Polish Society of Composite Materials
Polskie Towarzystwo Materiałów Kompozytowych

22: 2 (2022) 114-119

COMPOSITES THEORY AND PRACTICE
ISSN: 2084-6096
ISSN (online): 2299-128X

Sintija Zeltiņa[1], Mārtiņš Kleinhofs[1], Irena Chatys[2*]

[1] Riga Technical University, Institute of Aeronautics, Faculty of Mechanical Engineering, Transport and Aeronautics, 6B Ķīpsalas St., Kurzemes rajons LV-1048, Riga, Latvia
[2] Kielce University of Technology, Faculty of Mechatronics and Machine Design, al. 1000-lecia PP 7, 25-314 Kielce, Poland
**Corresponding author. E-mail: chatys@tu.kielce.pl*

Received (Otrzymano) 7.04.2022

DIFFERENT TECHNIQUES OF REPAIRING COMPOSITE STRUCTURES USED IN AIRCRAFT

The paper presents a fairly extensive review of the procedures and recommendations for checking the damaged repair area of various composite structures (such as sandwich or honeycomb structures) of airframe elements in military and civil airplanes (especially Boeing and Airbus airplanes and sailplanes). The presented requirements for the inspection and assessment of damage (e.g. delamination in layers or in the honeycomb, internal delamination in the laminate, and damaged fibers) should not exceed the permissible damage limits (PDL). The repair requirements (to ensure the continuing airworthiness of the structure) are largely dependent on the classification of the structure under consideration. In these analyses, a fairly important function is performed by non-destructive methods (such as ultrasonic bond testing or using eddy currents), thus assessing the degree of damage and the ongoing maintenance process.

Keywords: composite materials, components, repair ways

RÓŻNE TECHNIKI NAPRAWY STRUKTUR KOMPOZYTOWYCH STOSOWANYCH W SAMOLOTACH

Praca przedstawia dość obszerny przegląd procedur i zaleceń sprawdzenia uszkodzonego obszaru napraw różnych struktur kompozytowych (jak warstwowych czy o strukturze plastra miodu typu „sandwich") elementów płatowca, w samolotach wojskowych i cywilnych (szczególnie samolotów firmy Boeing i Airbus oraz szybowców). Przedstawione wymogi kontroli i oceny uszkodzeń (np. delaminacji w warstwach czy w plastrze miodu, wewnętrznych rozwarstwień w laminacie oraz uszkodzonych włókien) nie powinny przekraczać dopuszczalnych limitów uszkodzeń (PDL). Wymagania dotyczące napraw (aby zapewnić ciągłą zdatność konstrukcji do lotu) są w znacznym stopniu uzależnione od klasyfikacji rozpatrywanej konstrukcji. W tych analizach dość ważną funkcję spełniają metody nieniszczące (jak testowanie ultradźwiękowego testera wiązania czy prądy wirowe), oceniając w ten sposób stopień uszkodzenia i bieżący proces obsługi technicznej.

Słowa kluczowe: materiały kompozytowe, komponenty, sposoby naprawy

INTRODUCTION

Carbon epoxy and similar composite airframe components are immune to the costly forms of deterioration that occur in aluminum and most of the alloys used in airframe structures: fatigue, cracking and corrosion. However, composite components are much more easily damaged in service by mechanical impact and are also more prone to heat damage. Thus, the ability to repair such damaged structures is a critical requirement to maintain their service life and for the continued airworthiness of composite components in military and civilian aircraft.

Various outside factors can have a major effect on part damage, such as:

1. Delamination damage caused by inadequate shimming during component assembly
2. Local overheating caused by the impingement of hot exhaust gases or from a lightning strike
3. Impact from dropped tools, bird strikes
4. Mishandling, etc.

Lightweight honeycomb structures in particular require many repairs because the thin face skins are easily damaged by mechanical contact. Moisture penetration can occur through damaged face skins as well as through badly-sealed elements, resulting in corrosion damage if the core is an aluminum alloy. Damage is more of a problem with older generation composite materials involving thin skin sandwich and brittle matrices. The new composite materials for Boeing and Airbus are more complex, integrally stiffened structures and tougher matrices.

In the new structures, damage is more difficult to repair in the matrices due to the postbuckled design. Nevertheless, larger more integrated structures also make part replacement difficult and costly, thus increasing the need for reliable high performance repairs. To ensure maintenance and repair, structural renewal materials must be divided in fibers, binders, monolayer blanks, etc. Based on the statistics of faults detected in the operation process, we can make certain predictions.

