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COMPOSITE REPAIR ENGINEERING CASE STUDIES FOR U.S. ARMY AEROSTRUCTURES

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ABSTRACT

The U.S. Army fields a multitude of aircraft mission design series (MDS) developed by several different original equipment manufacturers with varying mission requirements and flight profiles. The H-64 Apache, H-60 Blackhawk, and H-47 Chinook are the U.S. Army's primary rotary-wing workhorses in the enduring fleet. These aircraft primarily have airframes composed of high strength aluminum alloys due to a provenance beginning within the second half of the 20th century. Advanced composite materials include portions of these aerostructure designs but mainly comprise secondary structures such as covers, fairings, spoilers, and antenna. More structurally critical applications include rotor blades, bulkheads, stringers, and stabilators. This paper will present examples of U.S. Army damaged composite aerostructures and will document an analysis approach for battle damage repairs using both scarf patch and scab patch methods. As this evaluation is presented, Classical Laminated Plate Theory (CLPT), finite element analysis (FEA), and bonded joint analysis will be applied to a representative U.S. Army battle damaged and worn component to substantiate structural repairs. This work will focus on a static analysis of composite structure repairs utilizing maximum allowable strain, stiffness, and bonded joint analysis through an equivalent strength methodology. The structural analysis in this work assumes the materials, tooling, skillsets, and capabilities are organically available and proper at the repair location.

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I. INTRODUCTION

The purpose of this paper is to identify current advanced composite applications and history pertaining to U.S. Army aircraft, highlight the typical battle damage and wear found in the fleet, discuss maintenance and engineering processes used for sustainment, and to discuss future aircraft considerations. Additionally, a case study is performed on a composite repair to a carbon fiber laminate sandwich structure on a H-47 Chinook crown skin. The case study outlines classical laminated plate theory (CLPT) to present an equivalent (or better) strength and stiffness comparison. Bonded joint analysis of the repair is also shown to verify structural adequacy and load transfer. This paper demonstrates the overall engineering processes, analysis, and challenges associated with the repair of U.S. Army advanced composites on aircraft applications in the field and depot. The long-term goal of this effort is to demonstrate and collaborate on a U.S. Army approach to advanced composite aerostructure repair substantiation with the objective of sharing a comprehensive method through a series of technical works. This paper provides the initial foundation to begin this series by presenting the first milestone of static structural analysis.

The U.S. Army operates and maintains several aircraft MDS to meet the warfighter's multi-domain mission. Aircraft fielded by the U.S. Army originate from multiple equipment manufacturers. These aircraft include rotary-wing configurations such as the AH-64D/E Apache, CH-47F Chinook, and H-60A/L/V/M Blackhawk aircraft which significantly vary in mission parameters and flight profiles. These aircraft contain structures made from a majority aluminum, steel, and titanium alloys which have dominated aircraft designs for much of the history of powered flight. However, the use of advanced composite material systems such as fiberglass, carbon, and aramid fiber reinforcement with high performance epoxy resins has steadily increased to optimize structural designs and improve mission capability.

Fiberglass was first developed in the 1930s, with carbon fiber materials subsequently developed in the 1980s (reference 10). The utilization of polymer matrix composite material systems such as carbon-epoxy fabrics and aramid honeycomb core has significantly increased in civilian and military aircraft designs. This is a result of the significant increase in specific strength and stiffness capabilities and resistance to corrosion and fatigue that composites offer to the enterprise. While composites certainly are not new to aviation, their uses were primarily limited to secondary and lightly loaded structures such as covers, fairings, spoilers, and antenna.

During the RAH-66 Comanche helicopter development effort, the U.S. Army made attempts to build a fiber-reinforced crashworthy airframe structure through the Advanced Composite Airframe Program (ACAP) (reference 15). The ACAP effort, conducted by Bell Helicopter and Sikorsky Aircraft, was intended to determine the feasibility of utilizing composites for fuselages and primary structure. The driver for this effort was to reduce weight, eliminate metal corrosion, build more tolerant battle damage structures, and reduce aircraft radar cross sections. While the program was cancelled in February 2004, the Comanche and ACAP programs demonstrated that composites would be acceptable for use on military aircraft. Additionally, these programs helped to set a precedence for future repairs of composite structures. The repair methods of that effort included standard wet layup, for smaller damage, and replacement of modular components for larger damage. Replacement of modular sections accomplished by removing smaller sub-sections of the structure and effectively splicing in a replacement part (reference 13).

As technologies develop, aircraft mature, and strides are made in advanced materials development and analysis, aircraft have increased use of advanced composites within the aerostructure. These advanced materials are now being used in highly loaded structures, including

aerodynamic profiles.

Evolving mission requirements led to the use of unmanned fixed wing aircraft, such as the MQ-1 Grey Eagle and RQ-7 Shadow, which employ carbon fiber composites as the bulk of the fuselage structure. Structurally critical applications include main rotor blade spars, bulkheads, fuselage skins, stringers, walkways, and stabilators. Additionally, it is suspected that future U.S. Army aircraft designs will have structures primarily constructed from advanced composite materials. The U.S. Marine Corps' MV-22 Osprey tiltrotor aircraft has an airframe constructed primarily from carbon fiber reinforced epoxy material systems, which enables a more structurally efficient aircraft. The majority of the aircraft's exterior is composite, in addition to the proprotor blades and many of the internal structural components. In order to increase payload, range, airspeed, and other critical mission capabilities, it is likely that the U.S. Army and aircraft manufacturers will begin incorporating and preparing to maintain a much greater number of these types of structures on future aircraft.

Figures 1 through 4 provide overviews of composite applications on several U.S. Army aircraft and differentiate areas made from traditional composite materials such as fiberglass and/or aramid (light grey) and more modern carbon fiber (dark grey) reinforced epoxy composites.

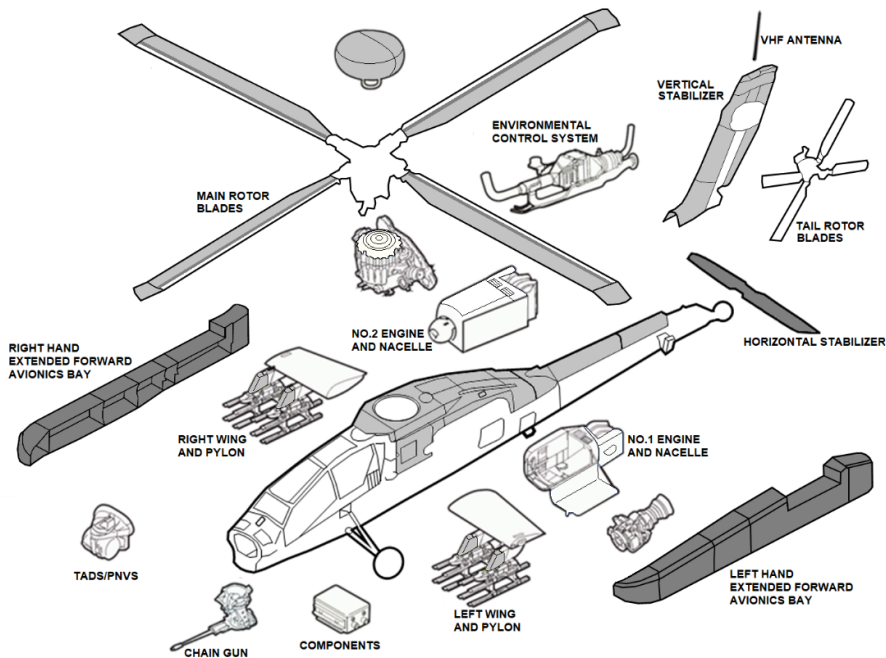


Figure 1. H-64 Apache Composite Layout.

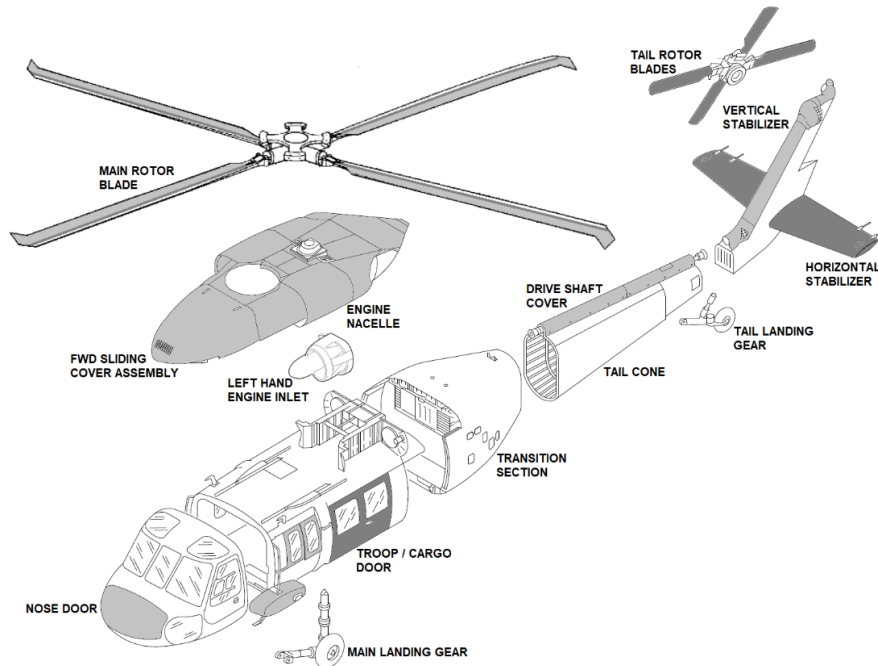


Figure 2. H-60 Blackhawk Composite Layout.

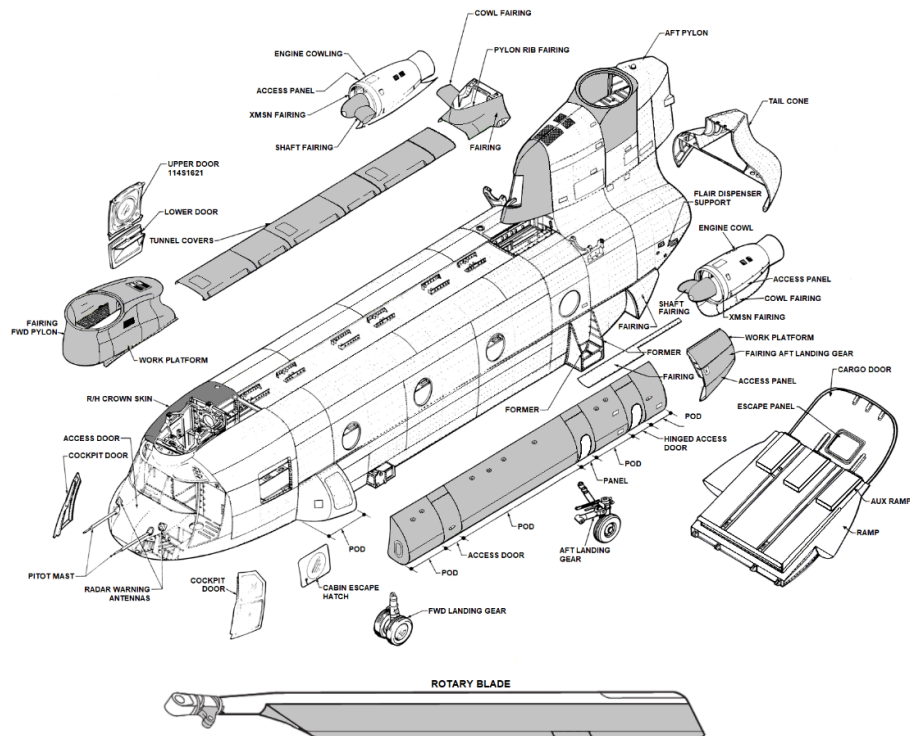


Figure 3. H-47 Chinook Composite Layout.

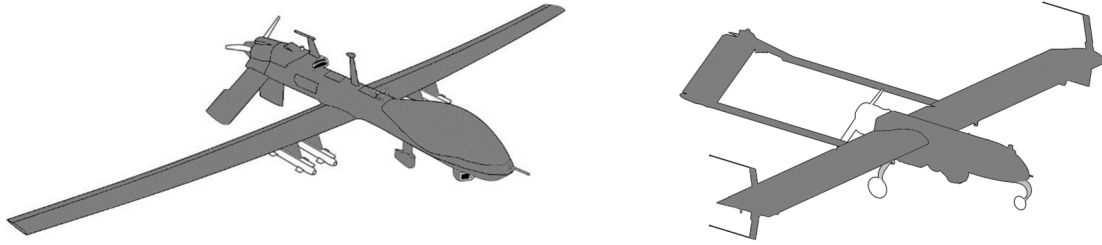


Figure 4. MQ-1 Grey Eagle and RQ-7 Shadow Composite Layout.

As seen in the figures, U.S. Army aircraft currently employ advanced composites as part of load bearing structure and aerodynamic surfaces. This is explicitly shown by the right-hand upper crown skin on the H-47 aircraft, Extended Forward Avionics Bays (EFABs) on the H-64, horizontal stabilator and cargo/troop door of the H-60, and fuselage profiles of the unmanned aircraft systems (UAS). Therefore, it is imperative that the U.S. Army recognize the future expansion of composite applications and the maintainability considerations of these materials in fielded and depot environments.

II. DISCUSSION

The U.S. Army deploys aircraft into hostile locations throughout the world from both an environmental and combat related perspective. These locations subject U.S. Army aircraft to harsh conditions in which aircraft and components deteriorate and are damaged at an accelerated rate. Figure 5 shows the magnitude of typical debonding between aramid laminates and honeycomb on a H-60 transmission access bay door; this debonding is a result of the normal operating conditions. Figure 6 shows sudden impact damage to an H-64 Apache EFAB close-out panel and sub-structure, exposing the carbon fiber/epoxy fibers. Figure 7 shows significant debonding damage to a H-47 Chinook upper crown skin, exposing the carbon fiber/epoxy internal structure.

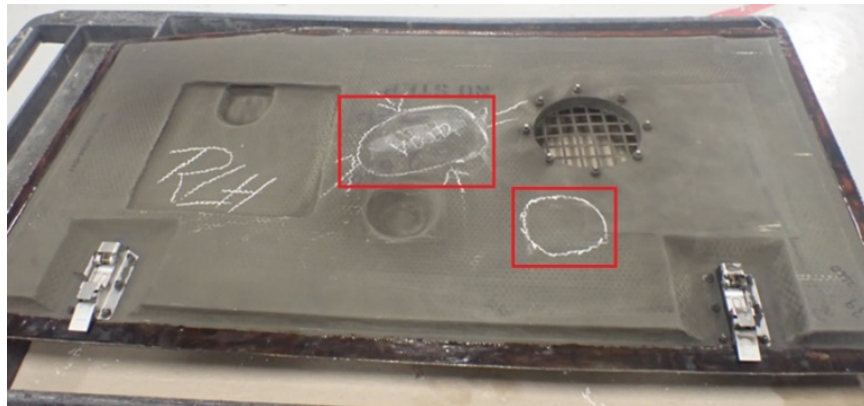


Figure 5. H-60, Access Bay Door Debonding.

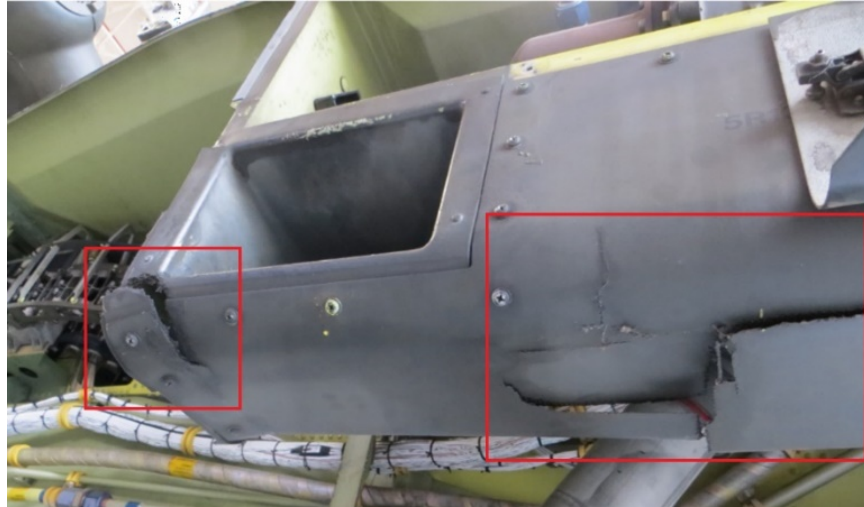


Figure 6. H-64, EFAB Close-Out Panel Impact Damage.

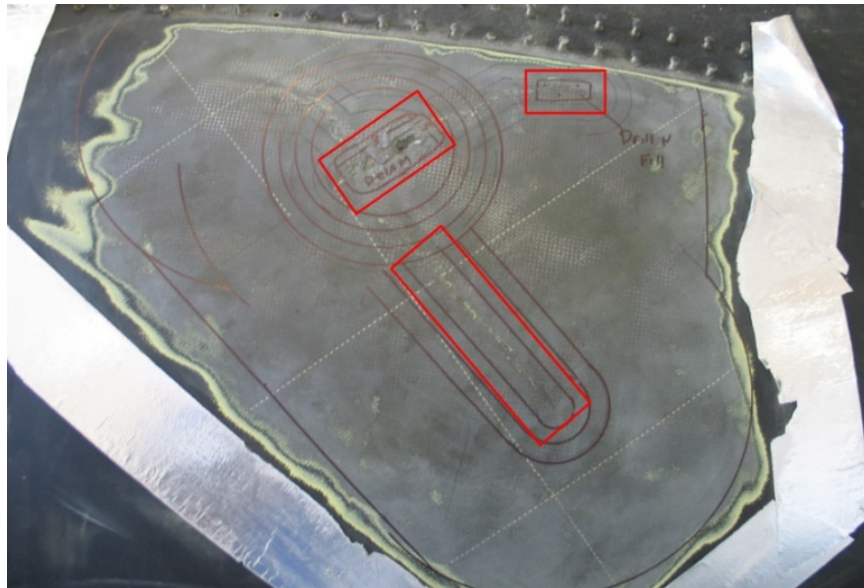


Figure 7. H-47, Right Hand Upper Crown Skin Damage.

Typical damage of this magnitude would normally require replacement of the respective components on aircraft; however, this is not always an option. Whether through supply availability, excessive replacement costs, mission requirements, or magnitude of labor required, it has become paramount for the U.S. Army to increase the repair capability of composites in the field. Currently, the U.S. Army separates aircraft maintenance into two main categories: field and sustainment level maintenance (reference 19). Field-level maintenance is typically performed by the operating units, or their supporting maintenance personnel, and usually includes tasks whose procedures can easily be performed on aircraft or its components. Sustainment-level maintenance is performed at specific facilities that specialize in deeper levels of maintenance and/or overhaul. Facilities include depots, field support brigades, field support battalions, and logistics readiness centers (reference 19). These sites provide aircraft inspections, preventative maintenance, repairs,

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modifications, assembly of aircraft to aid the warfighter in maintaining airworthiness, and operational readiness.

Army Regulations (AR) 70-62 and 750-1 (references 3 and 4) are the governing documents used to administer airworthiness and maintenance processes by prescribing the policies and procedures for life cycle sustainment of manned and unmanned aircraft systems. The U.S. Army Combat Capabilities Development Command (DEVCOM), Aviation & Missile Center (AvMC), Systems Readiness Directorate (SRD), is the organization that exercises airworthiness cognizance of U.S. Army aircraft. This airworthiness authority is delegated to the SRD Director by the commanding general of U.S. Army Aviation and Missile Command. However, the challenge the U.S. Army currently faces is the maintainability and sustainability of advanced composites given the metallic preponderance of its largest aircraft fleets.

