A CASE STUDY OF AIRCRAFT COMPOSITE COMPONENT REPAIR UTILIZING AUTOCLAVE CURING

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Abstract: This paper presents the evaluation process of aircraft composite component structural repair of flap fairing origin for Boeing 737-800 aircraft. The intended repair has been carried out due to a wide area of damage that requires utilization of an autoclave. All preparations are carried out in accordance to the aircraft structural repair manual of Boeing 737-800 aircraft. The damage is removed via the step cutting method on the outer surface of the component prior to the inspection verification via tap test inspection. In order to prevent profile bending and torsion, the part is placed on a male mold that will be vacuumed together. The damaged area is repaired using similar materials and orientation, but with an additional layer as per prescribed in the manual. The component is then cured inside the autoclave at 121°C for two hours at a pressure of four bars. Once cured, the component is inspected through the post-repair inspection for visual and structural integrity prior to painting. The post-repair inspection has indicated that the repair has preserved the component and it is now fit to be used.

Keywords: aerospace structure; aircraft maintenance; advanced composite; curing process

1. Introduction

The utilization of advanced composite materials has increased tremendously in aircraft structural application. The capability to offer high strength and high stiffness to weight ratio, resistance to fatigue and corrosion resistance are some of the benefits acquired in utilizing these materials. As for structure materials, the carbon fiber reinforced plastic (CFRP) has been mostly used due to its reduced structural weight compared to the aluminum alloy material. Although CFRP offers significant benefit, however, it is also highly susceptible to the impact and heat damage, especially when it is used near to the aircraft engine exhaust. In general, the advanced composite materials pose challenges compared to traditional metallic material used in aircraft structure. Their failure characteristics behave differently and there is a need for new set of skills in order to repair the components. A damaged composite part will require an extensive assessment, competent staff and advanced materials to be dealing with. This paper looks into the process in carry out such repair.

The world's aircraft fleet is expected to increase from 25,900 to 49,405 aircraft between 2019 and 2039. While the more established markets of Europe and North America are predicted to increase by around 76 and 42 percent, respectively, within that period, the Asia Pacific fleet has also been expected to increase by about 139 percent to 18,770 aircraft in 2040 [1]. It should be noted that new generation aircraft are having more than 50% of its structural weight contributed by advanced composite materials. With the increasing percentage of these materials, more aircraft structural parts or components made from composites need to be maintained and repaired. Damages can be found during the manufacturing and in service of the aircraft, in which the latter will accumulate the largest portion of defect findings and rectification. During manufacturing, defect on composite parts can be originated from micro-cracks and delamination, scratches, gouges, hole damage and impact damage [2]-[3]. These damages cannot be prevented and therefore, the acceptable thresholds are incorporated in the design by demonstrating the attainment of the intended ultimate strength with the damage present in the component through the principle of fracture mechanics. Some of these damages may go undetected or beyond the specification limits. Hence these damages should be assumed to exist within a composite component with a damage tolerant design [4]-[5]. In the meantime, during operation, aircraft are subjected to both scheduled and unscheduled maintenance inspections. For scheduled inspection, the aircraft will undergo a periodical inspection as part of a maintenance schedule to look for possible defect or damage. These maintenance interval inspection is categorized as Check A, Check B, Check C and Check D or Heavy Maintenance Visit (HMV) for continuing airworthiness. For unscheduled inspection, the aircraft need to be grounded and inspected to evaluate and rectify any identified damage for assessment. For example, impact damage such as due to foreign object debris (FOD) on the runway, ground handling, dropped parts, aircraft-handling accidents, collisions with airport structures and/or environmental factors are among the main causes of the defects [6]. More than 80% of damages to composite structures have been attributed to impact, with the remainder being attributed to environmental factors [7]. Moreover, damages occurred on the aircraft's wing area accounted for up to 13% of the total area damages, which is the fourth from the accounted location of damaged structure of a typical aircraft [8]. Figure 1 shows the typical damage extended to the laminated structure.

