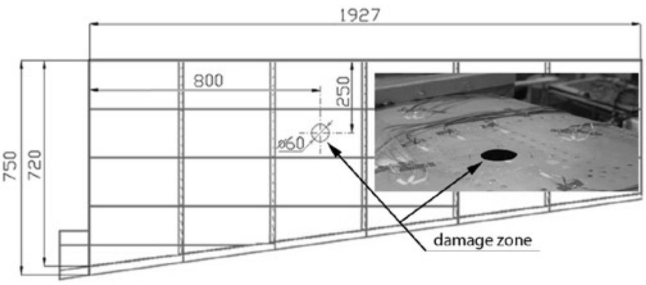

Fig. 7. The geometry of the damage and view of the hole made in the upper skin of tail

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 (case no. 1) and in damaged and loaded construction (case no 2). The results are presented in the Fig. 8.

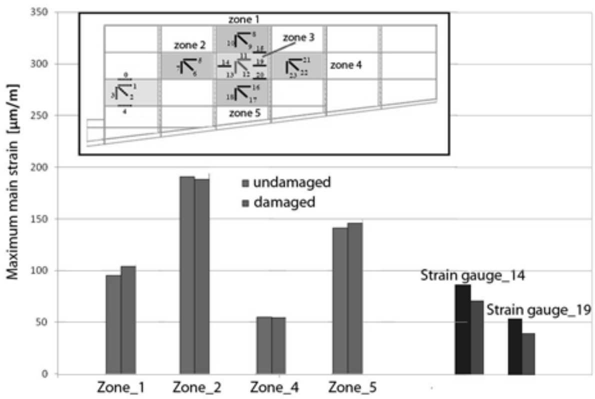

Fig. 8. Maximum main strain in the chosen zones of loaded construction

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.

3.3. 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 skin repair should be stiffness reconstruction of a part of skin in the damage zone (between supporting elements) and therefore, stability sustaining of the repaired element. With the use of results concerning repaired plates [8] which were loaded by shear, the method of tail skin repair was proposed. It included material loss supplemented by a disk-shaped insert of duralumin 2024T4 and formation of one-sided strengthening composite patch achieved by means of 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 of 6 layers of glass fabric Synglass E81weighing 101 g/m^2 was formed. The layers of fabric in a disk-shape were arranged according to the scheme $(0/45/90)_2$ and the successive layers had a diameter of 5 mm less than each previous one in order to obtain a stepped effect. The first layer had a diameter of 100 mm. The last layer with a diameter of 110 mm was made of aramid fabric with the 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 join all elements of the patch. Because there was access to only one side of the tail skin, the problem of holding up of duralumin insert had to be solved during composite patch formation. In order to achieve it a special composite ring was formed on the inside of tail skin. The composite ring consisted of 3 layers of glass fabric and its outside layer had a diameter of 100 mm and its inside layer had a diameter of 40 mm. Sealing the gap between the repaired skin and the metal insert was possible due to the application of the inside ring. In the Fig. 9 there is presented the scheme of layers in the repaired zone and the view of repairing zone prepared to curing process.

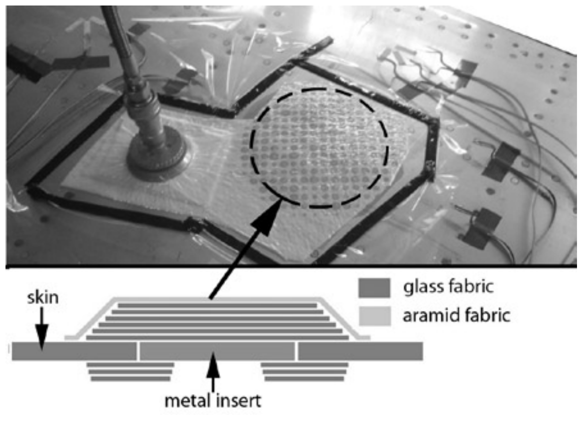

Fig. 9. The scheme of the composite patch application and the repaired area

The applied patch was cured at normal temperature for 24 hours and then heated up at temperature of 80°C within 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 was research on durability. The cyclic load was applied by shifting a hydraulic actuator with the 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 the adjacent area to the repaired zone. The changes of maximum principle strain during durability tests are presented in the Fig. 10. Strain changes after 2000, 5000 and 20 000 load cycles are presented.

