Composite Bonded Joint: Repair Development

Daniel Perez
NASA Kennedy Space Center
Mechanical Engineering
NE-L4 Engineering Fall Session
13 November 2018

Composite Bonded Joint: Repair Development

Daniel Perez.[1]
California State University of Long Beach, CA, 90804

Sarah Cox[2]
NASA Kennedy Space Center, Merritt Island, FL 32899, USA

and

Susan E. Danley[3]
NASA Kennedy Space Center, Merritt Island, FL 32899, USA

This report documents the details of my work as a NASA KSC intern for the Fall Session from August 27th to December 14th, 2018. My efforts and contributions were with the Materials Science Branch, a staffed organization within the Labs, Development, & Testing (NE-L) division of the Engineering Directorate. The principle responsibilities of the Materials Science group are to support the design, development, and operations activities for materials and processes with the purpose of providing unique solutions for flight hardware, ground support equipment, and customer requests. My role as an intern focused on assisting engineers in developing repair processes to mature bonded joint technology in support of SLS – scale hardware. My primary goals for this internship were to become more familiar with composite materials and learn more about processes I was unfamiliar with, such as prepregs and out-of-autoclave processing. This project allowed me to learn new skills such as scarfing and curing composites. Additional goals I had were to learn more about NASA's laboratories and projects under development. This internship not only provided me with those experiences, but also allowed me to build relationships with inspiring engineers; a takeaway I will never forget.

I. Introduction

The National Aeronautics and Space Administration (NASA) is researching and developing advanced composite technologies to provide lightweight structures to support future long duration missions. The Composite Technology for Exploration project is an interagency project focused on maturing composite bonded joints used on payload adapters and fittings, while demonstrating successful stress-strain analyses that predict joint failure and residual strength. While composite structures are emerging in the aircraft and aerospace industry, the use of secondarily bonded structures is still not trusted. Composite materials have a long history of usage. Original forms began with straw-mud bricks. Straw-mud bricks were used for their compression and tensile strength. Today, fiber-reinforced composite materials offer much higher strength to weight ratios. These qualities are important in weight-sensitive applications such as aircraft and space vehicles. For this reason, the aerospace industry has advanced research and development of composite technology. To gain a better understanding of this development, research and analysis of previous composite processes will be examined, current studies and results will be discussed, and challenges for advancement will be addressed.

Reinforced fiber composites are a combination of fiber and resin materials. Each material has a distinct material property. Fibers are known for their high tensile strength while resins are useful for their compressive strength. Fibers and resins are combined through wet lay-ups or prepregs. A wet lay-up is a traditional method of combining layers of fibers with liquid resin against a mold. Prepregs are fabrics pre-impregnated with resin. Out of the two processes,

---

[1] NIFS Intern, Mechanics of Materials & Structures, Materials Science Branch, Kennedy Space Center.
[2] AST, Mechanics of Materials & Structures, Materials Science Branch, Kennedy Space Center.
[3] AST, Mechanical Experimental Equipment, Analytical Laboratories Branch, Kennedy Space Center.

prepregs typically produce higher quality parts as they have a more consistent resin content. Some sandwich structures combine composite facesheets with a core. Sandwich structures are effective for their bending and buckling stiffness. Composite structures can be processed through autoclave or out-of-autoclave methods. An autoclave process applies a compressed gas into a pressure vessel containing the composite layup. An out-of-autoclave process applies pressure and heat using technologies such as vacuum pumps and hot bonders. Out-of-autoclave processes hold more advantages such as lower costs and more efficient equipment. Nonetheless, a reinforced fiber composite structure is durable, lightweight, corrosion resistant, and flexible.

II. Repair Development

Composites have been used in industrial applications for many years. They were first used in aerospace applications for wing and fuselage load bearing structures [5]. Common issues during composite development are repair processes. The Commercial Aircraft Composite Repair Committee (CACRC) was established to evaluate composite structure standards and materials used for repair. CACRC quantified the variability between technician experience and the resulting effects on the composite repairs achieved. CACRC evaluated the technicians for their ability to identify composite parts, lead mechanic computer skills, wet lay-up ability, prepreg repair ability, safety adherence, and risk prevention. CACRC standards are supplementary to structural repair manuals (SRM) [5]. Repair process innovation has stagnated. Repair kits use the same step back methodologies [2]. The repair approach removes-damaged material then replaces the material to restore structural properties. Detailed Structural Repair Manuals (SRM) were created to document failures, lessons learned, and guidance for future repairs.