CLASSIFICATION OF STRUCTURE TYPES

To ensure a correct flow of work, repair damage evaluation and repair, it is necessary to establish the criticality of the structure. The classification of structures is:

- Primary structure: Pressurization loads, carrier loads. Failure would reduce the structural integrity of the aircraft.
- Secondary structure: If the structure was to fail, it would affect operations but not its load capacity.
- Critical structure: The integrity of the structure is essential to maintain the overall flight safety of the aircraft.
- Tertiary structure: Failure of the structure would not significantly affect the operation of the aircraft.

Inspection, damage assessment and repair requirements differ significantly among these classifications; however, even within a single component the allowable damage type and size, and consequently the acceptable repair action, vary according to the criticality of the damage region. The component is generally zoned by the original equipment manufacturer (OEM) in the structural repair manual (SRM) to indicate these regions. SRM typically addresses repairs to a nonprimary structure. Repairs outside the scope of SRM, particularly to critical regions of a primary structure, require engineering design and approval by OEM [1].

Composite material inspection

Composite material repairs are made if damage is not more than the permitted damage limits (PDL). If this limit is exceeded, we need to have a permit to make repairs. According to the related damage, we can choose the repair technologies, tools, and processes.

For an easy way to inspect a damaged area in need of repair, we can use the following inspection methods: visual, eddy current, tap test, ultrasonic testing, and ultrasonic bond testing. Composite repairs can be performed with a repair tool package. Hand tools can cut material and for that we use special scissors, cutters and knives. Items made from Kevlar are more difficult to cut than fiberglass or carbon. A squeegee and a brush are used to impregnate dry fibers with resin for wet layup. Markers, rulers, and circle templates are used to make a repair layout. There are numerous suppliers that can provide the right tool kits.

Why we need to make repairs

External object impact is the main type of damage in composite material structures. To ensure continuing operation, it is necessary to identify the damage severity and detectability as part of the ongoing maintenance process. Methods of analytical assessment of residual strength in damaged composite components are needed to ensure that only the necessary repairs are undertaken. Damage assessment results in one of the following decisions: we can choose to do no repairs when the damage is negligible or we can perform cosmetic repairs to correct minor damage. For more extensive damage, we need to carry out structural repairs because the strength has decreased because of subsequent inspections. Repair is not acceptable and we need to replace the whole component. When there is penetration damage, the need for structural repair is obvious. The decision to conduct repairs is more difficult as we have to measure damage.

Allowable damage or allowable manufacturing damage/defects that do not degrade the structural integrity can go undetected by scheduled inspections. They can be limited delamination, porosity and small scratches, for which no repairs are needed. Visible impact damage, deep gouges, delamination or debonding must be inspected periodically and should not grow or, if it is slow, the residual strength of the damaged structure during the inspection must be sufficiently greater than the limit load capability. Repairs of damages like this can increase the load capability. This damage can be readily detected within a few flights by operations or maintenance personnel without special skills in composite inspection. The structure can still maintain the limits. Repair is required immediately to restore the designed ultimate load capability.

Discrete source damage reduces the strength below the designed limit load and flight maneuvers become limited. These structures can maintain safe flight at reduced levels. The most popular are bird strikes, tire bursts, and severe in-fight hail [2]. The last category of damage beyond the design limits, visible and noticeable in operation, is atypical collisions with service vehicles, flight overloads, etc., requires prompt repair.

Requirements for repair

The repair scheme selected for structural restoration must be the simplest and least intrusive to restore the stiffness and strain capability of the structure to the required design strength without compromising other functions of the component or structure and can be implemented in either a depot repair facility or a field-repair environment. It is necessary to restore the capability of the structure to withstand design ultimate loads and to maintain this capability (or higher) for full-service life. Structural requirements for the repair vary according to the component or structural element. For the safety of critical structures, existing regulations require the repaired structure to be able to match the

damage tolerance and fatigue durability of the original structure. It means that the repaired damage should be able to carry loads and reach the ultimate load fatigue endurance limits [3].

In addition to strength, stiffness and stability are other key properties that must be restored. They include different things like aerodynamic shape, balance, clearance and moving parts, as well as resistance to lightning strikes. The type of structure and its accessibility are major considerations that determine the repair. The repairs adopted for skin honeycomb structures are very different to those required for damage to a highly loaded thick-skin primary structure. Naturally, the repairs apply to all kinds of composite structures [4-7] and are not limited to the aviation applications.