Since advanced composite materials are costly, have expiration dates, and require specialty equipment, materials, training, and processes, many repair designs are difficult to implement in the field. Repair sites may not have the necessary facilities, tooling or skillsets required to process composite repairs to a manufacturing level standard. Typical in-theater repairs include traditional wet-layup and ambient temperature cure methods. This is in stark contrast to the prepreg, vacuum bag, autoclave, and elevated temperature cures necessary to achieve the highest level of repair quality.

The U.S. Army does have field facilities that have increased repair capabilities such as Theater Aviation Sustainment Maintenance Groups (TASMGs) and maintenance sites which report under the AMCOM Field Maintenance Directorate (AFMD). The AFMD primarily focuses on performing aircraft scheduled phase inspections and RESET, of which the primary focus is to improve readiness, quality, and availability of U.S. Army aircraft. These locations provide higher standards of repair and are usually sought after when heavier maintenance is required on the aircraft.

During an aircraft phase or RESET cycle, components that have been damaged are removed from the aircraft and subsequently replaced or sent off for repair. This includes composite components, which would typically be sent to a local composite shop. However, repair availability becomes dependent on the available technical data, materials, skillsets, equipment, and repairable damage limits authorized for that respective component.

The case study provided in this paper identifies one such example where a TASMG unit removed a H-47 Chinook upper crown skin, simulated damage on the internal surface, and performed a structural repair through the local composite shop. The subject skin is made from a prepreg carbon fiber laminate and Nomex honeycomb core. The field level repair included a tapered scarf type patch design made from a carbon fiber prepreg laminate and a bonded lap joint. The repair also included one additional overply layer, as well as a fiberglass cover-ply, which is typical per current Army technical data. This paper reflects SRD's ongoing efforts to enhance analytical capabilities for the structural evaluation and disposition of composite repairs.

III. LITERATURE REVIEW

SRD is the U.S. Army's delegated cognizant airworthiness authority and is responsible for assuring that U.S. Army aircraft and their modifications meet airworthiness standards. From a maintenance perspective, these standards are established through published technical data including Technical Manuals (TMs), Depot Maintenance Work Requirements (DMWRs), and original equipment manufacturer (OEM) drawings. Each MDS has its own set of TMs, DMWRs,

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and OEM drawings, however, the U.S. Army has also published generalized maintenance documents that apply to all U.S. Army aircraft.

The U.S. Army uses two documents which encompass the sustainment of these materials. The first is TM 1-1500-204-23-11 (reference 17), “*General Aircraft Maintenance Advanced Composite Material General Maintenance and Practices*”, which provides aircraft maintainers with proper instructions for maintaining advanced composite materials at a field level. The second document is DMWR 1-1500-204-1 (reference 9), “*Depot Maintenance Work Requirement for Advanced Composite Repair*”, which provides similar data, but from a depot facility perspective. These two documents provide standardization for the types of materials, tools, repairs, methods, and processes of handling advanced composite materials. Figures 8 and 9 show the cover page for each document.

TM 1-1500-204-23-11 TECHNICAL MANUAL AVIATION UNIT MAINTENANCE (AVUM) AND AVIATION INTERMEDIATE MAINTENANCE (AVIM) MANUAL GENERAL AIRCRAFT MAINTENANCE ADVANCED COMPOSITE MATERIAL GENERAL MAINTENANCE AND PRACTICES VOLUME 11 DISTRIBUTION STATEMENT A: Approved for public release; distribution is unlimited. HEADQUARTERS, DEPARTMENT OF THE ARMY 30 AUGUST 2013
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Figure 8. TM 1-1500-204-23-11, General Field Maintenance.

DMWR 1-1500-204-1 DEPOT MAINTENANCE WORK REQUIREMENT FOR ADVANCED COMPOSITE REPAIR REPORTING OF ERRORS AND RECOMMENDING IMPROVEMENTS You can help improve this IETM. If you find any mistakes or if you know of a way to improve the procedures, please let us know. If your IETM supports online forms, fill in the electronic publication change request and when connected to the internet, transmit the form. If your IETM does not support online forms, obtain a copy of a DA Form 2028, Recommended Changes to Publications and Blank Forms. Your IETM may include a partially completed DA Form 2028. Print out the form and complete filling in the pertinent information. For IETMs without a printable DA Form 2028, blank forms should be available through your publications system. Complete the DA Form 2028 and mail it directly to: Commander, U.S. Army Aviation and Missile Command, ATTN: AMAM-MAL-AS, Redstone Arsenal, AL 35898-5000. If you are unable to obtain a DA Form 2028, you may provide the recommendations by letter to the above address. You may also send in your recommended changes via electronic mail or by fax. Our fax number is DSN 788-6546 or Commercial 256-842-6546. Our email address is 2028@redstone.army.mil. You may also submit your recommended changes at the following Web site https://amcom2028.redstone.army.mil. A reply will be furnished to you. AVAILABILITY NOTICE – This publication is not available through U.S. Army Publication Distribution Centers. It must be obtained from U.S. Army Aviation and Missile Command, ATTN: AMAM-MAL-AS Redstone Arsenal, AL 35898-5000. DISTRIBUTION STATEMENT C – Distribution authorized to U.S. Government Agencies and their contractors. Critical Technology, 15 July 2007. Other requests for this document shall be referred to Commander, U.S. Army Aviation and Missile Command, PEO Aviation, ATTN: SP4E-AU/C4T, Bldg. 9678, Hicks Road, Redstone Arsenal, AL 35898-5000. WARNING – This document contains technical data whose export is restricted by the Arms Export Control Act (Title 22, U.S.C., Sec 2751, et seq.) or the Export Administration Act of 1979 (Title 50, U.S.C., App. 2401 et seq.), as amended. Violations of these export laws are subject to severe criminal penalties. Disseminate in accordance with provisions of DoD Directive 5230.25. DESTRUCTION NOTICE – For classified documents, follow the procedures in DOD 5220.22-M, National Industrial Security Program Operating Manual and/or DODM 5200.01, Information Security Program. For unclassified, limited documents, destroy by any method that will prevent disclosure of contents or reconstruction of the document. US ARMY AVIATION AND MISSILE COMMAND REDSTONE ARSENAL, AL 2 March 2023

Figure 9. DMWR 1-1500-204-1, General Depot Maintenance.

These documents are intended to be used in conjunction with the aircraft or component specific technical manuals, which provide detailed damage limits and repair instructions for their respective platform. Each aircraft MDS retains its own respective technical manual and airframe structural overhaul DMWR, which provides detailed repair instructions specific to that aircraft configuration. However, there are instances where clarifications, deviations, and/or exceedances from these technical sources are required to overcome schedule and supply constraints and support mission readiness.

The SRD's Maintenance Engineering Call (MEC), a document governed by AR 70-62, is used to provide instructions, clarifications, and deviations to Army aircraft, components, or processes. MECs are submitted by field or depot personnel to SRD for investigation, evaluation, and disposition. While the MEC is a versatile tool used by the Army, there are limitations to what it may cover. MECs are not authorized to accept aircraft/crew risk, implement aircraft configuration changes, or alter aircraft operations/performance. However, MECs may be authorized to disposition Critical Safety Items (CSIs) and primary structural components, with the coordination through appropriate engineering functions. Figure 10 shows a brief overview of the MEC flow process.

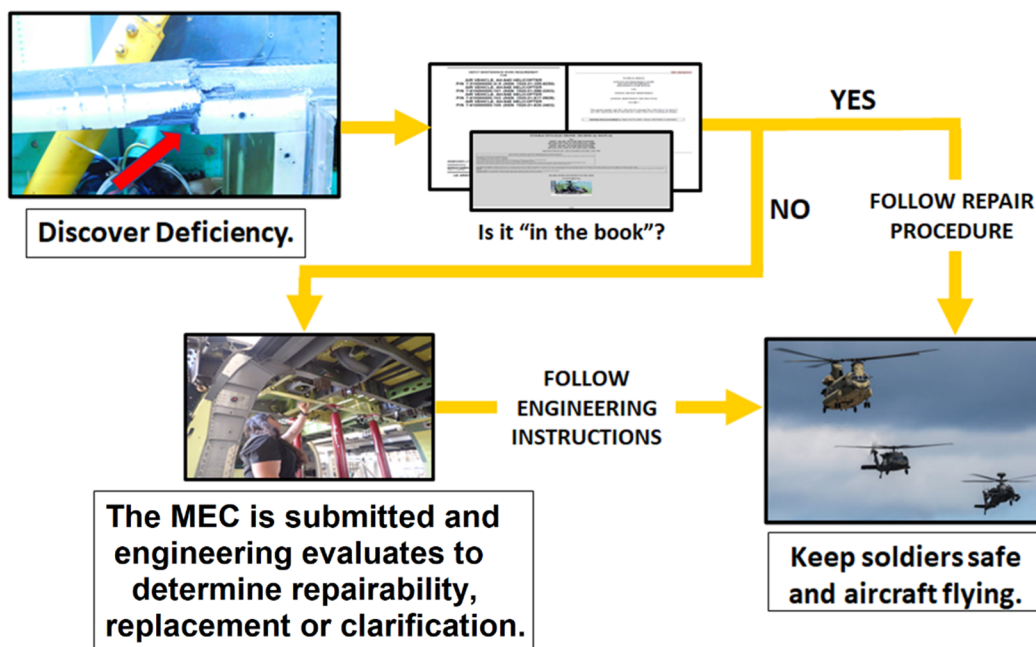


Figure 10. General MEC Process.

In the context of advanced composites, MECs are usually submitted to SRD engineers for repair authorizations that are beyond the allowed procedures in the platform specific repair manual. Additionally, MECs may provide specific stand-alone repair instructions that do not fall within the TM or DMWR. An example MEC would include a set of instructions for repairing a skin-to-core de-bond that exceeds the threshold limits established within the repair manual. The maintainers would then implement the repair based on the instructions provided within the MEC by SRD engineers.

Furthermore, if repeated MECs are seen on the same issue, or new repair processes are

developed, a Maintenance Engineering Order (MEO) may be published to make permanent changes to the manual. This vehicle allows for elimination, incorporation, modifications, or implementation of new procedures into Army technical data. However, for this case study, SRD has not, at this time, approved this repair into the DMWR for future use. This case study repair was developed as a separate effort led by SRD as part of a validation and verification (val/ver) project to confirm the repair process and structural adequacy. It is the intent of SRD to work towards incorporating a similar repair into the DMWR, via an MEO, for use on future overhaul programs.

IV. CASE STUDY DESCRIPTION

The H-47 Chinook has an advanced composite structural panel called the crown skin which is located on the right hand side of the aircraft, near the cabin cockpit and above the crew access door, which is shown in Figure 11.



Figure 11. H-47 Crown Skin Specimen.

The subject crown skin is a carbon fiber epoxy composite laminate sandwiched over Nomex honeycomb core structure. Each face of the sandwich panel is made from several layers of carbon fiber plain weave prepreg and one external layer of style E-glass prepreg. The E-glass composite is an external ply used for damage protection, galvanic isolation, and an interface for attaching electrical discharge material. This ply was excluded from evaluation as it does not impose significant strength to the part. There were no structural repair instructions available for this carbon fiber component, meaning that significant damage beyond the limits often leads to replacement of the assembly. However, due to cost, availability, and maintenance burdens, the U.S. Army recognized that repairs to these components are prudent. As a result, SRD developed repair instructions for the purposes of implementing into DMWR 1-1520-271. This DMWR is for the repair and overhaul of the H-47 Chinook airframe structure.

The specific repairs for carbon fiber composites on the H-47 were developed based off similar repair methods established in TM 1-1500-204-23-11, but specifically tailored to this aircraft and its materials. To demonstrate proper repair techniques and structural integrity of carbon fiber composite repairs, a Val/Ver effort was performed on the crown skin.

To initiate the process, the subject crown skin was removed from an aircraft and used as a repair specimen. A one-inch circle of simulated damage was imposed onto the inner surface of the panel as shown in Figure 12. The damage included full laminate thru thickness and honeycomb core damage, requiring the complete removal of material to the opposite laminate face.

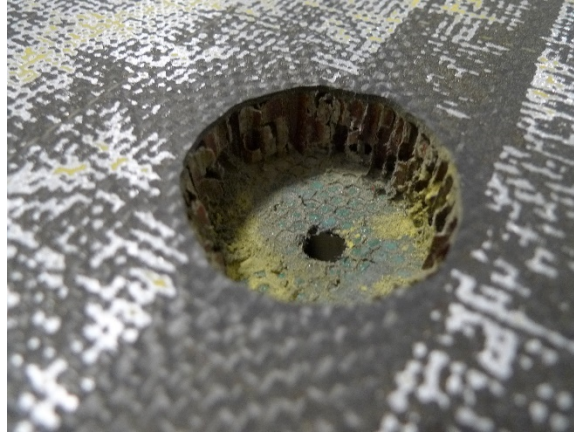


Figure 12. Damage Simulation.

To repair the panel, the core material was removed to the opposite facing laminate surface and a ½ inch overlap taper scarf repair configuration was sanded onto the internal laminate face per work instructions and TM 1-1500-204-23-11 (reference 17); the result of a sanded scarf repair is shown in Figure 13. Implementation of a ½ inch overlap per ply resulted in a six-inch scarf length (i.e. 63-1 taper ratio) per side. New replacement core material was cut to size, bonded to existing structure, and trimmed flush to match the original profile of the core face. After cleaning and preparing the area, a layer of Cytec FM300-2M film adhesive was applied to the scarf surface and core face repair joint surface. Pre-impregnated repair plies were then trimmed to size and progressively laid-up over the center of the damage and film adhesive. The same number of repair plies, plus one additional ply of carbon fiber pre-impregnated repair fabric, was used. Once properly laid-up over the damage, the carbon fiber repair was vacuum bagged and co-cured using a heat blanket and temperature monitoring thermocouples; see Figures 14 and 15. Once cured, the composite repair was sanded smooth to match the existing structural profile and prepared for follow-on bonding. Upon completion of the carbon fiber repair, an external layer of E-style fiberglass was applied to the structure using the wet layup methods provided in TM 1-1500-204-23-11 (reference 17).

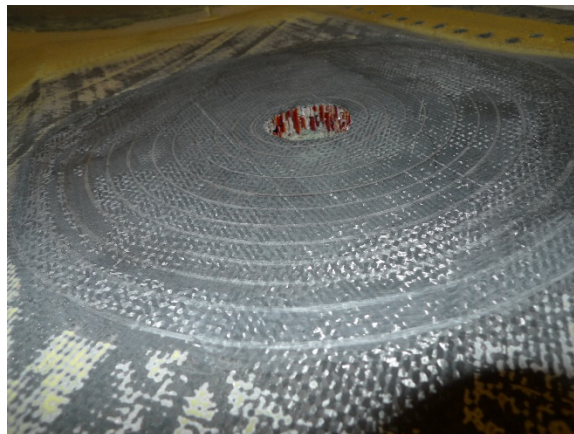


Figure 13. Scarf Repair Preparation.

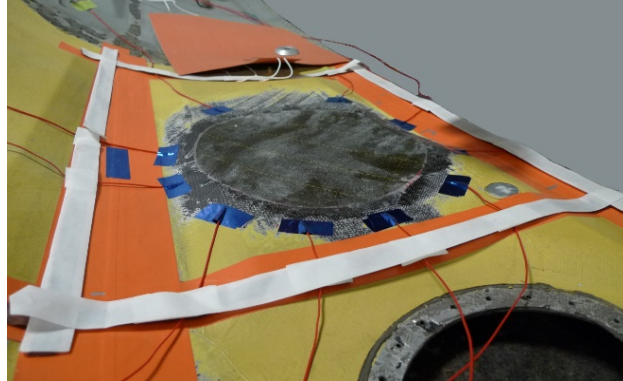


Figure 14. Repair Prepreg Layup.



Figure 15. Repair Cure.



Figure 16. Final Repair.

The repair provided in this case study has not been included into the Chinook DMWR or any other U.S. Army technical data as of this writing. This is due to the need for testing (static and fatigue) of the laminate and bonded joint structure, further verification of the repair process, and damage tolerance development of the base structure. However, it served as an opportunity for further analysis from a statics perspective; based on publicly available material allowables. This

paper presents a limited static structural analysis using classical laminated plate theory (references 10 and 14), scarf joint analysis, and Volkersen Shear Lag Theory (reference 10) of the lap joint. Additionally, a finite element analysis (FEA) of the laminates was performed to compare stiffness and strain responses obtained using CLPT; that analysis can be found in Appendix B.

However, fully substantiated repairs to composite primary structures include fatigue analysis (when applicable) and coupon testing to verify the results. Fatigue and coupon testing are beyond the scope of this paper, but are a part of U.S. Army's recommended repair substantiation doctrine for advanced composite primary structures. Further efforts, to include fatigue analysis and testing, are planned for inclusion within subsequent and related work.

V. LAMINATE STIFFNESS ANALYSIS

The following initial assumptions are made:

1. The style 108 E-glass layer of the original laminate is not included in the analysis as it is assumed to not contribute sufficient strength to the structure and is only acting as a protective and electrically insulative layer.
2. The curvature of the crown skin is neglected within CLPT analysis and is assumed as a flat laminate.
3. Acceptable taper sanding of the original laminate for bonded patch installation was performed per TM 1-1500-204-23-11 and DMWR 1-1520-271 instructions.
4. Acceptable surface preparation was performed prior to bonding.
5. The repair layup matches the original structure lamina orientations and assumes no ply orientation error during layup.
6. The vacuum bagging and elevated temperature cure cycle was performed as required in the work instruction.
7. The laminates are considered in a state of plane stress and that in-plane displacements vary linearly through the thickness and transverse displacement is constant through the thickness.
8. The nominal ply thicknesses are constant throughout the laminate.
9. Assumed that the repair was deemed acceptable per the provided inspection procedures and obtained the necessary material properties.
10. Fatigue analysis and failure testing were not performed and are beyond the scope of this paper.
11. The outer fiberglass repair ply laid up in the 0/90-degree laminate orientation.
12. This analysis is only focused on one half of the laminate panel, which is considered unsymmetrical. The core material and opposite laminate face sheet is excluded from analysis.

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Analysis of aircraft structures begins with obtaining material properties of the original base laminate structure and the repair materials to be used. Table 1 provides the lamina material properties used in this analysis. IM7/8552 is an intermediate modulus carbon fiber plain weave fabric and epoxy resin pre-impregnated material system. The material properties (reference 5) for this pre-preg are available to the public. The repair materials include carbon fiber plain weave pre-preg M20/40%/G904 which is qualified under AMS 3970, Cytec FM 300-2M film adhesive, and an outer mold line (OML) 8-harness satin weave E7781/EA9396 fiberglass epoxy wet-layup.

ORIGINAL STRUCTURAL PROPERTIES	
Material Properties, IM7/8552 Pre-Preg PW	Values
Tensile Modulus, E1	9.250 MSI
Tensile Modulus, E2	9.250 MSI
Averaged Shear Modulus, E12	0.520 MSI
Poisson's Ratio, ν_{12}	0.032
Fiber Areal Weight (FAW)	196 g/m ²
Nominal Ply Thickness, t	0.0079 inch
REPAIR MATERIAL STRUCTURAL PROPERTIES	
Material Properties, M20/40%/G904 Pre-Preg PW	Values
Tensile Modulus, E1	8.635 MSI
Tensile Modulus, E2	8.405 MSI
Averaged Shear Modulus, E12	0.713 MSI
Poisson's Ratio, ν_{12}	0.059
Fiber Areal Weight (FAW)	197 g/m ²
Nominal Ply Thickness, t	0.0083 inch
Material Properties, E7781/EA9396 Wet Layup	Values
Tensile Modulus, E1	3.420 MSI
Tensile Modulus, E2	3.390 MSI
Averaged Shear Modulus, E12	0.758 MSI
Poisson's Ratio, ν_{12}	0.115
Fiber Areal Weight (FAW)	295 g/m ²
Nominal Ply Thickness, t	0.0085 inch

Table 1. Composite Lamina Material Properties.