On July 22, 2017 a Boeing 787 aircraft was struck by lightning after departure. Upon landing, it was found that the aircraft had about 46 holes in the fuselage due to lightning strike damage. The Boeing 787 is made of “more than 50% composite material”. Repairing the damages was a complex process, “requiring the removal of damaged composites from affected areas, replacing them with new material, and curing and bonding them to the existing composites” [1]. During the Apollo and Space Shuttle programs, lightning strikes occurred during ground/launch pad operations, launch, and entry. Catastrophic failures required careful attention on repair and interfaces between composite and metallic meshes.

III. Project Overview

The Composite Technology for Exploration project recently completed design, analysis, fabrication and testing of longitudinal bonded joints. These testing results will be used to determine what damages and repair concepts need to be considered. The project overview includes current practices and data from the NASA Kennedy Space Center Materials Science group. The primary goal of the KSC task is to develop repair processes and requirements for launch site repair. Key takeaways will include lab safety, sandwich panel cutting, scarfing, and completing at least one composite repair on a jointed panel.

A. Lab Safety Preparation

Before any repair processes could be started, lab safety training was required. Gloves are required to be worn while handling uncured prepreg materials as they are known to be skin irritants. Masks, goggles, coveralls, gloves and hearing protection are always to be worn during sanding due to harmful composite particles. Any used cloths and wipes containing any solvent (i.e. isopropanol, ethanol, acetone, etc.) are to be placed into designated blue waste bins. The lab surface preparation involves wiping any surface with isopropanol (IPA) that has been used for preparing the laminates before beginning, ensuring to remove any foreign debris. During the process of cutting prepreg sheets, it is suggested to wipe the cutting surface with IPA in between cuts.

B. Panel and Core Cut Out

Before the panel cutting, the first steps of this project were to create a cut plan. Two 12 in x 12 in panels were dimensioned on a panel board using a sharpie. Both panels were labeled (Panel 1 and Panel 2). The panels were cut out using the Rockwell Model 20 Vertical Band Saw and the results are shown in Figure 1.

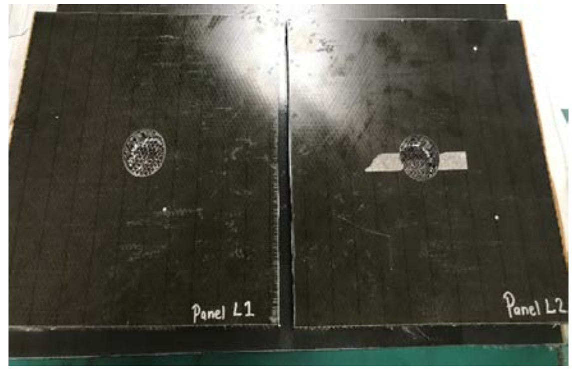

Figure 1. Two fiber-reinforced composite sandwich panels. *The dimensions of each panel are 12 in x 12 in.*

Once the panels are cut out, the core cut out preparation started. One end of a standard drill was attached to the 5" core cutter. Another end was attached to the standard drill to the Bridgeport Prototrak M2 Drill Machine. Panel L1 was placed under the core cutter and the center core was drilled on the panel. An important note to keep in mind is that the core cut process leaves an inner and outer hole. An issue came across while trying to remove the remaining core (plug). The solution was brought up by the technicians. An automated drill operation was programmed to cut out the remaining core in a layer by layer process. First, Panel 1 was stabilized using C-Clamps on two opposite ends of the panel. The drill operation was started at a feed rate of 1.5 and spindle speed of 500 rpm. These lasts steps are repeated for Panel L2.

C. Panel Scarfing

The first steps of panel scarfing include pneumatic orbital hand sander and vacuum preparation. The main pressure valve was manually turned clockwise to close off pressure. The air supply line was connected to the main pressure line. Once that was complete, the pneumatic orbital hand sander and vacuum ports were connected to the air supply. The hand sander was also connected to the vacuum using a larger, corrugated tube, as shown in Figure 2a. An additional vacuum was suggested to collect any airborne particles not collected by the hand sander vacuum. Before starting panel scarfing, 2 inches were marked and tapped out from the edge of the 1 inch center hole. Eight 0.25 inche increments were marked from the edge of the hole to account for each ply cutout. The face sheet was taper scarfed using 80 – 120 grit sand paper. The process was a tedious technique; therefore, the removal of individual layers was closely watched to avoid cutting core/adhesive level. An important note to keep in mind is since the panel is co-cured, dimpling starts to occur from the core a couple of plies in. Also, due to it being fiber placement, thin plies make it more challenging to expose layer by layer during the scarfing process. The approach was to be very gentle scarfing the closer it got to the last layer. Once the last layer is reached, all layers are smoothened out by brushing over the tapered scarf in a circular motion. The final product is shown in Figure 2b.