On the basis of the research included in the paper [4], it can be assumed that re-damage of repair zone (including disbonding of metal insert) causes significant change of maximum principle strain in the composite patch and in the 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 presented durability research. An interesting phenomenon of strain decrease in the composite patch was noticed in relation to the number of load cycles. The strains of the composite patch measured directly after durability tests were higher than the strains measured a few hours after completion of tests.

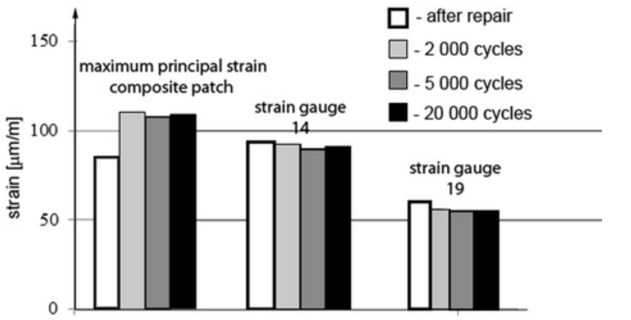

Fig. 10. Maximum principle strain in the composite patch and in the repaired zone

4. Research on composites materials used in repairing aircraft girder

4.1. Numerical calculation of a damaged girder

An important criterion of girder repair effectiveness is restoration of its initial stiffness before damage. Therefore, in the first step numerical calculations were executed in order to assess stiffness changes caused by damage. A numerical model of the girder was created on the basis of the girder geometry of tail-plane of the "Su-22" Aircraft. The created model is presented in the Fig. 11. The following features of materials (duralumin B93) were assumed during analysis: longitudinal modulus of elasticity – 72GPa, Poisson's ratio – 0,3.

The load was modeled by applying force to the middle node of the group creating a stanchion (Fig. 12). The stiffness of elements which creating stanchion was increased due to defining their modulus of longitudinal elasticity two orders of magnitude greater. Additionally, the stanchion's nodes were immobilized along the X and Y axis. The value of the loaded force was 8.3 kN and it was caused by assumption that maximum strain of girder in the loaded part is 2 mm.

The following damages during research were assumed:

– damage in the middle part of the girder partition in the shape of a hole of 29 mm diameter located in the axle of force loading;
– damage in the girder flange in the shape of a rectangle of 30 x 40 mm.

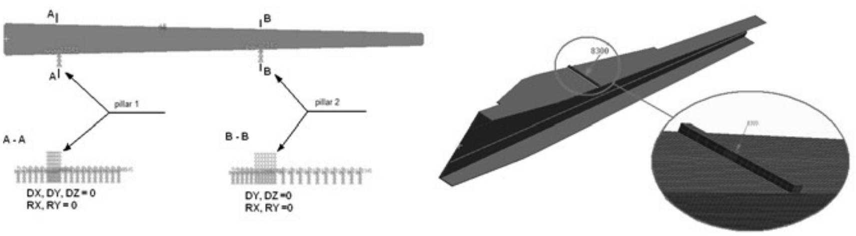

Fig. 11. Method of the girder anchoring and loading

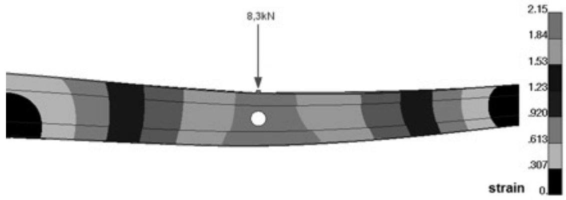

Fig. 12. Method of the girder loading and strain distribution for damage in shape of a hole

According to obtained results, it could be claimed that the damage in the shape of a hole did not influence significantly the stiffness of the girder – the value of strain changed slightly. It was also determined that the damage of the girder flange caused higher loss of stiffness than the damage of the girder partition. Taking into consideration the obtained results it seems that repair of a girder partition should be much easier to execute and more effective. The above conclusions were checked in experimental tests.

4.2. Experimental tests of the repaired girder

Effectiveness of the horizontal tail repair of the "Su-22" Aircraft's girder (Fig. 13) was tested. The girder is produced with duralumin PA7 (partition) and PA30 (flanges). The girder was a type of convergent girder and consisted of a partition (wall) made with sheet metal of 2 mm thickness and riveted to it convergent flanges.

