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BEARING RESPONSE CHARACTERIZATION OF BOLTED HYBRID COMPOSITE JOINTS

**John S. Brewer, Jr.
Air Force Institute of Technology**

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14. ABSTRACT Composite materials are used in ever increasing quantities as aerospace and automotive applications demand lighter and stronger structures. However, a major issue that plagues composite materials is the process of attaching them to other structural elements. In many cases, fasteners are employed in addition to adhesives to make a more predictable joint. In other cases, fasteners are solely employed to create a serviceable joint or access panel. These fastener sites create stress concentrations and crack initiation sites in the brittle composite materials. In some systems, locally adding additional plies of the composite material is used. This is commonly called a “pad-up.” This does increase the local strength, but causes a large variation in thickness and fails to address the brittle failure characteristics of the material. The Steel Reinforced Advanced Thin Unitized Structure (STRATUS) Program lead by the Air Force Research Lab Aerospace Systems Directorate (AFRL/RQ) proposes the addition of stainless steel foils between and in place of layers of composite near fastener sites (Figure 1). This technology seeks to combine the high strength to weight ratio of carbon fiber composites with the predictable toughness and ductile failure nature of stainless steel.									
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PROGRESSIVE FAILURE IN BOLTED HYBRID COMPOSITE JOINTS

I. Introduction and Background

1.1 Motivation

Composite materials are used in ever increasing quantities as aerospace and automotive applications demand lighter and stronger structures. However, a major issue that plagues composite materials is the process of attaching them to other structural elements. In many cases, fasteners are employed in addition to adhesives to make a more predictable joint. In other cases, fasteners are solely employed to create a serviceable joint or access panel. These fastener sites create stress concentrations and crack initiation sites in the brittle composite materials. In some systems, locally adding additional plies of the composite material is used. This is commonly called a “pad-up.” This does increase the local strength, but causes a large variation in thickness and fails to address the brittle failure characteristics of the material. The Steel Reinforced Advanced Thin Unitized Structure (STRATUS) Program lead by the Air Force Research Lab Aerospace Systems Directorate (AFRL/RQ) proposes the addition of stainless steel foils between and in place of layers of composite near fastener sites (Figure 1). This technology seeks to combine the high strength to weight ratio of carbon fiber composites with the predictable toughness and ductile failure nature of stainless steel.



Figure 1: Notional STRATUS Structure

1.2 Research Objective

The objective of this work is to answer the following research questions. How does the inclusion of stainless steel foils in a composite laminate near a fastener affect the bearing strength and failure characteristics of the structure in single-shear and double-shear geometries through quasi-static and fatigue loading scenarios? What are the most effective methods by which readymade finite element software can be employed to model the complex failure of this localized fiber metal laminate structure near a fastener load?

1.3 Background

This section seeks to describe the background and development of composite materials and hybrid composite materials as related to characterization and analysis of fiber metal laminate bolted joints. This document does not purport to present an unabridged history of composite materials, metals, or hybrid material systems. Rather, history and findings upon which current fiber metal laminate and joint technology rely is presented.

1.3.1 Composite Materials

Composite materials are typically thought of as some constituent material reinforced with a fibrous material. Commonly seen as a modern technological

development, composite materials are as old as creation. Wood and bone are both naturally occurring composite materials. Wood is composed of cellulose fibers and a bonding agent called lignin, while bone consists of the protein collagen and a brittle component known as apatite. [1] Manmade composites are also as old as recorded history. Mud bricks reinforced with straw are early examples. Concrete, made up of aggregate and cement, is a common manmade composite which found widespread use in the Roman Empire [2].

In modern times, the term “composite” has become synonymous with polymer matrix composites (PMC). Glass fibers were first developed in the 1930s and used as insulation. This material, the pink insulation, is still used in many home insulation applications to this day. Early applications of PMC were developed in the 1940s as an attempt to create lightweight aircraft components. However, due to limitations in manufacturing processes, the parts contained high resin ratios and uneven fiber distributions leading to more early applications in the marine industry. Many components of the landing craft used in the D-Day invasion were made of glass fiber reinforced polymers. Post war, composite materials made their way into the commercial market. The first Chevrolet Corvette, sold in 1953 consisted of an all glass fiber body [3], [4].



Figure 2: Corvette Subassemblies [3]

Also in the 1950's, Roger Bacon discovered "graphite whiskers" which are now known as carbon fibers. These were first applied as high temperature insulation and filters. The first application in composite form was in high temperature jet nozzles. Despite cost and complexity in manufacturing, carbon fiber, praised for its light weight and stiffness, has become mainstream in the automotive, aerospace and even the sporting goods industries. By 2013 the Corvette hood, fenders and floor panels were all carbon fiber [5], [6]. The aerospace industry is by far the largest consumer of polymer matrix composites. The majority of the Airbus A-380 XWB is composed of composite materials and boasts one of the largest continuous parts ever used in a commercial aircraft. The lower wing cover of the A-380 XWB, seen in Figure 3, is roughly 32 meters long (104 feet). [6], [7]



Figure 3: Airbus A-380 XWB Lower Wing Cover [7]

The largest composite aircraft to date is the Stratolaunch built by Scaled Composites (Figure 4). Built largely of carbon fiber, this aircraft boasts a wingspan of 385 feet. Special traveling ovens were designed to cure the wing spars that measured over 200 feet long. The goal of this vehicle is to begin normalizing access to near earth orbit [8].



Figure 4: Scaled Composites Stratolaunch [8]

1.3.2 Hybrids and Fiber Metal Laminates

It is important to define the terms “Hybrid Composite” and “Fiber Metal Laminate” before developing a discussion around the topics. Typically, a polymer matrix composite is thought of as being composed of two elements: a single type of reinforcement such as glass or carbon fibers, and a polymer matrix. The definition of what constitutes a hybrid composite varies amongst authors, but hybrids are generally accepted as those materials which include more than a single type of reinforcement [9],[10]. In some cases these include other fibers such as aramid fibers (Figure 5A) or polypropylene fibers [11],[12]. Hybrids can also be thought of as reinforcements that differ in scale and geometry compared to the conventional reinforcement. These may include materials such as aligned nanofibers (Figure 5B) or z-pins [4][5]. Finally, this extends to the concept of including metal sheets into composite material, also known as

Fiber Metal Laminates (FML)[14]. Thus, FML should be considered a specific type of material under the umbrella of hybrid composites.

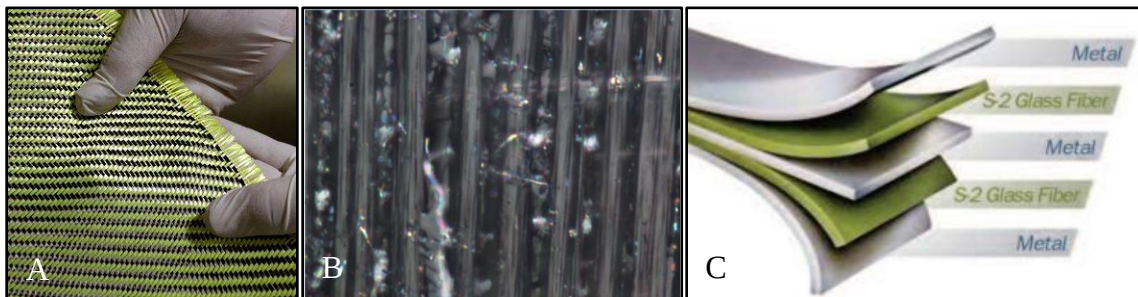


Figure 5: Examples of Hybrid Composites: A) Carbon/Kevlar® [11], B) Carbon Fiber with z-aligned Carbon Nanofibers[6], C) Glass fiber/Aluminum[14]

Typically FML are thought of in three main categories as defined by the parent composite material: glass fiber, aramid fiber, and carbon fiber. Then, they are further defined by the selection of the metal to be included. The most commonly seen Fiber Metal Laminates referenced are:

- ARALL – (Aramid Reinforced Aluminum Laminates) – Aramid/Aluminum hybrid
- GLARE – (Glass Reinforced) – Glass Fiber/Aluminum hybrid
- TiGr – (from “Titanium” and “Graphite”) – Carbon Fiber/Titanium hybrid

The origins of FML can be traced to post World War II Fokker aircraft facilities that had been ravaged in wartime. An engineer named Rob Schliekelmann, a graduate of Delft University in the Netherlands, which is known for the study of bonded metals and FML, had studied the bonding of metals while working for DeHavilland and brought that technology to Fokker. He refined the process by optimizing surface preparation techniques and introducing processes such as autoclaving. This metal bonding

technology was first applied to the Fokker F-27 Friendship. This model of aircraft is still used worldwide and is used by the United States Army Golden Knights parachute demonstration team under the military designation of C-31 as seen in Figure 6 [15], [16], [17]



Figure 6: US Army Golden Knights C-31 (F-27) [15]

Research by Jacobus Schijve at Netherlands National Aerospace Centre (NLR) and Delft University in the 1960s and early 1970s showed that laminated metallic sheets demonstrated reduced fatigue crack growth rates when compared to monolithic materials of the same thickness [18]. While at Delft, Schijve collaborated with Boud Vogelsang. Upon seeing composite research occurring in Britain and the United States, Schijve and Vogelsang began to incorporate composite materials into their bonded metals. The two proved that crack propagation was slowed through fiber bridging and delamination. While a student at Delft, Roelof Marissen conducted tests on hybrids made from aluminum and aramid fibers. He used fracture mechanics principles to develop an analytical model for fatigue crack growth in FML. In the early 1980s, Schijve,

Vogelsang and Marissen received two patents: for the process of bonding metallic plies with fibrous adhesives and for the material that would become ARALL [19], [20]. The first parts tested were wing panels for the F-27. Also during this time, variants of ARALL were developed with different aluminum alloys and adhesives. The first ARALL parts certified were wing hatches on the Fokker 50. In 1988, ALCOA (Aluminum Company of America) won a contract to build United States Air Force C-17 cargo doors. However, only 40 doors were constructed and the Air Force finally settled on traditional aluminum construction due to weight and complexity. [16]

AKZO (Algemene Koninklijke Zout Organon –Dutch: General Royal Silk Organization) filed a patent for GLARE in 1987. From a partnership of AKZO and ALCOA the Structural Laminates Company (SLC) was formed. SLC studied structural applications of GLARE with Boeing. In 1990 GLARE was applied to the floor of the Boeing 777. GLARE rose in appeal, because in addition to the other properties of hybrids, it would display a dent after damage. When compared to pure composite structures, this is appealing because it allows damage detection upon visual inspection. GLARE also has improved behavior in the presence of fire. The aluminum and polymer matrix burn away in a jet fuel fire, but the fibers stay intact and act as a barrier. Based on qualities of fire and impact resistance, GLARE was used to construct the ECOS-3 blast resistant cargo container (Figure 7). Certified by the FAA, this was an answer to the onboard bombing of Pan Am Flight 103 over Lockerbie, Scotland on 21 December 1988. [16], [17]



**Figure 7: ECOS-3 Blast Resistant Cargo Container
Before and After Blast[16]**

While seeking his PhD at Delft University, United States Air Force Officer Robert Fredell studied structural repair using hybrid composites. While assigned to the United States Air Force Academy, Fredell continued his work and applied the technology to a structural repair on a C-5A Galaxy[21][16]. As of 2007, the repair showed no crack extensions or debonding around the patch and no damage to the patch itself [22].

1.3.3 Modern Applications of Fiber Metal Laminates

The most famous and largest application of hybrid composites has been the application to main fuselage sections of the Airbus A380[23] as noted in Figure 8 and pictured in Figure 9.

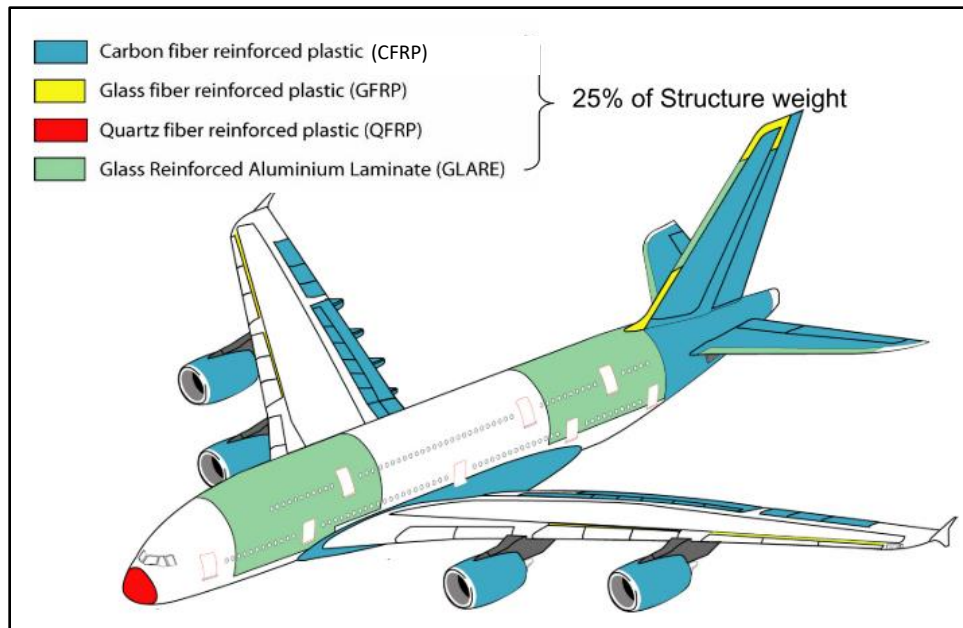


Figure 8: Airbus A380 Composite Structures[24]



Figure 9: Airbus A380 Composite Structure [22]

Praised for their light weight and high stiffness, carbon fiber composites continue to grow in use. Thus, this is a critical time in hisory continue development of Fiber Metal

Laminate with carbon fiber to construct the most robust structures possible. Carbon fiber in FML has been investigated and has been analyzed in bolted joints [25]–[27], but most of the research investigates the combination of carbon fiber with aluminum or titanium. There is little evidence to show the investigation of stainless steel foils in combination with carbon fiber.

1.3.4 Fasteners in Composites

Ideally, composite structures would be manufactured as completely continuous structures. However, many times component size, design complexity, logistical concerns or even political and economic concerns force structural interconnections. For example, the size of a composite part may be driven by the availability of an autoclave appropriately sized to cure that part, leading to a joint. Joints can also be bonded, but this often complicates airworthiness certification. In some cases, fastened joints exist without adhesive because the joint is designed to be serviceable or the component needs to be removed as part of a service or inspection routine.

The problem arises in how fastened joints fail. Fastener sites create high stress concentrations in brittle composites and are prone to fatigue due to flight dynamics and cyclical cabin pressurization. Figure 10 shows the three failure modes expected in a fastened composite joint: shear-out, net tension and bearing. Bearing is the most desirable failure mode since shear-out and net tension tend to be catastrophic failures while hole elongation due to bearing is likely to be discovered upon inspection. Studies have shown that current methods of predicting fatigue life in composite bolted joints are lacking[28].

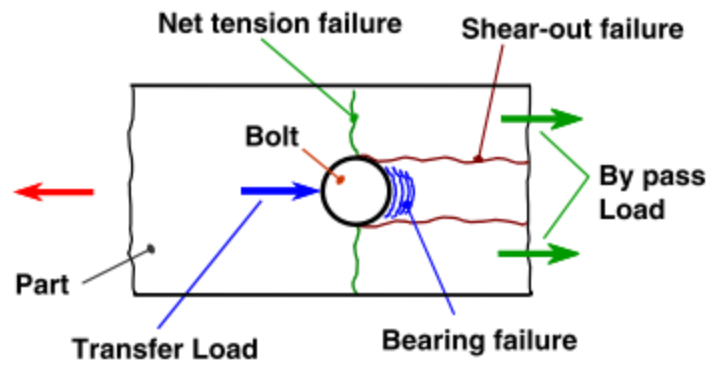


Figure 10: Bolted Joint Failure Modes [26]

Efforts to improve and control failure near these fastener sites have included tailoring and additional composite layers near fastener sites. Cylindrical inserts have been employed and even other constituent FML have been studied. FML have proven to increase the static strength and fatigue life of bolted joints. [17], [26], [28], [29]

II. Theory on Critical Topics

This chapter seeks to layout the theoretical background needed to undergird and complete the work proposed in this document. These topics will not be discussed in an exhaustive nature, but in enough detail that the application to this study is explained and supported.

A bolted joint in FML involves many diverse topics. Finite element modeling of this system brings about even more complexity. Since the material includes a composite material, failure mechanisms and failure criteria of composites must be considered. Also, different methods of modeling composites must be considered. For example, in many cases composites are modelled using cohesive elements to represent interlaminar bonding. Furthermore, the addition of an adhesively bonded metal foil will likely require a cohesive representation. While addressing the foil, plasticity associated with stainless steel will be considered. Since the load is resisted through the fastener, a complex contact relationship must be developed between the fastener and the FML part. Finally, plasticity in the fastener will need to be considered to properly represent bending in the fastener and how this bending affects load capacity in the specimen. This load scenario brings about numerous, complex contact relationships.

In the standard bearing test methods (ASTM D5961 and D6873), the fixture is constrained and the load is applied to the coupon. Thus, the stress is transferred through the coupon and into the foil/composite transition zone. The fastener hole in the foil-composite zone transfers stress into the bolt via the contact interface between the bolt and the coupon. This contact relationship creates the bearing stress of concern in this study. A contact interaction also occurs between the coupon and fixture. Contact interactions

are required between the cohesive layers and material layers to prevent penetration. Furthermore, self-contact must be established to prevent components from penetrating themselves under large deformation.