Figure 2a. Panel Scarfing. *Pneumatic orbital hand sander and vacuum port used for scarfing.*

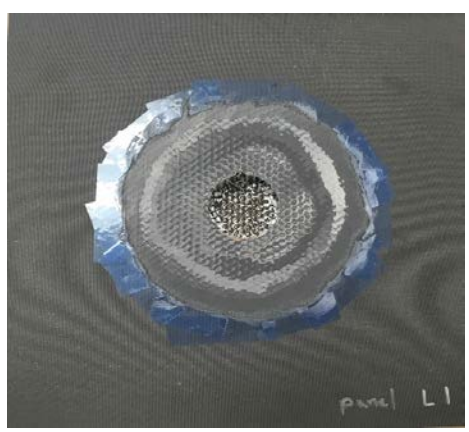

Figure 2b. Final Scarfed Panel. *2 inch scarf exposing 8 plies.*

D. Bonded Joint Panel

Before the panel cutting of the bonded joint, the first steps were to create a cut plan. Four panels were dimensioned on a panel board using a sharpie, as shown in Figure 3a. All panels were labeled (Panel J1, J2, J3, and J4). Panel J1 was dimensioned at 23 inches x 6 inches. Panel J2 was dimensioned at 23 inches x 10 inches. Panel J3 and J4 were both dimensioned at 23 inches x 16 inches. The panels were cut out using the Rockwell Model 20 Vertical Band Saw. Bonded joint panel J1 was used to practice scarfing and create a plan for repair placement, as shown in Figure 3b.

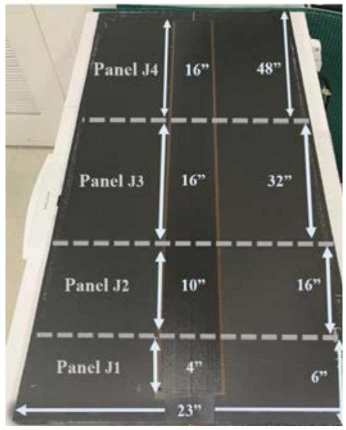

Figure 3a. Bonded Joint Panel. *Four panels are dimensioned for cut.*

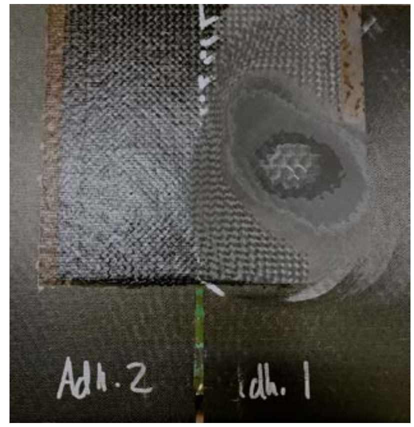

Figure 3b. Bonded Joint Panel J1. *Scarfing and repair placement.*

E. Debulk of the Patch

The first steps of the debulk process included removing the prepreg and film adhesive from the freezer and allowing them to thaw so that no condensation forms on the plastic bagging. The pregreg and film adhesive were thawed for about 1-2 hours in room temperature. Once thawing was completed, the plies were cut for the patch, which correspond to the scarfed region of the panel. Solid Nylon Bagging Film was placed on the tooling surface. Teflon PTFE release film was placed over the solid nylon bagging film. The first ply was placed over the Teflon PTFE release film. A plastic shim was used again to ensure flat application. The backing paper was removed and the steps were repeated for the remaining plies. The following step was to fabricate the vacuum bag over the part. This was done by first, placing a solid nylon bagging film over the patch. A breather cloth was placed over the part. The bottom component of the vacuum port was placed over the breather. Once that was done, high temperature tacky tape was applied on the glass tool around the breather cloth, ensuring that the tacky tape does not contact the breather. One continuous sheet of solid nylon vacuum bagging was placed over the part and secured with the tacky tape. A small slit was cut in the vacuum bag material over the vacuum port in order to securely attach the top component of the vacuum. Any open edges were sealed with tacky tape to ensure there were no leaks. Once that was complete, 25-28 inHg of vacuum was applied over the part and held for 15 minutes, as shown in Figure 4. After the debulk was completed, the patch was put in a sealed bag and placed in the freezer until patch bonding.