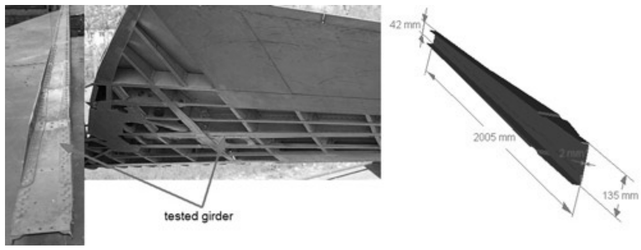

Fig. 13. The tested girder and the horizontal tail from where the girder was dismantled (left) and geometric dimensions of the girder (right)

In the first step the undamaged girder was tested. As a result of the bending tests a graph was determined which presents the girder's strain in function of the force (Fig. 14). The maximum deflection of the undamaged girder was 1.74 mm at the force of 8.3 kN.

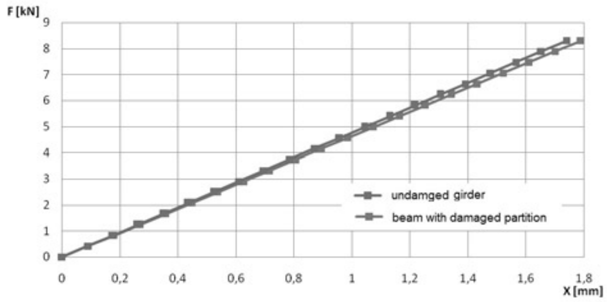

Fig. 14. Dependence of the girder's strain in function of the force changes – F=f(x)

Then the damaged girder was tested. The damage had a shape of the hole of 29 mm diameter and was located in the axle of the load (Fig. 15). The value of the force did not change. As a result of the damage a change of girder's stiffness was noticed. The maximum deflection of the researched part of girder was 1.79 mm (Fig. 14).

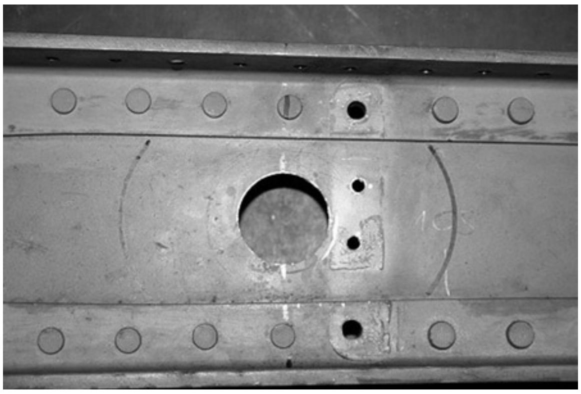

Fig. 15. The view of the girder's damaged partition

Subsequently, a repair of the damaged partition of the girder was executed. The repair was fixed with the use of composite materials and epoxy resin (Epidian 57/Z1). At the beginning, a metal insert (duralumin 2024) was set in the hole. Afterwards, a strengthening composite patch was formed with use of several layers of carbon and glass fabrics, which were laminated with the use of the mentioned epoxy resin (Fig. 16).

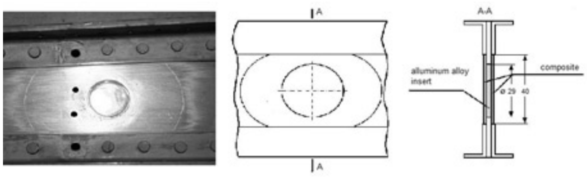

Fig. 16. Method of the girder's partition repair

The strengthening patch was formed in the elliptic shape with dimensions of 140 × 48 mm. Two sorts of plain wave fabrics were used to form the patch. The first one was glass fabric of Belgian company Synglass weighing 160g/m^2 and carbon fabric of KDL company weighing 160g/m^2 as well as. The strengthening patch was formed with five layers of 1.2 mm thickness. The first and last layers were formed with glass fabric and the middle layers were formed with carbon fabric (Fig. 17). The layers of fabric were laminated with epoxy resin. The surface for bonding was cleaned with extraction petrol. The repaired zone was cured in a thermal chamber at the temperature of 100°C within 30 minutes. The presented repair was completed within 120 minutes.

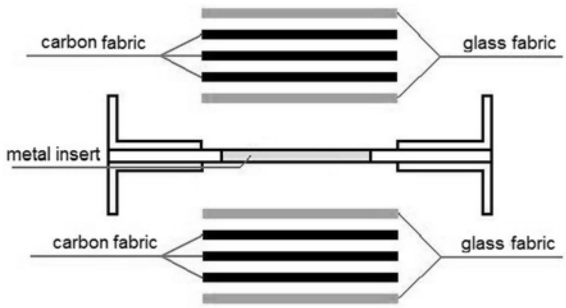

Fig. 17. Method of composite patch forming

Then stiffness was determined by means of bending test which was conducted in the same conditions as the undamaged girder. As a result, a curve F=f(x) was determined (Fig. 18). It was noticed that the applied repair allowed us to restore the initial stiffness of the damaged girder.

