Structural Out-of-Autoclave Repair of Aircraft Fuselages via Adhesive Bonding

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ABSTRACT

The presented paper describes the development of an out-of-autoclave repair process for structural components of civil aircrafts using adhesive bonding. In detail, it demonstrates the application of a structural repair at a typical aircraft fuselage section including mechanical testing through compression as well as a comparison with the test results of a similar intact structure. For civil, commercial aircrafts the technique of rivetless out-of-autoclave structural repair via adhesive bonding is neither qualified nor permitted, the standard way to repair structural components being riveting through tailored metallic doublers. This is, however, not the best method for fibre-reinforced structures as fibres are broken through drilling and stresses are concentrated around these holes, but it also represents a cost-intensive solution. Hence, the repair community is looking for solutions that take advantage of the typical characteristics of fibre reinforced plastics (FRP) and also reduce costs caused through rivet-repair.

1.0 INTRODUCTION

The increased use of carbon fibre reinforced plastics (CFRP) in civil aircraft structures has become evident especially through Airbus's A350 programme and through Boeing's B787. According to [1], Airbus A350's structure is made of 53% CFRP while the B787 uses 50% of CFRP for its new long range aircraft. As former repair methods focussed on removal of damages followed by a stabilization via doubler which is riveted to the repaired structure, which is a very good and suitable method to repair metallic structures, the necessity arises to define and validate a more fibre-friendly repair technique for CFRP structures. Riveting of doublers is not always feasible due to skin thicknesses as well as an unbalancing of stress distribution due to differences in mechanical properties of the materials. Furthermore, each hole that is drilled into the CFRP structure implies stress concentration around holes, which is, by the way, the case for metals, too. Thus, on the one hand a repair solution that makes use of a repair material with similar material properties to those of the original structure is desirable; on the other hand an areal load transfer to avoid stress concentrations appears to be a more fibre-friendly bonding method. Consequently, an out-of-autoclave repair process that relinquishes riveting but makes use of structural adhesive bonding is the aim of a lot of research and development projects.

2.0 STANDARD REPAIR METHODS FOR CIVIL AIRCRAFT STRUCTURES

During an aircraft's service time there happen to occur different types of damages that have to be restored either directly or when detected at prescribed regular service intervals. This depends on the damage severity. Various manuals define solutions for airline (A/L) and Maintenance, Repair and Overhaul (MRO) companies to repair the damaged area and bring it into a required and satisfactory state. These manuals are referred to as "Structural Repair Manuals" (SRM), being applicable to the aircraft programme they have been associated with. The SRM is "provided to A/L in the aim to restore structure in [to] conditions required to fulfil primary function[s]" [2], further being approved by the European Aviation Safety Agency (EASA) [2]. Within these SRMs among other issues as allowable damage limits solutions are described and defined for repairs of structural parts. One common simple and suitable approach for civil aircraft structures often applied is the doubler integration via riveting ([3], [4]). While being suitable for metallic structures where drilling and riveting is the most applied assembly technique it does not appear to be the method of choice for composite structures as fibres are damaged and stresses concentrate around the holes [5]. However, due to regulations there is no alternative repair method other than the combination of cutting out, drilling, riveting and integrating a doubler for composite structures if the damage size may cause severe failure, which is transferrable in terms of loading with the so called limit load level [6]. Hence, when a composite aircraft structure becomes damaged either on shop floor level or in service and the damage size extends the structure's limit load level the only way to repair is to use rivets and doublers. Unfortunately, this solution is not always applicable for fuselage made out of carbon fibre reinforced plastics (CFRP). E.g., due to skin thicknesses or stress distribution unbalances the responsible and authorised engineer may refuse from this solution, consequently resulting in scrap in case of shop floor level or full component exchange in case of in service level. Both consequences are obviously not economical. Furthermore, the customer himself may refuse a fastened repair on shop floor level due to its optical appearance. An example of a suggested rivet repair solution in comparison to an intact structure is shown in Figure 2-1.