Egan et al have investigated many of these topics in their work on lapped bolted joints. Figure 11 shows a single lap joint investigated by Egan et al. This joint was comprised of two coupons of unidirectional carbon fiber fastened with a single countersunk fastener. This image displays the strengths of modeling techniques such as advance failure models, layerwise modeling, and contact modeling. These topics and others will be explored in depth in this chapter. The remainder of this chapter seeks to present the theory on these topics, relevant research in which these theories have been employed, and how they will be employed in this study. In some cases, a breadth of theory and research are presented to make the case for selecting a certain method for this study. This chapter concludes with a discussion on how these theories are collectively employed. The computational work in this study will be carried out using Abaqus software. Thus the theory represented is compliant with or referenced from Abaqus. [30]–[32]

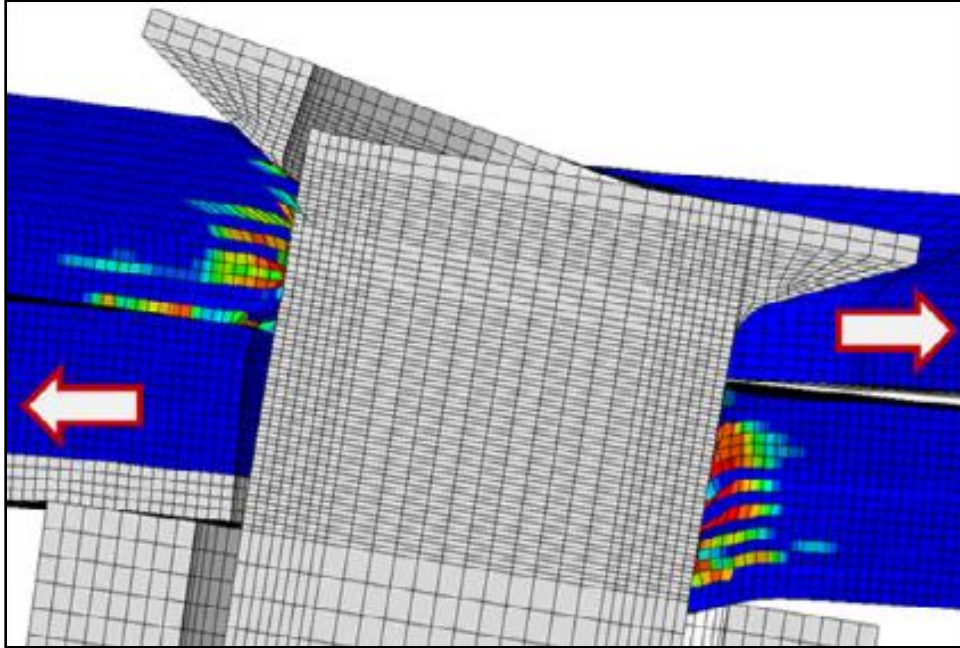


Figure 11: Compressive Fiber Damage in Double Lap Joint [32]

2.1 Explicit vs Implicit Modeling

Many decisions in the modeling process are based on the type of analysis being considered. Since this study will include simulation of a dynamic progressive failure, explicit and implicit dynamic analysis methods should be considered. Generally, explicit methods obtain values by solving for a given variable at time $t + \Delta t$ by using only values at time t . Explicit methods are conditionally stable based on a maximum time step based on wave propagation through the system. This stable time step is calculated prior to the first iteration of the explicit analysis. Implicit methods remove the time step constraint, by solving for a given variable at $t + \Delta t$ by using inputs from both time t and $t + \Delta t$. [33]

2.1.1 Implicit Modeling

The typical method of implicit calculation is the Hilber, Hughes and Taylor operator as a generalization of the Newmark Operator which allows for numerical damping. Equilibrium is replaced with a balance of d'Alembert forces with weighted averages of internal and external forces at the beginning and end of the time step.

$$M^{NM}\ddot{u}^M|_{t+\Delta t} + (1 + \alpha)(I^N|_{t+\Delta t} - P^N|_{t+\Delta t}) - \alpha(I^N|_t - P^N|_t) + L^N|_{t+\Delta t} \quad (1)$$

Displacement and velocity are given by

$$u|_{t+\Delta t} = u|_t + \Delta t \dot{u}|_t + \Delta t^2 \left(\frac{1}{2} - \beta \right) \ddot{u}|_t + \beta \ddot{u}|_{t+\Delta t} \quad (2)$$

$$\dot{u}|_{t+\Delta t} = \dot{u}|_t + \Delta t ((1 - \gamma) \ddot{u}|_t + \gamma \ddot{u}|_{t+\Delta t}) \quad (3)$$

In equations 1 through 3, M is mass, L is the sum of all Lagrange multiplier forces, P is comprised of forces, I is made of internal forces, t is time and $u, \dot{u}, \ddot{u}$ are position, velocity and acceleration.

Numerical damping parameters are given as

$$\beta = \frac{1}{4}(1 - \alpha)^2, \quad \gamma = \frac{1}{2} - \alpha \quad \text{and} \quad -\frac{1}{2} \leq \alpha \leq 0 \quad (4)$$

It is important to note that this system is *implicit* in the sense that displacement and velocity depend on information from both time t and $t + \Delta t$. [33]–[36] Abaqus Standard employs implicit modeling and defaults to employing Newton's Method to solve these

equations to ensure convergence, but Quasi-Newton Methods can also be chosen to save computational cost if the stiffness is fairly stable between iterations. [33]–[36]

2.1.2 Explicit Modeling

Abaqus Explicit relies on central difference integration combined with a “lumped” mass matrix which is mathematically diagonalized. Central difference first satisfies equilibrium at time t . Accelerations at time t are then used to calculate the velocity at time $t + \frac{\Delta t}{2}$ and displacement at $t + \Delta t$. In the central-difference integration theory, the next ($i + 1$ st) position of a given degree of freedom is

$$u_{(i+1)} = u_{(i)} + \Delta t_{(i+1)} \dot{u}_{(i+\frac{1}{2})} \quad (5)$$

where u is a displacement or rotation and i is an increment in an explicit step. Half step velocity is given as

$$\dot{u}_{(i+\frac{1}{2})} = \dot{u}_{(i-\frac{1}{2})} + \frac{\Delta t_{(i+1)} + \Delta t_{(i)}}{2} \ddot{u}_{(i)} \quad (6)$$

Acceleration is given by

$$\ddot{u}_{(i)} = (M)^{-1} (P_{(i)} - I_{(i)}) \quad (7)$$

M is the lumped mass matrix and $P_{(i)}$ is a vector of applied loads. $I_{(i)}$ is an internal force vector which is assembled from elemental contributions meaning a global stiffness matrix is not required to be calculated. The construction of the internal force vector is discussed

further in Stress-Strain Relations. This process is *explicit* because values of velocity and acceleration from the previous step are used to calculate the next increment. [33]–[36]

Belytschko et al concluded that explicit methods are well suited for complex contact relationships like the ones expected in this study. Egan et al concluded that for a “simple” model, explicit and implicit methods in Abaqus produced similar results, while the explicit methods took about 3.5 times longer to run. However, they suggest that as damage focused models become more complex, the explicit method is advantageous. Much of Egan’s follow-on research on countersunk bolted joints includes explicit modeling. Explicit solution methods also allow for a wide range of contact definitions and automatic or fixed time incrementation. [30]–[32]

Explicit methods are expected to be used throughout this study because of applications in similar research and the ability to handle complex contact and damage and failure scenarios. Similar to the findings of other researchers, in preliminary finite element modeling for this study, explicit methods have been much more likely to produce convergent results given complex contact and materials interactions. Because explicit methods will be employed where applicable for each theory proposed for this study, the implementation into an explicit analysis is discussed.

2.2 Stress-Strain Relations and Definitions

In the finite element method, first element strains are calculated using the strain displacement matrix, B_{sd} , as shown in Equation 8. This matrix is comprised of derivatives of admissible shape functions assumed for each type of element. This equation outputs total strain, ε^{tot} , using the strain-displacement matrix, and displacements.

$$\boldsymbol{\varepsilon}^{tot} = \mathbf{B}_{sd}\mathbf{u} \quad (8)$$

With total strain calculated, the elastic strain must be calculated so that stress may be determined. This requires a discussion on types of strains. Strain can be divided into two types. Elastic strain is deformation that is fully recoverable and is not dependent on the deformation history. Conversely, plastic strain is not recoverable and is dependent on the deformation history. The conclusion has been drawn that stress relies completely on elastic strain, and that elastic strain and plastic strain are additive to make total strain. Thus, in order to define elastic strain, ε^e , one need only know the plastic strain, ε^p , and subtract it from the total strain as seen below.

$$\boldsymbol{\varepsilon}^e = \boldsymbol{\varepsilon}^{tot} - \boldsymbol{\varepsilon}^p \quad (9)$$

The issue then becomes knowing the value of this plastic strain. Typically, an equivalent plastic strain is output as a result of the selected plasticity model and then expressed as components. These models are discussed more under Damage and Failure. With elastic strain known, the stresses can then be calculated using Hooke's Law by relating stress, σ , to elastic strain using the constitutive matrix, $\mathbf{C}$.

$$\boldsymbol{\sigma} = \mathbf{C} : \boldsymbol{\varepsilon}^e \quad (10)$$

In Equation 10, the double dot operator is the double inner product. $\mathbf{C}$ is a fourth order tensor and ε^e is a second order tensor, which results in a stress output as a second order tensor.

If damage (discussed in Damage and Failure Criteria) has been initiated, and damage is in the evolution stage, these damage stresses are updated here before the internal forces are calculated.

Finally, internal forces, I , necessary for equilibrium are determined by

$$I = \int_v B_{sd}^T \sigma dV \quad (11)$$

where B_{sd}^T is the transpose of the strain-displacement matrix and σ is stress determined from Equation 10. In an incremental finite element approach, these values of displacements, strains, and stresses are incremental values instead of total values. [33], [34], [37], [38]

An understanding of types and implications of different stresses and strains are also important to this work. Stress can be divided into two types, hydrostatic stress (also known as pressure stress) and deviatoric stress. Based on the assumption that metals are incompressible, hydrostatic stress does not cause deformation. The link is made with the idea that an incompressible material submerged in a fluid will have equal pressure (stress) in each of the three principal directions, but due to the incompressible nature, deformation does not occur. The naming conventions “hydrostatic stress” and “pressure stress” often differ by a sign since stress is assume to be tensile when positive and pressure is assumed to be compressive when positive. Hydrostatic stress is given as the average of the three principal stresses as

$$\sigma_{hyd} = \frac{1}{3} tr(\sigma) \text{ or } \sigma_{hyd} = \frac{1}{3} (\sigma_{kk}) = \frac{1}{3} (\sigma_{11} + \sigma_{22} + \sigma_{33}) \quad (12)$$

The component of stress that causes deformation is known as deviatoric stress. This is simply the total stress with the hydrostatic component subtracted out. It is given by,

$$\boldsymbol{\sigma}' = \boldsymbol{\sigma} - \sigma_{hyd} \text{ or } \sigma'_{ij} = \sigma_{ij} - \frac{1}{3}\delta_{ij}\sigma_{kk} \quad (13)$$

where σ' is the deviatoric stress, and δ_{ij} is the Kronecker delta defined in Appendix 2.

Comparable ideas and values exist for strains where hydrostatic strain is given by

$$\varepsilon_{hyd} = \frac{1}{3}tr(\boldsymbol{\varepsilon}) \text{ or } \varepsilon_{hyd} = \frac{1}{3}(\varepsilon_{kk}) = \frac{1}{3}(\varepsilon_{11} + \varepsilon_{22} + \varepsilon_{33}) \quad (14)$$

And deviatoric strain is given by

$$\boldsymbol{\varepsilon}' = \boldsymbol{\varepsilon} - \varepsilon_{hyd} \text{ or } \varepsilon'_{ij} = \varepsilon_{ij} - \frac{1}{3}\delta_{ij}\varepsilon_{kk} \quad (15)$$

Because three dimensional loading is complex, there needs to be some effective stress value to predict when a metal fails. Von Mises stress is widely employed in this case. Von Mises stress, which can be derived from deviatoric strain energy, is given by

$$\sigma_{vm} = \sqrt{\frac{3}{2}\boldsymbol{\sigma}':\boldsymbol{\sigma}'} \text{ or } \sigma_{vm} = \sqrt{\frac{3}{2}\sigma'_{ij}\sigma'_{ij}} \quad (16)$$

Here, the double dot operator is the double inner product of two second order tensors. Thus, the output is a scalar value. This is convenient because it gives a one dimensional equivalent value that can be compared to known experimental values of yield. For this reason, this is also referred to as an equivalent stress. [33], [34], [37], [38]

It is also helpful to have a related effective strain. This value, sometimes known as Von Mises Effective Strain, is given as

$$\varepsilon_{eff} = \sqrt{\frac{2}{3} \boldsymbol{\varepsilon}' : \boldsymbol{\varepsilon}'} \text{ or } \varepsilon_{eff} = \sqrt{\frac{2}{3} \varepsilon'_{ij} \varepsilon'_{ij}} \quad (17)$$

These effective stresses and strains are commonly written using a bar notation instead of a subscript notation as seen below. [33], [34], [37], [38]

$$\varepsilon_{eff} \rightarrow \bar{\varepsilon} \quad \text{and} \quad \sigma_{vm} \rightarrow \bar{\sigma} \quad (18)$$

In classical plasticity methods, once the yield stress has been exceeded, Abaqus employs associated flow to develop the components of plastic strain from the equivalent plastic strain in a uniaxially defined stress-strain curve. The components of plastic strain for the current increment are given as

$$\Delta \varepsilon_{ij}^p = \Delta \bar{\varepsilon}^p n_{ij} \quad (19)$$

where $\Delta \bar{\varepsilon}^p$ corresponds to the equivalent plastic strain in the user defined curve and n is the normal to the yield surface given as

$$n_{ij} = \frac{3}{2} \frac{\sigma'_{ij}}{\bar{\sigma}} \quad (20)$$

where σ'_{ij} is the tensor form of the deviatoric stress and $\bar{\sigma}$ is the von Mises Stress.

This relationship as illustrated by Mendelson is shown below. Mendelson uses σ_e (equivalent stress) to denote von Mises stress. [33], [39]

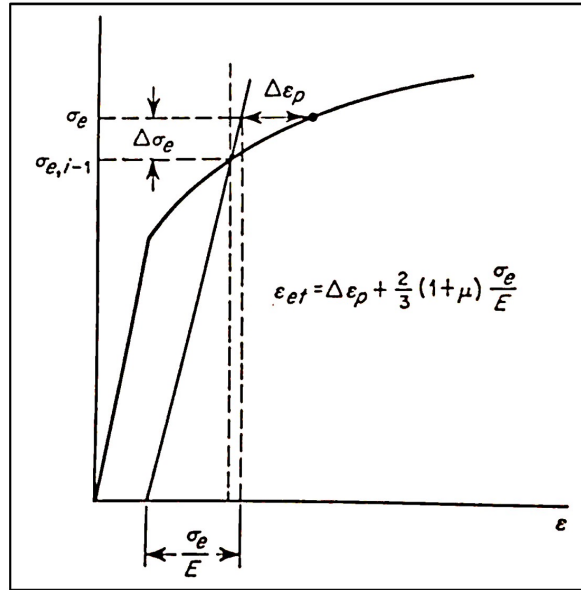


Figure 12: Incremental Stress and Plastic Strain [39]

An additional phenomenon that is important to understand is viscoplasticity. Viscoplasticity describes plastic deformation that is dependent on the strain rate. This is commonly experienced in impact settings but may occur in settings where local strain rates are dynamic. The presence of viscoplasticity is heavily dependent on the strain rate and materials properties. The significance of viscoplasticity in this study is presented later. [40]

2.3 Composite Modeling Methods

One of the first decisions to make when constructing a model containing composites is how to represent the laminate. Numerous methods have been studied, and may be placed into four main categories: Three Dimensional elasticity (3D), Equivalent Single Layer (ESL), Zig-Zag and Layerwise (LW). 3D theories treat the composite as a solid. Equivalent Single Layer modeling constructs a representative shell that mimics the properties of a given layup. ESL models effectively create a two dimensional solution to a three dimensional problem and are unable to properly represent the transverse stress

field. Zig-Zag models employ the superposition of a polynomial displacement field with a Zig-Zag function. These methods still have trouble with transverse stresses. Layerwise modeling constructs individual layers with at least one, but often many elements through the thickness of each layer. This method represents interlaminar relationships, discontinuities and stress concentrations well. For this reason, the layerwise approach is predominant in the study of bolted and countersunk joints. [41], [42]

A common approach in modeling composite layups is to define the solid as 90% of the actual thickness of the composite and to model the interlaminar region as the remaining 10% using cohesive or solid elements. Hereafter, this is referred to as 90/10 modeling. This allows finely tuned control over this region, and failure of these cohesive elements is indicative of delamination. This approach effectively forces the employment of a layerwise modeling. Layerwise modeling is typically computationally expensive, which can be overcome by efficient meshing practices and increasing computing power. As seen in Figure 11, Layerwise models enable characterization of complex failure around bolts. [32], [43]

Abaqus contains a “Composite Layup” tool that is a complex ESL-type system. It tends to model macroscopic composite shell parts well. However, it does not provide the detail and mesh refinement required when the focus is on progressive failure near a loaded discontinuity. [33], [34]

This study is expected to utilize a layerwise modeling approach made up of continuum shell elements representing 90% of the thickness while cohesive elements will be incorporated to represent the interlaminar matrix relationship. The details of cohesive elements are discussed later. Based on similar research and preliminary methods, this

method will be computationally expensive, but appears to be the only way to properly model the complex failure modes in bearing failure of a FML.