The original base laminate pre-impregnated material (IM7/8552) is a 350°F cure material, while AMS 3970 is a 250°F curing material. It is important to identify that during the repair process, the cure of the repair is limited to its design cure temperature; the temperature is not brought back to the original base laminate cure temperature and potentially damaging the adjacent structure. Additionally, as the E7781/EA9396 material has a much lower cure temperature (i.e. 150°F), it is not co-cured with the pre-impregnated repair materials. The material properties for these composite pre-impregnated materials are also available to the public (references 5, 6, 12, and 18). In each material case, the tensile and compression modulus values in each local axis (i.e., 1 & 2 direction) were averaged to obtain respective modulus strengths.

Table 2 provides the crown skin inner laminate ply stack-up while Table 3 provides the mating repair laminate ply stack-up. Matching the original number of plies, plus one additional repair ply, is standard repair protocol per TM 1-1500-204-23-11. Additionally, the external fiberglass wet-

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layup ply was incorporated using a 0/90 ply orientation to match the structural pattern of the laminate.

PLY #	Material	Nominal Thickness	Angle
1	IM7/8552 (CF-EP)	0.0079	45
2	IM7/8552 (CF-EP)	0.0079	45
3	IM7/8552 (CF-EP)	0.0079	0
4	IM7/8552 (CF-EP)	0.0079	45
5	IM7/8552 (CF-EP)	0.0079	45
6	IM7/8552 (CF-EP)	0.0079	0
7	IM7/8552 (CF-EP)	0.0079	45
8	IM7/8552 (CF-EP)	0.0079	45
9	IM7/8552 (CF-EP)	0.0079	0
10	IM7/8552 (CF-EP)	0.0079	45
11	IM7/8552 (CF-EP)	0.0079	45
12	IM7/8552 (CF-EP)	0.0079	45

Table 2. Crown Skin Original Inner Laminate Ply Stack-up.

PLY #	Material	Nominal Thickness	Angle
E.P. 1	E7781/EA9396 (FG-EP)	0.0085	0
R.P. 13	M20/40%/G904 (CF-EP)	0.0083	45
R.P. 12	M20/40%/G904 (CF-EP)	0.0083	45
R.P. 11	M20/40%/G904 (CF-EP)	0.0083	45
R.P. 10	M20/40%/G904 (CF-EP)	0.0083	0
R.P. 9	M20/40%/G904 (CF-EP)	0.0083	45
R.P. 8	M20/40%/G904 (CF-EP)	0.0083	45
R.P. 7	M20/40%/G904 (CF-EP)	0.0083	0
R.P. 6	M20/40%/G904 (CF-EP)	0.0083	45
R.P. 5	M20/40%/G904 (CF-EP)	0.0083	45
R.P. 4	M20/40%/G904 (CF-EP)	0.0083	0
R.P. 3	M20/40%/G904 (CF-EP)	0.0083	45
R.P. 2	M20/40%/G904 (CF-EP)	0.0083	45
R.P. 1	M20/40%/G904 (CF-EP)	0.0083	45

Table 3. Repair Laminate Ply Stack-up.

Classical laminate plate theory (CLPT) is used to analyze this structural repair case study. This method is deemed acceptable for this configuration for two reasons; 1) The thickness of the original base laminate structure (0.0948 inch) and repair (0.1164 inch) laminate faces are less than 0.300 inch, as identified in Reference 14, page 69. Additionally, the number of plies utilized is within the number of laminate plies deemed acceptable for thin plate theory as provided in reference 7 (Vol, 3, Chapter 8, section 8.3.1), and 2) the ratio of laminate thicknesses to panel

width (approx. 20 inches) is sufficiently low (i.e., 0.00474 and 0.00582 respectively) to neglect shear stresses through laminate thicknesses. Additionally, there are no tension, compression, or bending forces that are applied to the laminate face; this analysis neglects any aerodynamic forces on skin panel as the repair is located on the inner surface. Using the material properties provided in Table 1 and equation 1 (references 10 and 14) provided below, the components of the Reduced Stiffness Matrix [Q] are first obtained for each of the ply materials in the local (i.e. lamina) 1-2 direction.

$$\begin{aligned} Q_{11} &= \frac{E_1}{1-\nu_{12}\nu_{21}} & Q_{12} &= Q_{21} = \frac{\nu_{12}E_2}{1-\nu_{12}\nu_{21}} \\ Q_{22} &= \frac{E_2}{1-\nu_{12}\nu_{21}} & Q_{33} &= E_3 \\ Q_{13} &= Q_{23} = Q_{31} = Q_{32} = 0 \end{aligned} \quad (1)$$

Resulting in the following reduced stiffness values for each material in the lamina coordinate system:

$$\begin{aligned} [Q_{0/90}]_{IM7/8552} &= \begin{bmatrix} 9.259 & 0.296 & 0 \\ 0.296 & 9.259 & 0 \\ 0 & 0 & 0.520 \end{bmatrix} \text{MSI} \\ [Q_{0/90}]_{M20/40\%/G904} &= \begin{bmatrix} 8.664 & 0.498 & 0 \\ 0.498 & 8.434 & 0 \\ 0 & 0 & 0.713 \end{bmatrix} \text{MSI} \\ [Q_{0/90}]_{E7781/EA9396} &= \begin{bmatrix} 3.465 & 0.395 & 0 \\ 0.395 & 3.435 & 0 \\ 0 & 0 & 0.758 \end{bmatrix} \text{MSI} \end{aligned} \quad (2)$$

The subscripts used designate the angle orientation of the material Reduced Stiffness Matrix and the specific material identification. The lamina Reduced Stiffness Matrix values obtained in equation 2 are then rotated from the 0/90-degree axis into the +/-45-degree orientation, to accommodate the checking of off-axis loading. Using equation 3 below, the resulting rotated Reduced Stiffness Matrix values in the lamina coordinate system are obtained. Equation (3), transformation matrix [T], and Reuter's matrix [R] can be found in references 10 and 14.

$$[Q_{45}] = [T_{45}]^{-1}[Q_0][R][T_0][R]^{-1} \quad (3)$$

Note that the fiberglass/epoxy ply layer remains in the 0/90-degree lamina orientation during repair application, therefore it is unnecessary to rotate into the laminate (i.e. global) coordinate system, for ease of calculations.

$$\begin{aligned}
 [Q_{+/-45}]_{IM7/8552} &= \begin{bmatrix} 5.298 & 4.258 & 0.000 \\ 4.258 & 5.298 & 0.000 \\ 0.000 & 0.000 & 4.482 \end{bmatrix} \text{MSI} \\
 [Q_{+/-45}]_{M20/40\%/G904} &= \begin{bmatrix} 5.236 & 3.810 & 0.058 \\ 3.810 & 5.236 & 0.058 \\ 0.058 & 0.058 & 4.026 \end{bmatrix} \text{MSI} \\
 [Q_{0/90}]_{E7781/EA9396} &= \begin{bmatrix} 3.465 & 0.395 & 0 \\ 0.395 & 3.435 & 0 \\ 0 & 0 & 0.758 \end{bmatrix} \text{MSI}
 \end{aligned} \tag{4}$$

Using the ply Reduced Stiffness Matrices $[Q]$, nominal ply thicknesses, stack-up sequences (Tables 2 & 3), and equation 5 (references 10 and 14), the full Laminate Stiffness Properties $[ABD]$ are then calculated. Detailed analysis and methodology used on this case study are provided in Appendix A.

$$\begin{aligned}
 [A] &= \sum_{i=1}^m [\bar{Q}]_i \cdot (z_i - z_{i-1}) \\
 [B] &= \frac{1}{2} \sum_{i=1}^m [\bar{Q}]_i \cdot (z_i^2 - z_{i-1}^2) \\
 [D] &= \frac{1}{3} \sum_{i=1}^m [\bar{Q}]_i \cdot (z_i^3 - z_{i-1}^3)
 \end{aligned} \tag{5}$$

Calculating the above Laminate Stiffness Matrices $[A]$, $[B]$, and $[D]$ gives the overarching elastic properties $[ABD]$ of the laminate structures in the global coordinate system; where $[A]$ is for in-plane stiffness, $[B]$ is for the coupling of the in-plane and out of plane stiffnesses, and $[D]$ is for bending and twisting stiffness relations. The resulting $[ABD]$ matrices, for the original and repair structure, in the rotated and non-rotated matrices are provided of the form:

Original Structure, Non-Rotated:

$$[ABD]_{\text{orig } (0/90)} = \begin{bmatrix} 0.5961 & 0.3098 & 0.0000 & -0.0004 & 0.0004 & 0.0000 \\ 0.3098 & 0.5961 & 0.0000 & 0.0004 & -0.0004 & 0.0000 \\ 0.0000 & 0.0000 & 0.3310 & 0.0000 & 0.0000 & 0.0004 \\ -0.0004 & 0.0004 & 0.0000 & 0.0004 & 0.0003 & 0.0000 \\ 0.0004 & -0.0004 & 0.0000 & 0.0003 & 0.0004 & 0.0000 \\ 0.0000 & 0.0000 & 0.0004 & 0.0000 & 0.0000 & 0.0003 \end{bmatrix} \tag{6}$$

Original Structure, Rotated:

$$[ABD]_{\text{orig } (+/-45)} = \begin{bmatrix} 0.7839 & 0.1220 & 0.0000 & 0.0004 & -0.0004 & 0.0000 \\ 0.1220 & 0.7839 & 0.0000 & -0.0004 & 0.0004 & 0.0000 \\ 0.0000 & 0.0000 & 0.1432 & 0.0000 & 0.0000 & -0.0004 \\ 0.0004 & -0.0004 & 0.0000 & 0.0006 & 0.0001 & 0.0000 \\ -0.0004 & 0.0004 & 0.0000 & 0.0001 & 0.0006 & 0.0000 \\ 0.0000 & 0.0000 & -0.0004 & 0.0000 & 0.0000 & 0.0001 \end{bmatrix} \tag{7}$$

Repair Structure, Non-Rotated:

$$[ABD]_{\text{repair } (0/90)} = \begin{bmatrix} 0.6798 & 0.3320 & 0.0048 & 0.0012 & 0.0012 & 0.0000 \\ 0.3320 & 0.6738 & 0.0048 & 0.0012 & 0.0012 & 0.0000 \\ 0.0048 & 0.0048 & 0.3583 & 0.0000 & 0.0000 & 0.0011 \\ 0.0012 & 0.0012 & 0.0000 & 0.0007 & 0.0004 & 0.0000 \\ 0.0012 & 0.0012 & 0.0000 & 0.0004 & 0.0007 & 0.0000 \\ 0.0000 & 0.0000 & 0.0011 & 0.0000 & 0.0000 & 0.0004 \end{bmatrix} \quad (8)$$

Repair Structure, Rotated:

$$[ABD]_{\text{repair } (+/-45)} = \begin{bmatrix} 0.8532 & 0.1461 & 0.0015 & 0.0023 & 0.0000 & 0.0000 \\ 0.1461 & 0.8723 & 0.0015 & 0.0000 & 0.0024 & 0.0000 \\ 0.0015 & 0.0015 & 0.1724 & 0.0000 & 0.0000 & 0.0000 \\ 0.0023 & 0.0000 & 0.0000 & 0.0009 & 0.0001 & 0.0000 \\ 0.0000 & 0.0024 & 0.0000 & 0.0001 & 0.0010 & 0.0000 \\ 0.0000 & 0.0000 & 0.0000 & 0.0000 & 0.0000 & 0.0001 \end{bmatrix} \quad (9)$$

The above $[ABD]$ matrices (equations 6 thru 9) are identified as the laminate stiffness values for the respective orientation and laminate. The Laminate Compliance Matrix $[abd]$ values are then obtained by taking the inverse of the $[ABD]$ matrix, shown in equation 10 (references 10 and 14).

$$[abd] = [ABD]^{-1} = \begin{bmatrix} a_{11} & a_{12} & a_{13} & b_{11} & b_{12} & b_{13} \\ a_{21} & a_{22} & a_{23} & b_{21} & b_{22} & b_{23} \\ a_{31} & a_{32} & a_{33} & b_{31} & b_{32} & b_{33} \\ b_{11} & b_{12} & b_{13} & d_{11} & d_{12} & d_{13} \\ b_{21} & b_{22} & b_{23} & d_{21} & d_{22} & d_{23} \\ b_{31} & b_{32} & b_{33} & d_{31} & d_{32} & d_{33} \end{bmatrix} \quad (10)$$

The resulting Laminate Compliance Matrices $[abd]$ for each component and orientation can be found in Appendix A. From the Laminate Compliance Matrix, the actual laminate stiffness and flexural values can be obtained and compared for structural adequacy with the calculations shown in Table 4. This is performed on both the non-rotated and rotated laminates.

Axial Stiffness Modulus	Flexural Stiffness Modulus
$E_X^A = 1/(a_{11})$	$E_X^F I_{yy} = 1/d_{11}$
$E_Y^A = 1/(a_{22})$	$E_Y^F I_{yy} = 1/d_{22}$
$E_{XY}^A = 1/(a_{33})$	$E_{XY}^F I_{yy} = 1/d_{33}$

Table 4. Actual Laminate Stiffness Modulus Calculation.

The corresponding comparison of stiffness values are then taken as the ratio of the repair stiffness to the original base laminate stiffness. These calculations (equation 11) are summarized and shown in Table 5.

$$\text{Stiffness Ratio} = \frac{E_{\text{Repair}}}{E_{\text{Original}}} \cdot (100) \quad (11)$$

Parameter		Original Laminate	Repair Laminate	Stiffness Ratio
Non-Rotated Axial Stiffness (10 ⁶ lbf/in)	E _X ^A	0.4352	0.5162	118.6%
	E _Y ^A	0.4352	0.5116	117.6%
	E _{XY} ^A	0.3310	0.3583	108.3%
Rotated Axial Stiffness (10 ⁶ lbf/in)	E _X ^A	0.7649	0.8287	108.3%
	E _Y ^A	0.7649	0.8473	110.8%
	E _{XY} ^A	0.1432	0.1724	120.4%
Non-Rotated Flexural Stiffness (10 ⁶ in-lbf)	E _X ^F	0.0002405	0.0004681	194.7%
	E _Y ^F	0.0002405	0.0004659	193.7%
	E _{XY} ^F	0.0002807	0.0004083	145.5%
Rotated Flexural Stiffness (10 ⁶ in-lbf)	E _X ^F	0.0006143	0.0009097	148.1%
	E _Y ^F	0.0006143	0.0009311	151.6%
	E _{XY} ^F	0.0000731	0.0001500	205.2%

Table 5. Original vs. Repair Laminate Stiffness Comparison.

As shown in the stiffness ratio column of Table 5, the repair configuration meets and exceeds the structural capacity of the original laminate. This demonstrates that the in-plane and flexural stiffnesses of the repair laminate exceed the original structure. However, this does not complete structural substantiation of the repair. It is necessary to determine the maximum load carrying strength capability of each laminate in each direction. This is achieved by determining the corresponding running load [N] necessary to achieve maximum strain-to-failure extension of the composite ply lamina. Table 6 provides the warp and fill strain allowables for the original and repair lamina (references 5, 6, 12, and 18). For this study, the lowest strain allowables between M20/40%/G904 and E7781/EA9396 were used, as it is expected to fail first. Table 6 provides the strain-to-failure values for each material.

ORIGINAL STRUCTURAL PROPERTIES	
Material Properties, IM7/8552	Strain Values
Warp Tension Strain Allowable, ϵ_1^{tu}	0.0113
Fill Tension Strain Allowable, ϵ_2^{tu}	0.0113
Warp Compression Strain Allowable, ϵ_1^{cu}	0.0101
Fill Compression Strain Allowable, ϵ_2^{cu}	0.0101
REPAIR MATERIAL STRUCTURAL PROPERTIES	
Material Properties, M20/40%/G904	Strain Values
Warp Tension Strain Allowable, ϵ_1^{tu}	0.0136
Fill Tension Strain Allowable, ϵ_2^{tu}	0.0125
Warp Compression Strain Allowable, ϵ_1^{cu}	0.0136*
Fill Compression Strain Allowable, ϵ_2^{cu}	0.0125*
Material Properties, E7781/EA9396	Strain Values
Warp Tension Strain Allowable, ϵ_1^{tu}	0.0177
Fill Tension Strain Allowable, ϵ_2^{tu}	0.0182
Warp Compression Strain Allowable, ϵ_1^{cu}	0.0147
Fill Compression Strain Allowable, ϵ_2^{cu}	0.0119
*No data available. Values duplicated for convenience for this study.	

Table 6. Lamina Strain-to-Failure Values.