Repair levels

Repair activities are performed at one of the following levels:

- Field level repairs – performed directly on the aircraft in situations where skilled personnel and adequate facilities are unavailable. These repairs are mostly performed on nonprimary structures or are noncritical repairs to the primary structure. Emergency field repairs may be undertaken in a battle situation to make the aircraft operational or enable it to return back to the base. Battle damage repairs may be carried out rapidly by relatively unskilled personal.
- Depot level repairs – performed when skilled personnel and facilities are available. If the component is too big and difficult to remove from the aircraft, repairs are conducted directly on the elements onboard [2].

DIFFERENT TYPES OF REPAIRS

Nonstructural repairs (Cosmetic repairs)

When damage does not degrade the structural strength below the designed ultimate load, repairs are performed to limit damage growth and the need for further repairs.

Mechanically-fastened repairs with pre-cured patches

When the proper facilities or curing and bagging equipment are not available for on-line work, a pre-cured patch inserted with blind fasteners may be used. This type of repair usually does not restore the maximum strength and may cause vibration when performed on critical parts. This type of repair may be considered a temporary repair until the damage can be ground down and the patches are correctly laminated at the correct pressure, taking into account the exothermic reaction (temperature). Many times, these repairs are performed with common repair materials such as sheet metal plates and rivets. If composite patches are required, kits with pre-cured patches may be available.

Pre-cured patches come in several sizes: two-inch, and four-inch. These patches are produced to have the fibers of each layer in the correct orientation. Such a patch may have a peel ply layer that indicates the orientation in which it should be laid into the routed-out area.

Some manufacturers offer various sizes of core material that is bonded to pre-cured laminates. These premade patches are available, so the technician can simply route out the damage area and insert the core and laminate patch. This type of repair may have a type of adhesive pre-applied to help it bond.

Usually, the patch uses some type of mechanical blind fastener that is drilled through the patch into the surface of the original part to hold the patch in place, then it is further stabilized with blind fasteners. The problem with using blind fasteners in a core structure is that they tend to crush the core structure. This may cause the core to delaminate from the plies. Again, this type of repair may be considered a good temporary repair.

Potted repair

Potted repair does not give as much strength to the composite structure as refitting the hole with a new core. Filling the hole with a resin/microbalance mixture adds weight to the part and decreases the flexibility. Further flexing of the part might cause the potted plug to dislodge, but many structural repair manuals still recommend this type of repair for advanced composite structures. Most potted repairs are appropriate for foam core sandwich structures. In some cases it may be permissible to drill a small hole into the delaminated area and inject resin into a honeycomb disbanded area [8].

Delamination

Delamination occurs when the laminate layers become separated or when the plies separate from the core material. Delamination is sometimes referred to as unloading or disbanding of the plies. Sometimes delamination can be detected by shining a light over the part and looking at the damaged area at an angle. The damage can be recognized as a bubble or an indentation. Internal delamination is the separation of plies that do not extend to the edge or a drilled hole area. It is important to properly assess the extent of an internal delamination using the appropriate NDI method. If the delamination is over a core area, the delamination can move even further because of resin injection. If you perform a resin injection, use a low viscosity resin and apply pressure to the area so that the skin attaches to the core [9, 10].

Delamination injection repair

If the internal delamination is minor, it can sometimes be repaired by simple resin injection into the cavity formed by the ply separation. Injecting resin

fills the area and reattaches to the skin. The repair may not be approved by the manufactures because the delamination cavity is filled with resin, which adds extra weight. This will not be significant for the prime surfaces of the aircraft. After this repair, we need to understand that this will cause the resin to be ineffective in restoring the strength and could cause brittleness. During flight, the extra resin in this area would not be as flexible and may cause further delamination.

Delamination of honeycomb

This simple repair does not require vacuum bagging and is rarely cured with heat. In some cases, the manufacturer supplies a component with a potting compound installed to accommodate a fastener. If it is not correctly positioned, it maybe repositioned. If the part failed because the fasteners pulled out, filling the damaged hole and re-drilling it may not result in a quality repair. It can be pulled out if the repair condition and strength are not appropriate. An insert or grommet may be installed permanently with an adhesive. The fastener can be used without causing further damage to the component structure. It is only necessary to ensure that the adhesive does not change its microstructure under the influence of temperature. It would increase the brittleness and volume of the joint and may result in the formation of stresses and micro-cracks [11].

Damage removal and ply replacement

This repair calls for the removal of the damaged laminated plies and subsequent replacement of the removed plies with new ones. The replacement plies are cured with heat and pressure to restore the original component strength. The new impregnated and pre-cut patches are laid into the routed-out area with the weave of the new patches in the same orientation as those of the original part. An overlap patch is usually one inch larger than the repair ply. It is used as bridge between the repair and the original part. The overlap patch initially sits on top of the part but with the heat and pressure that are applied during the curing cycle, it compresses to the level of the surface.