2.4 Geometric Nonlinearity

Due to the complex nature of the failure modes of these materials and the intention to characterize progressive failure, nonlinearity in both geometry and materials will need to be considered. Materials nonlinearity is covered under damage and failure criteria. Mathematically, the nonlinear terms in strain come from a Taylor Series Expansion employed during the derivation. The equation seen below including the quadratic terms is commonly known as Green Strain. Here, u terms are displacements and x^0 terms are the undeformed positions.

$$E_{ij} = \frac{1}{2} \left[\frac{\partial u_i}{\partial x_j^0} + \frac{\partial u_j}{\partial x_i^0} + \sum_n \frac{\partial u_m}{\partial x_i^0} \frac{\partial u_m}{\partial x_j^0} \right] \quad (21)$$

To simplify calculations, engineers often make an assumption of small displacements and rotations and only consider the linear terms. This results in what is known as the small or infinitesimal strain tensor. [44], [45].

$$\epsilon_{ij} = \frac{1}{2} \left(\frac{\partial u_i}{\partial x_j^0} + \frac{\partial u_j}{\partial x_i^0} \right) \quad (22)$$

Egan, McCarthy, and Frizzell concluded that in a double-lap countersunk joint, it was most efficient to omit geometric nonlinearity in implicit modeling to ensure convergence and computation efficiency. It is important to note that Egan et al were

working with a standard composite panel instead of a hybrid. Thus, they did not face large deformations and plasticity in metals. However, they noted that in an explicit model, the Abaqus default state of including geometric nonlinearity produced stable simulations.[31]

In the case of this study, the large deformations that induce failure are expected to cause large localized strains. ASTM recommends a 50% hole elongation. Thus, nonlinear strain components are expected to be significant in modeling this hybrid material and will be considered as the models are developed.

2.5 Damage and Failure Criteria

Damage and failure criteria drive modern-day modeling capabilities. In general, there are three phases of damage and failure: initiation, evolution, and element deletion. Damage initiation is typically thought of as the point at which the material has yielded. However, in many cases, the user is allowed to define the exact definition of damage initiation. Damage evolution typically describes how the material behaves through some deformation or degradation resulting in failure. The user is typically able to define failure, and then the user is allowed to delete the element when failure criteria are met if so desired. In the different criteria presented below for specific components, the method by which each criterion fits into these three phases is discussed. [33], [34]

2.5.1 Foil and Bolt Damage

Since this study is examining progressive failure, it is assumed that the fiber metal laminate and bolt will yield and local fracture may occur. Figure 13 shows a notional model of how Abaqus handles a ductile material. The user is able to use various methods to define, elasticity, yield, plasticity, damage initiation and damage evolution.

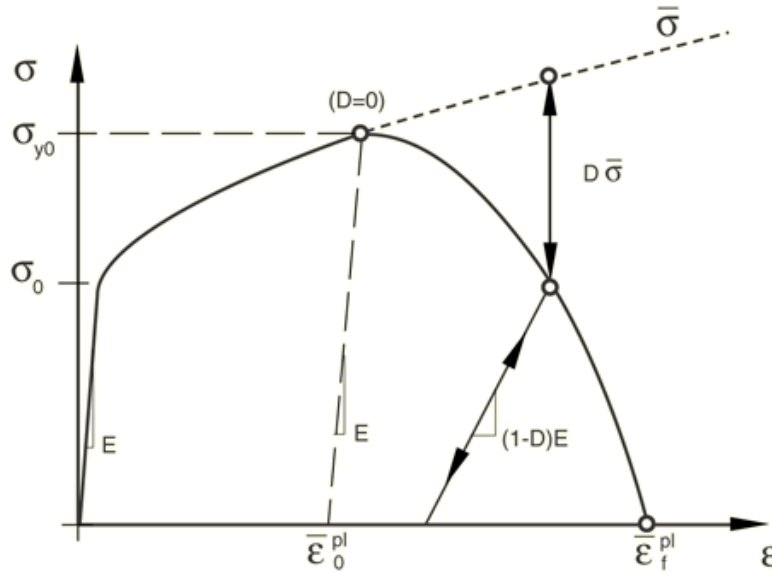


Figure 13: Stress-Strain and Damage Evolution of a Ductile Material [34]

There are numerous decisions to make about how to properly model each of these characteristics for a given material. For bearing testing, ASTM recommends a constant head speed of .05 in/min (or 2 mm/min) up to a 50% hole elongation, which is the measure of bearing strain. This equates to a .667/min (.011/s) average strain rate which is assumed to be quasi-static. Despite the quasi-static assumption, bearing failure in this system causes complex failure. Thus it is important to investigate whether or not viscoplasticity should be considered. Zubelewicz and Oliferuk and Lu et al showed that

stainless steels may demonstrate viscoplasticity even at quasi-static strain rates. This is shown by Zubelewicz and Oliferuk in Figure 14.

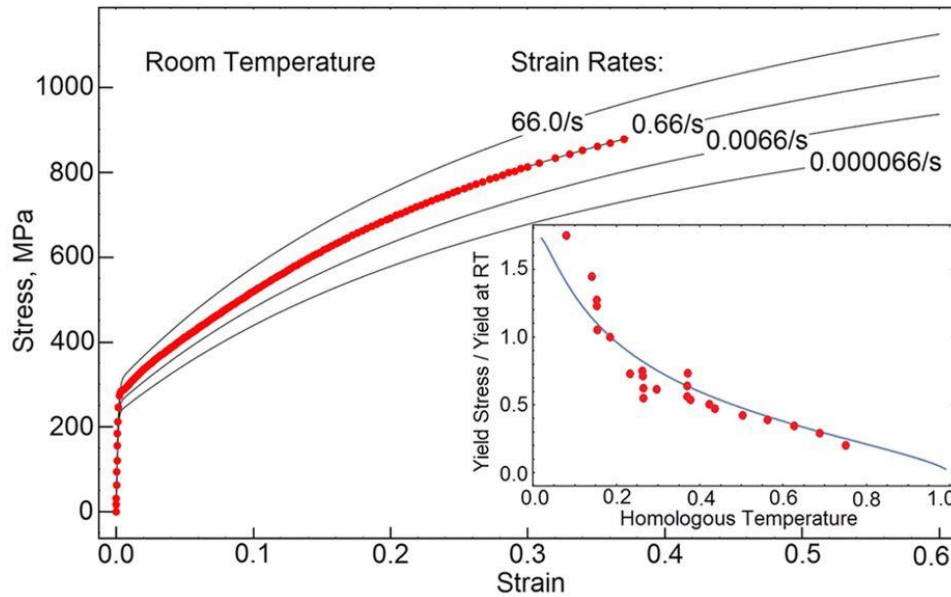


Figure 14: Strain Rate Dependence of 304 Stainless Steel [46]

When analyzing Figure 14, it is important to note that each data set represents a strain rate variation of two orders of magnitude compared to the next set. Despite the low strain rates, these are very large variations. In order to compare smaller strain rate steps, a comparison to Johnson-Cook plasticity was used. Johnson and Cook developed a model which is now widely used to represent viscoplasticity and temperature dependence in metals. Representative Johnson-Cook plasticity curves were plotted for the displacement rate expected in the study as well as an order of magnitude change up and down. [47] This is shown in Figure 15. A bilinear representation of the metal plasticity is included for reference. This shows that for a 10-fold increase or decrease in strain rate, in the expected strain range, one could expect about a 7 percent change in stress.

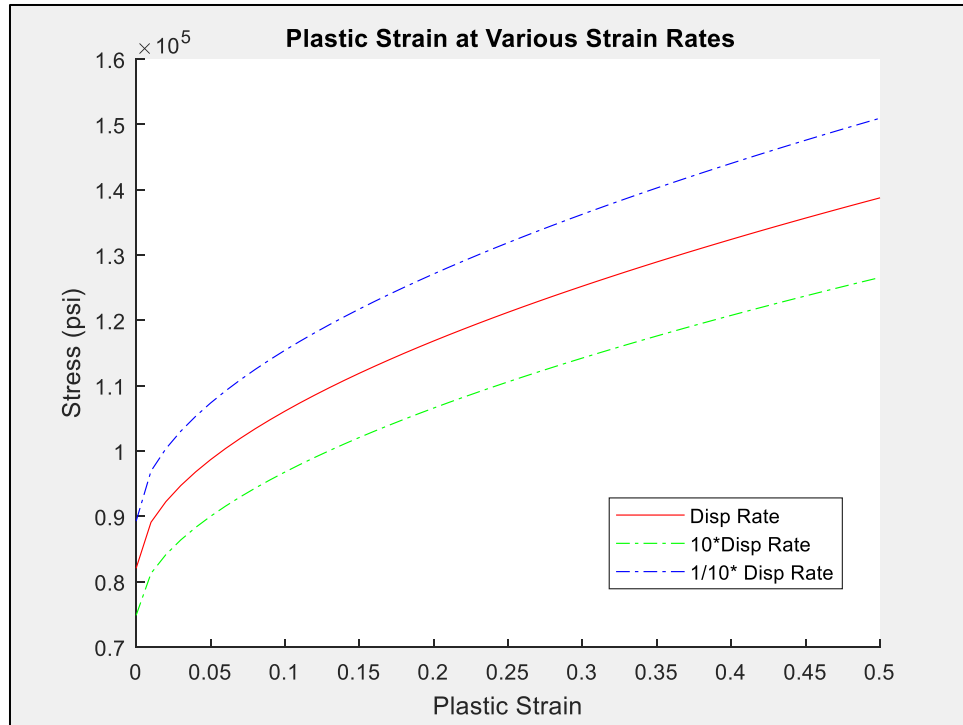


Figure 15: Johnson-Cook Estimate of Plastic Deformation

In order to understand the strain rate variations expected in this study, a finite element model was constructed to represent the foil using a bilinear elastic-plastic model. This will be presented more completely in the Results and Discussion Chapter. The conclusion of this preliminary finite element study is that the local strain rates around critical nodes are very close to the commanded bearing strain rate as seen in Figure 16. Furthermore, this configuration should be a worst case scenario. Because of this low deviation from the commanded strain, the stress value is only expected to vary by about 1 to 2 percent due to viscoplasticity. Thus, viscoplasticity will not be considered in this study.

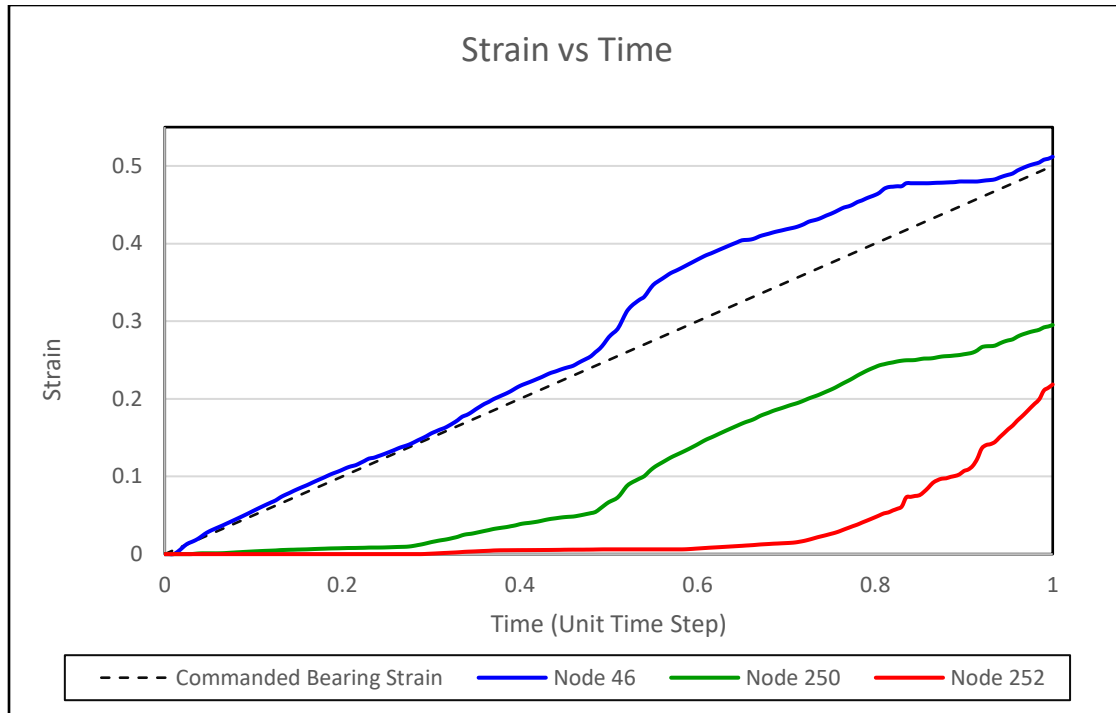


Figure 16: Strain at Critical Nodes

A secondary reason to discard viscoplasticity is that published Johnson-Cook parameters (or other viscoplasticity models) for the specific alloys and tempers used in this study do not seem to exist in available publications. To complicate this further, even if another alloy was representative, most parameters are developed at strain rates too high to be relevant to this study [37], [47]–[49]. Thus, parameters for similar alloys are still not applicable to this study. Testing could be conducted to determine these parameters, but the cost and timeline to perform those tests are well outside the scope of this study.

Since viscoplasticity is not required, plasticity in this study will be represented with a user defined tabular model. It is most likely that a bilinear model will be used as depicted in Figure 17. In most cases, the only data available for a given material are elastic modulus, yield stress, ultimate stress and elongation. This provides enough

information to define a line representing the elastic region and a line representing the plastic region. If more points are found to be available during the course of this research, they will be included to help smooth the plastic region. This user defined curve, as seen in Figure 17, is essentially defining a yield surface based on effective stress and strains.

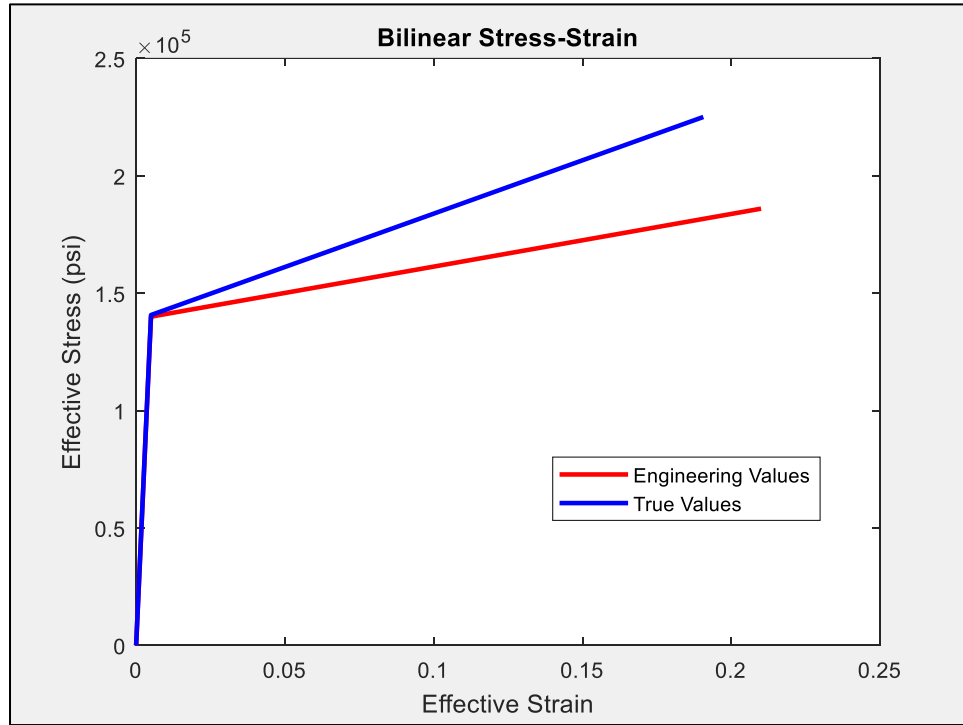


Figure 17: Bilinear Stress-Strain Model

2.5.1.1 Damage Initiation and Evolution

For materials exhibiting plastic deformation, damage is initiated at a user defined strain. Then, the damage variable, d , evolves with respect to time as

$$\dot{d} = \frac{L_c \dot{\bar{\epsilon}}^{pl}}{\bar{u}_f^{pl}} \quad (23)$$

where, L_c is the element characteristic length based on element geometry. For a first order element, it is the typical length of a line across an element. For a second order element, it is half of this length. For membrane and shell elements, it is calculated as the square root of area. Characteristic length is established for each element at the beginning of the analysis and is held as a constant reference throughout the analysis. $\dot{\epsilon}^{pl}$ is the equivalent plastic strain rate which is the strain rate with respect to the time step. $\bar{u}_f^{pl}$ is the user specified effective plastic displacement at failure. Buentello employed a value of $\bar{u}_f^{pl}$ as 1/6th of the characteristic length of critical elements [37]. Stress, which can be thought of as reduced load carrying capacity, is then calculated based on this damage variable as

$$\sigma = (1 - d)\bar{\sigma} \quad (24)$$

Thus, when the damage evolution variable is equal to one, the material is fully degraded. The user may choose to delete the element at a given value of d .

2.5.1.2 Explicit Step Implementation

In an explicit time step, the equivalent stress in an element is compared to the user input yield stress. If yield has not occurred, equivalent plastic strain is set to zero and the stresses are output. If yield has occurred, the equivalent plastic strain is calculated based on the user defined stress-strain curve, and damage initiation is checked by comparing the current equivalent plastic strain with the user input damage initiation strain. If no damage has occurred, the stresses and equivalent plastic strains are output. If damage has occurred, the damaged stresses are fed into the central difference approximation. If

element deletion is used, the element is deleted and the status is updated at the user defined value of the damage evolution variable.