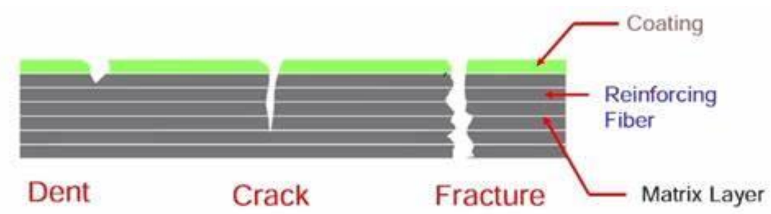


Figure 1: Typical damage on a laminated composite structure

It should be noted that the traditional repair methods based on the sheet metal philosophy cannot be used for composite parts or composites as they have transformed from drilling and cutting to laying up and trimming. A new set of skills are required and during one of the trainings conducted in UniKL MIAT, sheet metal technicians have been trained to do composite repair. This is to ensure an effective and optimum manpower usage. Furthermore, the authorities such as Federal Aviation Administration (FAA), European Union Aviation Safety Agency (EASA), Society of Automotive Engineers (SAE) have produced documents to recommend the competencies, facilities and requirement to carry out the repair of composite structure [8]-[10]. In this study, the presented case study of Flap Support Forward Fairing Panel of a Boeing 737-800 aircraft is used to demonstrate an example of repair process of the composite aircraft parts.

2. Methodology

As previously mentioned, the component of interest in this study is Flap Support Forward Fairing Panel that is situated at the bottom of the wing, which is housing the flap tracking system in order to control deployment and retraction of the flap. Due to its location, which is underneath the aircraft wing, the component is facing debris during taxing, landing and taking off, in which a high projectile of FOD can easily damage the structure. Figure 2 shows the location of the Inboard Forward Flap Fairing (IFFF) section.

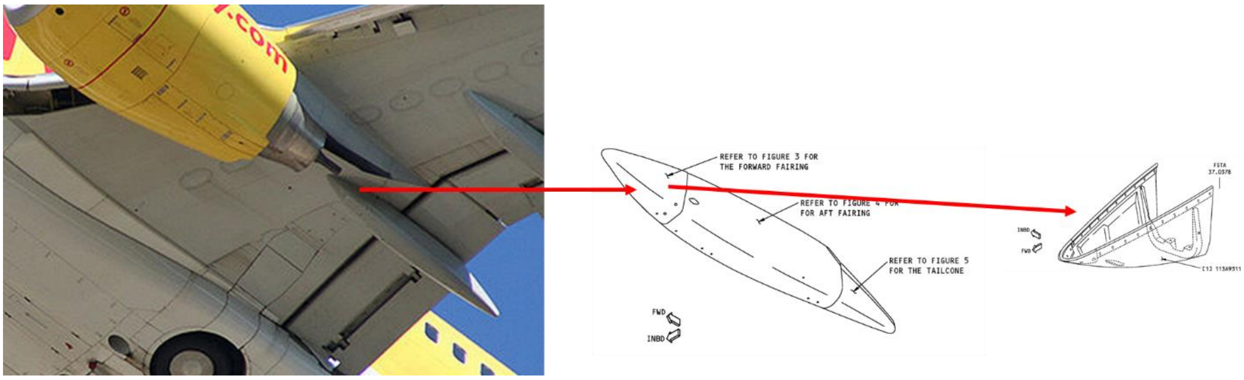


Figure 2: Location of the IFFF with respect to Boeing 737 series aircraft [11]

Upon inspecting the IFFF component, it has been found that the inboard side of the component is affected by heat damage due to its close vicinity of the exhaust area. The prolonged exposure has led to diminishing the component integrity and thus it is required to be repaired. Therefore, the objective of the task is to carry out the repair as outlined by Aircraft Structural Repair Manual (SRM) requirement. This involves utilization of the autoclave as the curing mechanism and the comparison of the outcome of curing with respect to the post-repair inspection. In short, the damage is evaluated in accordance to the corresponding SRM [10] and upon evaluation, the area subjected to damage is found to be repairable as documented in the SRM. Damage due crack has been found at the rear section of the component. It has been suggested that a major repair need to be carried out due to the fiber damage and the damage size. A preliminary evaluation is carried out to determine the extent of the damage through both physical and non-destructive testing. The damaged component is shown in Figure 3.