For this analysis, the lowest allowable strain values between the carbon fiber pre-preg and fiberglass wet layup layers were used. This is assumed as fiber failures would lead to disbonding of the laminate and, if observed during an aircraft inspection cycle, would lead to significant follow-on maintenance. Using the strain values (i.e. carbon fiber pre-preg) provided in Table 6, and Hooks' Law (where $[a *_{ij}]$ is the normalized reduced laminate compliance matrix), the stress values necessary to achieve the allowable maximum strains can be calculated for each orientation.

$$[a *_{ij}] = ((1/\Sigma_t) \cdot [A_{ij}])^{-1} \quad (12)$$

$$[\epsilon_{ij}] = [a *_{ij}][\sigma_{ij}] \quad (13)$$

Once stress values are obtained for each orientation, in tension and compression, it is then multiplied by the thickness of the laminate structure to determine the corresponding running load [N] capability through the laminate. This is shown in equation 14 (references 10 and 14).

$$[N_i] = t \cdot [\sigma_i] \cdot (10^6) \quad (14)$$

Note that the running load is in units of lbf/inch. Figure 17 provides the corresponding calculation necessary to achieve the running loads through the original and repair laminates. The corresponding structural capability ratios (equation 15) are then taken as the ratio of the repair load capability to the original base laminate capability. Table 7 provides the resulting calculations.

$$\text{Structural Capability Ratio} = \frac{N_{\text{Repair}}}{N_{\text{Original}}} \cdot (100\%) \quad (15)$$

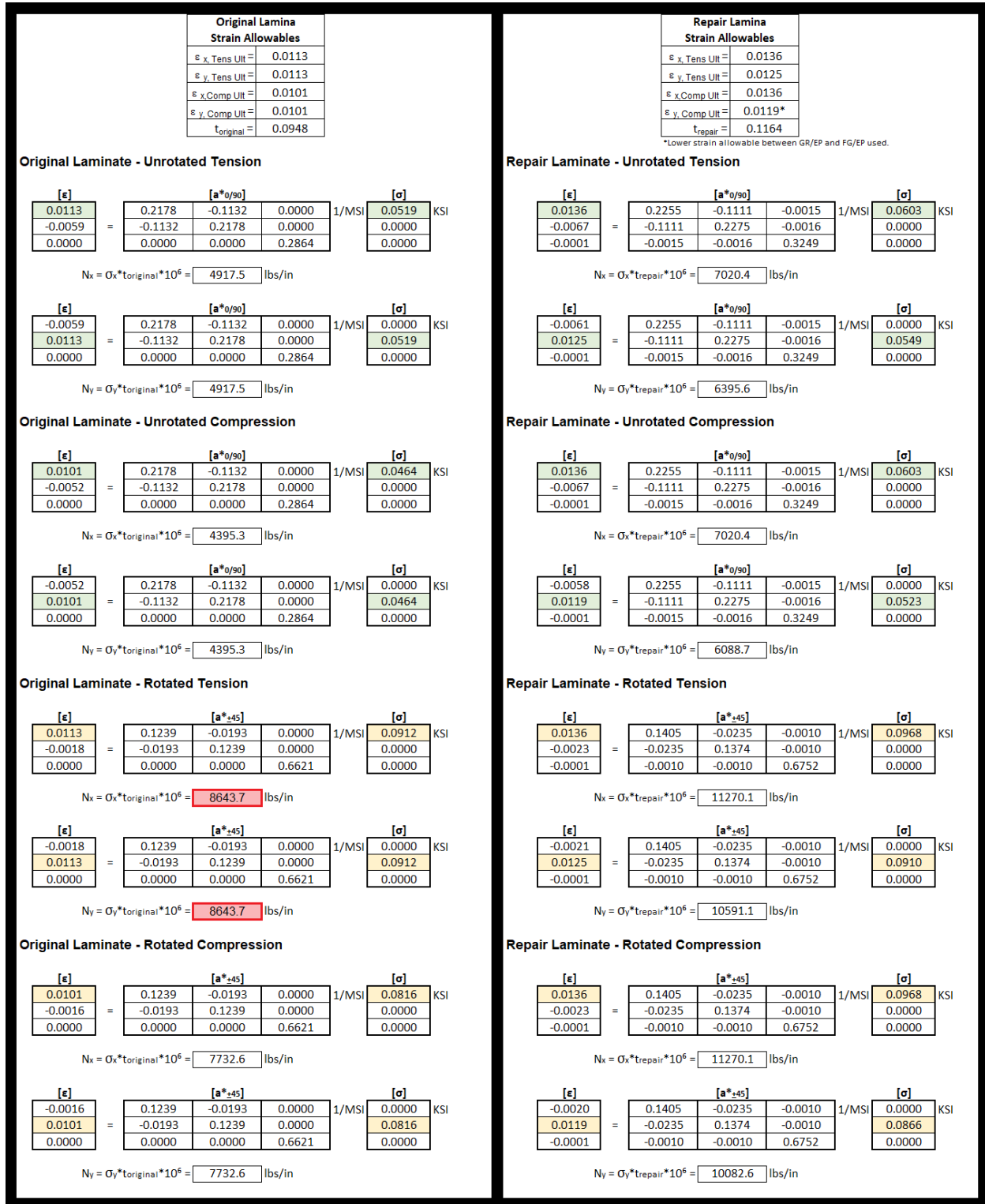


Figure 17. Running Load Calculations.

Parameter	Running Load, N		Structural Capability Ratio
	Original Laminate (lbf/in)	Repair Laminate (lbf/in)	
Non-rotated Tension	4,917.5	7,020.4	142.8%
	4,917.5	6,395.6	130.1%
Non-rotated Compression	4,395.3	7,020.4	159.7%
	4,395.3	6,088.7	138.5%
Rotated Tension	8,643.7	11,270.1	130.4%
	8,643.7	10,591.1	122.5%
Rotated Compression	7,732.6	11,270.1	145.7%
	7,732.6	10,082.6	130.4%

Table 7. Original vs. Repair Running Load Comparison.

As shown in the structural capability ratio column in Table 7, the repair is considered structurally adequate from a load carrying capability for the load orientations considered. This establishes that the capability of the repair laminate exceeds the capability of the original crown skin laminate and is therefore adequate. However, this still does not complete structural acceptability of the repair. This is because the analysis, up until this point, only determines whether the repair laminate and stacking sequence meet (or exceed) the capability of the original laminate. It does not account for the bonded joint between the two parts. Therefore, an investigation into the load carrying capability of the bonded joint is necessary.

While the repair performed on the subject Chinook crown skin employed a scarf type configuration as shown in Figure 18, this analysis will also consider a typical external bonded scab patch as shown in Figure 19 for comparison. Although the image is representative and not to scale, an additional carbon repair ply and external fiberglass ply were included in this repair and case study. This scarf repair uses a filler ply (included in analysis), and a 1/2-inch overlay per ply for a total scarf length of 6 inches per side. Alternatively, Figure 19 demonstrates the typical one-sided bonded scab patch repair. This configuration is assumed to include the same ply stacking and orientation sequence as the scarf repair but practically truncates the edges of the plies and uses a one inch overlay beyond edge of damage. In both repair cases, the opposite face of the skin panel is considered undamaged, there are no out-of-plane loading effects, and the repair laminate has been stabilized as is it bonded to the core and filler material.

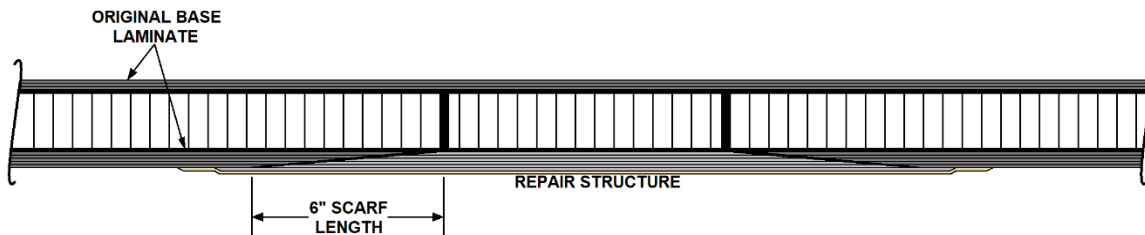


Figure 18. Scarf Repair Configuration.

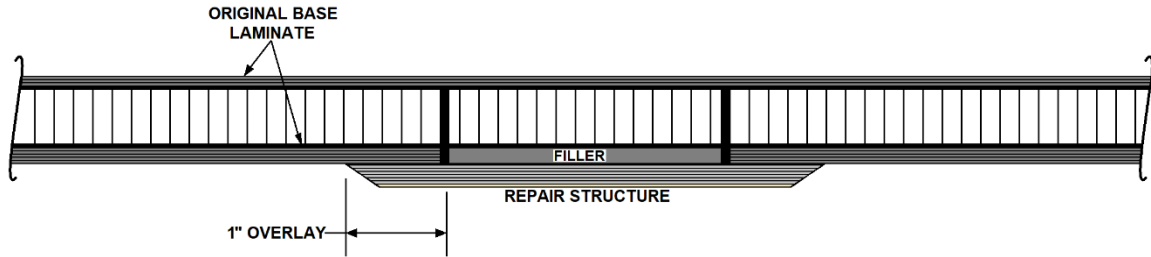


Figure 19. Scab Patch Repair Configuration.

VI. SCARF PATCH REPAIR JOINT ANALYSIS

For this repair, commercially available Cytec FM 300-2M film adhesive was used to co-bond the scarf repair to the original base laminate structure during cure. Material properties for this adhesive are found in the adhesive specification (reference 8) and provided in Table 8 for convenience.

FM300-2 Film Adhesive	
Material Property	Values
Nominal Thickness, t	0.008 INCH
Wide Area Lap Shear (75°F), τ_s	2,510 PSI
Flatwise Tensile Strength (75°F), σ_t	780 PSI
Shear Modulus, G	131,500 PSI
Linear Shear Limit (75°F), γ	0.0156

Table 8. Cytec FM300-2, Adhesive Material Properties.

Kress Scarf Joint Analysis (Reference 14)

The scarf repair (one side shown) and load transfer across the joint is shown in Figure 20. For clarity, the honeycomb core, opposing laminate face, and external repair plies have been removed. This analysis method only considers the length of the joint common to the original and repair laminate faces along the scarf angle. It is assumed that running loads are transferred across the joint through adhesive shear (parallel) and flatwise tension (normal) relative to the plane of the scarf face. Additionally, this presumes that if one side of the repair (as shown in Figure 20) is determined as structurally adequate, then the opposing scarf joint is acceptable by comparison.

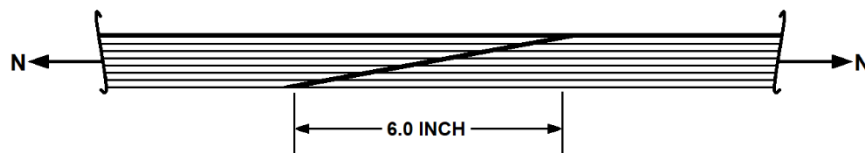


Figure 20. Scarf Repair Overlap Length Detail.

Using methodologies provided by G. Kress (reference 14), it is first necessary to determine the scarf length, taper ratio, and scarf angles. This is achieved through the product of the number of

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repair plies and the individual ply overlap length. The repair in this case study uses twelve repair plies with a ½ inch overlap per ply. Using these characteristics, the following values are obtained:

$$\text{Scarf Length } (L_S) = (\# \text{ of Plies}) (\text{Overlap Length}) = (12) (0.5 \text{ inch}) = 6 \text{ inches} \quad (16)$$

$$\text{Tapper Ratio: } TR_S = \frac{L_S}{t_{\text{laminate}}} = \frac{6 \text{ inch}}{0.0948 \text{ inch}} = 63.29:1 \quad (17)$$

$$\text{Scarf Angle: } \theta_S = \text{atan}\left(\frac{1}{TR_S}\right) = \text{atan}\left(\frac{1}{63.29}\right) = 0.905 \text{ degrees} \quad (18)$$

Confirmation of the necessary scarf ratio is performed by using the scarf ply criteria provided in TM 1-1500-204-23-11 (reference 17). Using the corresponding ply thicknesses (i.e., 0.0079 & 0.0083 inch) and ½ inch overlap criteria, an approximate scarf ratio of 63-1 was obtained; as shown in Figure 21.

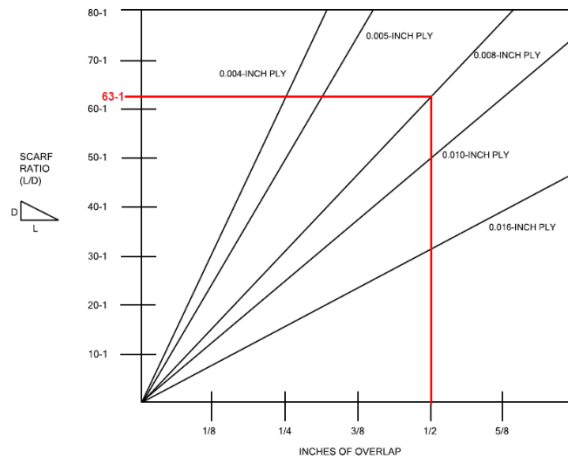


Figure 21. Ply Overlap Chart.

The methods described in reference 14 are derived from a summation of forces relative to the plane of the scarf bondline. Resolving the two equations for the resultant running loads, and using the adhesive material properties (Table 8) for shear and normal stresses, we obtain the following ultimate strength properties:

Ultimate Adhesive Shear Strength:

$$q_{\text{shear_ult}} = \frac{2 \cdot \tau_s \cdot t}{\sin(2\theta)} = \frac{2(2,510 \text{ psi})(0.0948 \text{ inch})}{\sin(2 \cdot 0.905)} = 15,067.1 \text{ lbf/in} \quad (19)$$

Ultimate Adhesive Normal (Tension) Strength:

$$q_{\text{tension_ult}} = \frac{\sigma \cdot t}{\sin^2(2\theta)} = \frac{(780 \text{ psi})(0.0948 \text{ inch})}{\sin^2(2 \cdot 0.905)} = 74,120 \text{ lbf/in} \quad (20)$$

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Using the maximum allowable running load, $N_{\text{Original}} = 8,643.7 \text{ lbf/in}$, calculated in Figure 17, we can then determine the corresponding Margin of Safety (M.S.) values for each loading state. Within this equation, a joint design factor of $k=1.15$ (reference 16) is included for use on military aircraft.

Adhesive Shear Strength Check:

$$M.S._{\text{shear}} = \frac{q_{\text{shear_ult}}}{k \cdot N_{\text{Original}}} - 1 = \frac{15,067.1 \text{ lbf/in}}{(1.15)8,643.7 \text{ lbf/in}} - 1 = +0.515 \quad (21)$$

Adhesive Normal (Tension) Check:

$$M.S._{\text{normal}} = \frac{q_{\text{tension_ult}}}{k \cdot N_{\text{Original}}} - 1 = \frac{74,120 \text{ lbf/in}}{(1.15)8,643.7 \text{ lbf/in}} - 1 = +6.456 \quad (22)$$

The resulting M.S. values indicate that the bondline joint will satisfactorily sustain the maximum applied load without failure.

Esp Scarf Joint Analysis (Reference 10)

For additional analysis and comparison, the bondline methodology provided by B. Esp (reference 10) was also used. This method assumes that the adherends are of relatively equal stiffnesses (less than 4 percent difference in this case study), are for an idealized scarf joint, and stresses are uniform throughout the adhesive layer. First, the uniform stress through the laminate is calculated using the maximum applied running load (Figure 17) and laminate thickness as follows:

$$\sigma = \frac{P}{w \cdot t} = \frac{N}{t} = \frac{8,643.7 \text{ lbf/in}}{0.0948 \text{ inch}} = 91,179 \text{ psi} \quad (23)$$

Using this applied stress value and equations 24 and 45 (reference 10), the shear and peel stresses in the adhesive are derived:

Adhesive Shear Stress:

$$\tau_a = \sigma \sin(\theta) \cos(\theta) = (91,178.3 \text{ psi}) \sin(0.905) \cos(0.905) = 1,440 \text{ psi} \quad (24)$$

Adhesive Peel Stress:

$$\sigma_{\text{normal}} = \sigma \cdot \sin(\theta)^2 = (91,178.3 \text{ psi}) \sin(0.905)^2 = 22.7 \text{ psi} \quad (25)$$

From the allowable adhesive properties provided in Table 8, we can then determine the corresponding M.S. values similar to the methods above. Additionally, a joint design fitting factor of $k=1.15$ (reference 16) is used.

Adhesive Shear Strength Check:

$$M.S._{\text{shear}} = \frac{\tau_s}{(1.15) \cdot \tau_a} - 1 = \frac{2,510 \text{ psi}}{(1.15)1,440 \text{ psi}} - 1 = +0.515 \quad (26)$$

Adhesive Normal (Tension) Check:

$$M.S._{\text{normal}} = \frac{\sigma_t}{(1.15) \cdot \sigma_{\text{normal}}} - 1 = \frac{780 \text{ psi}}{(1.15)22.7 \text{ psi}} - 1 = +\text{HIGH} \quad (27)$$

Based on the two methods, one using running (traction) loads and the other using laminate stress values, relatively similar M.S. values were achieved. In both cases, the adhesive film layer was determined to adequately transfer loads from the original base laminate to the repair laminate through shear and flatwise (normal) tension. Both analysis methods achieved the same M.S. values in shear, but also calculated relatively high M.S. values in tension.

However, this analysis further assumes that the two adherend laminate faces are uniformly tapered and allow for a constant adhesive thickness through the length of the bondline. Unless manufacturing a pre-cured tapered repair patch and subsequently bonding in place, we can reasonably assume that the face of repair plies results in a non-uniformity when laid up; at the edges of each repair ply. This non-uniform feature may result in peaking stresses at the edges of the repair plies and overall lower M.S. values.

VII. SCAB PATCH REPAIR JOINT ANALYSIS

To analyze the scab patch overply repair bondline (one side of repair), the Volkersen Shear Lag Theory (reference 10) method was used. This method calculates elastic bondline shear stresses for single lap joints of uniform adhesive thickness and varying adherend stiffness and thicknesses values. For this type of bondline, the peak (or maximum) shear stresses are located near the endpoints while lower shear stresses are seen towards the central position of the adhesive bondline. Essentially, the adhesive shear stress values are calculated as a function of its linear position along the overlap.

For this analysis, the stacking sequence and ply orientations of the original base (Table 2) and repair (Table 3) laminates remained unchanged. Meaning the actual engineering stiffness values (i.e., $[E_{ij}] = [1/a_{ij}^*]$) of the laminates were used as the stiffness values of the inner and outer adherends respectively. While both global laminate orientations (0/90° and +/-45°) and material properties were checked to determine peak shear stresses, it was determined that the 0/90° global orientation produced maximum stress values. Therefore, this is the method and material properties that were used; see Table 9 below for material properties used.

Property	Reference	Values
Inner Adherend Stiffness, E_i	Appendix A	4,590,447 PSI
Inner Adherend Thickness, t_i	Figure 17	0.094 INCH
Outer Adherend Stiffness, E_o	Appendix A	4,434,778 PSI
Outer Adherend Thickness, t_o	Figure 17	0.116 INCH
Adhesive Layer Thickness, t	Table 8	0.008 INCH
Adhesive Shear Modulus, G	Table 8	131,500 PSI
Width of Joint, w	Table 8	1.00 INCH
Maximum Applied Load, N	Figure 17	8,643.7 LBF/IN

Table 9. Volkersen Analysis Values.

Figure 22 provides the repair configuration of the scab patch, along with the respective material properties of the structure. It is important to point out that since both laminates are adhesively bonded to the honeycomb core, it is presumed that this precludes any out-of-plane deformations (i.e., rotation) as the structure is loaded. Additionally, the Volkersen method assumes idealized laminate and adhesive termination on the overply. However, legacy scab patch repairs and adhesive flows during curing result in material build-up at edges of the repair. This fillet feature is usually allowed to remain in place as it helps to transition the repair into the original structure, provides for environmental protection, and may be subsequently sanded to achieve aerodynamic profiles. For this analysis, the ply taper and presence of additional adhesive material at repair edges are neglected.

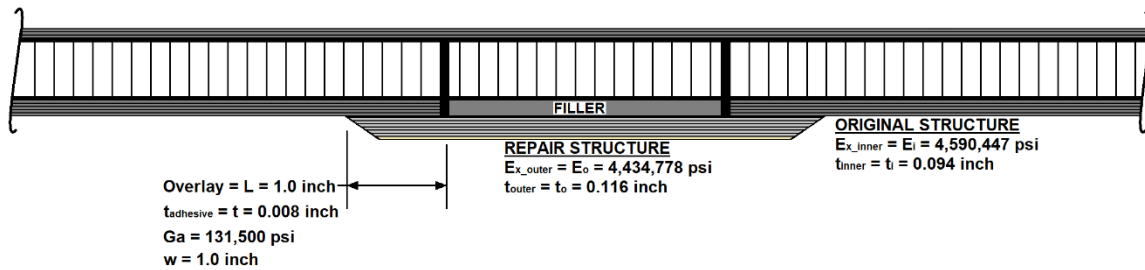


Figure 22. Scab Patch Repair Configuration and Properties.

As this method is for a single bonded overlay, and both sides of the repair patch are symmetrical, analysis of one side of the repair is considered sufficient for the bonded scab patch. This overlay is shown in Figure 23, where the honeycomb core and filler material have been removed for clarity. The shear stresses in the adhesive are a result of its position within the length of the overlap, which is provided by equations 28 and 29 (reference 10).