2.5.2 Composite Damage

Numerous failure criteria have been developed, studied, and applied as composite materials have been developed. A 1996 FAA study performed an exhaustive study on 26 of the most common criteria. These criteria can be split into three main categories: limit, interactive, and separate mode criteria. Limit criteria consider stresses with corresponding materials properties. Interactions are not considered. Interactive criteria employ a polynomial involving all stress components. Separate mode criteria consider matrix and fiber failure separately. The FAA study cites a survey by Burk of AIAA members that showed that at the time, the majority of users employed the limit style criteria of maximum stress and maximum strain criteria. However this study goes on to say, *“At the lamina level, those criteria (such as the Hashin-Rotem criterion) which separate the fiber failure mode from the matrix failure mode are the most reasonable and accurate[50].”* Hashin failure has become ubiquitous in unidirectional composite analysis and is built-in to Abaqus. [51]

2.5.2.1 Composite Damage Initiation

2.5.2.1.1 Hashin Failure

Hashin criteria (Hashin and Rotem, 1973 [52]; and Hashin, 1980 [53]) describe damage initiation and failure in unidirectional fiber reinforced composites. These methods have been widely employed in the modeling of polymer matrix composites and have been successfully applied to bolted joints in fiber metal laminates. As discussed in

the previously mentioned NASA study, Hashin failure is beneficial because it separately represents failure modes in the fibers and the matrix. It also separates tensile and compressive properties and loading of each. [43], [51], [54]–[57]. Thus, four separate criteria are observed and each may be analyzed separately in finite element software. The Hashin damage initiation criteria, as employed in Abaqus, are given below.

Fiber Tension ($\hat{\sigma}_{11} < 0$):

$$F_f^t = \left(\frac{\hat{\sigma}_{11}}{X^T} \right)^2 + \alpha \left(\frac{\hat{t}_{12}}{S^L} \right)^2 \quad (25)$$

Fiber Compression ($\hat{\sigma}_{11} < 0$):

$$F_f^c = \left(\frac{\hat{\sigma}_{11}}{X^C} \right)^2 \quad (26)$$

Matrix Tension ($\hat{\sigma}_{22} \geq 0$):

$$F_m^t = \left(\frac{\hat{\sigma}_{11}}{X^T} \right)^2 + \alpha \left(\frac{\hat{t}_{12}}{S^L} \right)^2 \quad (27)$$

Matrix Compression ($\hat{\sigma}_{22} < 0$):

$$F_m^c = \left(\frac{\hat{\sigma}_{22}}{2S^T} \right)^2 + \left[\left(\frac{Y^C}{2S^T} \right)^2 - 1 \right] \frac{\hat{\sigma}_{22}}{Y^C} + \left(\frac{\hat{t}_{12}}{S^L} \right)^2 \quad (28)$$

In the equations above

X^T is the longitudinal tensile strength

X^C is the longitudinal compressive strength

Y^T is the transverse tensile strength

Y^C is the transverse compressive strength

S^L is the longitudinal shear strength

S^T is the transverse shear strength

α is a coefficient determining the shear stress contribution

$\hat{\sigma}_{11}, \hat{\sigma}_{22}, \hat{\tau}_{12}$ are diagonal components of the effective stress tensor $\hat{\sigma}$

This effective stress tensor is related to the true stress by a damage matrix, M (Equation 29). This diagonal matrix consists of damage variables for the fibers, matrix and shear, as recorded in Equations 30 and 31.

$$\hat{\sigma} = M\sigma \quad (29)$$

Where

$$M = \begin{bmatrix} \frac{1}{(1-d_f)} & 0 & 0 \\ 0 & \frac{1}{(1-d_m)} & 0 \\ 0 & 0 & \frac{1}{(1-d_s)} \end{bmatrix} \quad (30)$$

where d_f, d_m , and d_s are defined from the relationship

$$d_f = \begin{cases} d_f^t & \text{if } \hat{\sigma}_{11} \geq 0 \\ d_f^c & \text{if } \hat{\sigma}_{11} < 0 \end{cases}$$

$$d_m = \begin{cases} d_m^t & \text{if } \hat{\sigma}_{22} \geq 0 \\ d_m^c & \text{if } \hat{\sigma}_{22} < 0 \end{cases} \quad (31)$$

$$d_s = 1 - (1 - d_f^t)(1 - d_f^c)(1 - d_m^t)(1 - d_m^c)$$

In Equation 31, the specific damage variables are the damage evolution variables presented later (Equation 43). Prior to damage initiation, these damage variables are zero. Thus, M is the identity matrix, meaning effective stress is equal to the actual stress

until damage is initiated by one of the four criteria (Equations 25-28) reaching a value of one. As damage occurs, M grows, increasing effective stress. This allows failure in a single mode to affect other modes. [34], [53]

Hashin Failure criteria or components thereof are ubiquitous in composite modeling and have been employed in composite bearing and FML bearing simulation. Zhou et al employed Hashin Criteria to model fiber failure in single-lap multi-bolt joints [58] and Zu et al demonstrated good characterization of a FML bolted joint using Hashin [54].

2.5.2.1.2 Puck Failure

Puck proposed that a third matrix failure mode was possible. Namely, that the matrix may fracture at some angle in addition to the two modes allowed in Hashin Criteria.

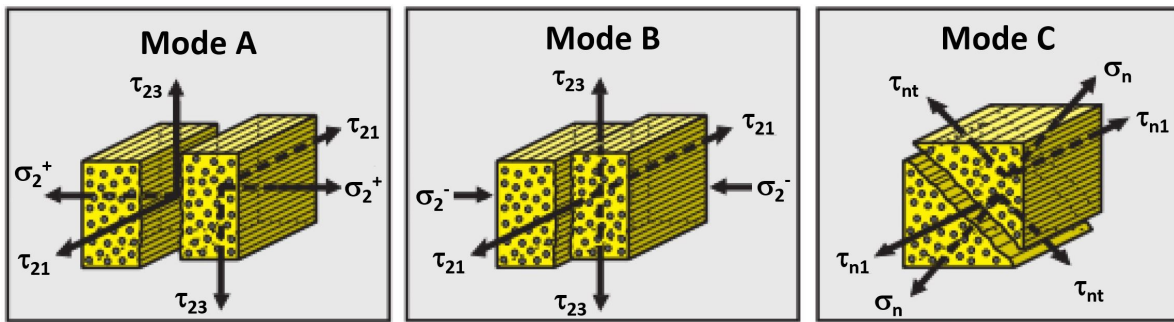


Figure 18: Possible Modes for Hashin (A and B only) and Puck Failure Criteria [59]

This seems like a vast improvement, but has been shown to not significantly change a failure prediction[59]. Working with Puck failure creates some complications. First, much of the related research is published in German. Second, this criterion has not been integrated into the standard Abaqus software package. It has been integrated into Abaqus

via subroutines, but this integration has been the complete focus of graduate research [60], [61]. Because of the complexities, Puck failure will likely not be considered for this study unless Hashin criteria are found to be insufficient.

In the last decade, a number of three-dimensional composite failure criteria have been studied by groups like Hundley et al [56], Zhou et al [58] and Donadon et al [57]. However, none appear to be widely accepted or applied to date and none are inherent to Abaqus.

2.5.2.2 Composite Damage Evolution

The proposed damage initiation criteria assume linear elastic behavior up until failure. Then, the load carrying capacity considered during damage evolution is also assumed to be linear. While these do omit nonlinearity, they are widely accepted as representing brittle composites well.

After damage has initiated via some criterion, such as Hashin, Abaqus uses a damage elasticity relationship where stress is related to strain through a damage elasticity matrix as seen in Equations 32 and 33.

$$\sigma = C_d \epsilon \quad (32)$$

C_d is the damage elasticity matrix given as

$$C_d = \frac{1}{D} \begin{bmatrix} (1 - d_f)E_1 & (1 - d_f)(1 - d_m)v_{21}E_1 & 0 \\ (1 - d_f)(1 - d_m)v_{12}E_2 & (1 - d_m)E_2 & 0 \\ 0 & 0 & (1 - d_s)GD \end{bmatrix} \quad (33)$$

$$D = 1 - (1 - d_f)(1 - d_m)v_{12}v_{21}$$

where E is the elastic modulus, G is the shear modulus, v is the Poisson's Ratio

d_f and d_m are the previously employed damage variables used in failure initiation described in Equation 43. [33], [34]

With the stress-strain relation developed, the damage variable evolves as seen in Figure 19.

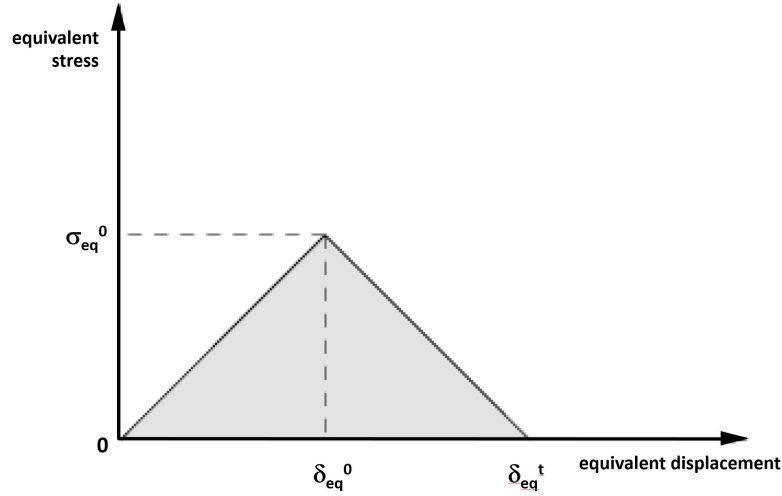


Figure 19: Equivalent Stress vs Equivalent Displacements

It is important to note that the values in Equation 32 are stress and strain matrices. Here, equivalent stresses and strains, as laid out in Equations 34-41, are derived from the components stress and strain matrices of the relationship in Equation 32. It is also important to note that the plot in figure represents only one mode of the four considered: fiber tension, fiber compression, matrix tension, and matrix compression. Equations 34-41 develop equivalent stresses and strains used to define damage evolution in each of the four described modes. These are calculated based on total stress and strain values accumulated at the current increment. These equations are also used in the calculation of the damage variable described later in Equation 43.

Fiber Tension ($\hat{\sigma}_{11} \geq 0$):

$$\delta_{eq}^{ft} = L^C \sqrt{\langle \varepsilon_{11} \rangle^2 + \alpha \varepsilon_{12}^2} \quad (34)$$

$$\sigma_{eq}^{ft} = \frac{\langle \sigma_{11} \rangle \langle \varepsilon_{11} \rangle + \alpha \tau_{12} \varepsilon_{12}}{\delta_{eq}^{ft} / L^C} \quad (35)$$

Fiber Compression ($\hat{\sigma}_{11} < 0$):

$$\delta_{eq}^{fc} = L^C \langle -\varepsilon_{11} \rangle \quad (36)$$

$$\sigma_{eq}^{fc} = \frac{\langle -\sigma_{11} \rangle \langle -\varepsilon_{11} \rangle}{\delta_{eq}^{fc} / L^C} \quad (37)$$

Matrix Tension ($\hat{\sigma}_{22} \geq 0$):

$$\delta_{eq}^{mt} = L^C \sqrt{\langle \varepsilon_{22} \rangle^2 + \varepsilon_{12}^2} \quad (38)$$

$$\sigma_{eq}^{mt} = \frac{\langle \sigma_{22} \rangle \langle \varepsilon_{22} \rangle + \tau_{12} \varepsilon_{12}}{\delta_{eq}^{mt} / L^C} \quad (39)$$

Matrix Compression ($\hat{\sigma}_{22} < 0$):

$$\delta_{eq}^{mc} = L^C \sqrt{\langle -\varepsilon_{22} \rangle^2 + \varepsilon_{12}^2} \quad (40)$$

$$\sigma_{eq}^{mc} = \frac{\langle -\sigma_{22} \rangle \langle -\varepsilon_{22} \rangle + \tau_{12} \varepsilon_{12}}{\delta_{eq}^{mc} / L^C} \quad (41)$$

where L^C is the characteristic length.

The $\langle \rangle$ symbol is the Macaulay operator which is defined for all $a \in \mathbb{R}$ real numbers as

$$\langle a \rangle = \frac{a + |a|}{2} \quad (42)$$

Finally, the damage variable for any given mode is given as seen in Equation 43. By referencing this variable, the user has a quick understanding and characterization of failure in each of the 4 modes in each element. If the damage variable exceeds 1, the element has failed. Abaqus will output the element STATUS which, for a selected mode, displays the elements that have failed with respect to this damage variable.

$$d = \frac{\delta_{eq}^f(\delta_{eq} - \delta_{eq}^0)}{\delta_{eq}^f(\delta_{eq}^f - \delta_{eq}^0)} \quad (43)$$

where, δ_{eq}^0 is the equivalent displacement at which the damage criterion was met and δ_{eq}^f is the fracture equivalent displacement for the given mode. Based on this damage variable, the user can decide to delete the element or keep it in place. [34], [62]

This study will most likely employ Hashin Failure to model the unidirectional composite since it is widely studied and applied, and inherent to Abaqus. Once damage occurs, damage evolution will be traced through the Abaqus evolution method described above. [63]

2.5.2.3 Explicit Step Implementation

In an explicit step, stresses from the stress-strain relations are used to calculate an effective stress tensor based on damage (Eq 29 and 30). If no damage has occurred, this is equal to the stress values from the stress-strain relations. Then, these effective stresses are run through the Hashin Failure criteria to predict failure (Eq 25-28). If any of the

criteria reaches a value of one, failure is initiated in that mode. If failure is not initiated, the elastically derived stresses and strains are output. If damage has occurred, effective stresses and strains are calculated (Eq 34-41) and then the damage evolution variable is calculated (Eq 43). Finally, the damaged stresses (Eq 32) can be calculated using the damage elasticity matrix (Eq 33). These damaged stresses are then used in the calculation of the internal elemental forces. If desired by the user, the element is deleted at a given value of d and the element status updated. [33], [34]

2.5.3 Cohesive Layer Damage

Cohesive layers are commonly employed in representing interlaminar matrix bonding (90/10 rule) in composites as well as adhesively bonded joints. Employment of cohesive layers are important to this study because it is expected to use the 90%/10% modeling technique where 10% of the thickness of each ply is modelled with cohesive elements to represent interlaminar matrix bonding. Additionally, the adhesive bonds between the stainless steel foils and composite layers are expected to employ cohesive elements. Figure 20 shows the constituent layup (discussed thoroughly later) with cohesive zones. The composite plies and their orientations are shown in blue, green and yellow. The cohesive layers representing the matrix are shown in red, while the cohesive layers representing the adhesive are shown in white. [33], [34]

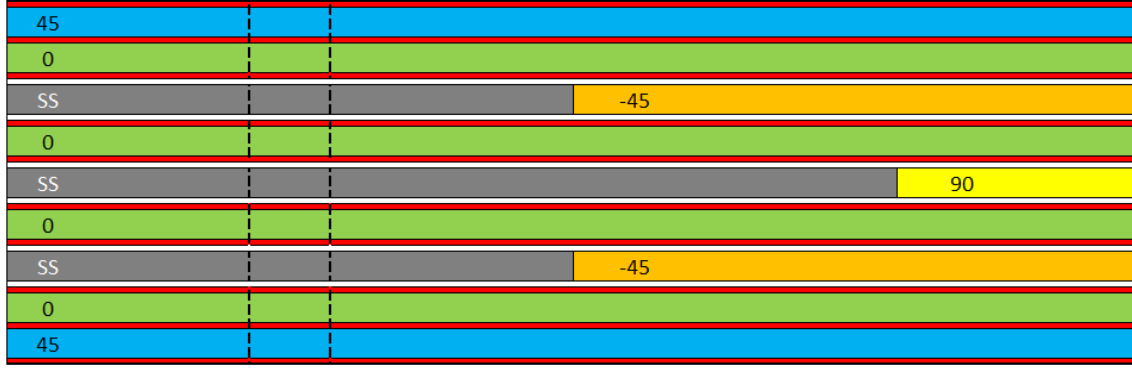


Figure 20: Constituent Layup Showing Cohesive Zones

Cohesive zones are typically modelled using traction-separation laws. Similar to the composite damage, cohesive damage involves damage initiation, damage evolution and element deletion.

2.5.3.1 Cohesive Damage Initiation

First, cohesive element strains are defined as

$$\varepsilon_n = \frac{\delta_n}{T_0}, \varepsilon_s = \frac{\delta_s}{T_0}, \varepsilon_t = \frac{\delta_t}{T_0} \quad (44)$$

where δ_n , δ_s , and δ_t are relative displacements between the top and bottom surface of the cohesive layer and T_0 is the original thickness of the layer. [33], [34]

Elastic behavior is given as

$$\begin{Bmatrix} t_n \\ t_s \\ t_t \end{Bmatrix} = \begin{bmatrix} E_{nn} & E_{ns} & E_{nt} \\ E_{ns} & E_{ss} & E_{st} \\ E_{nt} & E_{st} & E_{tt} \end{bmatrix} \begin{Bmatrix} \varepsilon_n \\ \varepsilon_s \\ \varepsilon_t \end{Bmatrix} = \mathbf{E} \boldsymbol{\varepsilon} \quad (45)$$

Figure 21 shows a typical traction-separation response. Below, t_n^0 , t_s^0 , and t_t^0 represent peak displacement in the normal, shear and transverse shear directions respectively.