Glider step cut repair

The step cut repair is probably one of most commonly used repair types. Nonetheless, there are differing opinions on how the plies should be laid in different materials and regarding the repair options. Some glider repair manuals call for this type of repair. This repair prevents delamination and detachment of the repaired surface in the event of a strong impact. Very often the repaired surfaces do not match the shape of the cut area. This creates an air gap around the edges of the patch. In such cases an air gap occurs and the repairs are not considered airtight [8].

Fiber repair

Damage that affects all of the laminate layers of a structure can be addressed in several ways depending on the number of plies, the location of damage, and the size of damage. Before the repair, we must prepare the surface and then remove by cutting the plies. After this step, it is necessary to select and mix the proper resin and repair material. Then we prepare bonging patches and will use a backing plate to support the structure from the backside if it is required. A vacuum bag or pressure are used.

Repairs to sandwich structure

Sandwich structure panels are vulnerable to impact damage primarily because these structures use relatively large face sheets. Delamination may occur at the point where the core is laminated to the skin. Punctures on one side that damage the face sheet and core may be repaired with a number of ways dependent on the size, extent of damage and location. Minor delamination between the skin and core can be addressed by resin injection similar to how laminate ply delamination is repaired. Holes are drilled into the skin, then resin is injected into the delamination cavity. The repair is then clamped and cured. It becomes more difficult as we need to cut out the delaminated skin, cut back the laminate skin and fill the core area with heat and pressure. This same repair may be accomplished without filling the core but rather by adding a layer of adhesive on top of the core material and then laying the patches in place and curing. This repair can be performed according to the supplier’s recommendations.

Repair to both face sheets of honeycomb

If one side is damaged and is inaccessible, the repair can be completed using a pre-cured patch, which is bonded to the opposing side and cured. Core inserts to the plies are used.

Process of composite material repair from SRM

There are risks when working with composite repair materials. To reduce these risks to a minimum, read and follow the indications given above. The task supporting data are given below.

The information given in this topic concerns standard repairs to metallic sandwich structures in TASK 51-77-13-911-013 through TASK 51-77-13-911-017. The repairs include skin and core damage, delamination and debonding. The repairs are for general use and they should be used only when reference to them is made in a specific repair, or if the identification section ('Action or Repair' column of a specific chapter) calls up the use of these repairs. The specific chapter will give the required allowable repair limits and applicable repair schemes. Before starting to repair the damage, follow

the steps given in the Inspection/Repair Flow Chart in Chapter 51-77-10 [12].

For repairs on composite sandwich structures in TASK 51-77-13-911-001 through TASK 51-77-13-911-012, the repair materials must – as a general rule – match the original material as follows:

- only carbon fiber material is used to repair a carbon fiber structure
- only glass fiber material is used to repair glass fiber and aramid fiber structures.

For repairs to metallic sandwich structures, refer to TASK 51-77-13-911-013 through TASK 51-77-13-911-017 for the repair materials. For information concerning temporary repairs, refer to TASK 51-77-13-911-009. In the case of delamination or damage to the skin only, refer to TASK 51-77-12-911-002 through TASK 51-77-12-911-007. The main pieces of information are:

A. Area of Applicability
B. Weight Variant Applicability
C. Limits/Restrictions
D. Repair Category
E. Inspections
F. Work Set Up Information
G. Material References, repair materials.

Skin Repair with Core Replacement (Scrim Cloth Procedure) covers the following steps: First, remove the damaged skin and prepare the repair area. Then lightly abrade the original adhesive and remove remaining the core on the bottom of the hole. Use a 400 grade abrasive cloth to remove the adhesive fillet and to receive a smooth matt surface. Be careful not to remove all of the bonding adhesive. Next, clean the area with a vacuum cleaner. Afterwards, clean the repair area with a cleaning agent. Prepare the honeycomb core plug. Put one ply of scrim cloth (Material No. 05-021) on the bottom of hole. Then prepare the adhesive paste and apply it onto the bottom of the hole and put in the scrim cloth. Brush the scrim cloth with a brush to make sure that it is fully soaked with the adhesive paste. If necessary, add more adhesive.

The important restrictions are: No metal bristle brushes (wire brushes) are permitted for this operation. Apply a thick layer of adhesive to the sides of the hole and to the mating surfaces of the core plug. Put the plug into the hole and add adhesive as necessary. Put the parting film (Refer to TASK 51-77-11-911-002) on the repair area and let it cure under pressure. To apply pressure, use a vacuum bag (Refer to TASK 51-77-11-911-022), weights or clamps. As an alternative method to make the curing procedure faster, heat can be applied.