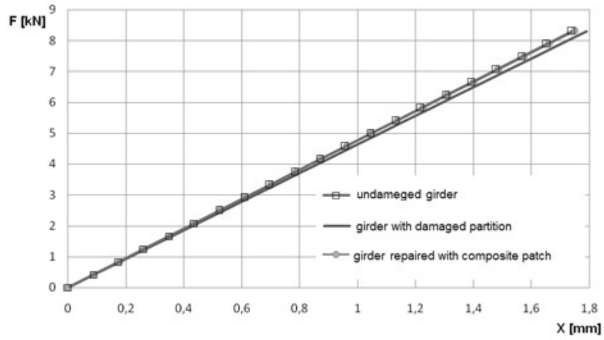

Fig. 18. Dependence of the girder's strain in function of the force changes – F=f(x)

Additionally, fatigue test of the repaired girder was executed. The girder was subjected to cyclical force in the range of 0…8.3 kN. The load cycle was repeated 200 times and organoleptic assessment of the repaired zone was executed. No re-damages were detected. After that stiffness of the researched girder was determined again and no changes were observed.

The next step was to conduct research on the girder which had damage in the flange. The damage was designed in a few steps and after each step the dependence of F = f(x) was determined. The damages were as follows (Fig. 19):

- case no. 1., a single cut in the middle of the flange which was located 120 mm from the axle of force loading;
- case no. 2, double cut in the middle of the flange which was located 80 and 120 mm from the axle of load;
- case no. 3, an extraction in the girder flange in the shape of rectangle of 30 mm × 40 mm.

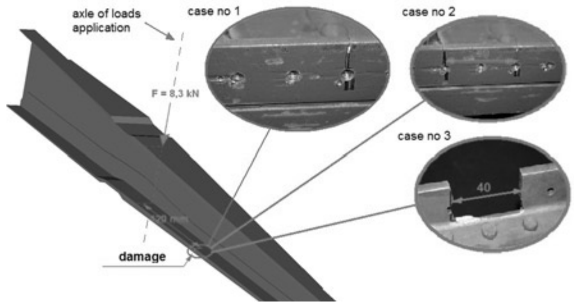

Fig. 19. A fragment of the girder with three damages in the flange

The obtained results of the dependence of F = f(x) are presented in the Fig. 20.

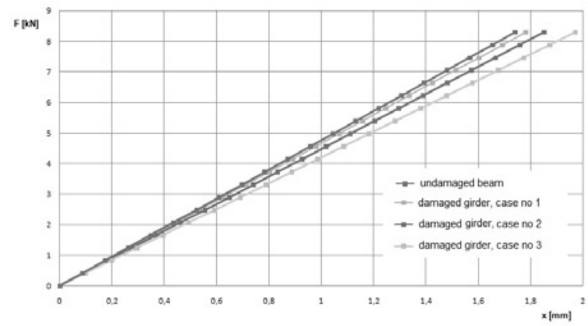

Fig. 20. Dependence of the girder's strain in function of the force changes – F=f(x)

Afterwards, the damaged flange of the girder was repaired with use of duralumin and a steel plate and rivets. The strengthening plate was riveted to the inside of the girder in the place of the damage. The dimensions of the plate were 140 × 25mm and thickness was 2.5mm. The thickness of the plate was selected taking into account the possibility of access to the rivets' heads during crimping. The strengthening plate was one-sided in order not to change external dimensions of the girder. The duralumin rivets were used in the repair, which had a diameter of 4 mm and a shank length of 10 mm. Furthermore, the metal insert which was made with the same material as the repaired girder was implemented in the place of damage (extraction). The view of the repaired zone is depicted in the Fig. 21.

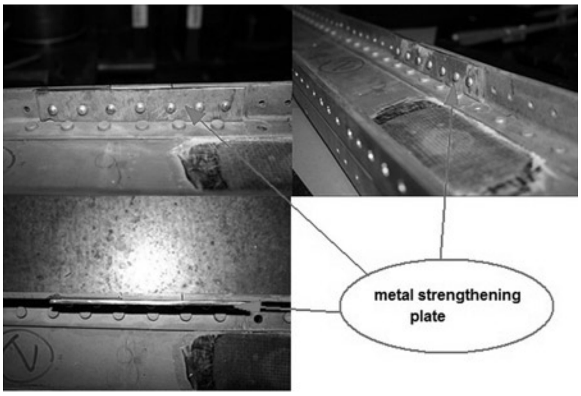

Fig. 21. The view of the repaired zone

The comparison analysis of the girder stiffness which was undamaged, damaged and repaired with riveted plates, is presented in the Fig. 22.