Simlary, ε_n^0 , ε_s^0 , and ε_t^0 represent peak strains. These values are based on known materials properties. Traction and strain values without the naught superscripts are values measured throughout deformation. [33], [34]

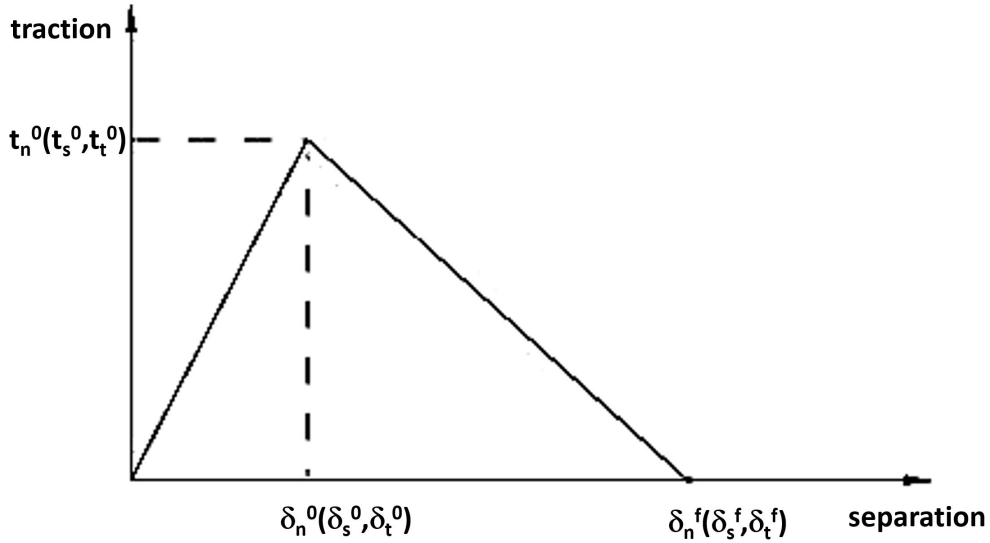


Figure 21: Traction-Separation Response [34]

Damage initiation, the positive sloped portion of Figure 21, is typically modelled using one of the following criteria: maximum strain, maximum stress, quadratic strain, quadratic stress. These are presented below. In all cases, damage is assumed to be initiated when the criteria is equal to one. The $\langle \ \rangle$ operator is the Macaulay operator as discussed. It is used here to show that pure normal compressive stress does not initiate damage in traction-separation methods. As the names imply, maximum stress and strain criteria take the maximum value in any component direction compared to its peak capacity input by the user. If at any time this value meets or exceeds one for any element, that element has sustained damage. These first two methods do not combine loading directions. The quadratic methods allow damage to grow quadratically and consider the combination of modes. [33], [34]

Maximum nominal stress

$$\max \left\{ \frac{\langle t_n \rangle}{t_n^0}, \frac{t_s}{t_s^0}, \frac{t_t}{t_t^0} \right\} = 1 \quad (46)$$

Maximum nominal strain

$$\max \left\{ \frac{\langle \varepsilon_n \rangle}{\varepsilon_n^0}, \frac{\varepsilon_s}{\varepsilon_s^0}, \frac{\varepsilon_t}{\varepsilon_t^0} \right\} = 1 \quad (47)$$

Quadratic nominal stress

$$\left\{ \frac{\langle t_n \rangle}{t_n^0} \right\}^2 + \left\{ \frac{t_s}{t_s^0} \right\}^2 + \left\{ \frac{t_t}{t_t^0} \right\}^2 = 1 \quad (48)$$

Quadratic nominal strain

$$\left\{ \frac{\langle \varepsilon_n \rangle}{\varepsilon_n^0} \right\}^2 + \left\{ \frac{\varepsilon_s}{\varepsilon_s^0} \right\}^2 + \left\{ \frac{\varepsilon_t}{\varepsilon_t^0} \right\}^2 = 1 \quad (49)$$

2.5.3.2 Cohesive Damage Evolution

To aid in calculation of damage evolution, Abaqus calculates an effective displacement as [64]:

$$\delta_m = \sqrt{\langle \delta_n \rangle^2 + \delta_s^2 + \delta_t^2} \quad (50)$$

This is rolled into the linear deformation based damage evolution variable as

$$D = \frac{\delta_m^f (\delta_m^{\max} - \delta_m^0)}{\delta_m^{\max} (\delta_m^f - \delta_m^0)} \quad (51)$$

where $\delta_m^{\max}$ is the maximum value of effective displacement through the deformation history. For the stress-strain methods, the user may also select an exponential damage evolution. Deformation based damage evolution can also be expressed exponentially as in Equation 52, where α is a material parameter. [33], [34]

$$D = 1 - \left\{ \frac{\delta_m^0}{\delta_m^{\max}} \right\} \left\{ 1 - \frac{1 - \exp\left(-\alpha \left(\frac{\delta_m^{\max} - \delta_m^0}{\delta_m^f - \delta_m^0} \right)\right)}{1 - \exp(-\alpha)} \right\} \quad (52)$$

Damage evolution can also employ various energy methods as opposed to the stress-strain methods above. These are convenient because many common tests are employed that yield these energies. Energy methods can also consider linear and exponential onset. The linear energy method employs the same basic equation as in Equation 51. However, in this case, the effective displacement at failure is defined as

$$\delta_m^f = 2G^C / T_{eff}^0 \quad (53)$$

where T_{eff}^0 is the effective traction at damage initiation and G^C is mixed mode fracture energy calculated differently in different methods. The calculation of this mixed mode fracture energy for various criteria is the subject of much research.

For exponential energy methods, Abaqus employs the following relation.

$$D = \int_{\delta_m^0}^{\delta_m^f} \frac{T_{eff} d\delta}{G^C - G_0} \quad (54)$$

where G_0 is the energy at damage initiation.

These selected damage variables feed into the following set of equations from which Abaqus calculates the properties of the cohesive layers at the step being considered.

$$t_n = \begin{cases} (1 - D)\bar{t}_n & \text{for } \bar{t}_n \geq 0 \\ \bar{t}_n & \text{otherwise (no damage to compressive stiffness)} \end{cases} \quad (55)$$

$$t_s = (1 - D)\bar{t}_s$$

$$t_t = (1 - D)\bar{t}_t$$

where $\bar{t}_n$, $\bar{t}_s$, and $\bar{t}_t$ are the stress components assuming no damage. [33], [34]

Figure 22 shows how energy methods are employed in traction-separation laws in Abaqus. In essence this is a multidimensional version of the plot shown in Figure 21. Here the Benzeggagh-Kenane (B-K) criterion is being used to calculate G^C . This model is geared toward situations where the critical fracture energies in both shear directions are expected to be the same. The B-K criterion are given as

$$G^C = G_n^C + (G_s^C - G_n^C) \left\{ \frac{G_s}{G_T} \right\}^\eta \quad (56)$$

where

$$G_S = G_s + G_t \text{ and } G_T = G_n + G_s \quad (57)$$

The user should input the critical energies G_n^C and G_s^C and the material parameter η . [33], [34]

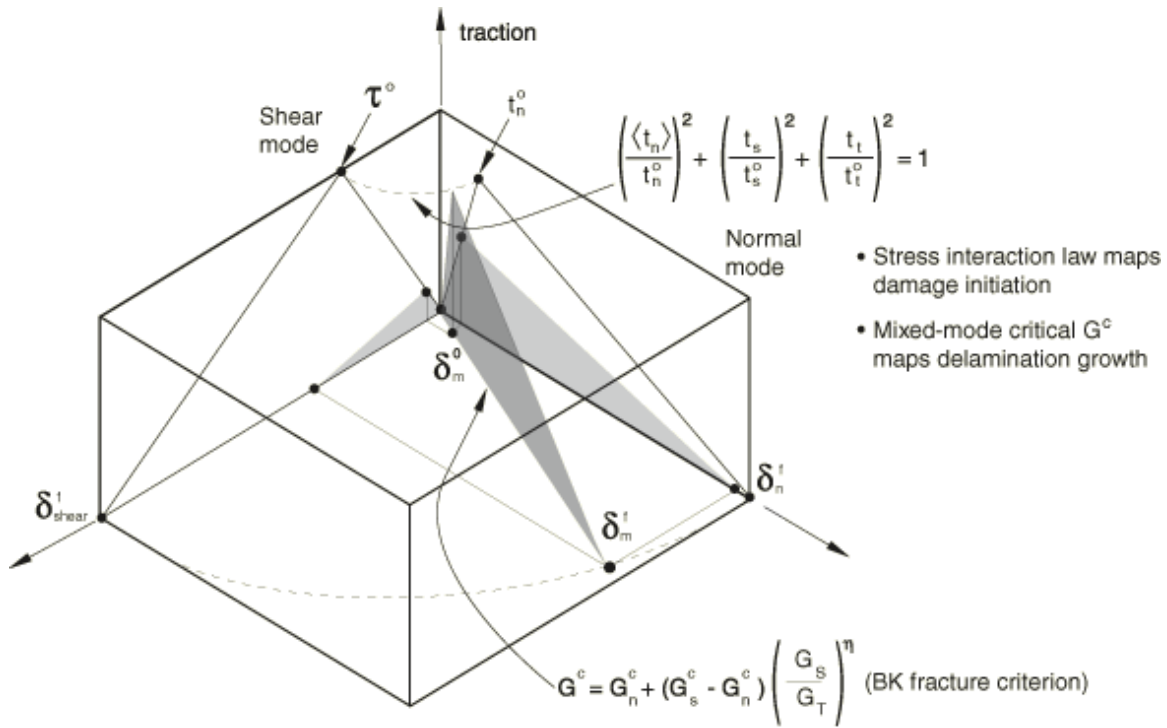


Figure 22: Mixed Mode Response In Cohesive Elements [34]

Figure 22 plots the traction mode along the 2 axis and the shear mode along the 1 axis. The damage initiation and progress are plotted in the third direction. The plots lying on the 1-3 plane represent the shear mode independently and the plot in 2-3 plane represents the tensile mode independently. The three dimensional plot represents the mixed mode damage criterion. [33], [34]

The use of cohesive elements is now inherent in much of the current composite fastener research. Frizzel et al employed an optimized cohesive layer shape with Benzeggagh–Kenane criteria in GLARE to model delamination at the critical area near the bolt while preserving computational efficiency. [65]

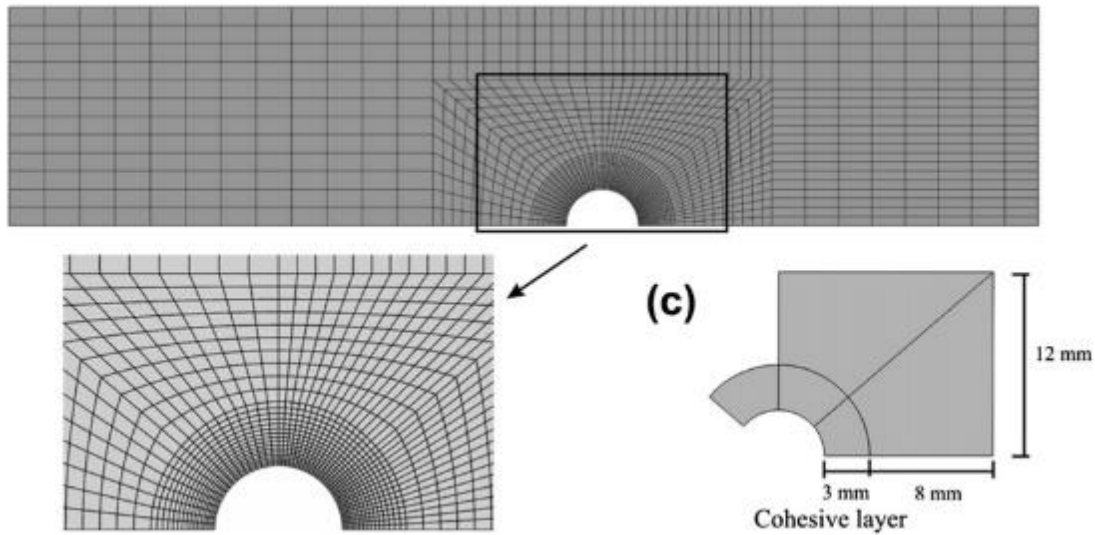


Figure 23: Optimized Cohesive Layer [65]

Given the 90/10 layup presented in Figure 20, this study will employ a quadratic nominal stress (Eq 48) damage initiation and will likely compare a linear deformation based damage evolution with a linear energy method (Eq 51,53) with B-K criterion (Eq 56,57, Figure 42). These cohesive layers will be used to represent the interlaminar matrix relationship and the adhesive. The method that best represents the experimental results in Phase 1 will most likely be continued throughout the study.

2.5.3.3 Explicit Step Implementation

In an explicit step assuming quadratic nominal stress damage initiation and a linear deformation damage evolution (Eq 51), cohesive strains are first calculated by comparing the top and bottom surfaces of the cohesive layer (Eq 44). Then, the cohesive stresses are calculated using a standard elastic relationship (Eq 45). Next, the quadratic nominal stress damage initiation value is calculated (Eq 48). If this value is less than one,

there is no damage and the stresses and strains are output. If damage is initiated (value greater than or equal to one), the effective displacements are calculated (Eq 50). These are then used to calculate the damage evolution variable (Eq 51). The damaged stresses are calculated (Eq 55) and sent to the internal force calculation. If selected, fully degraded elements are removed and the status is output. [33], [34]

2.6 Contact and Interaction

Contact is expected to play a crucial role in this study. Prior research and preliminary testing have shown that most of the failure in the structure should be expected to occur near the interaction of the bolt and the FML. The importance of this relationship is increased in the single shear and countersunk cases where fastener bending increases complexity.

In order to produce the most relevant multi-part models, one must consider how the parts and surfaces relate to one another when they contact and interact. Since explicit modeling methods are advantageous in damage focused models, the methods of contact and interaction definition discussed here are all compatible with the Abaqus explicit modeling environment.

Contact is generally described in pressure-overclosure relationships, where pressure is a function of the distance between two surfaces. If there is a measurable distance between two surfaces, this is known as clearance. The condition of two surfaces moving toward one another to remove this clearance is known as closure. Thus, the tendency to for two surfaces to penetrate one another after closure is known as

overclosure. This pressure-overclosure relationship is also commonly referred to as the contact stiffness. [33], [34], [66]

Contact is still a widely explored field and the exact calculation methods vary greatly depending on software package and pressure-overclosure relationship. Thus, a general process is presented here. [33], [34], [66]

A basic method of representing the pressure-overclosure relationship is called “hard” contact as seen in Figure 24. This assumes no pressure until the surfaces are in contact at which point any possible pressure is allowed. This however, creates a numerical discontinuity.

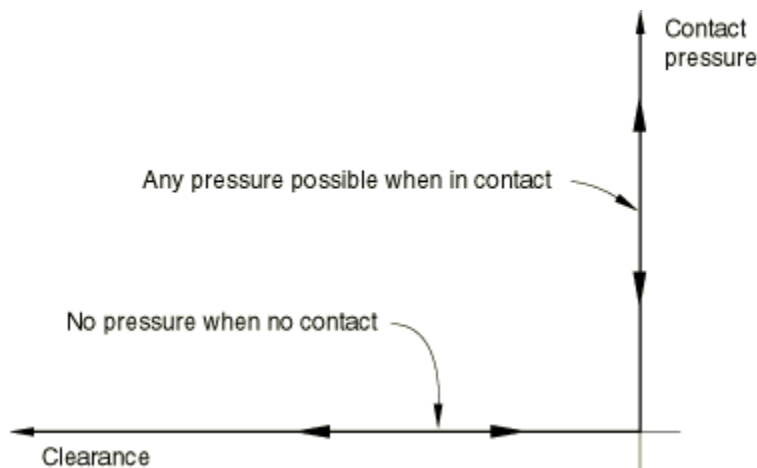


Figure 24: Hard Contact [67]

Thus, it is computationally advantageous for a user to be able to control more details in the pressure-overclosure relationship. The most common method is known as the penalty method.

Figure 26 shows a nodal penetration into an element with both normal and tangential components. The potential energy of the pressure-overclosure relationship is minimized using the first variation of potential energy as

$$\delta \Pi_C = f_n \delta g_n + f_t \delta g_t = k_n g_n \delta g_t + \text{sgn}(g_t) \mu_d k_n g_n \delta g_t \quad (58)$$

where

f_n is the normal force

f_t is the tangential force

k_n is the normal penalty term

k_t is the tangential penalty term

g_n is the normal penetration

g_t is the tangential penetration

μ_d is the coefficient of friction

sgn is the sign operator which takes the sign of the parenthetical quantity

From this relationship, the normal and tangential forces emerge as

$$f_n = k_n g_n \quad (59)$$

$$f_t = -\text{sgn}(g_t) \mu_d k_n g_n \quad (60)$$

These forces are distributed to nodes as external nodal forces. Abaqus allows the user to define the coefficient of friction, μ_k , with respect to slip rate and temperature. The slip

rate definition allows the user to tailor the relationship based on static and dynamic frictional relationships. [37], [66], [68], [69]

Abaqus allows the user to define contact as general contact or by using contact pairs. General contact is aptly named in that it handles most general contact situations well by allowing equal interpenetration between parts. Contact pairs employ a master-slave relationship in which there is no resistance to the master surface penetrating the slave surface, but there is resistance to the slave surface penetrating the master surface. In some modern version of software, the “master” surface is referred to as “surface 1” and the “slave” surface is referred to as “surface 2” or the “accessory” surface. These are typically useful in more complex contact situations such as the interaction between a part and a fastener. [34], [40]

Egan et al used the contact pair definitions for master-slave relationships in composite-composite single-lap joints. Here, the countersunk fastener is the dominant master surface [31].

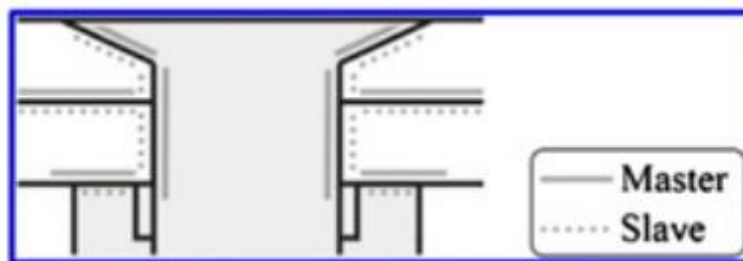


Figure 27: Master-Slave Surface Definitions (Egan) [31]

This method was also adopted by Liu et al and applied to a composite-steel lap joint. [62]

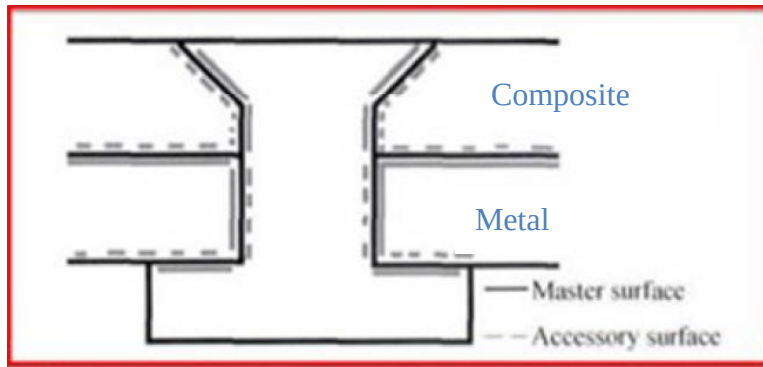


Figure 28: Master-Slave (Accessory) Surface Definitions (Liu) [62]
(Blue notation by author)

Kolks and Tserpes employed a similar contact pair relationship and employed penalty methods which produced representative results in a carbon fiber-titanium bolted joint [25].

