Optimization of the Composite Repair Work using 5H Satin Dry Glass Fabric and Epoxy Resin LY5052/HY5052 Materials through the Vacuum Bagging Technique

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Abstract. In composite repair terminology, the intermingled fiber joints (IFJs) concept is clearly identified as the most effective in transferring stress. However, the practical application leads to apply the laminated fiber joints (LFJs) concept for composite repair. The LFJs become almost as strong as the IFJs as the jointed zone length increases. The LFJs concept was implemented to set up the repair configuration for this work. It is identified that fracture tendency as the result of crack propagation starts from the free edges. This tendency is alleviated by applying vacuum bag to compact the laminate system. If necessary, additional accelerated curing was performed at low temperature for short time. However, this repair work preferably applying natural overnight cure to ensure the structure stability. Additionally, the repair handling and waiting purposes take 40% out of the effectively repair time. The waiting time should be anticipated in the composite repair to reduce wasting time for unproductive output.

1. Introduction

The composite repair skill and knowledge has significant role in the aircraft maintenance service providers. The composite repair system should accommodate several parameters to ensure the quality of the result. This includes damage identification, inspection techniques, repair scenario, repair materials, repair process and final inspection. The example of the repair procedures can be studied from the composite repair of the flap track fairing, a canoe shaped structure that rests under the wing. [1)]
Typical damage of the flap track fairing may be caused by an impact stress either cracks, delamination, holes, or moisture ingress. The lightning strike or dynamically stress, strain, and fatigue may also contribute to the possibility of this damage. The practical inspection may be conducted through tapping test and visually inspection. The additional thermal graphic inspection or through transmission ultrasonic (TTU) may strengthen in the verification of the repair quality.
The repairable scenario should consider an assess damage repair requirements in which possible maximum damage size is less or equal to 3.152 inches (SRM 57-55). As long as damage is within the standard repair manual (SRM) repairable limits, a core splice and wet lay-up repair is allowed. The reliable repair materials may be used such as 5H satin, 285 g/m^2 dry glass / carbon fabric, and epoxy laminating resin e.g.: Hysol 9396A/B or Araldite LY5052/HY5052. Alternately, it may apply dry repair method using 1/8", 4.0 lbs. hexagonal Nomex honeycomb core and 121°C cure, 5H satin weave carbon fiber prepreg, 45% resin.
The example of flap track fairing repair process, includes preliminary inspection and damage area preparation by locally paint removing and cleaning over the repair structure. Preliminary work consist of preparing taper or step-sand damaged skin, cutting the required plies, fabricate core plug (if required) and pre-setting before applying curing cycle. After ply repair materials are prepared, then

apply lay-up plies and completely vacuum bagged. Seal the repair zone before applying cure cycle. Inspection is needed during and after curing process.

In the composite repair definition, fracture failure is initiated by crack propagation starts from the free edges. The description of failure modes refers to ASTM D5573.[2)] This standard classifies seven failure types in fiber reinforce plastic (FRP) joints. These failure types are listed below.

- Adhesive failure, rupture of the adhesively bonded joint, such that the separation appears to be at the adhesive–adherent interface. This failure type sometimes referred to as interfacial failure).
- Cohesive failure, rupture of an adhesively bonded joint, such that the separation is within the adhesive. This parameter is used to qualify durability of adhesive bonded joint.[3)]
- Thin-layer cohesive failure (TLCF), failure similar to cohesive failure, except that the failure is very close to the adhesive–adhered interface, characterized by a ‘light dusting’ of adhesive on one adherent surface and a thick layer of adhesive left of the other. This failure type sometimes referred to as interphase failure.
- Fiber-tear failure (FTF), failure occurring exclusively within the fiber reinforced plastic (FRP) matrix, characterized by the appearance of reinforcing fibers on both ruptured surfaces.
- Light-fiber-tear failure (LFTF), failure occurring within the FRP adherent, near the surface, characterized by a thin layer of the FRP resin matrix visible on the adhesive, with few or no glass fibers transferred from the adherent to the adhesive.
- Stock-break failure. This occurs when the FRP substrate breaks outside the adhesively bonded joint region.
- Mixed failure where the failure is a mixture of different classes.