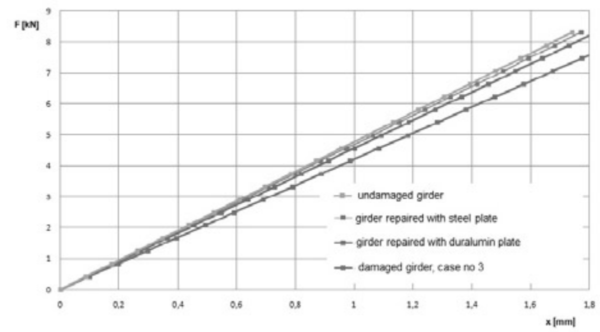

Fig. 22. The comparison analysis of the girder's stiffness which was undamaged, damaged and repaired with riveted metal plates – F=f(x)

As a result of the researched repairs it was not possible to restore initial stiffness of the girder. According to predictions the stiffness of the repaired girder was closest to the initial stiffness in case of using steel plate.

Furthermore, the repair was applied where a composite patch and adhesive joint were used. The damage (extraction) in the girder was implemented with a metal insert of the same material as the girder. Then, there were laminated subsequent layers of carbon, glass or aramid fabric to form a strengthening patch. The epoxy resin of Epidian 57/Z-1 was used to saturate and join layers of fabric. The strengthening patch was formed in the shape of a rectangle which had dimensions of 150 × 25 mm (joined to inside of the girder's flange) and 120 × 40 mm (glued to outside of the girder's flange). 10 layers of fabric were used from the inside and 3 layers were used from the outside of the girder's flange in order to form a composite patch. The number of outside layers was limited to keep the external dimensions of the girder (Fig. 23). The surface of repaired element was cleaned and degreased with extraction petrol and subsequent layers were laminated with epoxy resin, which was hardened in the temperature of 100°C within 45 minutes.

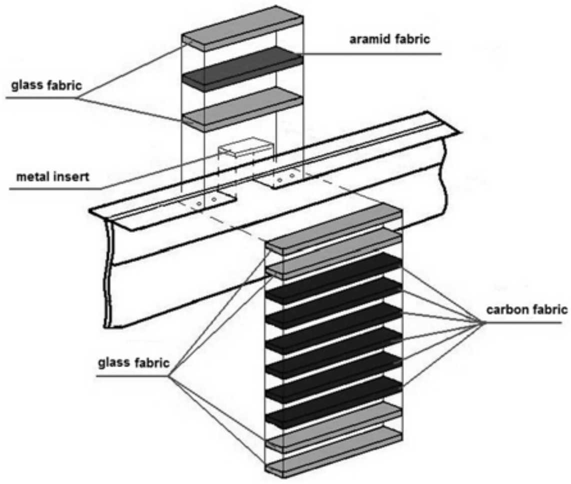

Fig. 23. Method of the strengthening composite patch forming

The view of the repaired zone is presented in the Fig. 24 and research on the girder's strain in function of the force changes (F = f(x)) is presented in the Fig. 25.

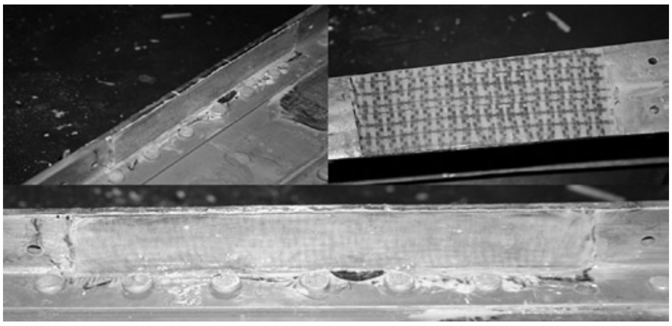

Fig. 24. The view of the repaired flange of the girder

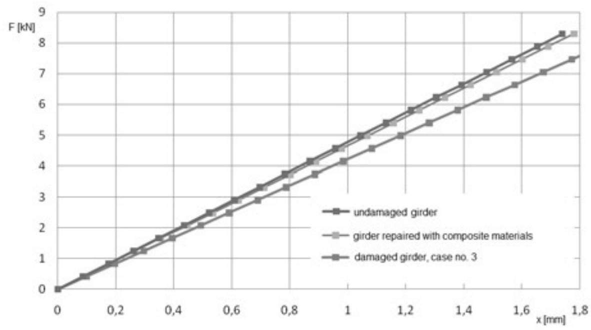

Fig. 25. The comparison analysis of the girder's stiffness which was undamaged, damaged and repaired with composite patch – F=f(x)