When the alternative method is used:

- Install the heating equipment and thermocouples.
- Let the adhesive gel before applying heat to prevent unwanted adhesive flow.
- Let the adhesive cure at the required time and temperature.
- Remove the equipment and parting film. Remove the surface of the honeycomb core plug until you get to the same level with the skin surface.
- Clean the repair area with a cleaning agent.
- Prepare the external doubler, of a thickness equal to the skin thickness or 0.2 mm (0.008 in.) thicker. If the doubler has been stored, clean the mating surface with a cleaning agent.
- Cut one ply of scrim cloth (Material No. 05-021) to put on top of the honeycomb core plug.
- Prepare the adhesive paste.
- Brush the scrim cloth with a brush to make sure that it is fully soaked with the adhesive paste. If necessary, add more adhesive.

No metal bristle brushes (wire brushes) are permitted for the above described operations.

Finally, apply adhesive paste to the mating surface of the doubler and put it on the repair area in position. Fasten the doubler in position with polyester tape. Apply equal pressure and remove unwanted adhesive. Put the parting film on the repair area and let it cure under pressure. To apply pressure, use a vacuum bag, weights or clamps.

The curing time can be decreased by applying heat (Refer to TASK 51-77-11-911-023). After the time of curing, remove the equipment and parting film. Inspect the adhesive fillet and repair area. Do not remove the adhesive fillet, and then seal the edge of the repair area with sealant. Restore the surface protection [13]. It is necessary to beware of preparing large volumes of resin at once because of the evident relation between the volume of curing resin and intensity of the curing process [14].

CONCLUSIONS

Repairs of composite structures in airplanes cover a very formalized group of procedures. Considering the high load-capacity and great responsibility of structural elements in aviation, repairs are not particularly welcome. However, as the review of procedures carried out above shows, very extensive and strict guidelines guarantee prolongation of the service life of the aircraft. In the opposite situation, the decommissioning of aircraft caused by unrepairable defects would be disastrously expensive for airlines.

REFERENCES

[1] Baker A.A., Murray L., Sco, 2016, 595-601.
[2] Ridha M., Wanga C.H., Chen, B.Y., Tay T.E., Modeling complex progressive failure in notched composite laminates with varying sizes and stacking sequences, Composites Part A 2014, 58, 16-23.
[3] Fawcett A., Trostle J., Ward S., 777 Empennage Certification Approach.
[4] Chatys R., Orman Ł.J., Technology and properties of layered composites as coatings for heat transfer enhancement,

Mechanics of Composite Materials 2017, 53(3), 351-360, DOI: 10.1007/s11029-017-9666-8.

[5] Orman Ł.J., Chatys R., Heat transfer augmentation possibility for vehicle heat exchangers, Proc. of Int. Conf. TRANSPORT MEANS, Lithuania 2011, 9-12.

[6] Otmianowski T., Antoszewski B., Żórawski W, Local laser treatment of tribological plasma sprayed coatings, Proc. of 15th Int. Thermal Spray Conf., 25-29 May, Nice, France 1998, 1333-1336.

[7] Vidinejevs S., Chatys R., Aniskevich A., Jamroziak K., Prompt Determination of the Mechanical Properties of Industrial Polypropylene Sandwich Pipes, Materials, 2021, 14(9), 2128. DOI: https://doi.org/10.3390/ma14092128.

[8] Foreman C., Advanced Composites 2002, 12-3. 12-21.

[9] Scendo M., Zórawski W., Góral A., Influence of Nickel Powders on Corrosion Resistance of Cold Sprayed Coatings on Al7075 Substrate, Metals 2019, 9, 890.

[10] Unicomposite, 2020 https://www.unicomposite.com/fiberglass-honeycomb-panel.

[11] Olesik P., Godzierz M., Kozioł M., Preliminary Characterization of Novel LDPE-Based Wear-Resistant Composite Suitable for FDM 3D Printing, Materials 2019, 12, 2520, DOI: 10.3390/ma12162520.

[12] Failure Mode and Effects Analysis, FMEA Handbook (with Robustness Linkages – version 4.2), ed. Ford Motor Company, 2011.

[13] SRM TASK 51-77-13-911-016 Skin Repair with Core Replacement.

[14] Kozioł M., Toroń B., Szperlich P., Jesionek M., Fabrication of a piezoelectric strain sensor based on SbSI nanowires as a structural element of a FRP laminate, Composites Part B 2019, 157, 58-65, DOI: 10.1016/j.compositesb.2018.08.105.