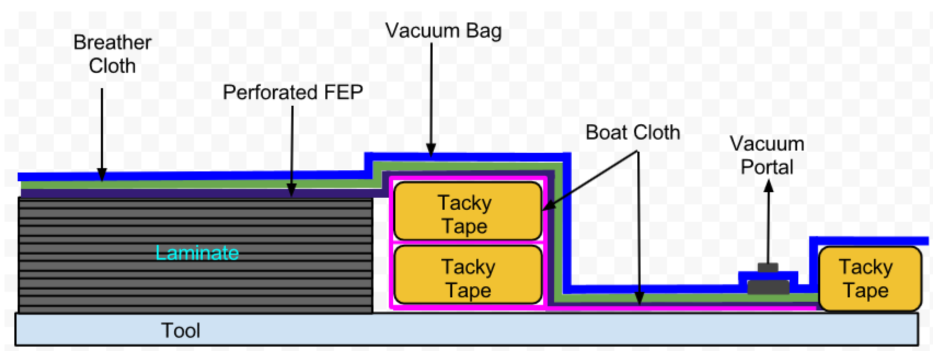

Figure 4. Debulk Cycle. *Vacuum was completed at 25-28 inHg and held for 15 minutes.*

The first steps of the cure cycle were similar to the debulk process. All materials that cannot withstand high temperature cure were removed. Foam blocks were placed around the sandwich panel in order to prevent the vacuum bag from pulling down on the edges of the panel. Thermocouples were placed adjacent to the patch and on the panel. The perforated film adhesive was placed over the part and secured with flash breaker tape. Once that was done, a non-perforated film adhesive was placed over the laminate. A breather cloth was placed over the part and the vacuum port was placed on top of the breather cloth. High temperature tacky tape was applied to the tool around the breather, ensuring that the tape does not contact the breather. One continuous sheet of vacuum bagging (high temperature Nylon) was placed over the part and secured with tacky tape. A small slit was cut in the vacuum bag material over the vacuum port in order to securely attach the top component of the vacuum. Any open edges were sealed with tacky tape, especially around the thermocouple wires, to ensure there were no leaks. Once that was complete, 25-28 inHg of vacuum was applied over the part until the leak rate was less than 2inHg over 5 minutes. The cure cycle started on the hot bonder using a preset program. The hot bonder ramped up to 250°F at 3°F per minute. Once it reached that temperature, it held for 3 hours then ramped up to 350°F at 3°F per minute. At 350°F, it held for 2 hours then ramped down to 100°F at 3°F per minute. Once the curing cycle was complete, the hot bonder was turned off and the part was debagged.

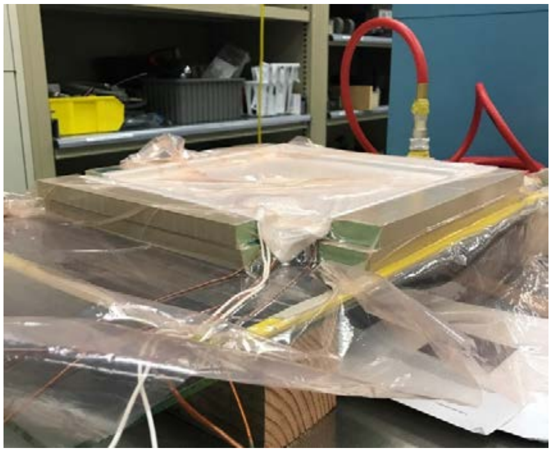

Figure 5. Cure Cycle. *Panel was cured using hot bonder heater blanket*

F. Thermal Survey Results

A thermal survey was performed to determine differences in the heating across the blanket. The data of the cure cycle was recorded through the thermocouple readings on the Fluke Data Acquisition. There were a total of six thermocouples connected to the Fluke and two thermocouples connected to the hot bonder. The data was recorded and shown on Figure 6. This figure also displays the locations of the thermocouples on the part. It was observed that Fluke Thermocouple Ch. 2 experienced the highest temperature reading of about 375°F. This could have been due to the center placement of the thermocouple on the part. The second and third highest thermocouple readings were Fluke Thermocouple Ch. 1 and Fluke Thermocouple Ch. 4. These were relatively close to the center as well, which justifies a reasoning that higher temperatures are experienced closer to the center of the heater blanket. Hot bonder thermocouples were also used in this experiment to show differences between the set temperature point and the highest thermocouple. Based on the graphed data, it can be seen that the hot bonder set point is closely similar to the highest hot bonder thermocouple reading. However, when compared to the highest fluke thermocouple reading (Ch.2), the difference is 25°F. This large difference could have been due to the fact that the hot bonder thermocouples were placed furthest away from the center of the heater blanket.