When using contact, there are special considerations when defining meshes and selecting contact pair surfaces. General convention is to make the stiffer surface master and model it with a coarser mesh. This prevents the master from being able to penetrate the slave surface during the initial kinematic calculation. Figure 29 shows a properly defined mesh relationship on the left and a poorly defined mesh on the right.

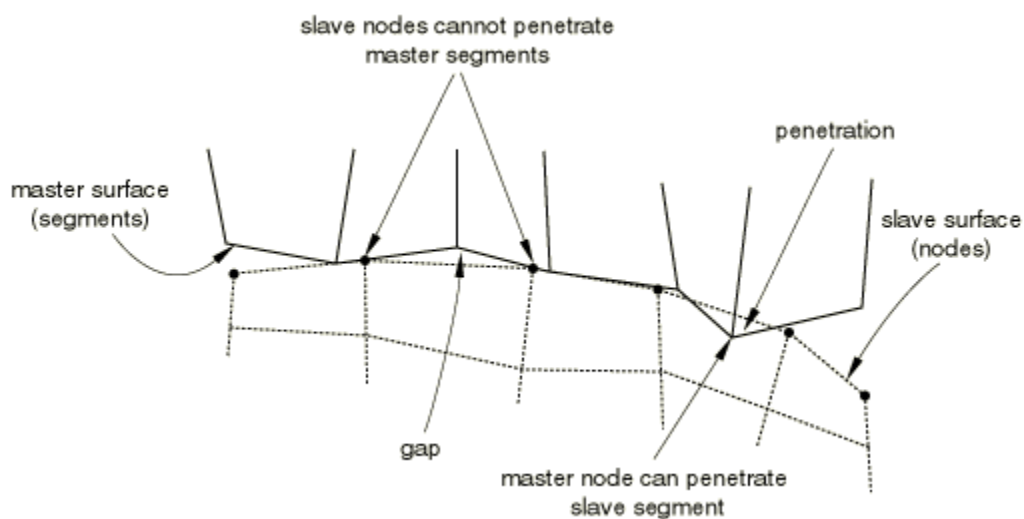


Figure 29: Master-Slave Discretization [33]

Since this study is expecting to see damage and failure, it is important to understand what happens to the contact relationship when an element fails. Abaqus defaults to applying the same contact relationships to elements newly exposed after an original element is removed due to failure.

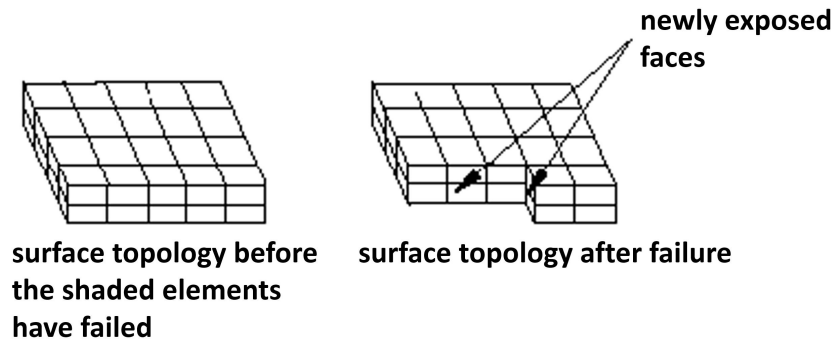


Figure 30: Eroding Contact Surface

In preliminary modeling for this study, general contact has been used throughout. As modeling gets more complex, general contact will likely still be employed between the layers of composite. However, a contact pair relationship similar to that used by Egan et al and Liu et al will be employed to represent the bolt/FML relationship with a linear penalty enforcement. [25], [31], [33], [34]

2.6.1 Explicit Step Implementation

In general, in an explicit step, the kinematic state of the system is first calculated without considering contact. Then, penetrations are mathematically sought out. Next, based on the penalty stiffness, a local solver is used to minimize the potential energy of the system as the penetrations are corrected so that the contact constraints are enforced.

The results of the minimization are restoring forces which are distributed as external nodal loads and corrected nodal positions. [33], [34], [66]

2.7 Residual Stress and Thermal Expansion

There are multiple sources of residual stress that can be considered in Fiber Metal Laminates. Two are most evident: residual stresses from manufacturing the foil and residual stresses due to mismatched coefficients of thermal expansion (CTE) and high cure temperature.

During metal forming procedures, residual stresses are imparted into materials. This can be seen in many machining operations, where a material deforms during machining due to the release of this residual stress. Despite the presence of these stresses, they are expected to play much less of a role in this study compared to the thermal stresses. This is discussed further in the Results and Discussion chapter.

Thermal stresses are imparted during cure cycles. Then, aircraft often see large temperature cycles on each flight which can range from over 100°F on the ground to around -50 °F at altitude. Prussak et al showed the formation of residual stress in FML as seen in Figure 31. This plot shows the progression of temperatures and resultant strains through the cure cycle. At time t_0 just before the cure cycle begins, the samples are at ambient room temperature, T_r . Then, as temperature increases and the matrix begins to cure, the materials are joined by the matrix. This cure point is known as the stress free temperature, T_{sf} . Then, as the cure temperature increases, the compressive and tensile stresses are induced. Finally, as the material cools and exits the cure cycle, the final

residual stresses and strains are realized as the comparison between the material at the stress free temperature, T_{sf} , and the environmental ambient temperature, T_r .

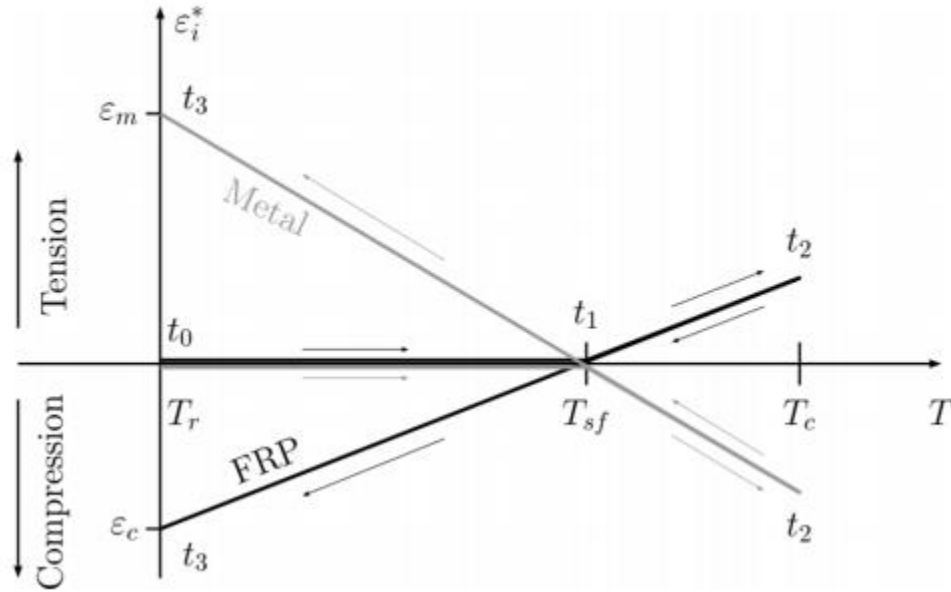


Figure 31: Formation of Residual Stress In Curing [70]

The process described above and shown in Figure 31 can be exacerbated when a large difference in coefficient of thermal expansion exists between materials. Table 1 shows the CTE for some of the materials referenced throughout this document. It is noteworthy that GLARE constituents have closer CTE than those FML with carbon fiber constituents, but the differences are on the same order of magnitude. [70]

Table 1: Coefficients of Thermal Expansion

Material	Approximate Coefficient of Thermal Expansion ($10^{-6}/^{\circ}\text{F}$)
Aluminum Alloys [71]	13
304 Stainless Steel [71]	9.6
Titanium Alloys [71]	5
Hextow IM7 Carbon Fibers [72]	-0.36
Glass Fibers [73]	5

Despite the existence of these cure induced stresses, it seems that they do not have major effects on the structures in operational or experimental settings. Frizzell stated that there were no negative effects from cure induced stresses in a GLARE repair to the C-5 Galaxy [21]. Hosseini-Toudeshky et al showed that varied residual stresses due to different cure cycles did not significantly affect fatigue crack propagation [74]. Da Costa et al showed that up to 2000 temperature cycles to simulate flight did not have significant effects on the tensile strength or interlaminar shear strength of a FML [75]. Fink et al predicted variation in thermal stress due to cure temperatures would have no significant effect on bearing strength in a carbon-titanium FML [27]. While there are studies to show that cure induced stresses in a carbon-steel FML are higher, as one might expect due to the wider difference in CTE [76], there is no evidence to show that these stresses are detrimental. Also there does not seem to be significant work on how to address these thermal stresses, if they indeed need to be addressed at all. Smart cure cycles have been proposed, but this has not been tested with the composite employed in this study [70]. Furthermore, initial STRATUS samples have shown a marked increase in bearing strength over control samples (as discussed further in Results and Discussion).

Thus, the main question to be asked with cure induced stresses is, how much does the consideration of thermal stresses drive the ability to model bolted joints in FML properly?

Hausmann et al proposes an analytical solution to predict thermal stresses in a fiber metal laminate in the following set of equations [76].

$$\sigma_M = \frac{\Delta T(\alpha_C - \alpha_M)}{\frac{1}{E_C} \cdot \frac{1 - \nu_C}{\nu_C} + \frac{1}{E_M}} \quad (61)$$

$$\sigma_C = -\frac{1 - \nu_C}{\nu_C} \cdot \sigma_M \quad (62)$$

Where

σ_M is the thermal stress in the metal

α_M is the metal coefficient of thermal expansion

E_M is the metal elastic modulus

σ_C is the thermal stress in the composite

α_C is the composite coefficient of thermal expansion (longitudinal)

E_C is the composite elastic modulus

ΔT is the temperature change (negative for cooling)

ν_C is the volume fraction of the composite with respect to the thickness

Using this calculation method, the thermal stresses in the layup proposed in the study are calculated to peak at 44.6 ksi tensile in the metal and 22.6 ksi compressive in the composite, which equate to almost 32% of the yield stress of the metal and almost 9% of the ultimate stress of the composite. While these are substantial, they are high estimates in that they do not consider an adhesive layer between the composite and foil. The adhesive planned for use in this study is expected to be compliant enough that it absorbs some of these stresses as strains.

The plan for this study is to conduct a finite element study during Phase 2 to evaluate the impacts and importance of including these cure stresses in the model. Stresses will be calculated by constructing a model of the constituent layup and evaluating a thermal step. Then, these stresses will be carried into the overall model by either employing a thermal step before the displacement step or by including the residual stresses as boundary conditions. A model that does not include these stresses will also be created. Finally, these models will be compared to the experimental results. The model that best simulates the experimental results will be carried through the rest of the modeling efforts. Heat transfer and heat produced from plasticity will not be considered due to low strain rates and small metal volume fractions.

2.8 Consolidation of Theory

It is important to understand how the modeling methods explored to represent this scenario will relate to one another computationally. The flow chart in Figure 32 shows the logic flow during each iteration of an explicit dynamic analysis. Logically, at the beginning of each iteration, $time = t$, accelerations, velocities, and locations as well as equivalent plastic strains are coupled with user input materials properties, parameters for failure criteria, and boundary conditions. These values are first used to develop stress-strain relationships. These stress-strain relationships feed the central difference system of equations and are input into each failure criterion. The failure criteria develop the stress values experienced in the material and process the damage initiation, evolution and element removal. Meanwhile, the contact algorithm takes in the updated kinematic state from the central difference calculation and first allows penetration. Once these contact or

penetration points are located, the pressure-overclosure relationship is solved using a local solver which minimizes potential energy. Pressures are distributed as external forces to nodes, and element positions for the current iteration are updated. Final outputs are an updated kinematic state, updated equivalent plastic strain, stresses, strains, and element status. [33], [34], [37], [66]

For space efficiency, equations throughout the paper are referenced by the flow chart. Dashed outlines indicate a logical decision point.

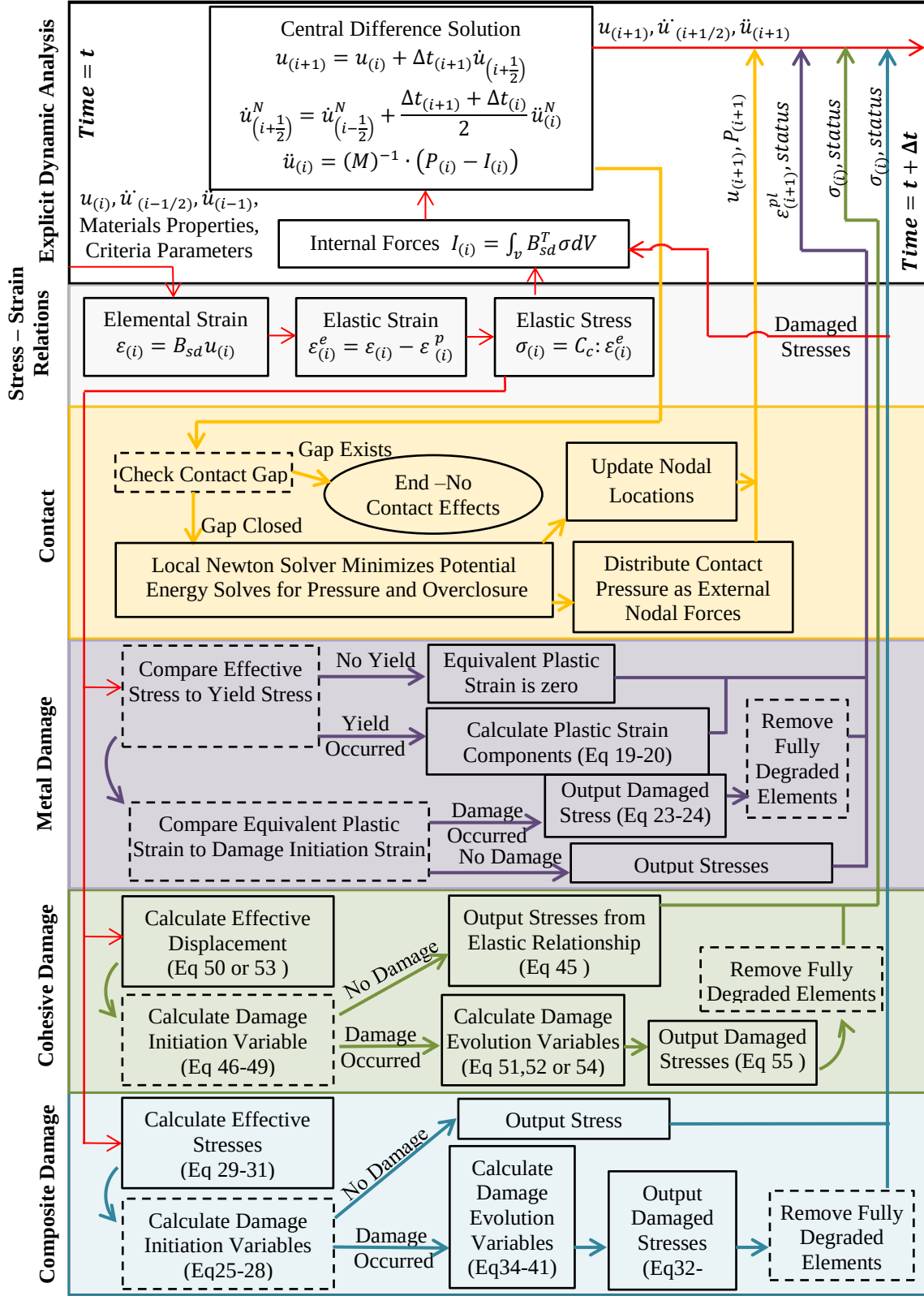


Figure 32: Finite Element Method Logic Flow Chart [33], [34], [37]

III. Methodology

This chapter outlines the methodology for approaching this area of research and producing useful results and conclusions. For clarity, this chapter is intended to outline the plan and explain decisions made based on the theories discussed in Chapter II.

3.1 Material and Process Selection

The selection of 304 stainless steel for the foils was made by engineers at AFRL/RX. This was largely due to the inherent toughness of stainless steel coupled with its corrosion resistant properties. Also, it avoids the galvanic corrosion effects of introducing aluminum into a carbon fiber structure. For many alloys, titanium has a higher yield strength, but stainless steels demonstrate more ductile failure enabling detection through inspection or vibration instead of catastrophic failure [77]. The IM7/977-3 unidirectional carbon fiber was selected because it has been studied in depth by AFRL in the Composite Airframe Life Extension programs [28]. This incorporates Hexcel IM7 carbon fiber pre-impregnated with CYCOM 977-3 thermoset epoxy resin by Solvay [72], [78].

The metal preparation process is that recommended by AFRL/RX. This process has been proven by Kondash et al to provide a consistent surface roughness for bonding while limiting distortion in the material [79]. While much of the research has been done on aluminum bonding, where this process produces an average surface roughness of $24\mu in$, it has been proven to produce a similarly consistent surface on stainless steel with only slightly reduced surface roughness ($19\mu in$). This grit-blast procedure is followed by a Sol-Gel solution. Finally, the foil is coated with a primer to seal and stabilize the

surface. During the process, technicians use a sacrificial “traveler panel” that accompanies the main panels being processed. This allows engineers to quantify and quality check the processes without risk of contaminating the main panels. Figure 33 shows a main panel (left) and traveler panel (right) just after application of primer.