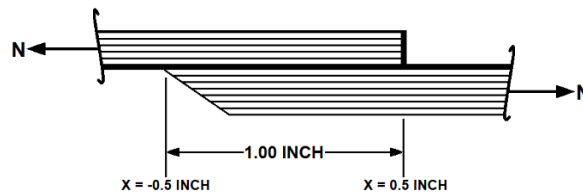


Figure 23. Scab Patch Repair Overlap Length Detail.

$$\lambda^2 = \frac{G}{t} \left(\frac{1}{E_i t_i} + \frac{1}{E_o t_o} \right) \quad (28)$$

$$\tau_a(x) = \frac{(P/w)\lambda}{2 \cdot \sinh(\lambda \cdot L/2)} \cosh(\lambda x) + \frac{(P/w)\lambda}{2 \cdot \cosh(\lambda \cdot L/2)} \sinh(\lambda \cdot x) \left[\frac{E_i t_i - E_o t_o}{E_i t_i + E_o t_o} \right] \quad (29)$$

After incorporating all related material properties associated with this repair, the resulting functions that define this bondline case are achieved:

$$\lambda^2 = 69.65$$

$$\tau_a(x) = 1,111.9 \cosh(8.35x) - 91.6 \sinh(8.35x), \text{ where } -\frac{L}{2} < x < \frac{L}{2} \quad (30)$$

Using equation 30 and the overlap length (Figure 23), the shear stress profile is plotted and peak stresses at both ends of bondline can be calculated. As shown in Chart 1, the peak shear stress ($\tau_{\max} = 39,056$ psi) is located common to the original base laminate structure and edge of the repair patch. This is expected as the original base laminate structure will deform faster than the repair structure, effectively increasing the strain near the edge.

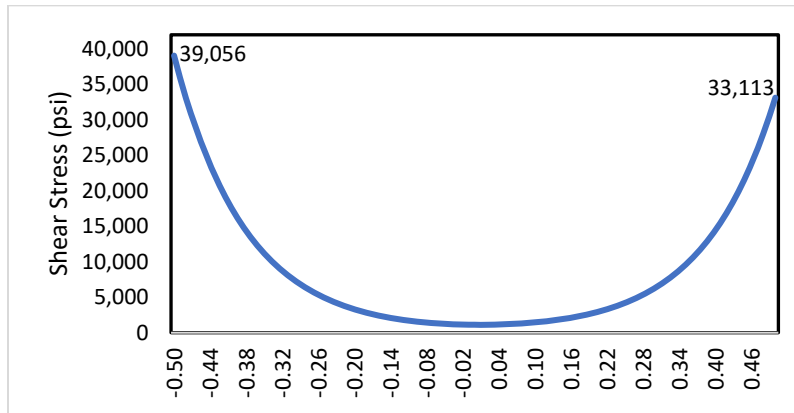


Chart 1. Bondline Elastic Shear Stress.

While the peak shear stresses calculated in Chart 1 exceed the elastic-plastic transition for this adhesive, this load case was derived from maximum ultimate fiber strain values. The subject structure is not expected to see true adhesive plastic deformation while in operation and shear stresses in the adhesive are expected to be much lower during normal flight operations. In addition, increasing the overlap length of the scab patch repair does not significantly reduce the peak stresses that build up in the adhesive. To verify this, the overlap length, used in equation 29, was subsequently increased to two inches; the resulting peak shear stresses in the adhesive were reduced by less than one percent. Also, while increasing the thickness of the adhesive bondline layer did reduce the shear stresses, it did not reduce the peak stresses enough to be below the elastic-plastic threshold.

Additionally, plastic deformation of the adhesive under ultimate loading is not necessarily cause for rejection of the repair, provided operational loads result in elastic deformations only.

High fidelity loads data for repair locations would allow for analysis based on required capability rather than qualification by similarity, especially for locations subjected to oscillatory flight loads. Composite repairs with critical adhesive bond lines would include representative fatigue and static loading test data coverage as part of the demonstration of repair limits, which can be found in the Army Military Airworthiness Certification Criteria (AMACC, reference 2). Section 5.4.2.1.4 of AMACC specifically discusses Structural Substantiation of Damage Repair Limits and Processes.

However, if the adhesive experiences plastic deformation under ultimate loads, this would be considered identifiable and would be captured through the frequent aircraft inspection intervals and evaluated by SRD engineering for acceptability (via MEC). Furthermore, due to the resulting shear peak stresses and stress concentrations that develop at the edges of the scab patch, this method of repair is recommended to be avoided on structural composite components. Due to these issues, alternate repair configurations such as the upside-down wedding cake, stepped scarf, or pre-cured scab patches with tapered edges are recommended. These type of repair configurations may provide lower stress concentration development and gradual load transfer. However, further analysis would be required on subsequent efforts to determine structural acceptability.

VIII. CONCLUSION

The use of advanced composite structures in U.S. Army aircraft designs is likely to drastically increase to gain critical capabilities in areas such as payload, range, and speed through structural weight optimization. Currently fielded U.S. Army aircraft have some composite applications which may be leveraged to further develop organic maintenance and engineering approaches to improve sustainment of enduring fleet aircraft while simultaneously posturing for success with future aircraft systems. The U.S. Army's aircraft structures are subjected to extreme wear and tear through service in austere combat environments and adverse mission profiles. The variety of aircraft types and original equipment manufacturers requires a structural repair analysis approach which relies primarily on an original structure equivalent comparison methodology. The case study presented in this work provides a strong initial foundation to a repair engineering approach through CLPT, FEA, and bondline based static substantiation of the repair.

As identified in this case study, CLPT demonstrated a statically acceptable scarf repair configuration by comparing the repair laminate capabilities to the original aircraft structure. This equivalent strength approach confirmed that the stiffness and load carrying capabilities of the repair laminate met and, in some cases, exceeded the capabilities of the original laminate. Additionally, two separate scarf bond line analysis approaches were used to approximate the load carrying capability of the adhesive joint. In both cases the bondline was deemed acceptable for the maximum allowable running loads calculated. However, when analyzing the bondline joint of the scab patch repair using the same applied loads, it was determined that the adhesive would exceed the elastic-plastic threshold of the material. Additionally, higher shear stresses in the adhesive correlated to the development of significant stress concentrations at the edges of the repair patch. While this may not be cause for rejection of the repair, scab patches are usually avoided as they may cause follow-on maintenance actions in the field and, if failed, adversely affect the adjacent structure beyond the point of serviceability. This is not the case with the scarf repair, which gradually transfers load across the joint and prevents any out of plane rotation of the structure due to tension, compression, and shear loads. It is for these reasons shown that scarf type repairs are recommended over scab batch repairs, especially for primary structural elements.

Further developments are required on subsequent project efforts to develop this composite

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repair engineering case study. To substantiate the repair of an advanced composite primary structure multiple enhancements are needed to the initial approach explained here. These include fatigue analysis, mechanical testing to verify analysis, additional analysis using Open Hole Tension (OHT), Closed Hole Tension (CHT), Elevated Temperature Wet (ETW), Cold Temperature Dry (CTD), and Elevated Temperature Dry (ETD) material allowables. Additionally, hygrothermal analysis and the investigation of failure theories are required to provide a comprehensive approach.

The U.S. Army Aviation enterprise will continue to evolve advanced composite repair engineering and maintenance techniques through studies and repair verifications like this one. These initiatives will develop the workforce and inform investments to ensure capabilities are available to meet the sustainment mission for years to come. To that end, it is key to continue development of these methods through subsequent works and application to other aircraft types and components. This will promulgate advanced composite repair excellence to U.S. Army Aviation stakeholders to better support the enduring fleet and future aircraft acquisitions. Collaborating on this subject within the Department of Defense and with industry, academia, and other governments is key to developing advanced composite repair engineering to support the warfighter.

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LIST OF ABBREVIATIONS, ACRONYMS, AND SYMBOLS

ACCRONYMS

ACAP	Advanced Composite Airframe Program
AFMD	AMCOM Field Maintenance Directorate
AMACC	Army Military Airworthiness Certification Criteria
AMCOM	U.S. Army Aviation and Missile Command
AR	Army Regulations
AvMC	Aviation & Missile Center
CF-EP	Carbon-Fiber Epoxy
CHT	Closed Hole Tension
CTD	Cold Temperature Dry
DEVCOM	Combat Capabilities Development Command
CCAD	Corpus Christi Army Depot
CLPT	Classical Laminated Plate Theory
CSI	Critical Safety Item
DMWR	Depot Maintenance Work Requirement
ETD	Elevated Temperature Dry
ETW	Elevated Temperature Wet
EFAB	Extended Forward Avionics Bays
FG-EP	Fiberglass Epoxy
FEA	Finite Element Analysis
IM	Intermediate Modulus
MEC	Maintenance Engineering Call
MEO	Maintenance Engineering Order
M.S.	Margin of Safety
MDS	Mission Design Series
OHT	Open Hole Tension
OEM	Original Equipment Manufacturer
OML	Outer Mold Line
N	Running (Traction) Load (N)
SRD	Systems Readiness Directorate
TASMG	Theater Aviation Sustainment Maintenance Group
TM	Technical Manual
UAS	Unmanned Aircraft System
Val/Ver	Validation and Verification

APPENDIX A

CLASSICAL LAMINATE PLATE THEORY (CLPT) ANALYSIS

Original Base Laminate, [ABD] Matrix

IM7/8552 (K6-70-PW)										Transformed Reduced Stiffness Matrix (Original)									
t= 0.0079 in										[Q] ₁ = 9.259 0.296 0									
E1= 9.250 MSI										0.296 9.259 0									
E2= 9.250 MSI										0 0 0.520									
E12= 0.52 MSI										[T ₄₅] ⁻¹ [Q][R][T ₄₅][R] ⁻¹ = [Q] ₄₅ = 5.298 4.258 0.000									
V12= 0.032										4.258 5.298 0.000									
V21= 0.032										0.000 0.000 4.482									
Q11= 9.259 MSI										[T] ₄₅ = 0.5000002 0.4999998 -1									
Q22= 9.259 MSI										0.4999998 0.5000002 -1									
Q12= 0.296 MSI										-0.5 0.5 3.268E-07									
Q66= 0.520 MSI										[T] ⁻¹ = 0.5000002 0.4999998 -1									
										0.4999998 0.5000002 1									
										0.5 -0.5 3.268E-07									
										[R] = 1 0 0									
										0 1 0									
										0 0 2									
										[R] ⁻¹ = 1 0 0									
										0 1 0									
										0 0 0.5									
										[Q][R][T][R] ⁻¹ = 4.777894 4.7778911 4.4815891									
										4.7778911 4.777894 -4.481589									
										-0.52 0.52 1.699E-07									
										[R][T][R] ⁻¹ = 0.5000002 0.4999998 0.5									
										0.4999998 0.5000002 -0.5									
										-1 1 3.268E-07									
										[T][R] ⁻¹ = 0.5000002 0.4999998 0.5									
										0.4999998 0.5000002 -0.5									
										-0.5 0.5 1.634E-07									

ROTATED AXIAL/FLEXURAL COUPLING									
Laminate Axial/Flexural Coupling Reduced Stiffness Matrix					Normalized Laminate Axial/Flexural Coupling Reduced Stiffness Matrix				
$[B]_{ax}$ =					$[B^*]_{ax}$ =				
in-lbs					MSI				
0.0003709 -0.0003709 0.0000000					0.0825331 -0.0825331 0.0000000				
-0.0003709 0.0003709 0.0000000					-0.0825331 0.0825331 0.0000000				
0.0000000 0.0000000 -0.0003709					0.0000000 0.0000000 -0.0825331				
Laminate Axial/Flexural Coupling Reduced Compliance Matrix					Normalized Laminate Axial/Flexural Coupling Reduced Compliance Matrix				
$[b]_{ax}$ =					$[b^*]_{ax}$ =				
1/in-lbs					1/MSI				
-0.9981581 0.9981587 0.0000053					-0.0044852 0.0044852 0.0000000				
0.9981587 -0.9981587 -0.0000053					0.0044852 -0.0044852 0.0000000				
0.0000053 -0.0000053 35.4470602					0.0000000 0.0000000 0.1592821				
FLEXURAL STIFFNESS, [D]									
thickness	z	ZI-1	(1/8)*6*(n-lb)	Q11	Q22	Q12	Q66	Q16	Q26
1	0.0079	-0.0395	-0.0474	0.000015	5.298	5.298	4.258	4.482	0.000
2	0.0079	-0.0316	-0.0395	0.000010	5.298	5.298	4.258	4.482	0.000
3	0.0079	-0.0237	-0.0316	0.000006	9.259	9.259	0.296	0.520	0.000
4	0.0079	-0.0158	-0.0237	0.000003	5.298	5.298	4.258	4.482	0.000
5	0.0079	-0.0079	-0.0158	0.000001	5.298	5.298	4.258	4.482	0.000
6	0.0079	0.0000	-0.0079	0.000000	9.259	9.259	0.296	0.520	0.000
7	0.0079	0.0079	0.0000	0.000000	5.298	5.298	4.258	4.482	0.000
8	0.0079	0.0158	0.0079	0.000001	5.298	5.298	4.258	4.482	0.000
9	0.0079	0.0237	0.0158	0.000003	9.259	9.259	0.296	0.520	0.000
10	0.0079	0.0316	0.0237	0.000006	5.298	5.298	4.258	4.482	0.000
11	0.0079	0.0395	0.0316	0.000010	5.298	5.298	4.258	4.482	0.000
12	0.0079	0.0474	0.0395	0.000015	5.298	5.298	4.258	4.482	0.000
0.0948				4.132E-04	4.132E-04	2.652E-04	2.811E-04	7.979E-11	-7.979E-11
NON-ROTATED FLEXURAL STIFFNESS, [D]									
Laminate Flexural Reduced Stiffness Matrix					Normalized Laminate Flexural Reduced Stiffness Matrix				
$[D]_{xx}$ =					$[D^*]_{xx}$ =				
in-lbs					MSI				
0.0004132 0.0002652 0.0000000					5.8206022 3.7351829 0.0000011				
0.0002652 0.0004132 0.0000000					3.7351829 5.8206022 -0.0000011				
0.0000000 0.0000000 0.0002811					0.0000011 -0.0000011 3.9588795				
Laminate Flexural Reduced Compliance Matrix					Normalized Laminate Flexural Reduced Compliance Matrix				
$[d]_{xx}$ =					$[d^*]_{xx}$ =				
1/in-lbs					1/MSI				
4158.395253 -2684.421132 -0.001943					0.2952361 -0.1905875 -0.0000001				
-2684.421132 4158.395253 0.001943					-0.1905875 0.2952361 0.0000001				
-0.001943 0.001943 3563.088007					-0.0000001 0.0000001 0.2529708				
Dxx= 0.0002405 10 ⁶ in-lbs					Dxx= 3.3871 MSI				
Dyy= 0.0002405 10 ⁶ in-lbs					Dyy= 3.3871 MSI				
Dxy= 0.0002807 10 ⁶ in-lbs					Dxy= 3.9530 MSI				
v12= 0.6455426					v12= 0.6455				
ROTATED FLEXURAL STIFFNESS, [D]									
Laminate Flexural Reduced Stiffness Matrix					Normalized Laminate Flexural Reduced Stiffness Matrix				
$[D]_{xx}$ =					$[D^*]_{xx}$ =				
in-lbs					MSI				
0.0006203 0.0000581 0.0000000					8.7367739 0.8190150 -0.0000011				
0.0000581 0.0006203 0.0000000					0.8190150 8.7367720 0.0000011				
0.0000000 0.0000000 0.0000740					-0.0000011 0.0000011 1.0427097				
Laminate Flexural Reduced Compliance Matrix					Normalized Laminate Flexural Reduced Compliance Matrix				
$[d]_{xx}$ =					$[d^*]_{xx}$ =				
1/in-lbs					1/MSI				
1627.758772 -153.785269 0.001943					0.1155670 -0.0109184 0.0000001				
-153.785269 1627.759127 -0.001943					-0.0109184 0.1155670 -0.0000001				
0.001943 -0.001943 13685.632769					0.0000001 -0.0000001 0.9716479				
Dxx= 0.0006143 10 ⁶ in-lbs					Dxx= 8.6530 MSI				
Dyy= 0.0006143 10 ⁶ in-lbs					Dyy= 8.6530 MSI				
Dxy= 0.0000731 10 ⁶ in-lbs					Dxy= 1.0292 MSI				
v12= 0.0944767					v12= 0.0945				
[ABD]									
NON-ROTATED									
$[ABD]_{xx/90}$ =					$[ABD]^*_{xx/90}$ =				
in-lbs					MSI				
0.5961 0.3098 0.0000					6.2883 3.2675 0.0000				
0.3098 0.5961 0.0000					3.2675 6.2883 0.0000				
0.0000 0.0000 0.3310					0.0000 0.0000 3.4912				
-0.0003709 0.0003709 0.0000000					-0.2476 0.2476 0.0000				
0.0003709 -0.0003709 0.0000000					0.2476 -0.2476 0.0000				
0.0000000 0.0000000 0.0003709					0.0000 0.0000 0.2476				
$[abd]_{xx/90}$ =					$[abd]^*_{xx/90}$ =				
in-lbs					1/MSI				
2.3209 -1.2170 0.0000					0.2200 -0.1154 0.0000				
-1.2170 2.3209 0.0000					-0.1154 0.2200 0.0000				
0.0000 0.0000 3.0259					0.0000 0.0000 0.2869				
8.8618 -8.8618 0.0000					0.0398 -0.0398 0.0000				
-8.8618 8.8618 0.0000					-0.0398 0.0398 0.0000				
0.0000 0.0000 -3.9926					0.0000 0.0000 -0.0179				
ROTATED									
$[ABD]_{+45}$ =					$[ABD]^*_{+45}$ =				
in-lbs					MSI				
0.7839 0.1220 0.0000					8.2691 1.2867 0.0000				
0.1220 0.7839 0.0000					1.2867 8.2691 0.0000				
0.0000 0.0000 0.1432					0.0000 0.0000 1.5104				
-0.0004 -0.0004 0.0000					0.2476 -0.2476 0.0000				
0.0004 0.0004 0.0000					-0.2476 0.2476 0.0000				
0.0000 0.0000 -0.0004					0.0000 0.0000 -0.2476				
$[abd]_{+45}$ =					$[abd]^*_{+45}$ =				
in-lbs					1/MSI				
1.3084 -0.2045 0.0000					0.1240 -0.0194 0.0000				
-0.2045 1.3084 0.0000					-0.0194 0.1240 0.0000				
0.0000 0.0000 7.0758					0.0000 0.0000 0.6708				
-0.9982 0.9982 0.0000					-0.0045 0.0045 0.0000				
0.9982 -0.9982 0.0000					0.0045 -0.0045 0.0000				
0.0000 0.0000 35.4471					0.0000 0.0000 0.1593				
13685.6328					0.0000 0.0000 0.9716				