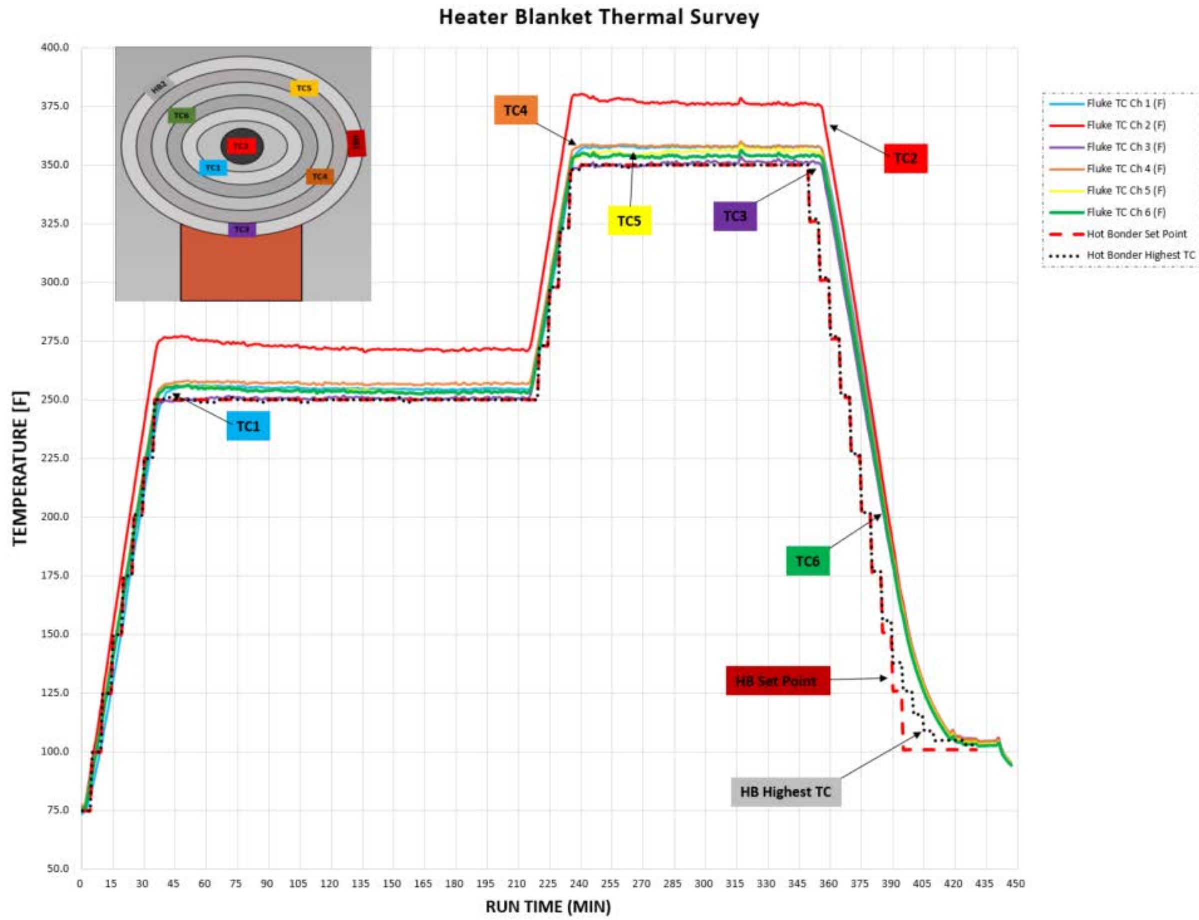

Figure 6. Thermal Survey. *Results of cure cycle.*

IV. Challenges and Future Work

Some of the challenges faced during repair processes were scarfing efficiency, proper vacuum sealing, and retrieving data from the fluke thermocouples. Little to no experience in scarfing caused minor setbacks due to uneven tapered scarfs and dimpled surfaces. After proper training and practice, these setbacks were fixed and repair processes progressed. Once acceptable scarfing conditions were met, attaining a proper vacuum seal was critical to test the prepreg repair patch. However, pressure loss was experienced after multiple attempts. It was noted that potential reasons could have been due to bad sealing and air escaping through small imperfections of the core hole. A separate challenge was retrieving data from the fluke thermocouples. The issue was nonfunctioning thermocouples due to overworked wiring. This was resolved by replacing them with new thermocouples. Additionally, further concerns were finding a vacant workplace to perform the repair processes. It was crucial to have a spacious area in order to place the bonded joint panel, vacuum pump, hot bonder, and thermal fluke reader all together. A space became available at the Mechanical Testing Lab during the middle of the project period.

Future work consists of completing additional repairs on the remaining jointed panels. Bonded joint Panel J2 will be the checkpoint of the project, as shown in Figure 7. Further repair work and documentation will be conducted for the remaining of the internship period.

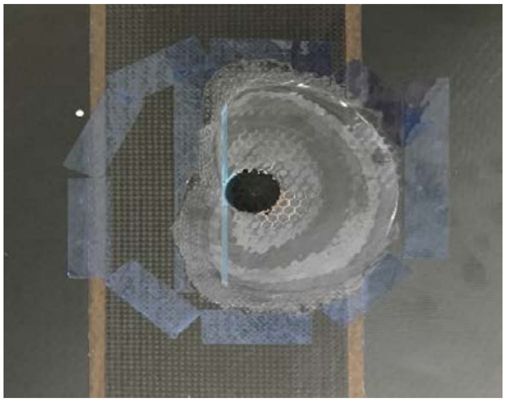

Figure 7. Bonded Joint Panel J2. *Scarfed and ready for future repair.*

V. Conclusion

The Composite Technology for Exploration project has completed design, analysis, fabrication and testing of longitudinal bonded joints. The primary goal of the KSC task was to develop repair processes and requirements for launch site repair. Working with the Materials Science group exposed me to lab safety, sandwich panel cutting, scarfing, debulking and curing, and completing at least one composite repair on a jointed panel. Our testing results will assist those continuing research studies on repair development. Ultimately, it will serve as guidance for repair processes and requirements in support of SLS – scale hardware.