Figure 33: Stainless Steel Priming

The adhesive, AF191U, is a film adhesive made by 3M commonly used for composite metal bonding. This material is 2.5 mils thick and is unsupported, meaning it contains no fibrous support material [80]. Hi-Lok fasteners are a standard removable fastener in aircraft.

3.2 Phased Testing and Analysis

3.2.1 Phase 0: Test Sample Preparation

Before any experimentation can be performed, a great deal of preparation is required. The testing and analysis plan presented here was developed with the help of engineers and other experts at AFRL/RQ, AFRL/RX and AFIT. This fed into an extensive contracting and scheduling effort due the long lead time of certain items, finite shelf lives of items, and fairly large quantities of diverse hazardous materials required. Dozens of man hours of material preparation and fabrication are required to produce these test articles. A schedule for this work is seen in Appendix 3.

3.2.2 Phase 1: Bolted Double-Shear Progressive Damage Study

The first test phase will examine progressive failure of control and hybrid STRATUS coupons in a bolted double shear configuration as seen in Figure 34.

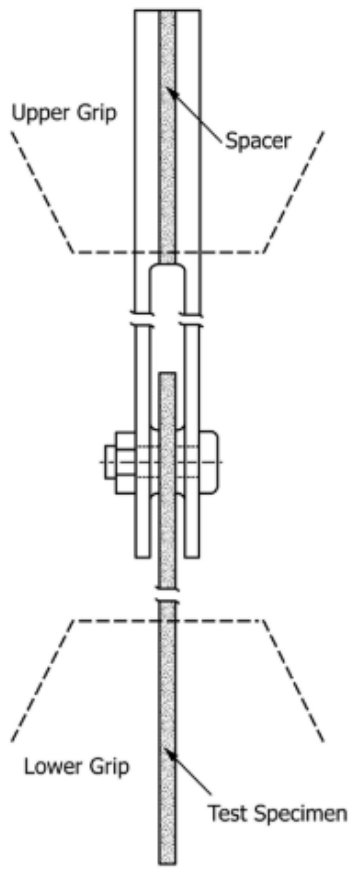


Figure 34: ASTM D5961 Procedure A Fixture Assembly

The baseline composite will be the IM7/977-3 employed in the Composite Airframe Life Extension Project 2. In the [44/44/12] family, the layup sequence is $[45/0/-45/0/90/0/-45/0/45]_3$. This 27 ply composite should have a final thickness of 0.1431 in. This layup is chosen since it is well studied in the CALE programs and is the thinnest possible layup that will allow a countersunk fastener for later phases. The dimensions for these samples are compliant with ASTM D5861 and D6873 and are shown in Figure 35. This type of coupon will be used for each phase. The coupons will be fabricated in 8" by 12" panels as detailed in the Fabrication section. Then the coupons are cut from the panels as seen in Appendix 8.

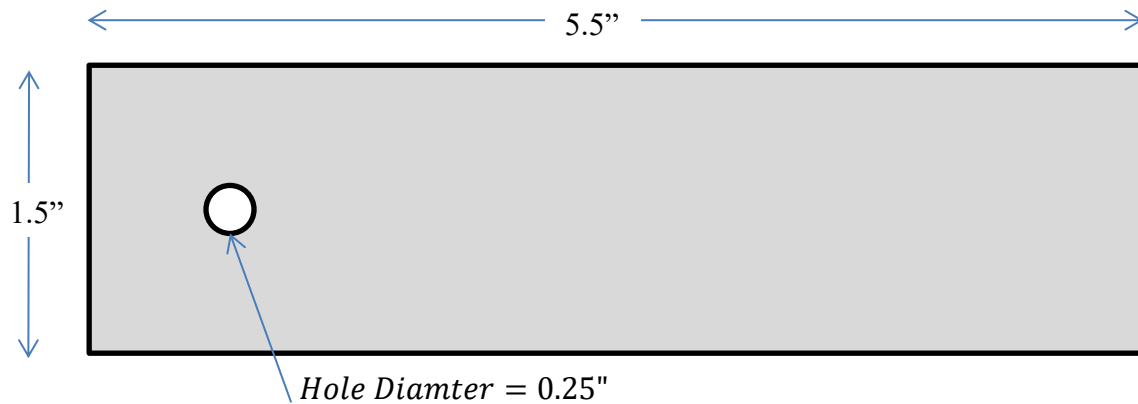


Figure 35: Sample Dimensions

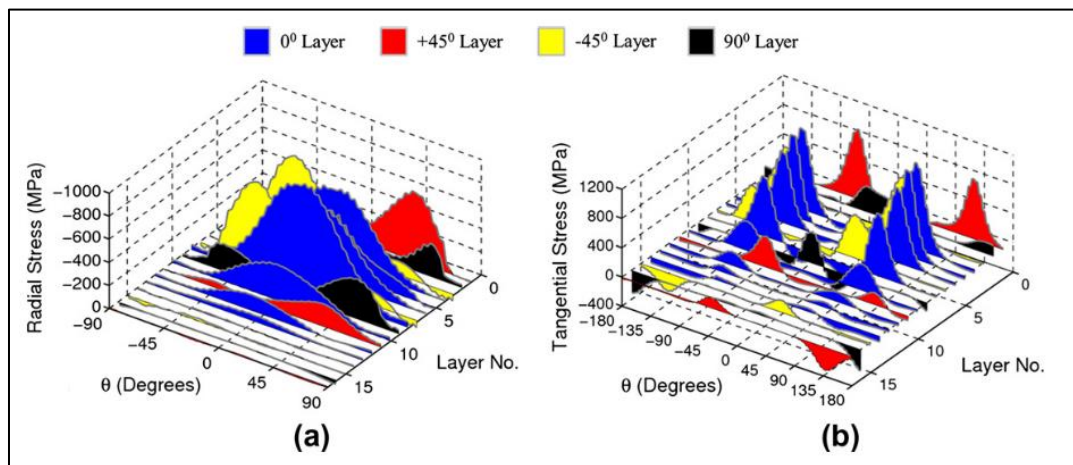


Figure 36: Stress Distributions at Countersunk Hole: (a) Radial (b) Tangential [31]

As seen in Figure 36, Egan et al showed that the 0° plies carry the majority of the bearing load at a fastener site, while the 45° and 90° plies carry little of the load [30], [31]. Thus, it is most structurally efficient to leave the 0° layers intact and replace the others with stainless steel foils. Only the internal layers of the constitutive layup

sequence are replaced with stainless foils resulting in the following layup sequence:
 $[45/0/SS/0/SS/0/SS/0/45]_3$. The constitutive layup is employed such that modeling efforts can first be focused on this 9-ply structure and then expanded to include the full thickness. This method also allows more straightforward layup sequencing and provides natural points to debulk the layup. The constitutive and hybrid layup sequences are seen below.

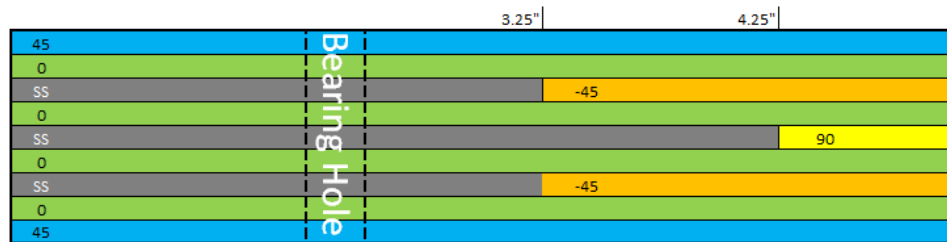


Figure 37: Hybrid Composite Constitutive Layup

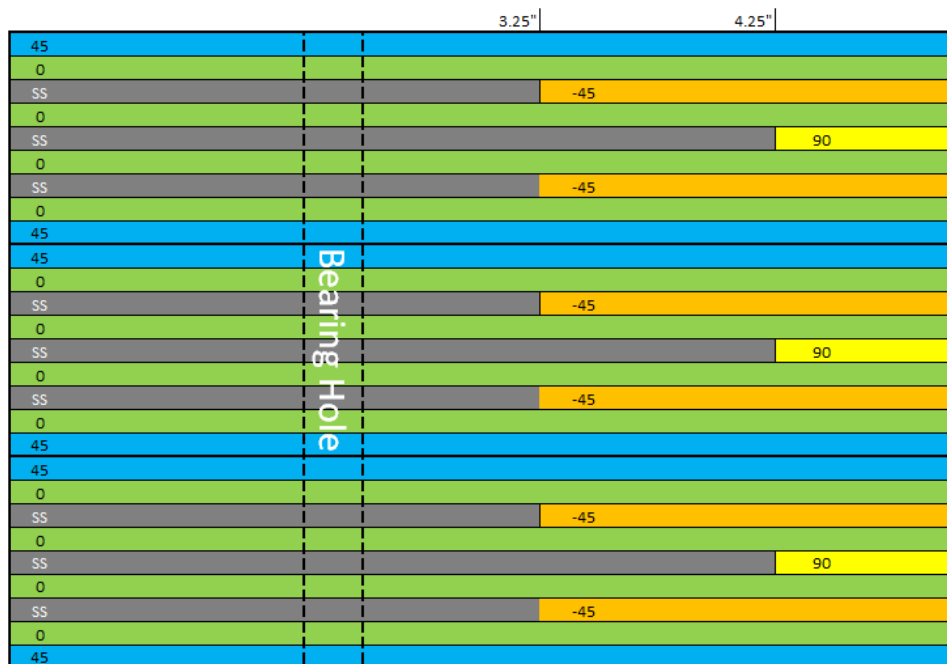


Figure 38: Hybrid Composite Stacking Sequence

The depths of the foils for this study are not assumed to be the optimum configuration. Rather, foil depths are selected based on previous STRATUS testing to ensure that the failure mode is bearing failure. A 1in step is made between the foils in the primary sequence to avoid a sharp stress concentration through the part thickness and to allow for a smooth transition in the event of minor thickness variations between the foil and composite after debulking.

Kolesnikov et al employed a similar stacking sequence with a carbon-titanium hybrid bearing test specimens by preserving the 0° composite plies and placing foils locally in the 45° and 90° plies. They also employed a transition scheme similar to that proposed in this study. [26]

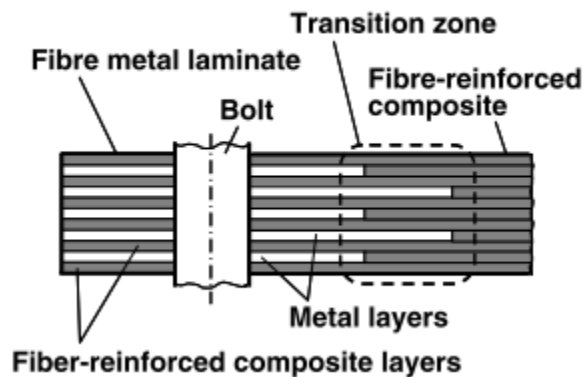


Figure 39: Ti-Gr Bearing Specimen Layup [26]

The first series of tests will be conducted on control samples, that is, composite samples that contain no stainless steel. Following this, the hybrid samples will be tested in similar fashion. Initially, three specimens would be tested to 30% hole elongation as directed by the ASTM standard. The statistical significance of three samples is discussed later. Then, specific points of interest will be determined from the stress-strain

relationships found in the three primary samples. For example, the results of previous double shear bolted testing are depicted below in Figure 40.

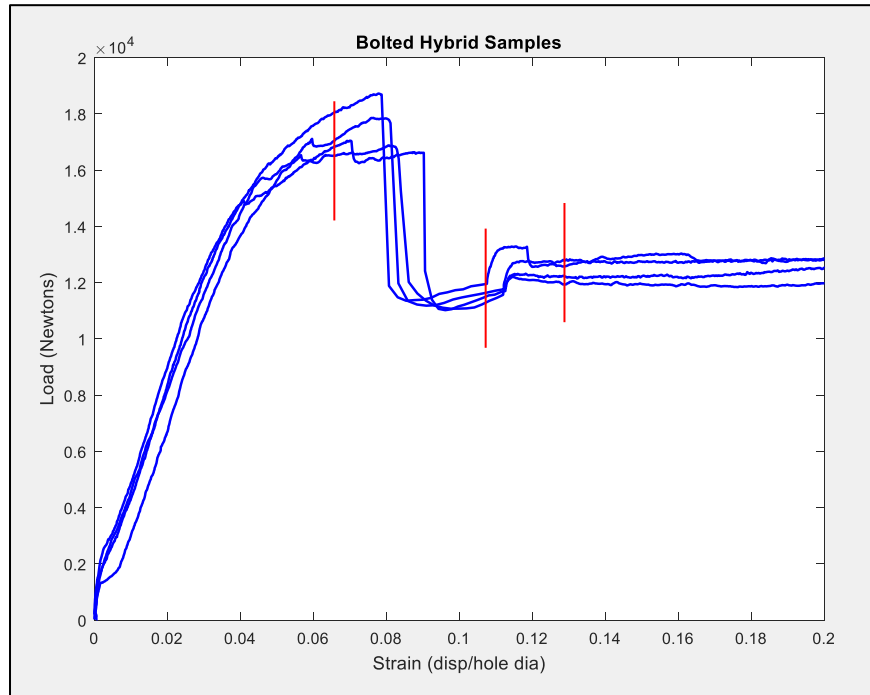


Figure 40: Bolted Hybrid Test Sample Results

The 3 vertical red lines indicate areas of interest in this data occurring just after major structural failure events. Understanding these events will enable understanding of failure modes of these structures. To limit the number of samples required, one sample would be tested at each of three points of interest. This is acceptable, because for these specific cases, the focus is not on the exact stress/strain value at the given point. Rather, the focus is on the failure mode that has caused the artifact in the stress strain curve. Though only one sample will be employed at each of these points of interest, establishing statistical significance can be done by comparing to primary samples at the specific strain

of interest and throughout the stress-strain curve. This is also discussed later in Statistical Relevance.

The fastener will be a Hi-Lok 13-8MO stainless steel bolt with a tensile stress of 220 ksi and a shear stress of 125 ksi. Tension heads will be employed to spread the load over the composite in later test phases. A conventional hex nut will be used to control preload precisely. Preload will be measured using a load washer throughout the experiment.

After testing, the samples will be sectioned through the 0° direction of the hole and examined through optical microscopy and scanning electron microscopy (SEM). Zhao et al showed that the details of fiber failure can be effectively studied using SEM as seen in Figure 41. Yamada et al showed that the failure modes in hybrid composite materials can be evaluated with optical microscopy (Figure 42) [81].

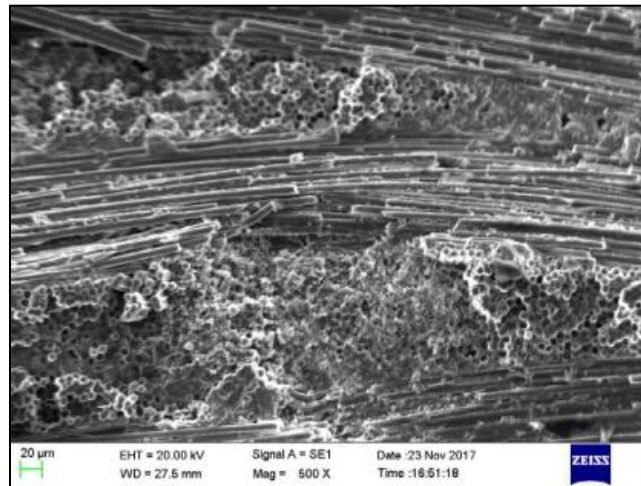


Figure 41: Fiber Failure in C/SiC Bolted Joint [82]

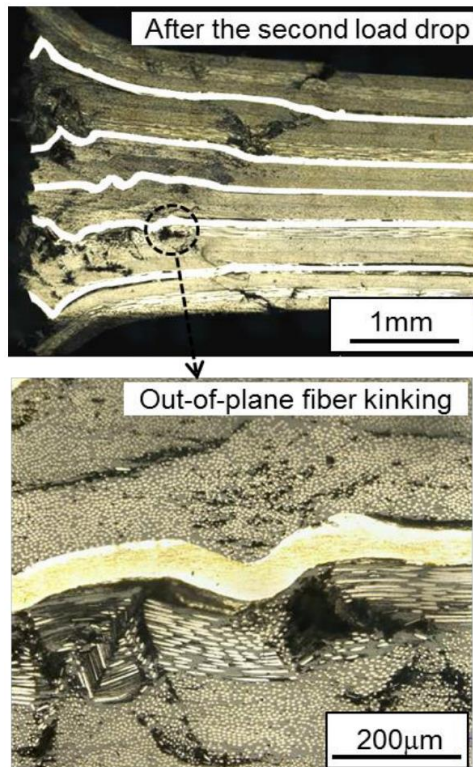


Figure 42: Optical Images of Failure in PMC/Titanium hybrid [81]

3.2.2.1 Phase 1 Test Summary

- Test Method: Quasi-Static, Double-Shear, Bearing
 - Control Samples – 6 total
 - 3 – Primary Samples - 30% hole elongation
 - 3 – Progressive Failure Samples - 3 different strains determined from primary samples
 - Hybrid Samples – 6 total
 - 3 – Primary Samples - 30% hole elongation
 - 3 – Progressive Failure Samples - 3 different strains determined from primary samples
 - Fastener: Protruding Head Hi-Lok (HL646-8-20).