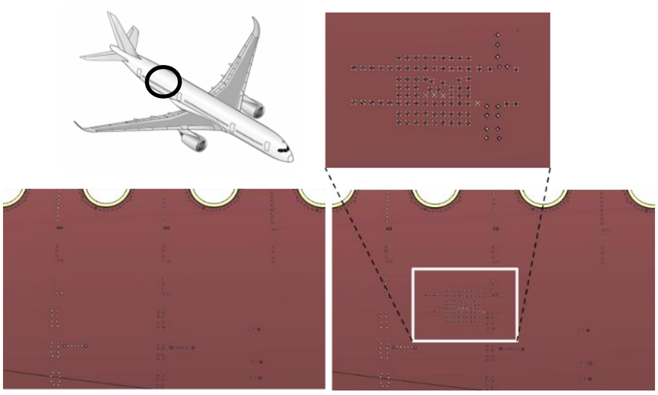

Figure 2-1: Damage position, drawing without repair (left) and with repair via riveting (right) including a magnification of the area.

Due to the aforementioned increased use of carbon fibre reinforced plastics (CFRP) in civil aircraft structures (compare Section 1.0) in addition to the explanations elaborated previously it becomes evident that there is the clear need for a repair method that is suitable and always applicable for composite aircraft structures. This topic has been focussed on within several publicly funded research projects ([7], [8]) and good results have been achieved in the past, aiming for a validation and authorisation of out-of-autoclave structural repairs also valid for below limit load level. Some results from these projects are presented in the following Sections.

3.0 OUT-OF-AUTOCLAVE REPAIR OF AN AIRCRAFT'S FUSELAGE

3.1 Repair Definition

To develop and validate the feasibility of an out-of-autoclave repair process without rivets and doublers not only on coupon size level but with respect to original structure size, a scrap left side shell of A350's section S16/18 was selected. The shell is approximately 14 m long and about 7 m wide and is built up in a conventional fuselage architecture consisting of a skin, stringers and frames. A representative area in terms of skin thickness, panel geometry, damage size and complexity was defined for conduction of the repair process, which aimed at a feasibility study of a fully penetrated fuselage. The defined area and an excert of the entire side shell is illustrated on the following figure.

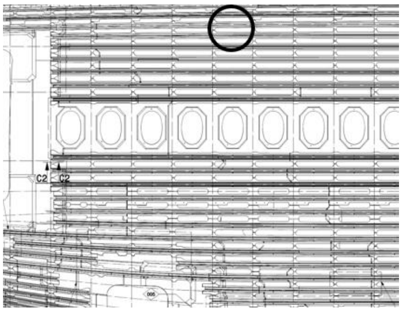

Figure 3-1. Excerpt of the left side shell of section 16/18 including the defined repair area.

The area that was cut out from the panel to demonstrate a penetration was sized 120 mm in circumferential and 150 mm in flight direction. To be able to conduct the repair two stringers had to be removed from the panel’s inside surface which is conducted by cutting out and manual grinding. The grinding operation is conducted until reaching the bond line which is shown in Figure 3-2.

Figure 3-2. Inside of a fuselage side shell showing the partly removed stringers.

3.2 Repair Design

After removal of stringers the cut out and removal of the damaged area can be conducted following a specified repair design. According to design guidelines a certain minimal overlap length must be secured for each ply that is restored [9], meaning that this must be considered when removing the damage. Further, each repair ply has to match the substrate’s ply orientation, i.e. a 0° repair ply must overlap with a 0° substrate ply, a +45° repair ply must overlap with a +45° substrate ply etc. Depending on the selected overlap length, whether using a minimum or a standard overlap length, the resulting ramp is defined as 55:1 or 94:1, respectively. The slope ratio depends on the ply thickness when an overlap configuration must be maintained.

To keep the repair area within a better handling size, the minimum overlap length and thus a slope ratio of 55:1 was defined. Additionally, two coverplies with a +45° orientation and an overlap length of 12 mm on each repair side had to be applied. Repair patch and adhesive film materials qualified and suitable in terms of thickness and mechanical properties were used. Altogether, 17 repair layers were used containing the entire original fibre stacking and additionally two coverplies, one on each side, as well as one filler ply to match the penetration, and finally one glass ply on each side to cover the repair and adhesive film fully.

Substrate, repair patch lay up, adhesive film, filler and cover plies and glass plies are sketched in Figure 3-3.