Repair Laminate, [ABD] Matrix

M20/4076/10904				E7781/EA996				Transformed Reduced Stiffness Matrix (Repair)				Transformed Reduced Stiffness Matrix (Repair)			
t=	0.0083 in			t=	0.0085 in			[Q] ₁ =	8.664	0.498	0	[Q] ₁ =	3.465	0.395	0
E1=	8.635 MSI			E1=	3.420 MSI				0.498	8.434	0		0.395	3.435	0
E2=	8.405 MSI			E2=	3.390 MSI				0	0	0.713		0	0	0.758
E12=	0.713 MSI			E12=	0.758 MSI			[T ₄₅] ⁻¹ [Q] ₁ [R] ₁ [T ₄₅] ⁻¹ = [Q] ₄₅ =	5.236	3.810	0.058		2.680629	1.164039	0.0076
V12=	0.059			V12=	0.115				3.810	5.236	0.058		1.164039	2.680629	0.007599
V21=	0.057			V21=	0.114				0.058	0.058	4.026		0.0076	0.007599	1.5276
Q11=	8.664 MSI			Q11=	3.465 MSI			[T] ₄₅ =	0.5	0.5	1	[T] ₄₅ =	0.5	0.5	1
Q22=	8.434 MSI			Q22=	3.435 MSI				0.5	0.5	-1		0.5	0.5	-1
Q12=	0.498 MSI			Q12=	0.395 MSI				-0.5	0.5	3.27E-07		-0.5	0.5	3.27E-07
Q66=	0.713 MSI			Q66=	0.758 MSI			[T] ₁₅ =	0.5	0.5	-1	[T] ₁₅ =	0.5	0.5	-1
									0.5	0.5	1		0.5	0.5	1
									0.5	-0.5	3.27E-07		0.5	-0.5	3.27E-07
								[R] ₁ =	1	0	0	[R] ₁ =	1	0	0
									0	1	0		0	1	0
									0	0	2		0	0	2
								[R] ₁₅ =	1	0	0	[R] ₁₅ =	1	0	0
									0	1	0		0	1	0
									0	0	0.5		0	0	0.5
								[Q] ₁ [R] ₁ [T] ₄₅ ⁻¹ =	4.58097	4.580968	4.083388	[Q] ₁ [R] ₁ [T] ₄₅ ⁻¹ =	1.930229	1.930228	1.5352
									4.465577	4.465579	-3.968		1.915029	1.91503	-1.52
									-0.713	0.713	2.33E-07		-0.758	0.758	2.48E-07
								[R] ₁ [T] ₄₅ ⁻¹ =	0.5	0.5	0.5	[R] ₁ [T] ₄₅ ⁻¹ =	0.5	0.5	0.5
									0.5	0.5	-0.5		0.5	0.5	-0.5
									-1	1	3.27E-07		-1	1	3.27E-07
								[T] ₁₅ ⁻¹ =	0.5	0.5	0.5	[T] ₁₅ ⁻¹ =	0.5	0.5	0.5
									0.5	0.5	-0.5		0.5	0.5	-0.5
									-0.5	0.5	1.63E-07		-0.5	0.5	1.63E-07

NON-ROTATED STIFFNESS											
Reduced Laminate Stiffness Matrix						Normalized Reduced Laminate Stiffness Matrix					
[A _{xx}]=	0.6798	0.3320	0.0048	10 ⁶ lb/in		[A _{xx}] _{norm} =	5.8403	2.8522	0.0411	MSI	
	0.3320	0.6738	0.0048				2.8522	5.7887	0.0411		
	0.0048	0.0048	0.3583				0.0411	0.0411	3.0784		
Reduced Laminate Compliance Matrix						Normalized Reduced Laminate Compliance Matrix					
[B _{xx}]=	1.9372	-0.9544	-0.0131	10 ⁶ in/lbf		[B _{xx}] _{norm} =	0.2255	-0.1111	-0.0015	1/MSI	
	-0.9544	1.9545	-0.0134				-0.1111	0.2275	-0.0016		
	-0.0131	-0.0134	2.7911				-0.0015	-0.0016	0.3249		
E _{xx} =	0.5162	10 ⁶ lb/in				E _{xx} =	4.4348	MSI			
E _{yy} =	0.5116	10 ⁶ lb/in				E _{yy} =	4.3956	MSI			
E _{xy} =	0.3583	10 ⁶ lb/in				E _{xy} =	3.0780	MSI			
v ₁₂ =	0.4927					v ₁₂ =	0.4927				

ROTATED STIFFNESS											
Reduced Laminate Stiffness Matrix						Normalized Reduced Laminate Stiffness Matrix					
[A _{xx}]=	0.8532	0.1461	0.0015	10 ⁶ lb/in		[A _{xx}] _{norm} =	7.3395	1.2549	0.0129	MSI	
	0.1461	0.8723	0.0015				1.2549	7.4941	0.0129		
	0.0015	0.0015	0.1724				0.0129	0.0129	1.4811		
Reduced Laminate Compliance Matrix						Normalized Reduced Laminate Compliance Matrix					
[B _{xx}]=	1.2067	-0.2021	-0.0087	10 ⁶ in/lbf		[B _{xx}] _{norm} =	0.1405	-0.0235	-0.0010	1/MSI	
	-0.2021	1.1802	-0.0085				-0.0235	0.1974	-0.0010		
	-0.0087	-0.0085	5.8005				-0.0010	-0.0010	0.6752		
E _{xx} =	0.8287	10 ⁶ lb/in				E _{xx} =	7.11928894	MSI			
E _{yy} =	0.8473	10 ⁶ lb/in				E _{yy} =	7.27913058	MSI			
E _{xy} =	0.1724	10 ⁶ lb/in				E _{xy} =	1.48109005	MSI			
v ₁₂ =	0.1674					v ₁₂ =	0.16744469				

AXIAL/FLEXURAL COUPLING, [B]											
thickness	z	Z1-Z2	(1/2)(Z2-Z1) ²	Q11	Q22	Q12	Q66	Q16	Q26		
1	0.0085	-0.04970	-0.05820	-0.00046	3.465	3.435	0.395	0.758	0.000	0.000	
2	0.0083	-0.04140	-0.04970	-0.00038	5.236	5.236	3.810	4.026	0.058	0.058	
3	0.0083	-0.03310	-0.04140	-0.00031	5.236	5.236	3.810	4.026	0.058	0.058	
4	0.0083	-0.02480	-0.03310	-0.00024	5.236	5.236	3.810	4.026	0.058	0.058	
5	0.0083	-0.01650	-0.02480	-0.00017	8.664	8.434	0.498	0.713	0.000	0.000	
6	0.0083	-0.00820	-0.01650	-0.00010	5.236	5.236	3.810	4.026	0.058	0.058	
7	0.0083	0.00010	-0.00820	-0.00003	5.236	5.236	3.810	4.026	0.058	0.058	
8	0.0083	0.00840	0.00010	0.00004	8.664	8.434	0.498	0.713	0.000	0.000	
9	0.0083	0.01670	0.00840	0.00010	5.236	5.236	3.810	4.026	0.058	0.058	
10	0.0083	0.02500	0.01670	0.00017	5.236	5.236	3.810	4.026	0.058	0.058	
11	0.0083	0.03330	0.02500	0.00024	8.664	8.434	0.498	0.713	0.000	0.000	
12	0.0083	0.04160	0.03330	0.00031	5.236	5.236	3.810	4.026	0.058	0.058	
13	0.0083	0.04990	0.04160	0.00038	5.236	5.236	3.810	4.026	0.058	0.058	
14	0.0083	0.05820	0.04990	0.00045	5.236	5.236	3.810	4.026	0.058	0.058	
	0.1164				1.175E-03	1.164E-03	1.216E-03	1.148E-03	2.035E-05	2.035E-05	

NON-ROTATED AXIAL/FLEXURAL COUPLING											
Laminate Axial/Flexural Coupling Reduced Stiffness Matrix						Normalized Laminate Axial/Flexural Coupling Reduced Stiffness Matrix					
[B _{xx}]=	0.0011748	0.001156	0.0000204	in-lbs		[B _{xx}] _{norm} =	0.1784218	0.1794352	0.0030043	MSI	
	0.001156	0.001164	0.0000204				0.1794352	0.178744	0.0030042		
	0.0000204	0.0000204	0.0011479				0.0030043	0.0030042	0.1694473		
Laminate Axial/Flexural Coupling Reduced Compliance Matrix						Normalized Laminate Axial/Flexural Coupling Reduced Compliance Matrix					
[B _{xx}]=	-0.8998902	-1.3490756	0.0167872	1/in-lbs		[B _{xx}] _{norm} =	-0.0060963	-0.0091393	0.0001137	1/MSI	
	-1.3381164	-0.8977785	0.0164718				-0.0060960	-0.0060820	0.0001116		
	0.0177934	0.0177010	-0.8455436				0.0001205	0.0001199	-0.0531495		

A-4

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Laminate Stiffness Capability Check

ORIGINAL STRUCTURE			REPAIR STRUCTURE					
NON-ROTATED STIFFNESS, [A] _{0/90}			NON-ROTATED STIFFNESS, [A] _{0/90}					
	E _{x_0/90} =	0.4352		E _{x_0/90} =	0.5162	M.S. =	0.186	118.6%
	E _{y_0/90} =	0.4352		E _{y_0/90} =	0.5116	M.S. =	0.176	117.6%
	E _{xy_0/90} =	0.3310		E _{xy_0/90} =	0.3583	M.S. =	0.083	108.3%
ROTATED STIFFNESS, [A] _{±45}			ROTATED STIFFNESS, [A] _{±45}					
	E _{x_±45} =	0.7649		E _{x_±45} =	0.8287	M.S. =	0.083	108.3%
	E _{y_±45} =	0.7649		E _{y_±45} =	0.8473	M.S. =	0.108	110.8%
	E _{xy_±45} =	0.1432		E _{xy_±45} =	0.1724	M.S. =	0.204	120.4%
NON-ROTATED FLEXURAL STIFFNESS, [D] _{0/90}			NON-ROTATED FLEXURAL STIFFNESS, [D] _{0/90}					
	D _{x_0/90} =	0.0002405		D _{x_0/90} =	0.0004681	M.S. =	0.947	194.7%
	D _{y_0/90} =	0.0002405		D _{y_0/90} =	0.0004659	M.S. =	0.937	193.7%
	D _{xy_0/90} =	0.0002807		D _{xy_0/90} =	0.0004083	M.S. =	0.455	145.5%
ROTATED FLEXURAL STIFFNESS, [D] _{±45}			ROTATED FLEXURAL STIFFNESS, [D] _{±45}					
	D _{x_0/90} =	0.0006143		D _{x_0/90} =	0.0009097	M.S. =	0.481	148.1%
	D _{y_0/90} =	0.0006143		D _{y_0/90} =	0.0009311	M.S. =	0.516	151.6%
	D _{xy_0/90} =	0.0000731		D _{xy_0/90} =	0.0001500	M.S. =	1.052	205.2%

Laminate Running Loads

Original Lamina Strain Allowables	
$\epsilon_{x, \text{ Tens Ult}} =$	0.0113
$\epsilon_{y, \text{ Tens Ult}} =$	0.0113
$\epsilon_{x, \text{ Comp Ult}} =$	0.0101
$\epsilon_{y, \text{ Comp Ult}} =$	0.0101
$t_{\text{original}} =$	0.0948

Original Laminate - Unrotated Tension

[ϵ]	[$a^*/90$]	[σ]
0.0113	0.2178 -0.1132 0.0000	0.0519
-0.0059	-0.1132 0.2178 0.0000	0.0000
0.0000	0.0000 0.0000 0.2864	0.0000

$N_x = \sigma_x * t_{\text{original}} * 10^6 =$ 4917.5 lbs/in

[ϵ]	[$a^*/90$]	[σ]
-0.0059	0.2178 -0.1132 0.0000	0.0000
0.0113	-0.1132 0.2178 0.0000	0.0519
0.0000	0.0000 0.0000 0.2864	0.0000

$N_y = \sigma_y * t_{\text{original}} * 10^6 =$ 4917.5 lbs/in

Original Laminate - Unrotated Compression

[ϵ]	[$a^*/90$]	[σ]
0.0101	0.2178 -0.1132 0.0000	0.0464
-0.0052	-0.1132 0.2178 0.0000	0.0000
0.0000	0.0000 0.0000 0.2864	0.0000

$N_x = \sigma_x * t_{\text{original}} * 10^6 =$ 4395.3 lbs/in

[ϵ]	[$a^*/90$]	[σ]
-0.0052	0.2178 -0.1132 0.0000	0.0000
0.0101	-0.1132 0.2178 0.0000	0.0464
0.0000	0.0000 0.0000 0.2864	0.0000

$N_y = \sigma_y * t_{\text{original}} * 10^6 =$ 4395.3 lbs/in

Original Laminate - Rotated Tension

[ϵ]	[$a^*\pm 45$]	[σ]
0.0113	0.1239 -0.0193 0.0000	0.0912
-0.0018	-0.0193 0.1239 0.0000	0.0000
0.0000	0.0000 0.0000 0.6621	0.0000

$N_x = \sigma_x * t_{\text{original}} * 10^6 =$ 8643.7 lbs/in

[ϵ]	[$a^*\pm 45$]	[σ]
-0.0018	0.1239 -0.0193 0.0000	0.0000
0.0113	-0.0193 0.1239 0.0000	0.0912
0.0000	0.0000 0.0000 0.6621	0.0000

$N_y = \sigma_y * t_{\text{original}} * 10^6 =$ 8643.7 lbs/in

Original Laminate - Rotated Compression

[ϵ]	[$a^*\pm 45$]	[σ]
0.0101	0.1239 -0.0193 0.0000	0.0816
-0.0016	-0.0193 0.1239 0.0000	0.0000
0.0000	0.0000 0.0000 0.6621	0.0000

$N_x = \sigma_x * t_{\text{original}} * 10^6 =$ 7732.6 lbs/in

[ϵ]	[$a^*\pm 45$]	[σ]
-0.0016	0.1239 -0.0193 0.0000	0.0000
0.0101	-0.0193 0.1239 0.0000	0.0816
0.0000	0.0000 0.0000 0.6621	0.0000

$N_y = \sigma_y * t_{\text{original}} * 10^6 =$ 7732.6 lbs/in

Repair Lamina Strain Allowables	
$\epsilon_{x, \text{ Tens Ult}} =$	0.0136
$\epsilon_{y, \text{ Tens Ult}} =$	0.0125
$\epsilon_{x, \text{ Comp Ult}} =$	0.0136
$\epsilon_{y, \text{ Comp Ult}} =$	0.0119*
$t_{\text{repair}} =$	0.1164

*Lower strain allowable between GR/EP and FG/EP used.

Repair Laminate - Unrotated Tension

[ϵ]	[$a^*/90$]	[σ]
0.0136	0.2255 -0.1111 -0.0015	0.0603
-0.0067	-0.1111 0.2275 -0.0016	0.0000
-0.0001	-0.0015 -0.0016 0.3249	0.0000

$N_x = \sigma_x * t_{\text{repair}} * 10^6 =$ 7020.4 lbs/in

[ϵ]	[$a^*/90$]	[σ]
-0.0067	0.2255 -0.1111 -0.0015	0.0000
0.0125	-0.1111 0.2275 -0.0016	0.0549
-0.0001	-0.0015 -0.0016 0.3249	0.0000

$N_y = \sigma_y * t_{\text{repair}} * 10^6 =$ 6395.6 lbs/in

Repair Laminate - Unrotated Compression

[ϵ]	[$a^*/90$]	[σ]
0.0136	0.2255 -0.1111 -0.0015	0.0603
-0.0067	-0.1111 0.2275 -0.0016	0.0000
-0.0001	-0.0015 -0.0016 0.3249	0.0000

$N_x = \sigma_x * t_{\text{repair}} * 10^6 =$ 7020.4 lbs/in

[ϵ]	[$a^*/90$]	[σ]
-0.0058	0.2255 -0.1111 -0.0015	0.0000
0.0119	-0.1111 0.2275 -0.0016	0.0523
-0.0001	-0.0015 -0.0016 0.3249	0.0000

$N_y = \sigma_y * t_{\text{repair}} * 10^6 =$ 6088.7 lbs/in

Repair Laminate - Rotated Tension

[ϵ]	[$a^*\pm 45$]	[σ]
0.0136	0.1405 -0.0235 -0.0010	0.0968
-0.0023	-0.0235 0.1374 -0.0010	0.0000
-0.0001	-0.0010 -0.0010 0.6752	0.0000

$N_x = \sigma_x * t_{\text{repair}} * 10^6 =$ 11270.1 lbs/in

[ϵ]	[$a^*\pm 45$]	[σ]
-0.0021	0.1405 -0.0235 -0.0010	0.0000
0.0125	-0.0235 0.1374 -0.0010	0.0910
-0.0001	-0.0010 -0.0010 0.6752	0.0000

$N_y = \sigma_y * t_{\text{repair}} * 10^6 =$ 10591.1 lbs/in

Repair Laminate - Rotated Compression

[ϵ]	[$a^*\pm 45$]	[σ]
0.0136	0.1405 -0.0235 -0.0010	0.0968
-0.0023	-0.0235 0.1374 -0.0010	0.0000
-0.0001	-0.0010 -0.0010 0.6752	0.0000

$N_x = \sigma_x * t_{\text{repair}} * 10^6 =$ 11270.1 lbs/in

[ϵ]	[$a^*\pm 45$]	[σ]
-0.0020	0.1405 -0.0235 -0.0010	0.0000
0.0119	-0.0235 0.1374 -0.0010	0.0866
-0.0001	-0.0010 -0.0010 0.6752	0.0000

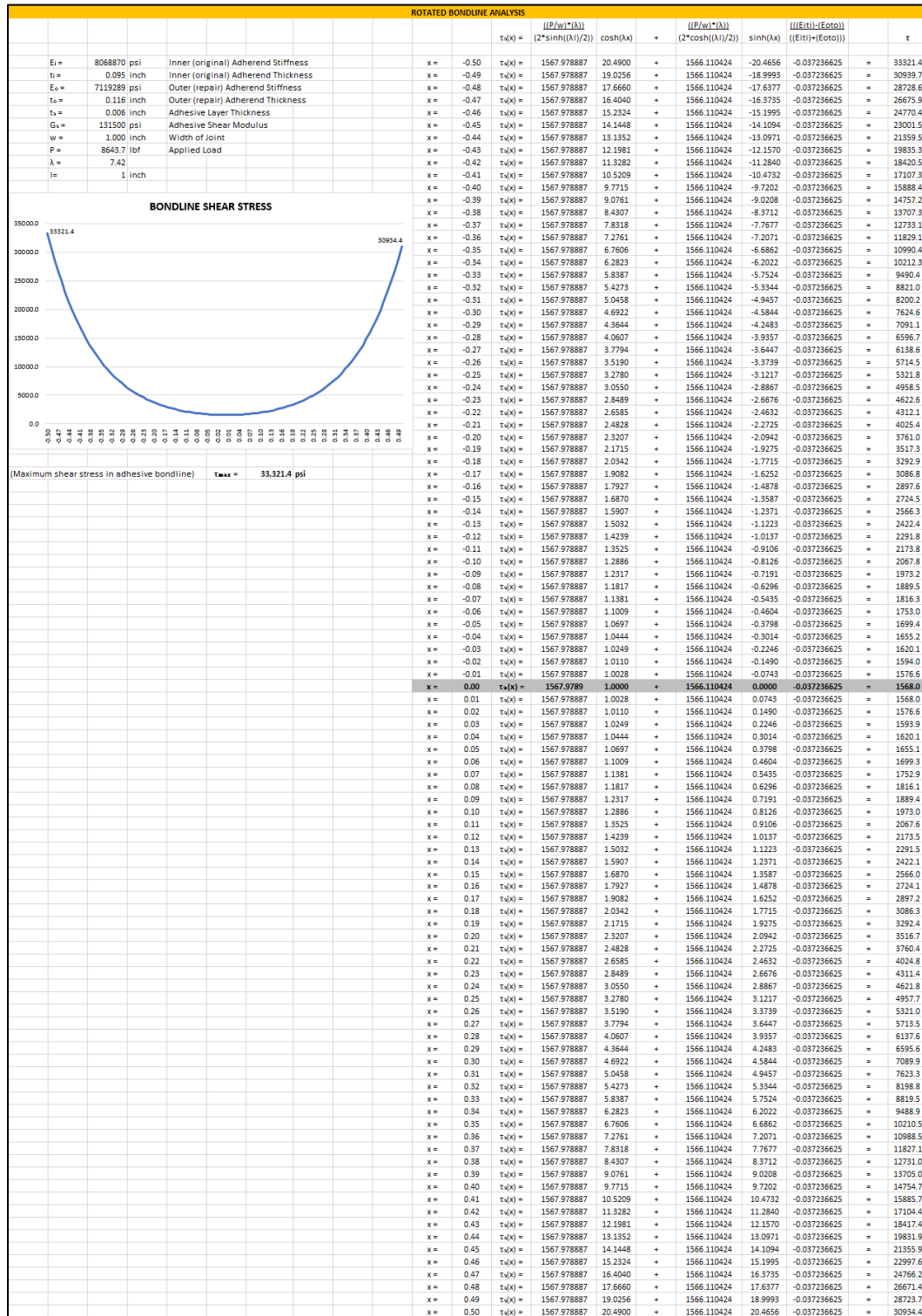
$N_y = \sigma_y * t_{\text{repair}} * 10^6 =$ 10082.6 lbs/in

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Running Load Capability Check

	Running Load, N		
Parameter	Original Laminate (lbs/in)	Repair Laminate (lbs/in)	Margin of Safety
Unrotated Tension	4917.5	7020.4	0.428
	4917.5	6395.6	0.301
Unrotated Compression	4395.3	7020.4	0.597
	4395.3	6088.7	0.385
Rotated Tension	8643.7	11270.1	0.304
	8643.7	10591.1	0.225
Rotated Compression	7732.6	11270.1	0.457
	7732.6	10082.6	0.304
(Max Running Load)	Nmax =	8643.7	lbs/in

Volkersen Bondline Analysis, +/-45 Degree Orientation



APPENDIX B

FINITE ELEMENT ANALYSIS

OBJECTIVES

This appendix documents the finite element analysis conducted in support of the composite material structural analysis of the base report and Appendix A. Finite element analysis models are pre- and post- processed with FEMAP version 2020.2 with a Simcenter Nastran solver and apply SOL101 static solutions.