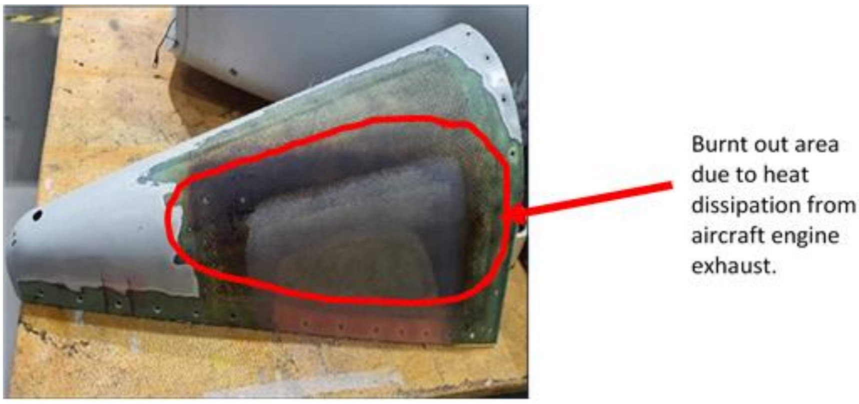
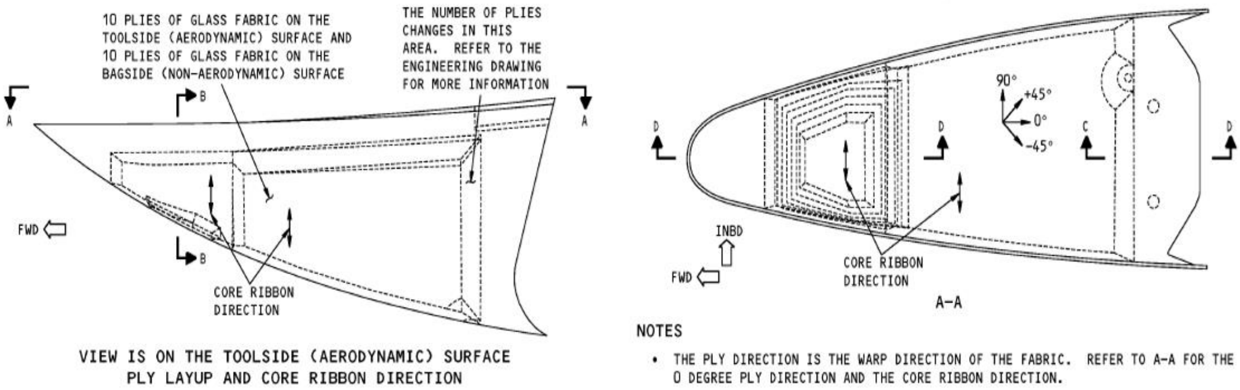


Figure 3: Extent of damage that is observed on the IFFF component

Furthermore, assessment from non-destructive testing has indicated that the damage is feasible to be repaired. However, due to its size characteristic, autoclave curing has been suggested for the curing of the component. This is to ensure the compaction from the pressure can assist in consolidating the repaired plies and subsequently provide a stronger and intact bonding between the original and repair layers. Further information is retrieved pertaining to related material specification and curing parameter. Figure 4 shows the configuration of the component. Upon investigation, P1 to P10 have been removed in order to go for the preparation process. Once the damaged plies have been removed, a male mold is fabricated using layer of fiberglass laminates saturated with high temperature curing resin in preparation for autoclave curing. Six layers of 400 gsm woven roving fiber glass are used, which lead to an average thickness of approximately 8 mm as depicted in Figure 5. It should be noted that the use of the male mold is to provide a positive pressure and reduce distortion whilst curing.