The initial stiffness of the girder was not restored as a result of the conducted repair. However, the obtained stiffness was close to the one which was determined for the girder that was repaired with steel riveted plate. Furthermore, fatigue tests of the repaired girder were executed. The girder was subjected to cyclical force in the range of 0…8.3kN. The load cycle was repeated 50 and 100 times and organoleptic assessment of the repaired zone was executed. No re-damages were detected but the stiffness of the repaired girder changed little according to the curve presented in the Fig. 26.

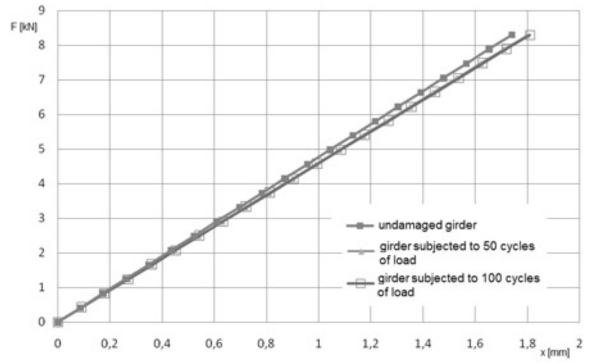

Fig. 26. The comparison analysis of the girder's stiffness which was undamaged, damaged and repaired and subjected to fatigue tests – F=f(x)

5. Conclusions

Taking into account the performed studies, the 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.
- Stiffness reconstruction of the damaged elements should be main criterion in designing aircraft structure in order to protect them 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 the skin of the semi-monocoque airframe. The use of an insert which has the same stiffness as a repaired element is a solution that allows restoration of its local stiffness. The composite patch joins all elements of repaired zone and increases its fatigue life.
- Damage of a girder's partition (wall) can be effectively removed with a composite patch. The conducted repair allowed us to restore the initial stiffness of the repaired girder. Moreover, this kind of repair enables us to sustain fatigue life of the repaired component.
- Damage of the girder's flange causes more trouble. The repair is more time consuming and it is difficult to restore the initial stiffness of the damaged component. The mentioned damage has a greatest impact on the stiffness of considered element. Repairs were executed with 3 technologies in this case. The best results were obtained for repair with use of steel riveted plate and composite patch. The conducted experimental tests confirmed the usefulness of composite patching as an effective repair technology.
- Various methods of expedient repairs were researched. Some assumptions were made while executing repairs. First of all, It should be possible to perform the assumed repairs in the field conditions with the use of basic tools and materials. Secondly, time of repair was limited to 120 minutes.

References

1. Chester RJ, Walker KF, Chalkley PD. Adhesively bonded repairs to primary aircraft structure. International Journal of Adhesion and Adhesives 1999; 19.
2. Cichosz E. External load of aircraft (In polish). Warsaw: WAT, 1968.
3. Jones R, Chiu WK, Smith R. Airworthiness of composite repairs: Failure mechanisms. Engineering Failure Analysis 1995; 2: 117–128.
4. Kijewski P, Rośkowicz M. Diagnose of a composite patch (In polish). Technology and Assembly Automation 2011; 2: 49–55.
5. Komorek A, Przybyłek P. Examination of the influence of cross-impact load on bend strength properties of composite materials, used in aviation. Eksploatacja i Niezawodnosc – Maintenance and Reliability 2012; 14(4): 265–269.
6. Kuczmaszewski J. Fundamentals of metal-metal adhesive joints design. Lublin: University of Technology and Polish Academy of Science, 2006, ISBN 83-89293-11-0.
7. Rośkowicz M, Smal T. The use of composite adhesives in repairing of aircraft semi-monocoque airframe. In electronic proceeding of the 15th European Conference on Composite Materials, Venice, 24–28.06.2012.
8. Rośkowicz M. Stability of composite repair plate (In polish). Bulletin WAT 2007; 4(648): 257–272.
9. Rudawska A, Dębski H. Experimental and numerical analysis of adhesively bonded aluminium alloy sheets joints. Eksploatacja i Niezawodnosc – Maintenance and Reliability 2011; 1(49): 4–10.

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