In this work, composite repair is applied to modify fairing skin with sandwich configuration by removing honeycomb partially along the stringer placement. The repair materials are selected and prepared. The repair work include preparation, wet lamination, mechanical work and painting. The optimization of process sequences will be examined to improve the work efficiency by considering the quality aspects.

2. Experimental Procedure

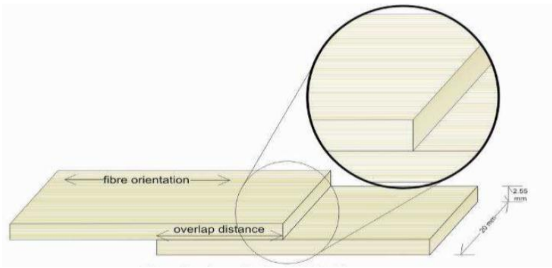
This work employed fiberglass fabric 7781 type-181 and type-120. The dry fiberglass was laminated in a balanced and symmetrically orientation using epoxy resin i.e. Araldite LY5052/HY5052. This transparent resin contains butanediol di-glycidyl ether (34-42%) and epoxy phenol novolak resin (60-72%).[4)] As a laminating resin, it has relatively low mixed viscosity of 800 mPa.s, and work life in 130 minutes. This resin will cure at room temperature or can be accelerated cure between 50 and 100 ^{o}C. It is typically applied with service temperature up to 80 ^{o}C. This laminating resin is different from the infusion resin with lower viscosity of 200 mPa.s, and longer gel time up to 460 minutes at room temperature.[5)]

The tensile specimen of glass/epoxy was prepared to verify the tensile properties used in the composite repair. The mechanical testing was determined according to the ASTM standard D 3039/D 3039M-00, “Standard Test Method of Tensile Properties of Polymer Matrix Composite Materials”. The scope of this standard is to determine the in-plane tensile properties of continuous or discontinuous fiber reinforced polymer matrix composite materials, which are required to be balanced and symmetric with respect to test direction.

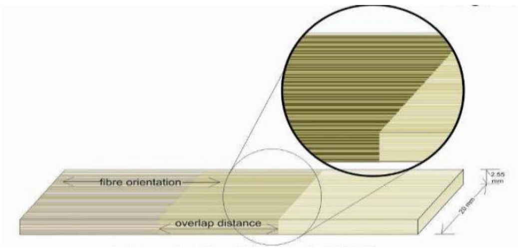
Although the specimen geometry is not dictated in the standard, but the general requirements states that the coupon must have a constant rectangular cross section, and the minimum length of coupon should be summation of gripping length, gage length and twice of the specimen width. It is also stated that, specimen must be flat, and tabs may be used at the grip locations as necessary. Test machine used in this study is Instron Universal Static Testing Machine. Tensile specimen geometry for balanced and symmetric is chosen with width 25 mm, overall length 250 mm, thickness 2.5 mm.

3. Composite Repair Process

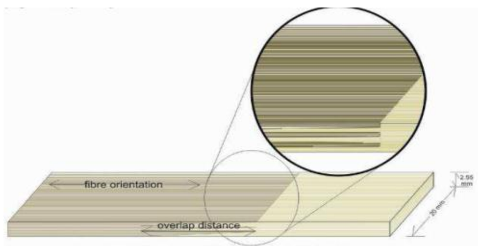
In this work, honeycomb section was partially removed in a taper shape. The open honeycomb will be covered by new composite skin, and the laminate side will be strengthen with additional plies. The composite repair configuration may follow one of three type of joining model proposed by Murillo, et.al (2007).[6)] By using the bio fiber (henequen and sisal fiber) they proposed three kind of composite joining model called Single lap joint (SLJ), Intermingled fiber joint (IFJ), and Laminated fiber joint (LFJ) as shown in Fig.1. The conventional SLJs contain an offset shear zone whereas the IFJs and LFJs have an intermingled or interleaved jointed region. Hence the force to fail the specimens per unit area of unjointed cross-section is one way to compare joint performance.