PLY MATERIAL AND DIRECTION OF THE FORWARD FAIRING BONDED SKIN FOR FIGURE 3 [1]		
PLY	**DIRECTION**	**MATERIAL**
P1, P10, P11, and P20	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 (Optional: Style 220)
P2 thru P4, P6, P8, P13, P15, and P17 thru P19	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 (Optional: Style 7781)
P5, P7, P9, P12, P14, and P16	+ or - 45 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 (Optional: Style 7781)
Filler Plies	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 (Optional: Style 7781) or Style 120 (Optional: Style 220)

Figure 4: Information pertaining to the component configuration [11]

Figure 5: Fabrication of in-house male mold to support the part during curing

The damaged area is removed by manually step sanded the affected area with respect to the outlined SRM procedure. The purpose of step sanding is to allow the stresses to be transferred from the original towards the repair areas and back to the original, thus minimizing stress concentration. The information with respect to the total number of layers and orientation are retrieved to replace the removed layers, inclusive with the repair plies and honeycomb material specifications. Once the plies and honeycomb have been replaced, the area are prepared for the vacuum bagging layers. In order to prevent distortion of the part during curing, the part has been placed on top of the mold for stabilization. The material of the mold needs to be fabricated by using high temperature resin to suite with the curing environment. This is to ensure that the resin works well with high temperature curing of the part. Figure 6 shows the placement of the part on the mold and the envelope type of vacuum bagging system used to encapsulate the part for curing. The curing has taken place in an autoclave with parameters set accordingly. Prior to curing in the autoclave, the part is vacuumed at 0.75 bar (25 in Hg). After that, the consolidated set up is transported to the autoclave.

Figure 6: Positioning of the repaired panel on top of the mold and the vacuum bagging encapsulation during curing

3. Results and Discussion

Table 1 shows the setup of the curing parameters for the autoclave. The autoclave applied in this study is located at UniKL MIAT Subang, which is manufacturerd by ARKAMAK Inc. with an internal size of 1 m x 2 m (i.e. diameter x depth). Once the part has cured, it is removed from the autoclave and ready for post-repair inspection. A tap test, which is then followed by ultrasonic NDT, has been carried out to ensure that the part is well-cured and also consolidated. The tap test is performed by tapping from the original surface to the repaired areas and back to the original surface. In this test, a consistent thudding sound registered shows an intact bond between the repair plies to the original surface.

Table 1: Curing parameters and actual comparison

	Setup	**Actual (Average)**
Curing Temperature	250 ^{0}C ± 10 ^{0}C	247 ^{0}C ~ 260 ^{0}C
Curing Time	120 min	120 min
Rate of Curing	5 ^{0}C/min	5 ^{0}C/min
Rate of Cooling	5 ^{0}C/min	3 ^{0}C/min
Pressure	5 bars	4.97 bars

Figure 7 shows the curing cycle during the process. The actual and estimated curing cycles show a good argument within the boundaries result. Therefore, the process is valid and in accordance with the desired perimeter. Once the vacuum bag system is separated from the part, crevices due to resin ridge presence on the outer surface of the part can be observed as shown in Figure 7. This line is caused by the wrinkling of the peel ply and it does not dislocate the fiber. A sanding process is later carried out to smoothen the surface. Dimension-wise, the component is undistorted due to the mold holding on its position from the surrounding pressure exerted on the surface. Finally, the tap test is carried out at the locality of the repair cured area. Neither disbonding nor delamination is found to be present within the component. Overall, the process shows positive result and the component is proceeded for painting.

(a) damage (b) repair (c) post repair

Figure 7: The repair process from damaged, repaired and post-repaired

4. Conclusion

The repair process has been carried out successfully within the set parameter. The outcome of the repaired surface shows crevices but this does not affect the strength of the component as it is within the aircraft structural repair limits. All objectives are attained and the parts have been certified to be put back in service. The part has undergone a permanent repair and have been installed back to the original position and location. The defect report has been closed and the certificate release to service is issued for it to be returned back to the respective aircraft safely. An interim inspection has been set to monitor for any degradation during its operation. Therefore, the process has been conducted successfully with respect to the manual procedures.

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