FEA Modeling Methods

Parametric Geometry Modeling:

Parametric geometry is modeled as a combination of solids and surfaces with SolidWorks version 2021 and exported as IGES multi-body files for import into FEMAP for finite element analysis. Geometry bodies are modeled with net-zero offsets from connecting surfaces in preparation for surface element offsets during finite element preprocessing. Split surface preparations are applied in SolidWorks prior to export. Geometry construction uses the convention that the primary in-plane loading direction is Global X, secondary in-plane loading direction is Global Y, and laminate surface normal is Global Z. SolidWorks geometry illustrated in Figure B-1.

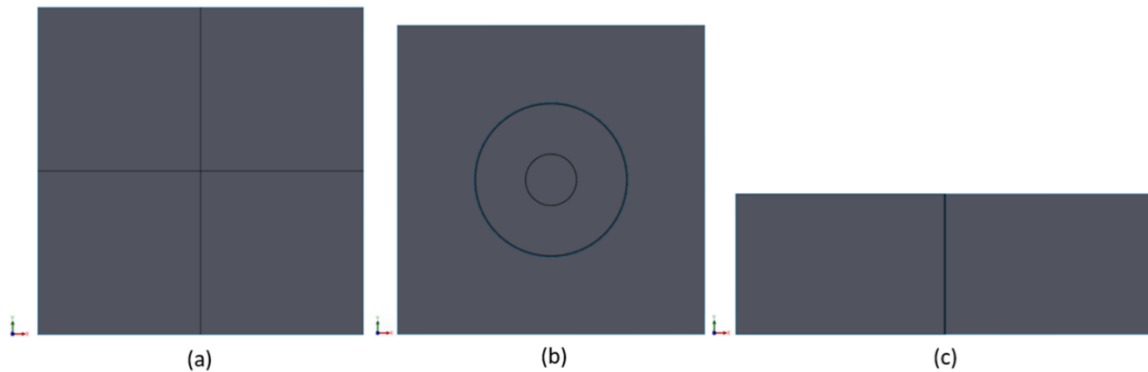


Figure B-1. Geometry for (a) Flat Plate, (b) Scab Patch Repair and (c) Lap Joint.

Material Modeling and Element Type Selection:

Composite materials are modeled as 2nd order 2D elements (QUAD for initial CLPT verification and TET for repair models) with 2D orthotropic material definitions, laminate layups, and laminate plate element property cards. Material properties for IM7/8552 (6K-70-PW), M20/40%/G904, and E7781/EA9396 are taken from Table 1.

Film adhesive elements are modeled as 2nd order 3D tetrahedral elements with an isotropic material definition and solid element property cards. The assumption of isotropic material behavior for an adhesive layer is leveraged in Appendix D of reference 11. Material properties for Cytec

FM300-2M film adhesive are taken from Table 8 of this report with a derived Young's Modulus value of $E = 341,900$ psi from the following relationship:

$$G = \frac{E}{2 \cdot (1 - \nu)} \quad (B.1)$$

Laminate definitions for original base laminate and scab patch repair shown in Figures B-2 and B-3.

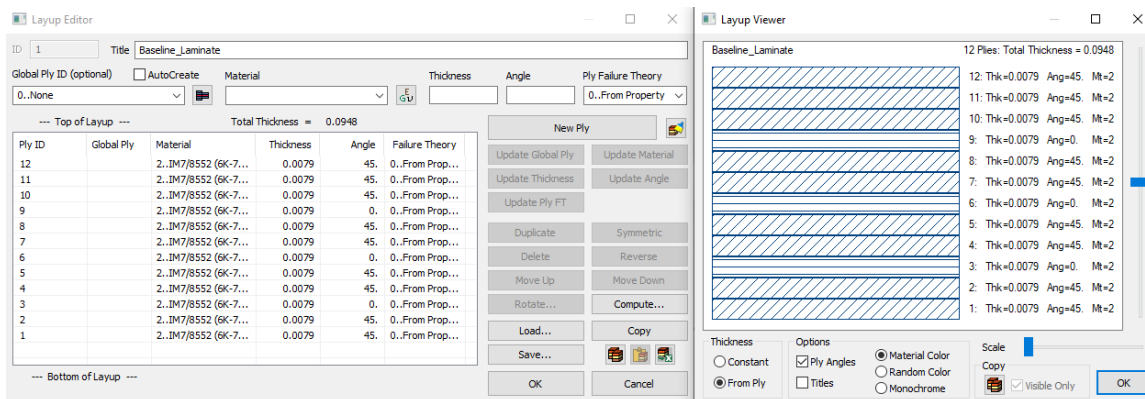


Figure B-2. Original Base Laminate Layout.

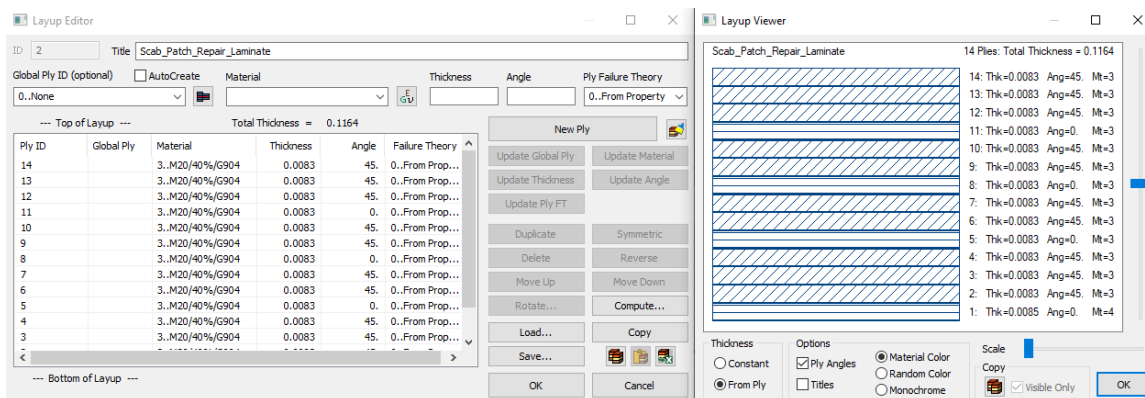


Figure B-3. Scab Patch Repair Layout.

CLPT Verification for Baseline Configuration

To verify the CLPT mechanics of the base report and Appendix A, a traction of $N_x = 4,917.5$ lbf/in was applied to the model shown in Figure B-4. As the plate measures 10-inch x 10 inch, the applied edge load was multiplied by a factor of 10 resulting in a total edge load of 49,175 lbf. The node located at the geometric center of the model is fixed with a 6-DOF constraint and mid-line edge nodes are constrained in the Global-X. Deformations with a scale factor of 5X are shown in Figure B-5; contour plot illustrates total deformations. Note that the very large deflections are consistent with ultimate lamina strains and out of plain coupling deflections.

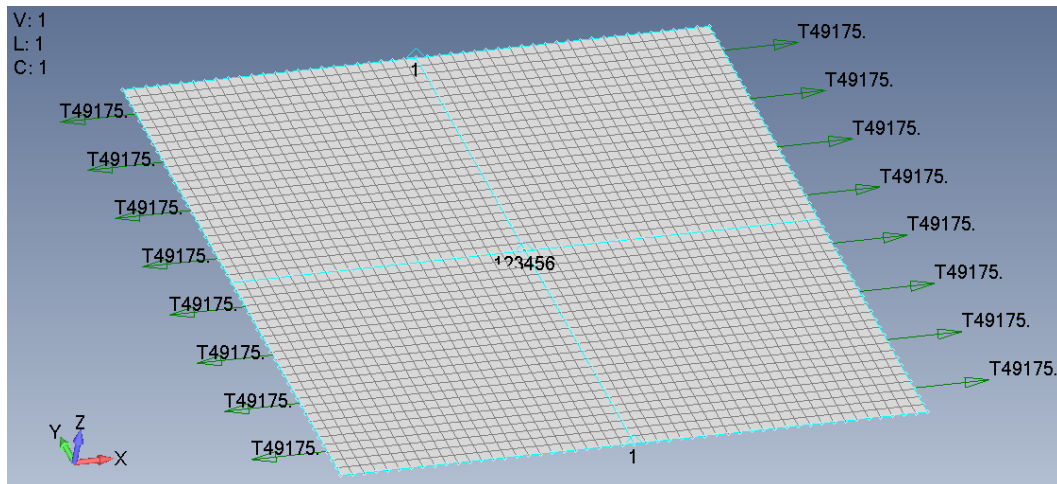


Figure B-4. Mesh with Applied Boundary Conditions.

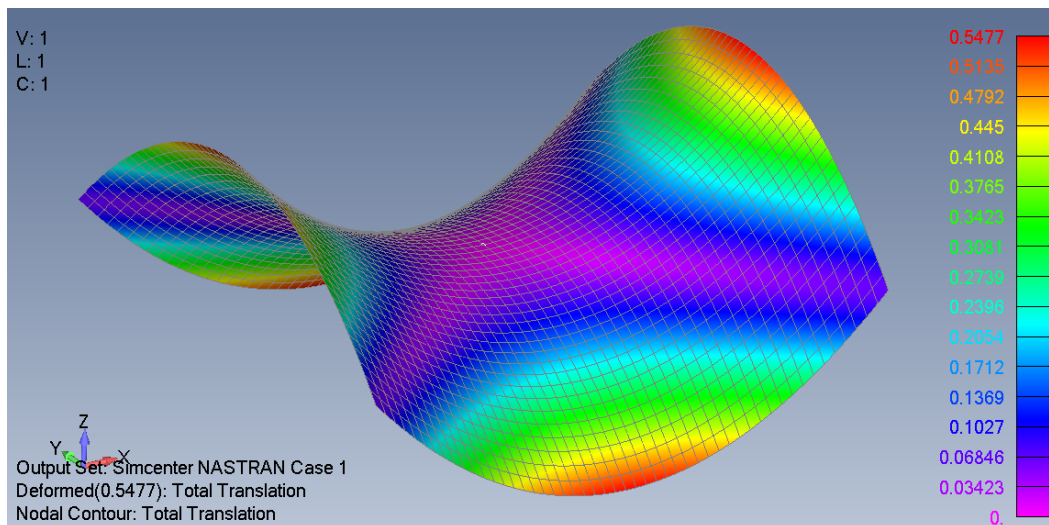


Figure B-5. Deformation Plot at Ultimate Load.

[ABD] matrix values from the CLPT analysis are reported in terms of MSI. These values are converted to psi and compared to the [ABD] matrix values of the finite element model. See Figures B-6 thru B-8. The difference between the two [ABD] matrices represent **negligible error**.

[ABD] (psi)	Closed Form CLPT (psi)					
	5.96E+05	3.10E+05	9.20E-02	-3.71E+02	3.71E+02	1.21E-04
	3.10E+05	5.96E+05	-9.20E-02	3.71E+02	-3.71E+02	-1.21E-04
	9.20E-02	-9.20E-02	3.31E+05	1.21E-04	-1.21E-04	3.71E+02
	-3.71E+02	3.71E+02	1.21E-04	4.13E+02	2.65E+02	7.98E-05
	3.71E+02	-3.71E+02	-1.21E-04	2.65E+02	4.13E+02	-7.98E-05
	1.21E-04	-1.21E-04	3.71E+02	7.98E-05	-7.98E-05	2.81E+02

Figure B-6. Closed Form CLPT [ABD] Matrix.

[ABD]	FEA (psi)					
	5.96E+05	3.10E+05	0.00E+00	-3.71E+02	3.71E+02	0.00E+00
	3.10E+05	5.96E+05	0.00E+00	3.71E+02	-3.71E+02	0.00E+00
	0.00E+00	0.00E+00	3.31E+05	0.00E+00	0.00E+00	3.71E+02
	-3.71E+02	3.71E+02	0.00E+00	4.13E+02	2.65E+02	0.00E+00
	3.71E+02	-3.71E+02	0.00E+00	2.65E+02	4.13E+02	0.00E+00
	0.00E+00	0.00E+00	3.71E+02	0.00E+00	0.00E+00	2.81E+02

Figure B-7. Finite Element [ABD] Matrix.

%Error						
	-0.000021%	-0.000145%	-	0.001124%	0.000045%	-
	-0.000145%	-0.000021%	-	0.000045%	0.001124%	-
	-	-	-0.000004%	-	-	-
	0.001124%	0.000045%	-	-0.000028%	0.000030%	-
	0.000045%	0.001124%	-	0.000030%	-0.000028%	-
	-	-	0.000045%	-	-	0.000002%

Figure B-8. % Error Between Finite Element [ABD] and Closed Form CLPT [ABD].

The in-plane strain response between the FEA and the closed form solution are compared for the prescribed N_x traction of 4,917.5 lbf/in. Figure B-9 reports a maximum X-displacement of 0.05706 inches.

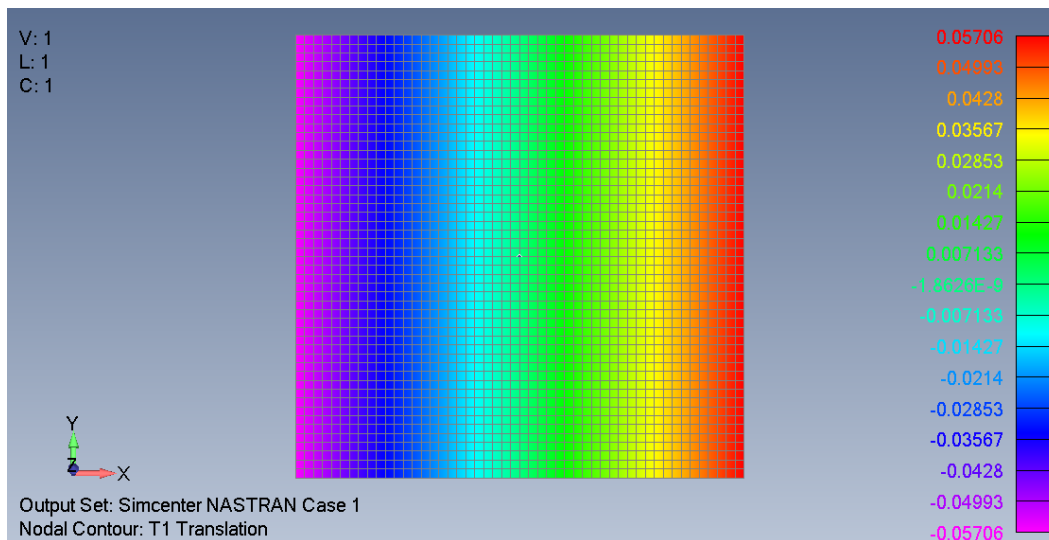


Figure B-9. X-Direction Displacements.

Effective length from the center line to the part edge is 5 inches. Strain in the x-direction is computed as follows:

$$\begin{aligned}\epsilon_{x_FEA} &= \frac{\text{Displacement}_x}{\text{Length}_{\text{effective}}} \\ \epsilon_{x_FEA} &= \frac{0.05706 \text{ in}}{5 \text{ in}} \\ \epsilon_{x_FEA} &= 0.0114 \frac{\text{in}}{\text{in}}\end{aligned}\tag{A.2}$$

The Strain-Traction relationship reported in Figure 17 of the base report indicates $\delta x = 0.0113$ in/in for an Nx traction of 4,917.5 lbf/in. The error between the FEA and CLPT results is **0.9%** showing a **strong correlation** between both methods:

$$\begin{aligned}\% \text{ Error} &= \frac{\epsilon_{x_FEA} - \epsilon_{x_CLPT}}{\epsilon_{x_CLPT}} (100\%) \\ \% \text{ Error} &= \frac{0.0114 - 0.0113}{0.0113} (100\%) \\ \% \text{ Error} &= 0.88\%\end{aligned}\tag{A.3}$$

Strong agreement between the [ABD] matrix values and in-plane x-direction strain response under Nx loading supports the verification of the CLPT closed form solution model used in the course of this analysis.

Scab Patch Analysis (Volkersen) Verification for Film Adhesive Shear Stress

The FEA model of Figure B-10 represents the configuration assumptions of the Scab Patch Repair Joint Analysis section of the base report. Edge widths in the global y-direction are modeled with 1 inch width so the magnitudes of Px and Nx match. Interfaces between the composite laminates (2D elements) and film adhesive (3D elements) are modeled as glued connections. Volkersen assumes zero displacements in the Z-direction under in-plane loading. Constraint boundary conditions pin the nodes along the edge opposite the load application edge. All composite elements include a global-Z displacement constraint to match the assumptions of the Volkersen model. Figure B-10 plots the element offsets applied during FEA pre-processing as indicated with the orange lines. The geometry itself is modeled with coincident surfaces to aid in contact connection detection. Figure B-11 plots a total displacement contour.

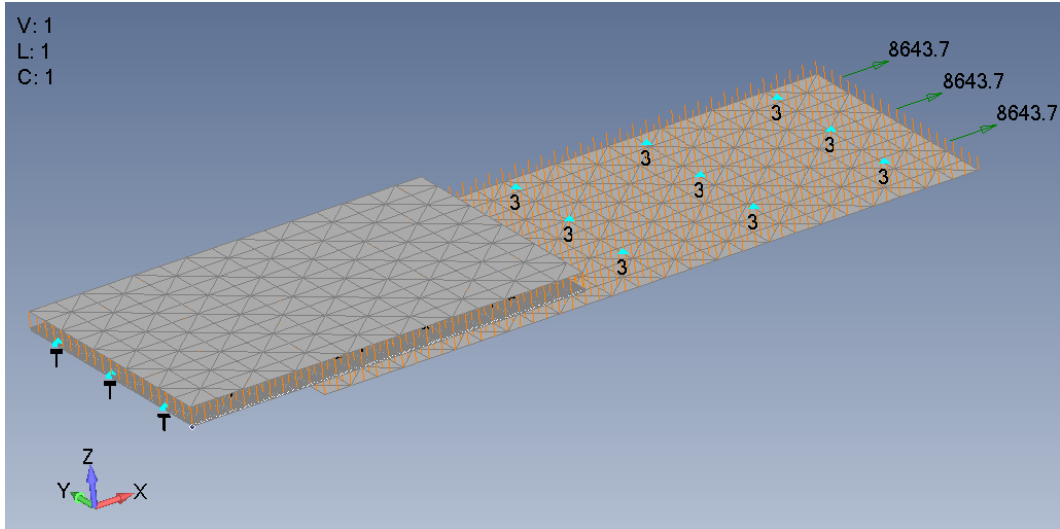


Figure B-10. Lap Joint Model (Volkersen Verification).