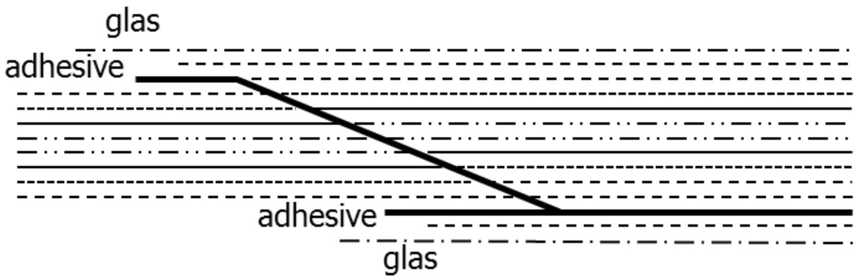

Figure 3-3. Representative sketch of a substrate's lay up and its repair patch including filler and cover plies as well as the adhesive film and glas plies.

Materials employed in the repair process were Hexcel Corporation's M20/IM7 as repair patch material and Cytec's FM300-2M as film adhesive. Compatibility with Hexcel Corporation's M21e/IMA, which is the main A350 structural CFRP material, was proven in advance [10].

3.3 Repair Process

Before the final repair process is conducted, a thermal profile must be performed in order to proof that the temperature in the entire repair area fulfils the requirements. For the thermal profile the originally planned repair patch lay up must be performed including positioning of thermocouples within the interface between adhesive film and substrate. A specified distribution of multiple thermocouples is given in repair process specifications [10]. Before the adhesive film is laid up for the thermal profile, a release film is placed on the interface to avoid adhesion. The entire lay including all debulking steps is performed as required for the original repair process. Once the lay up is finished, the vacuum bagging and heating is started. After curing, the temperature profile is analysed with respect to requirements. If all requirements are fulfilled, i.e. all recorded temperatures are within a tolerance range around target values, the actual repair process can be started. If not, the thermal profile must be repeated.

First, the surface quality of the grinded and activated bonding area is examined through a water break test. When a thin water film forms on the surface, it is ready for bonding. Then, after drying the bonding area, which can be conducted using a heat blanket or heat box, the adhesive film is applied on the surface. The original grinded substrate with perforation and ready for bonding before adhesive film application is shown in Figure 3-4.

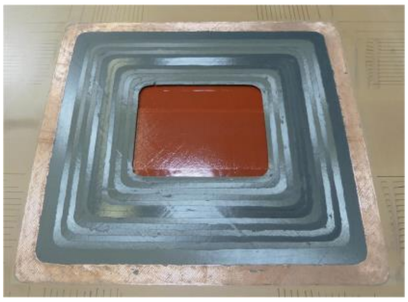

Figure 3-4. Penetrated fuselage structure with prepared bonding surface and mounting area before repair process.

Following the adhesive film repair plies according to repair design are laid up on the repair area being debulked at certain intervals. The debulking steps are performed at a certain vacuum pressure and time and aim for air removal from the repair patch. After the last ply was laid up a final debulking step is conducted. Figure 3-5 shows the repair area after lay-up of several repair plies.

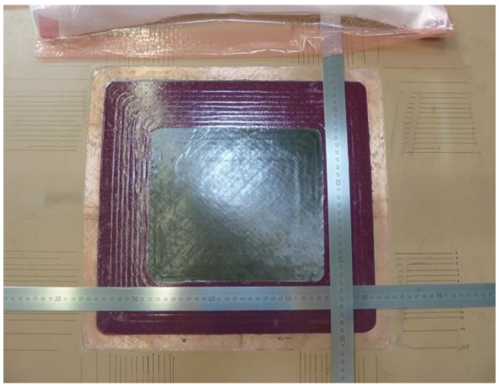

Figure 3-5. Repair patch after several repair plies have been laid up on the repair area.

Next, a vacuum chamber is constructed around the repair area and heating equipment is positioned in order to cure the adhesive film and repair plies. As the substrate to be repaired has been cured before and the adhesive film and the repair patch are not cured yet, the repair process applied is co-bonding. A leak test shows whether the vacuum chamber is suitable for starting the heating. If a certain vacuum pressure is not exceeded over test time the heating is started, else the vacuum bagging has to be repeated. Curing is conducted via a two step curing cycle, with the highest temperature applied being around 140°C.

A photograph of the cured repair patch laid up on the penetrated fuselage substrate is illustrated in Figure 3-6. The patch is visible in the middle, the edges around it showing the lightning strike protection.