(a) Single lap joint (SLJ)



(b) Intermingled fiber joint (IFJ)



(c) Laminated fiber joint (LFJ)

Figure 1. Three joint configurations (Murillo et.al, 2007)

They identified clearly that the IFJs are the most effective in transferring stress, especially at small overlap distances. For example, the 10 mm overlap IFJ is more than five times stronger than the 15 mm SLJ. Lot of specimen were prepared in which the tensile strength of single lap joints (SLJs), intermingled fiber joints (IFJs) and laminated fiber joints (LFJs) are 63 MPa, 240 MPa, and 203 MPa, respectively. This result refer to the standard value of sisal/epoxy LY5052/HY5052 of 311 MPa. The LFJs become almost as strong as the IFJs as the jointed zone length increases. This concept was implemented to set up the repair configuration for this work. The fiberglass type-180/epoxy LY5052/HY5052 refers to standard value of 400 MPa through the dry lamination process in the autoclave, refer to the design requirement of the aircraft composite structure.

The other study reported the similar laminate configuration using henequen fiber (Yucatan's sisal) density 1,477 kg/m^3 with epoxy Araldite LY5052 / Hardener HY5052. The test result of tensile strength was 228 MPa, while strain value at failure of 2.2%, and Young’s modulus was 17 GPa. [7)] From lots of specimen, they identified that chemically treated of sisal fiber, in sodium hydroxide solution, did not improve the mechanical strength while compared with the natural sisal fiber.

Turgut (2009) examined the effect of post-curing, and it was found that post-curing does not show significant effect on the mechanical properties of composites. He argued that post-curing as a process related with matrix material, does not affect the properties of the fibers in the composite. Since the elastic modulus and tensile strength are fiber dominated properties, post-curing does not improve these properties significantly. From lots of specimens, he obtained the tensile strength value of fiberglass/ epoxy LY5052/HY5052 are between 295 to 317 MPa.[8)]

The sixth class failure modes mentioned above is not relevant to the current study. This work describes the modification of fairing skin to place additional stringer in the middle side along the fiber composite fairing panel. In this case, the core section was cut along the stringer location. Principally, the work changed the sandwich configuration to become laminate side in the middle of sandwich panel (Fig.2).

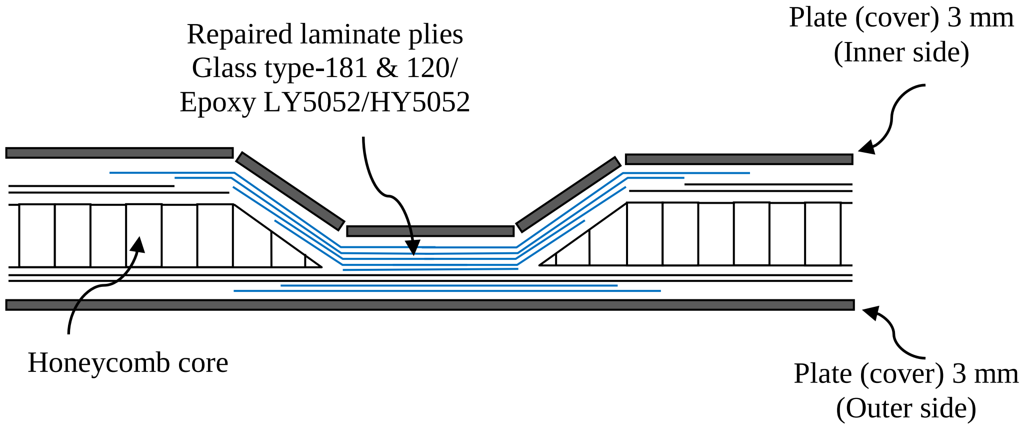


Figure 2. Cut section of sandwich panel after rework (no scale)

In this work, the fiber to resin ratio was controlled in a constant proportion at 55-60%. The characteristic of 3-D fabric composites has been known that elastic moduli and in-plane shear moduli tend to increase with the increase of fiber to resin ratio. In the other hand, coefficient of thermal expansion (CTE) decrease with the increase of fiber to resin ratio.[9)] The manually recording system was prepared to ensure the consistency of the parameter fiber to resin ratio.
The important feature is that for all joint configurations of the composite structure, fracture is always the result of crack propagation from the free edges. Any situation that crack initiation from the free edges will precipitate the crack propagation and the joint may rupture. The presence of incomplete fiber impregnation on the surfaces of the composite joints could be a cause of early fiber delamination and crack propagation during the tensile test. This constraint is overcome by applying vacuum bag to compact the laminate system. Additional accelerated curing can be performed at 50 to 100 ^{o}C during 15 to 30 minutes.