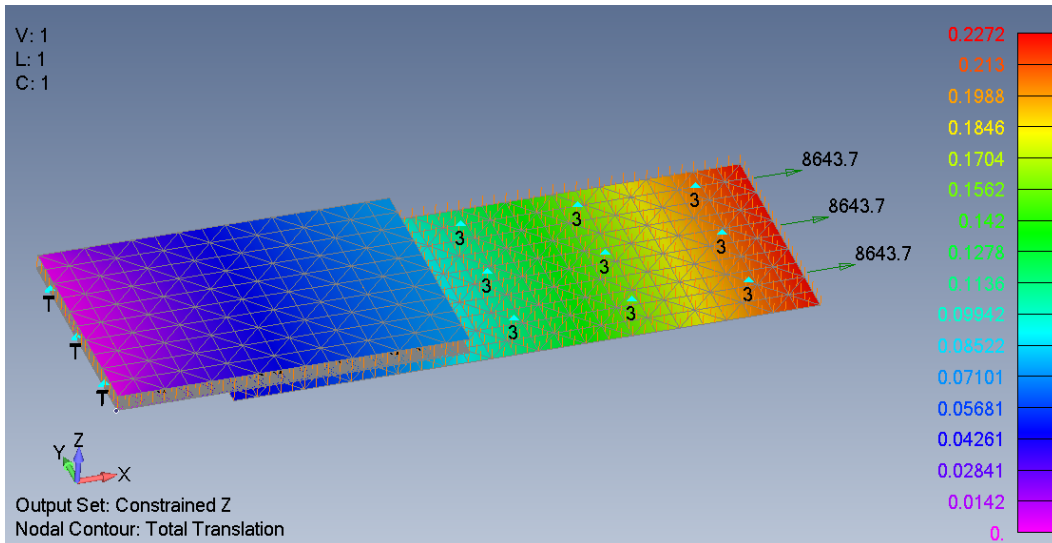


Figure B-11. Total Displacement Contour.

Figure B-12 shows the film adhesive elements only with the contour plot set to maximum shear stress. Central collections of elements are isolated to remove corner node peaking from the reported maximum shear stress.

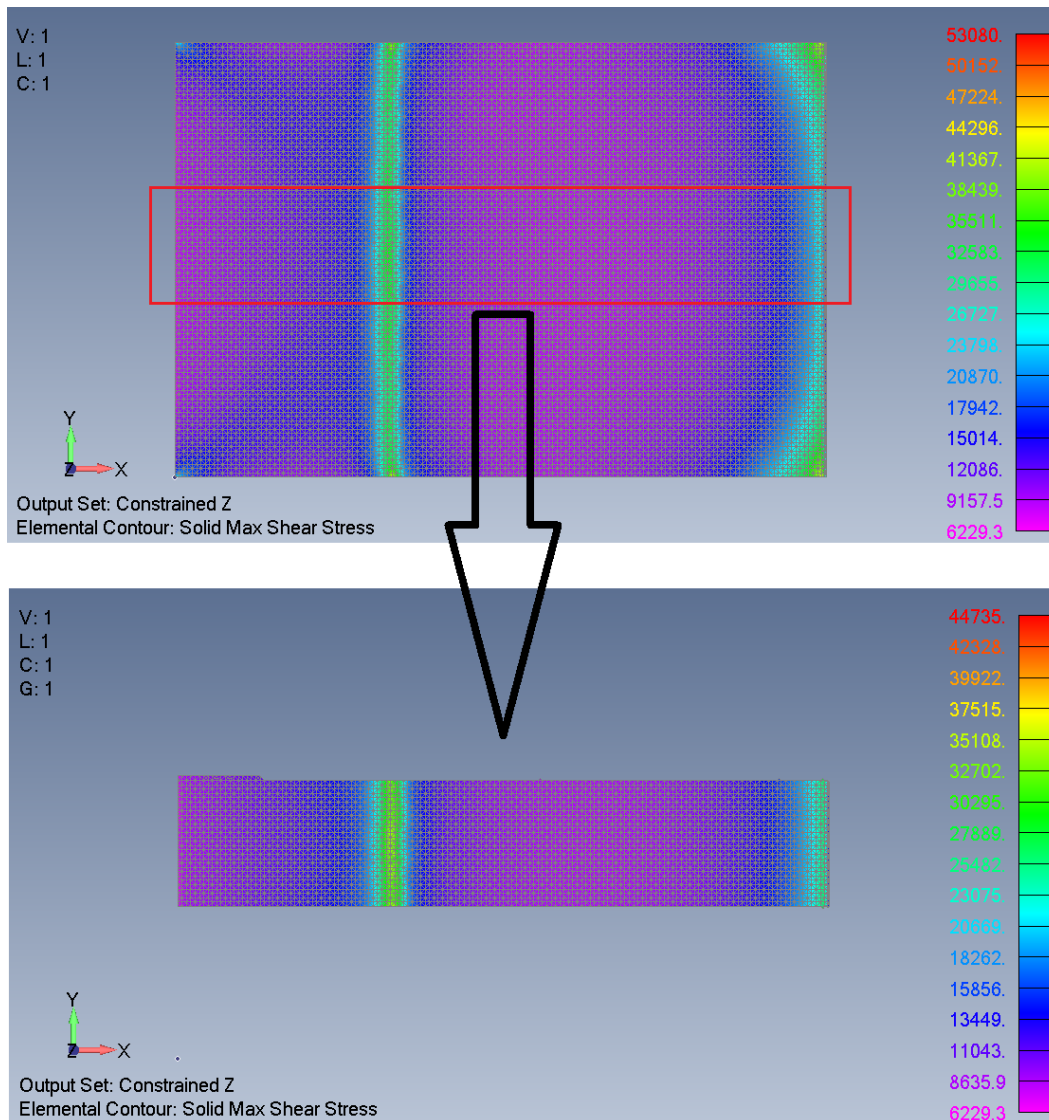


Figure B-12. Film Adhesive FEA Results.

The finite element model of Figure B-13 is a 6 inch x 6 inch sub-assembly representing the scab repair of a 1 inch diameter hole. Applied N_x consistent with previous models ($P_x = 51,862$ lbf total applied load; consistent with very large reported out of plane displacements). Models evaluated with both simply supported edges and constrained Z displacement. Figures B-14 and B-15 illustrate the simply support and Z-constrained deformation plots respectively; Figures B-16 and B-17 report the max shear stress contours.

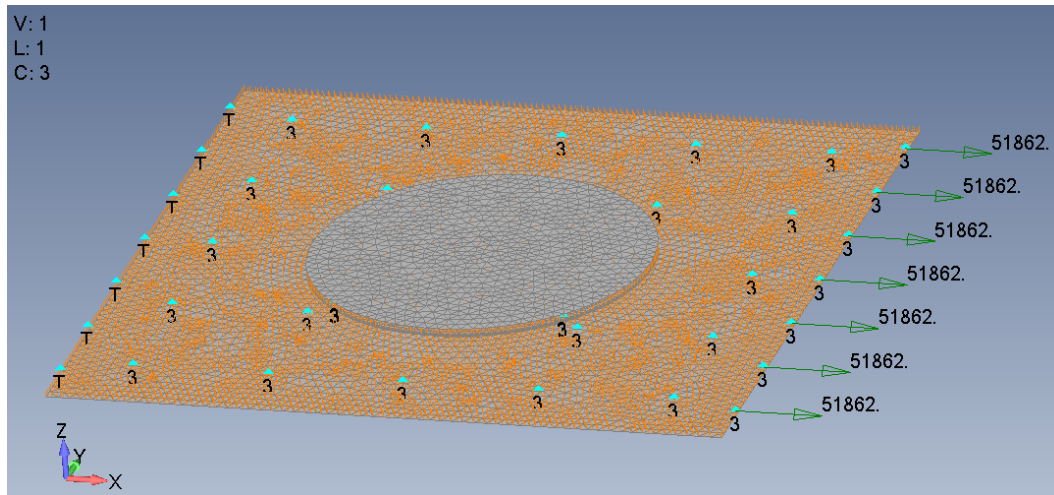


Figure B-13. Scab Repair Subassembly Model.

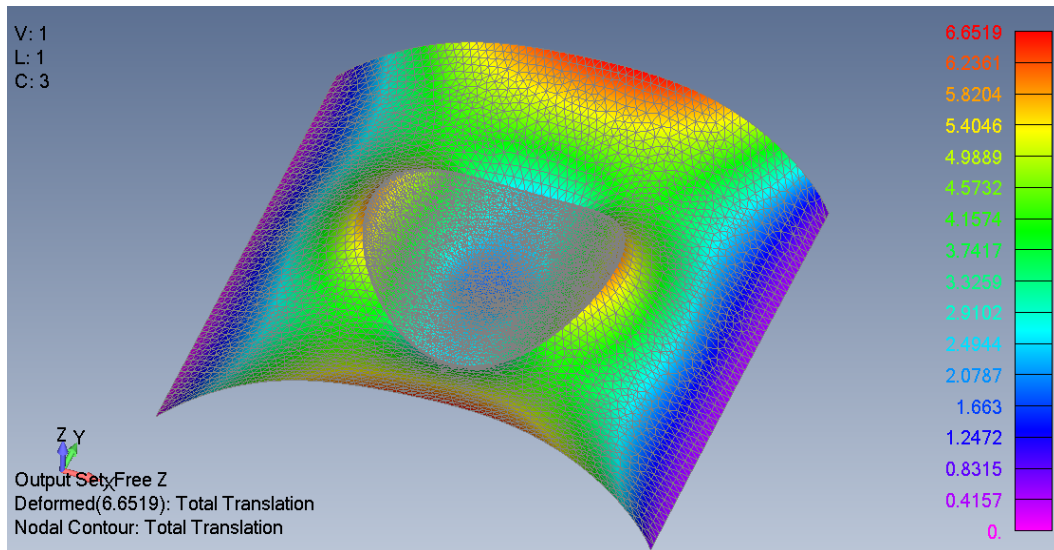


Figure B-14. Simply Supported Deformation Plot.

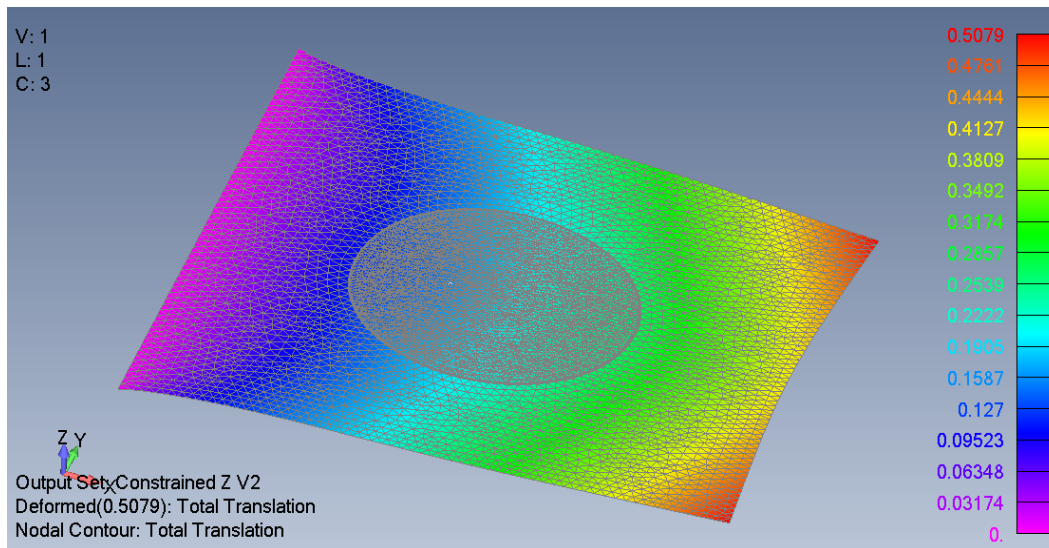


Figure B-15. Z-Constrained Deformation Plot.

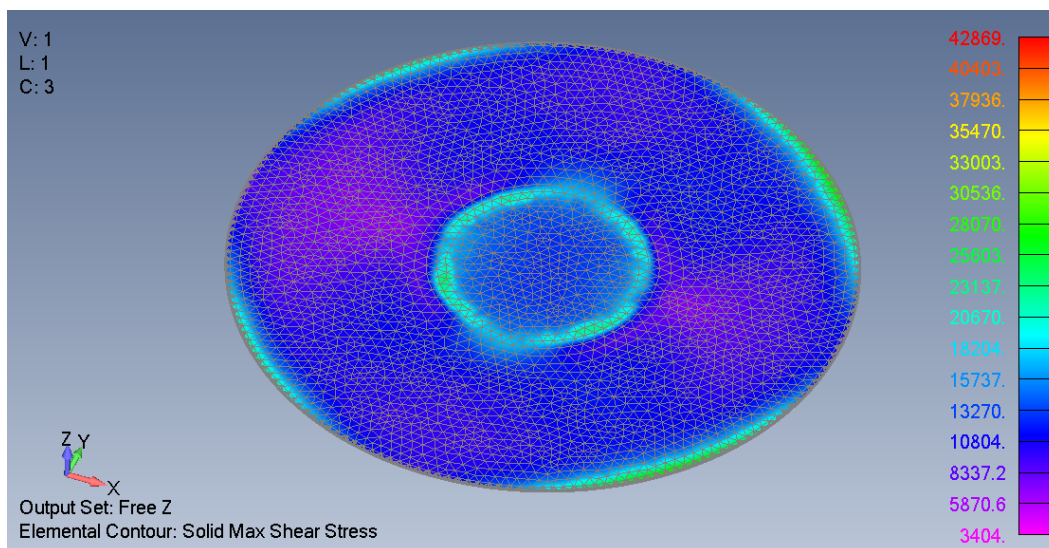


Figure B-16. Simply Supported Max Shear Stress Plot.

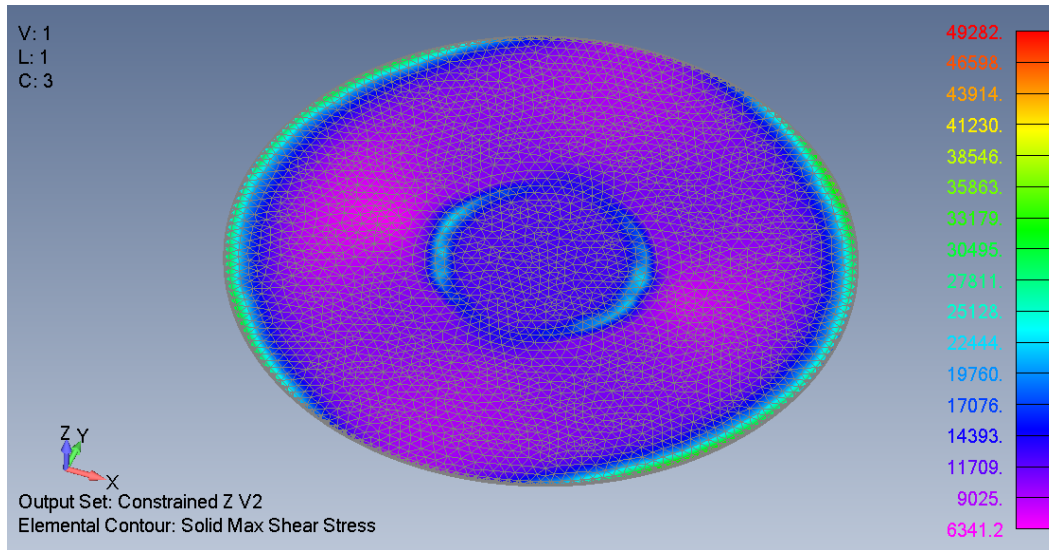


Figure B-17. Z-Constrained Max Shear Stress Plot.

Table B-1 reports summary data for peak stress values from each analytical model. Note that the highest peak of the lap joint is predicted on the edge common to the damaged hole while the Volkersen and sub-assembly models indicate the highest peak on the exterior edge of the repair bondline. Chart B-1 plots the Peak (1) & (2) pairs for each analysis case.

ID	Description	Peak (1) (psi)	Peak (2) (psi)	Range (psi)
1	Volkersen	39,056	33,113	5,943
2	Lap FEA – all elements	53,080	47,224	5,856
3	Lap FEA – exclude edges	44,735	30,295	14,440
4	Sub-Assembly – Z-Constrained	49,282	25,148	24,134
5	Sub-Assembly – Simply Supported	42,869	26,203	16,666

Table B-1. Summary of Bondline Peak Stress Analysis.

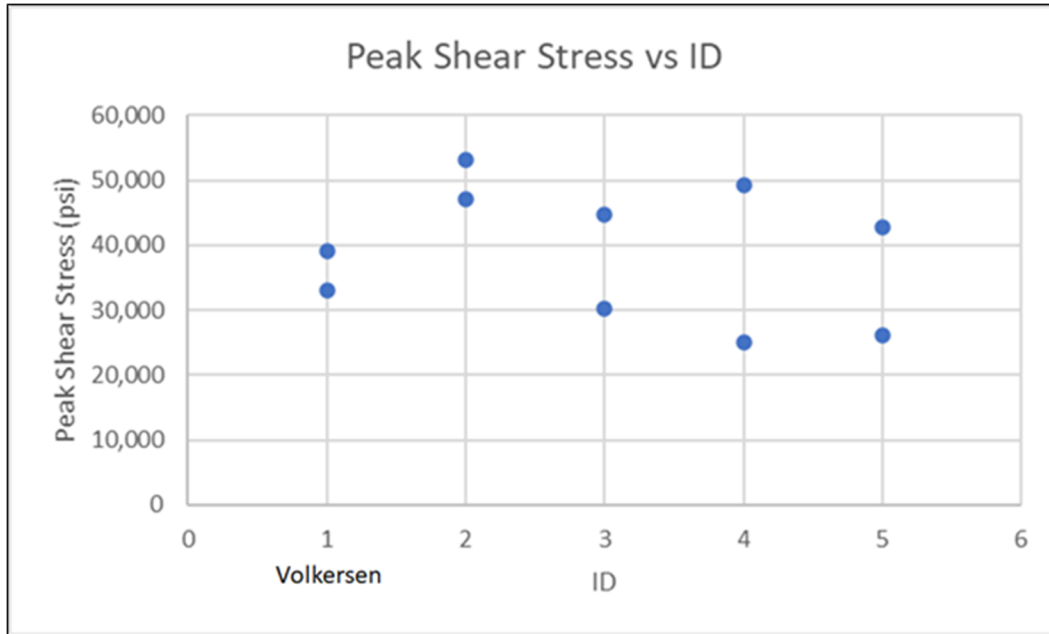


Chart B-1. Peak Shear Stress Pairs vs Case ID.

Analytical predictions for peak shear stress values along repair bond lines are compared for a variety of models. Except for the corner peaks of Case 2, the Volkersen peaks fall within the range of peaks for the alternate FEA models. Localized peaking reported by FEA models present a challenge for interpretation as the highest values appear to be artifacts of model limitations arising from discretization and connection error.

Both Volkersen and FEA bond line shear stress predictions generally agree for the in-plane load cases considered. Allowables were not considered in the FEA evaluation, all models assume linear elastic responses only.

Follow-on efforts for static and fatigue testing of representative coupons for the proposed repair configuration would be required to validate applicability of the proposed analytical models.