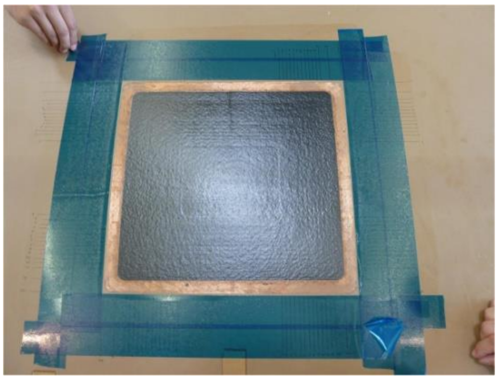

Figure 3-6. Fully laid up repair patch on fuselage structure.

The quality control checked all requirements in terms of geometrical measures and optical appearance as well as all recorded data, e.g. thermal profile and curing profile, which all were in line with requirements. Finally, to evaluate the structure's inner quality a non-destructive testing must be conducted through ultrasonic testing. The C-Scan of the repair area is shown in Figure 3-7.

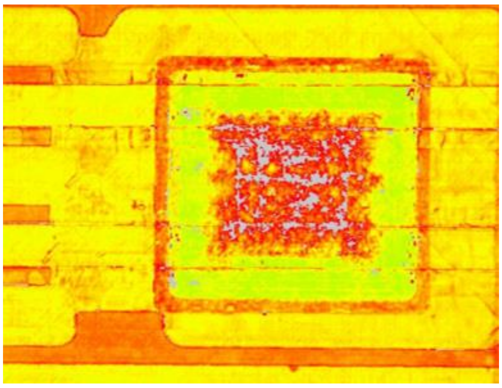

Figure 3-7. C-scan of repair patch on perforated fuselage structure showing some quality issues in the middle of the repair area.

The C-scan illustrates the amplitude drop after being reflected from the back wall and compared to a reference limit. Subtracting a defined allowed drop results in a C-picture that displays all weakenings below that limit in grey colour. This is evident in some areas of the sheer repair patch. The area where the repair patch overlaps the substrate as well as the sheer substrate area show no grey areas and thus can be interpreted as intact. The repair patch, however, might include some porous parts. To validate this assumption, microcuts of these parts and microscopic analysis must be conducted, which is planned for the future, after mechanical testings have been performed. A second explanation is that the reference amplitude level which was applied is not suitable for the sheer repair material. This may result from the material, although being a repair prepreg, not being cured under elevated pressure and therefore not being as compacted as a comparable prepreg that was autoclave-cured. Further investigations must be performed to understand whether a new reference amplitude level may be required.

4.0 MECHANICAL TESTING

In order to make a comparison between an intact and a repaired CFRP structure, an element test was defined and requested [6]. Extending coupon sizes to detail element level has the advantage of being able to analyse the entire repair patch effect on the mechanical properties. The test specified for comparison was a compression test of a panel containing four stringers. For comparison reasons, the two middle stringer were removed before the tests. The test panel is presented schematically in Figure 4-1.

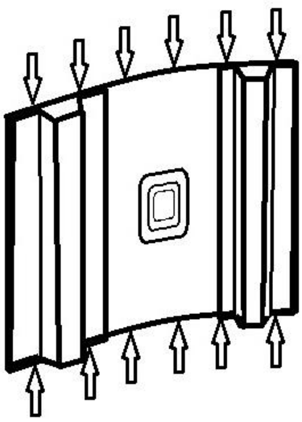

Figure 4-1. Schematical drawing of test panel with repair patch in the middle.

As stated, two compression tests were performed, one using a completely intact panel (P1) and one including a repaired area (P2) as shown in Figure 4-1 and as explained in Chapter 3.0. The test panel size was identical, being 800 [mm] in length and 710 [mm] in width for both. To analyse test results multiple strain gauges were applied as well as an Aramis system. The latter sprinkles a black-and-white dot pattern on the surface and measures the displacement of each dot. Hence, a displacement measurement with respect to time can be derived for *x*-, *y*- and *z*-direction. Test interpretation is ongoing. Testing machine measurement data was recorded and is presented below. Tests were performed on a servohydraulic testing machine with a static load capacity of 1600 kN. To introduce loading into testing panel, load introduction blocks made of aluminum alloys were designed and manufactured. A comparison between compression force with respect to displacement up to a value of 90 [kN] is illustrated in Figure 4-2.