4. Composite repair lead time

The modification work of composite skin was initiated by removing the honey-comb sandwich partially along the stringer placement line. The formation of laminate for stringer placement considered allowance 0.25 inch width on the both edge side. Pre-setting was performed on aircraft and laminate was formed through vacuum bagged wet layup technique. The distribution of repair time is shown as Table-1 herein. The tight control during repair work improves the allocated repair time significantly.

Table 1. Distribution of repair time

Repair work (3_personel)	Estimated hours (h)	Actual hours (h)	Time saving (h)	Time saving (%)
Preparation	30.1	20.1	10.0	33%
Main layup	32.3	27.3	5.0	16%
Fitter finishing	20.3	2.8	17.5	86%
Riveting step-1	35.0	15.5	19.5	56%
Layup contour gap	14.0	12.0	2.0	14%

Finishing	1.5	1.5	0.0	0%
Riveting step-2	2.0	2.0	0.0	0%
Top coat	2.5	1.5	1.0	40%
Grand Total	137.6	82.6	55.0	40%

The new laminate was performed with the placement of dry fiberglass type-180 (2 plies laminate), type-120 (2 plies covering honeycomb), type-180 (2 plies laminate), type-120 (2 plies covering honeycomb), and nylon peel ply. After overnight curing in vacuum bag condition, the nylon ply was removed, apply mixed resin and Aerosil. Aerosil was evaluated as a compatible nanofiller to be combined with epoxy resin for touch up repair area of the airframe structure.[10)] The layup was continued up to 8 plies of dry fiberglass type-180, and cover with nylon peel ply. The layup then cured overnight. The next turn was lay up on the upper side of composite panel. First, set the stringer and preview set of hole as required. Then the upper side of skin was laid up by 2 plies of dry fiberglass type-180 and peel ply. Then allowing to cure naturally overnight.

The next step was mechanical work including countersunk, preview hole, install round rivet on required place, cherry rivet, and hi-lock. The work also included preview and install anchor nut on required location. In this example, the skin gap was found on the radius / frame side between 1 to 3 mm. This gap was identified after setting this composite skin on aircraft, and was compensated with 9 plies of dry fiberglass type-180. Finishing work included sanding refer to original contour prior to manually solvent cleaning.

The behavior of composite to composite adhesive bonded joints was reported by Donaldson, et.al (1997). They examined the ultimate failure load for four surface preparation techniques, in which acetone wash provided the best result while compared with grinding, grit blasting and hand sanding, respectively. [11)] Then first layer of coat was conductive layer application before the panel installed on the aircraft. Finally top coat was applied refers to the color scheme.

An improvement to increase time effectiveness can be traced through the repair time allocation. It was found that 40% out of repair hours was consumed for handling and transportation purposes. The actual time for repair process is only 60% of the whole repair hours. Further, actual repair time only consumed 61% of totally man-hour as plan. Naturally overnight curing under vacuum bagged on repair zone, was selected to ensure the stability of the composite panel structure.

5. Conclusion

In this repair work, the laminated fiber joints (LFJs) concept for composite joining was applied. The LFJs become almost as strong as the IFJs as the jointed zone length increases. It was identified that fracture tendency is always the result of crack propagation from the free edges. Incomplete fiber impregnation could also be a cause of early fiber delamination and crack propagation. The failure tendency is overcome by applying vacuum bag to compact the laminate system. Although additional heating in low temperature is allowed to accelerate curing, but, this repair work preferably using natural overnight cure to ensure the structure stability. It is known, the repair work includes handling and transportation purposes at about 40% of the effectively repair time. The unnecessary waiting time should be anticipated first by accurate plan in composite repair.

Acknowledgement

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