VI. Acknowledgments

This opportunity would not have been possible without the support of my mentor, Sarah Cox. Her decision to select me for this position allowed me to engage and contribute to a valuable experience in my life. Sarah has done an amazing job at being available for questions every step of this project. She has ensured that I gain the most from this internship by connecting me with other engineers in the Materials group and inviting me to events that were unforgettable. Her communication and passion for her work is truly inspirational and has motivated me to look forward to my journey in Mechanical Engineering. I would also like to thank my co-mentor Susan Danley for her support and feedback during this opportunity. She did an amazing job at training me to become better both in the lab and with my technical writing. The relationships and conversations built here at the Kennedy Space Center were like no other. This experience has made the hugest impact on my life due to my professional and academic growth; a takeaway I will never forget.

VII. References

[1]Boeing. (2017). *Boeing 787 Grounded for a Week after Lightning Strike.* London: Airinsight.

[2]Epperson, J., Borgman, M., & Welch, J. (2015). *Examples of Substantiation Testing and Documents to Support Large Areas of Composite Repair.* Spirit Aerosystems.

[3]Hubert, P., Preau, M., Kothari, R., & Wilson, D. (2015). *Effect of Processing Parameters On Bonded Repair Quality and Strength.* McGill University. Bombardier Aerospace: Composite Transport Workshop on Damage Tolerance and Maintenance.

[4]Keller, R. (2004). Challenges in Composite Maintenance and Repair: A Perspective. (S. f. Engineering, Ed.) 1-14.

[5]Miller, S., Minaie, B., & Keshavanarayana, S. (2014). *Cure Management for Bonded Composite Repair.* Witchita State University.

[6]Tomblin, J., & Salah, L. (2014). *CACRC Depot Bonded Repair Round Robin Investigation.* Joint Advanced Materials & Structures Center of Excellence.

[7]Tomblin, J., Salah, L., Welch, J., & Borgman, M. (2004). *Bonded Repair of Aircraft Composite Sandiwch Structures.* National Institue for Aviation Research, Witchita State University. The Boeing Company.