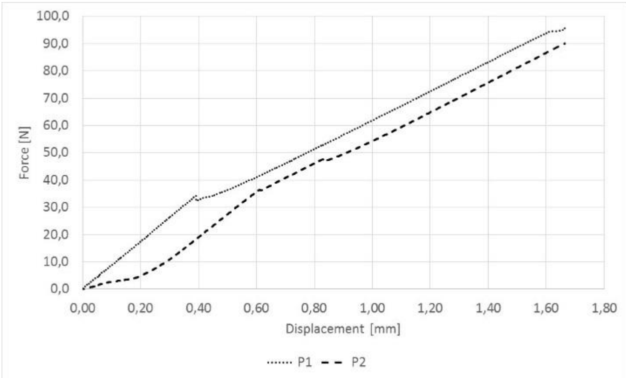

Figure 4-2. Force with respect to cross head displacement for panels P1 (intact) and P2 (repaired).

The recorded differences at the beginning are due to setting effects which are negligible. Once these are overcome (at about 0,2 mm for P2) both panels show a linear increase in compression force with respect to displacement. A first buckling is recorded at about 35 kN which is in line with numerical analysis data. The tests were terminated at 90 kN to validate numerical data and adjust strain gauges. Optical strain measurements were conducted for each test panel and compared to each other (Aramis). An example of these comparisons is shown in Figure 4-3 at a compression force of 85 kN.

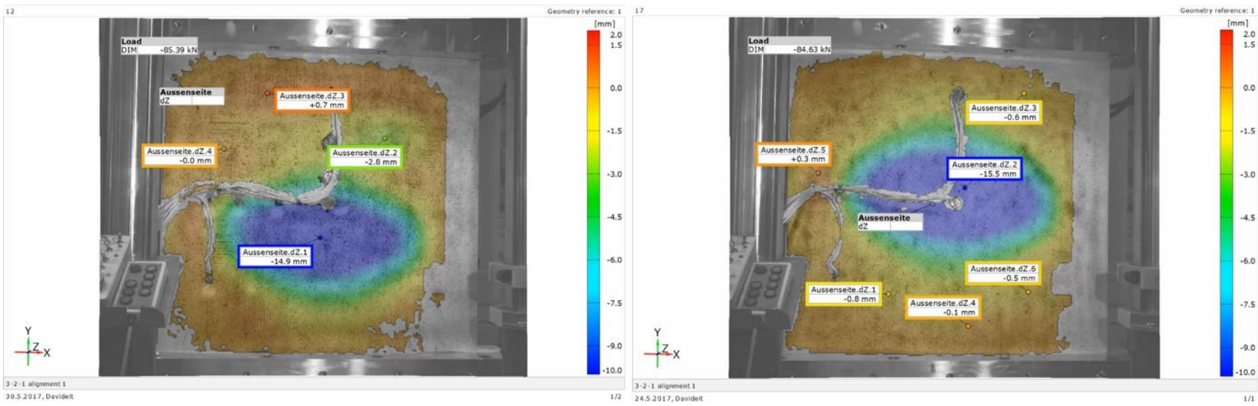

Figure 4-3. Plots of optical strain measurement for panel P1 and P2.

The optical strain measurements indicate a similar behaviour between the intact and the repaired panel. Data analysis and interpretation as well as numerical validation is ongoing and results are expected in the near future.

5.0 CONCLUSION

The presented work shows the development and validation of an out-of-autoclave rivetless CFRP fuselage structure repair process for an entire penetration. The necessity of an out-of-autoclave repair process was explained and an increase of use cases is expected due to the increased use of carbon fibre reinforcements for aircraft structures. Adhesives and prepreg repair materials were benchmarked and the most suitable for interaction with standard A350 CFRP selected. The here presented validation process proofed the feasibility in terms of inspection, interpretation, design, preparation, lay-up, heat application as well as non-destructive testing. Although first results of NDT implicated porous areas further investigation must be conducted to make sure whether the noticed amplitude drop indeed is caused by material imperfections. The repair patch quality, though, did not affect mechanical properties of the tested detail element for the tested range and loading case. Up to 90kN compressive loading hardly any differences between an entirely intact panel and a repaired panel were noticed. Further test interpretation is ongoing and will be used to validate structural integrity of structural out-of-autoclave repairs. In addition, several multi-directional mechanical testings (e.g. shear testing, four-point-bending, bi-axial tensile testing) are planned to better understand structural repair impact on performance and compare it with numerical predictions. Long-time goal is set to allow rivetless out-of-autoclave bonding (i.e. co-bonding) of primary aircraft CFRP structures by authorities.

ACKNOWLEDGEMENTS

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