3.2.3

Phase 2: Bolted Double-Shear Progressive Damage Analysis

This phase will refine finite element modeling based on information gained from Phase I experimentation. This will likely require that some specialized materials behaviors be controlled using UMAT in Abaqus and may also require development of other codes from MATLAB and/or Python to aid in efficient meshing and analysis. Due to the thickness of these coupons, bolt bending is expected to play a major role in the analysis. This phase is expected to be used to develop a majority of the modeling processes that will be used and refined throughout the study. It is in this phase that the applications of plasticity, Hashin failure, cohesive relationships and contact relationships should be fine-tuned.

3.2.4 Phase 3: Bolted Single-Shear Progressive Damage Study

Phase 3 will examine the progressive damage characteristics of the STRATUS material in a protruding bolt, single shear configuration. ASTM D5961 Procedure C will be employed to produce a quasi-static single-shear test. This phase is expected to produce some complexity due to the bending moment expected in the fastener. The load washer measuring bolt tensile load during the test is expected to aid in understanding the phenomena occurring in the fastener as well as in the delamination of the laminate.

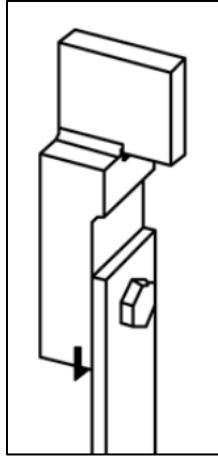


Figure 43: ASTM D5961 Procedure C

3.2.4.1 Phase 2 Test Summary

- Test Method: Quasi-Static, Single-Shear, Bearing
 - Control Samples – 6 total
 - 3 – Primary Samples - 30% hole elongation
 - 3 – Progressive Failure Samples - 3 different strains determined from primary samples
 - Hybrid Samples – 6 total
 - 3 – Primary Samples - 30% hole elongation
 - 3 – Progressive Failure Samples - 3 different strains determined from primary samples
 - Fastener: Protruding Head Hi-Lok (HL646-8-20).

3.2.5 Phase 4: Bolted Single-Shear Progressive Damage Analysis

As in Phase 2, a representative finite element model will be developed for the single shear bolted case employing knowledge gained from previous modeling efforts. This configuration is expected to introduce much more complexity than the double shear case due to bending moments in the fastener.

3.2.6 Phase 5: Countersunk Single-Shear Progressive Damage Study

Next, a similar progressive failure test matrix will be run on similarly constructed samples. Instead of employing countersunk fasteners, as in the CALE studies, a 75% countersink depth will be employed. Egan et al showed that, in a lap joint, little of the load is carried by the head of the fastener and that most of the load is carried in the 0° layer as seen in Figure 36 [30], [31]. For this reason, hybrid samples will be manufactured that include stainless steel plies throughout the thickness of the composite and then separate coupons will be manufactured that include stainless steel plies only near the shank of the fastener. This shank-only layup sequence is pictured in Figure 44.

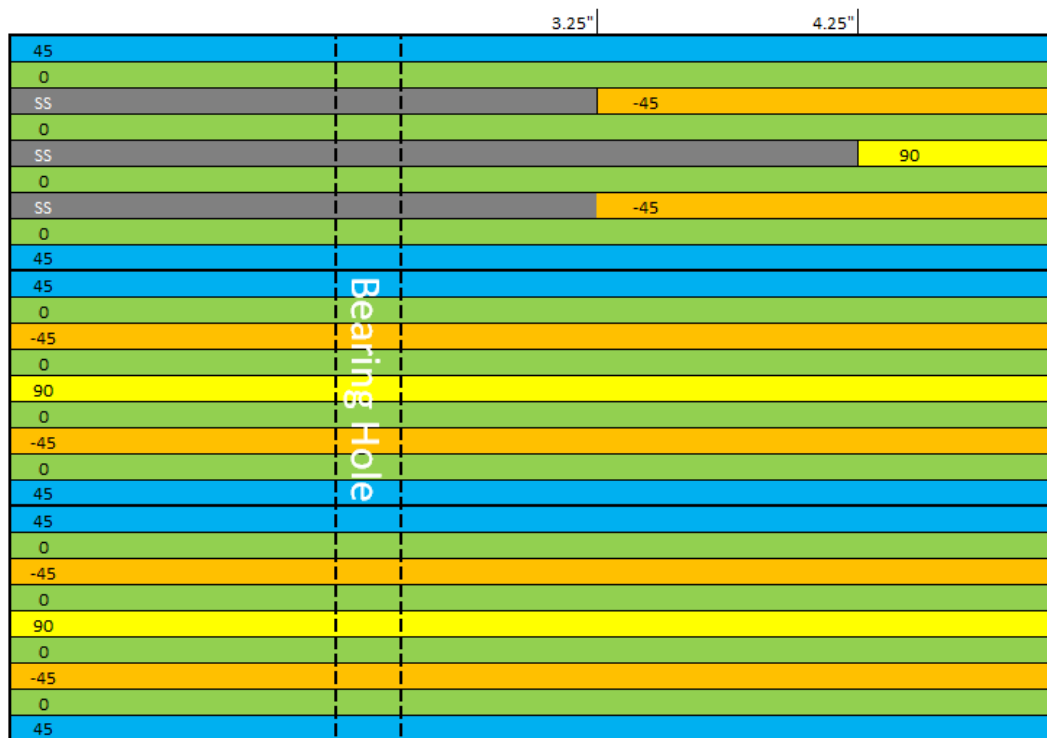


Figure 44: Shank Only Layup Sequence

3.2.6.1 Phase 5 Test Summary

- Test Method: Quasi-Static, Single-Shear, Bearing
 - Control Samples – 6 total
 - 3 – Primary Samples - 30% hole elongation
 - 3 – Progressive Failure Samples - 3 different strains determined from primary samples
 - Hybrid Samples – 6 total
 - 3 – Primary Samples - 30% hole elongation
 - 3 – Progressive Failure Samples - 3 different strains determined from primary samples
 - Shank-only Hybrid Samples – 6 total
 - 3 – Primary Samples - 30% hole elongation
 - 3 – Progressive Failure Samples - 3 different strains determined from primary samples
 - Fastener: Countersunk, 100° head, Hi-Lok (HL646-8-20).

3.2.7 Phase 6: Countersunk Single-Shear Progressive Damage Analysis

As in Phases 2 and 4, a representative finite element model will be developed for the countersunk case employing knowledge gained from previous modeling efforts. It is expected that the countersunk head will introduce a great deal of complexity into the modeling by creating more complex loading and failure.

3.2.8 Phase 7: Fatigue Progressive Damage Study

A final phase will investigate a progressive fatigue response using countersunk fasteners just like the coupons studied in Phase 3. In order to limit machine run time, a progressive fatigue test will be employed that loads the sample in a tension-tension to 85% of the yield determined in Phase 4, and is governed by the number of cycles as opposed to hole elongation. Six samples in each configuration would be run. One sample would be run at each of six cycle counts presented in Table 2. The table also shows the expected run times assuming a 4 Hz loading rate. To run six samples of two

configurations in both single shear and double shear (24 total samples) through this test matrix will require an estimated 4.62 hours of machine run time.

Table 2: Fatigue Cycles and Run Times

Number of Cycles	Estimated Run Time
1×10^4	42 min
5×10^3	21 min
1×10^3	4 min
5×10^2	2 min
10^2	30 sec
10	3 sec

These samples will not be tested for residual peak stress capacity so that they may be studied via microscopy and compared to the static progressive failure samples from Phases 1 and 3. The intention is to produce a curve as depicted in Figure 45 (from the CALE 2 study) to understand how including the stainless steel foils changes hole elongation under fatigue.

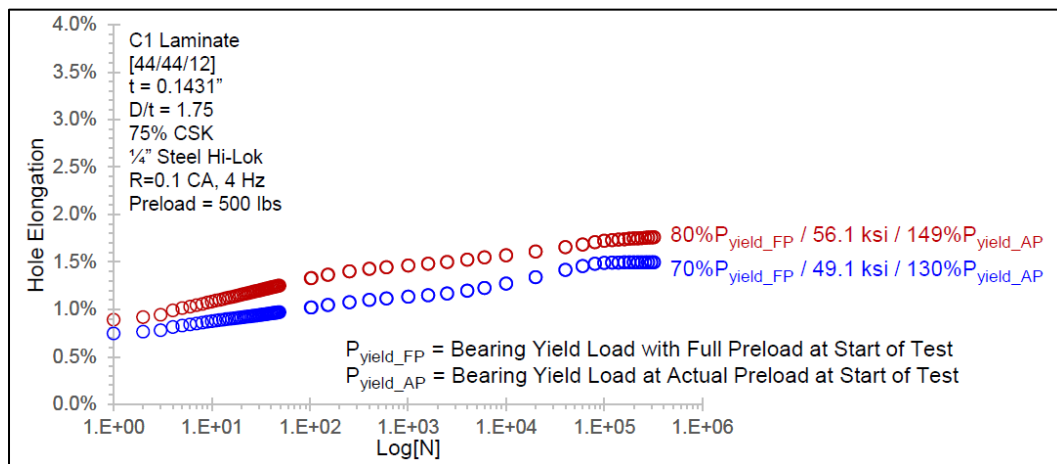


Figure 45: Hole Elongation vs Cycle Count [28]

3.2.8.1 Phase 7 Test Summary

- Test Method: Fatigue, Double-Shear, Bearing
 - Loading, Constant Amplitude, Tension-Tension, 85% of Static Yield
 - Control Samples – 6 total at cycles:
 - 1×10^4
 - 5×10^3
 - 1×10^3
 - 5×10^2
 - 10^2
 - 10
 - Hybrid Samples – 6 total at cycles:
 - 1×10^4
 - 5×10^3
 - 1×10^3
 - 5×10^2
 - 10^2
 - 10
 - Fastener: Protruding Head Hi-Lok (HL646-8-14).
- Test Method: Fatigue, Single-Shear, Bearing
 - Loading, Constant Amplitude, Tension-Tension, 85% of Static Yield
 - Control Samples – 6 total at cycles:
 - 1×10^4
 - 5×10^3
 - 1×10^3
 - 5×10^2
 - 10^2
 - 10
 - Hybrid Samples – 6 total at cycles:
 - 1×10^4
 - 5×10^3
 - 1×10^3
 - 5×10^2
 - 10^2
 - 10
 - Fastener: Protruding head, Hi-Lok (HL646-8-20).

3.2.9 Phase 8: Fatigue Analysis

The final modeling phase is expected to be the most complex. However, at the time this phase is conducted, a library of test and modeling data will have been produced to leverage in this modeling scenario. The CALE studies have investigated the use of Abaqus software for cycle life predictions in composite materials with mixed results.

3.3 Fabrication

Fabrication began first with preparing the metal foils with the previously described Grit-Blast, Sol-Gel, primer process. The specific process will be included in Appendix 4. Next the prepreg components will be cut by an automated cutter. Then the layups will be stacked as detailed in Appendix 5 and then debulked and bagged as described in Appendix 6. Finally, the layups will be cured per the manufacturer recommended process in Appendix 7. The specimens will be cut from the panels using a diamond saw and then the holes will be machined. The samples will then be examined with x-ray and c-scan to ensure no delamination occurred during machining. Finally, fiberglass grip tabs will be added to enable to sample to be gripped by the test machine.

3.3.1 Test Sample Overview

The test samples required for each phase are outlined in Table 3.

Table 3: Test Sample Overview

Phase	Test Type	Control Coupons	Hybrid Coupons	Shank Only Hybrid Coupons
Phase 1	Quasi-Static, Double Shear Bearing, Bolted	6	6	0
Phase 3	Quasi-Static, Single Shear Bearing, Protruding Bolt	6	6	0
Phase 5	Quasi-Static, Single Shear Bearing, Countersunk	6	6	6
Phase 7	Fatigue, Double Shear Bearing, Bolted; Single-Shear Bolted	12 (6-SS, 6-DS)	12 (6-SS, 6-DS)	0
	Total Samples	30	30	6

3.3.2**Test Panel Layup**

Test panels for Phases 1,3,5 and 7 are detailed below. These correspond with phases outlined above. The proposed series of tests will require 66 coupons of dimensions 1.5”x 5.5”.

Table 4: Layup Configurations

Panel Name	Priority	Layup	Coupon qty	Description
Phase 1: Quasi-Static, Double-Shear, Bolted				
C-01	1	[45/0/-45/0/90/0/-45/0/45] ₃	6	Control-Double- Shear Bolted
H-01	1	[45/0/ SS 0/ SS 0/ SS 0/45] ₃	6	Hybrid-Double- Shear Bolted
Phase 3: Quasi-Static, Single-Shear, Protruding Bolt				
C-02	2	[45/0/-45/0/90/0/-45/0/45] ₃	6	Control-Single-Shear, Protruding Bolt
H-02	2	[45/0/ SS 0/ SS 0/ SS 0/45] ₃	6	Hybrid-Single- Shear, Protruding Bolt
Phase 5: Quasi-Static, Single-Shear, Countersunk Bolt				
C-03	3	[45/0/-45/0/90/0/-45/0/45] ₃	6	Control-Single- Shear, Countersunk
H-03	3	[45/0/ SS 0/ SS 0/ SS 0/45] ₃	6	Hybrid-Single- Shear, Countersunk
H-04	4	[45/0/-45/0/90/0/-45/0/45] ₂ [45/0/- SS 0/ SS 0/ SS 0/45]	6	Hybrid, shank only- Single Shear, Countersunk

Phase 7 – Fatigue				
C-04/ C-05	5	[45/0/-45/0/90/0/-45/0/45] ₃	12	Control-Double-Shear, Bolted; Single-Shear, Bolted
H-05/ H-06	5	[45/0/ SS /0/ SS /0/ SS /0/45] ₃	12	Hybrid- Double- Shear, Bolted; Single-Shear, Bolted

Legend

- |SS| - Stainless Steel plies with layers of AF-191U on each side

3.3.3 Required Materials

AFRL/RQV will be responsible for the procurement of materials required to fabricate the test panels shown above in Table 3. The materials required for each panel type are shown in Table 5. The materials required for all panels are seen in

Table 6. Finally, an overview of all the materials required for this study is seen in Table 7. Length 1 is parallel to fibers in the 0° direction. Length 2 is perpendicular to fibers in the 0° direction.

Table 5: Cut Sheet For Each Panel Type

Material	Angle	Length 1	Length 2	1 Control Panel	1 Hybrid Panel	1 Hybrid, Shank-Only Panel
				Number	Number	Number
IM7	0	8	12	12	12	12
IM7	45	8	12	12	6	10
IM7	45	4.75	12	0	6	2
IM7	90	8	12	3	0	2
IM7	90	3.75	12	0	3	1
SS	-	3.25	12	0	6	2
SS	-	4.25	12	0	3	1
AF 191	-	8	12	0	18	6

Note: Similarly shaded rows correspond to a single layer in the layup.

Table 6: Cut Sheet For All Panels

Material	Angle	Length 1	Length 2	5 Control Panels	5 Hybrid Panels	1 Hybrid, Shank-Only, Panel
				Number	Number	Number
IM7	0	8	12	60	60	12
IM7	45	8	12	60	30	10
IM7	45	4.75	12	0	30	2
IM7	90	8	12	15	0	2
IM7	90	3.75	12	0	15	1
SS	-	3.25	13	0	30	2
SS	-	4.25	13	0	15	1
AF 191	-	8	12	0	90	6

Table 7: Total Material Requirements

Material	Area (sq ft)	Area (sq yd)
IM7	184	20.41
Stainless Steel Foil	15.6	1.73
AF 191 adhesive	64	7.12

3.4 Test Implementation

Testing will be conducted at the AFRL FIRST lab (Facility for Innovative Research in Structures Technology) in accordance with the phased approach presented above. The procedure and details are outlined in Appendix 9.

3.5 Post-Test Characterization

After the samples are tested, they will be scanned with x-ray and c-scan to quantify the delamination that has occurred. Then, they will be sectioned through the hole in the zero degree direction to enable microscopy. Optical and SEM will be employed to characterize the failure modes that occurred.

3.6 Statistical Relevance

In Phases 1, 3, and 5, the purpose of the 3 primary samples is to establish consistent stress-strain curve shapes, from which progressive failure test points can be determined. Also, loads are expected to be compared with control samples. Thus, the number of samples drives the ability to draw conclusions about the means of two different sample types. ASTM recommends five samples [83]. However, discretion is allowed to reduce this number if valid test results may be obtained. To estimate the number of samples required, previous STRATUS bolted test data was explored. Those data demonstrated a standard deviation in maximum stress of 42.5 pounds. That value is expected to be representative in this series of tests. In Table 8 below, a confidence interval length is calculated using this estimated standard deviation and based on a t-distribution [84]. The confidence interval length is also presented as a percentage of an estimated maximum load of 10,000 lbf. This is also seen graphically in Figure 46.

From this, it is understood that the primary three samples will allow a confidence interval length of less than 200 lbf or around 2% of the expected peak load. Furthermore, as the progressive failure samples are processed, the sample size will increase each time for the specified load range. For example, by the time the 6th sample is tested over the most narrow load range, the confidence interval is expected to be down to about 100 lbs. This is also expected to be a conservative estimate for lower strains since the samples typically display lower variation at lower strains. ASTM E122 has a method of calculating a required number of samples, but in this case, that method provides trivial solutions of less than one required sample [85]. Figure 46 shows graphically that

increasing the number of results should return slightly smaller confidence intervals, but it shows a diminishing return on the time and materiel resources to produce those samples.

Table 8: 95% Confidence Interval Estimates

n	$t_{\alpha/2, n-2}$	Estimated StDev (lbf)	Confidence Interval Length (lbf)	% of Expected Peak
2	4.303	42.5	365.725	3.66
3	2.776	42.5	192.691	1.93
4	2.447	42.5	147.069	1.47
5	2.306	42.5	123.968	1.24
6	2.228	42.5	109.345	1.09
7	2.179	42.5	98.993	0.99
8	2.145	42.5	91.153	0.91
9	2.120	42.5	84.943	0.85
10	2.101	42.5	79.863	0.80

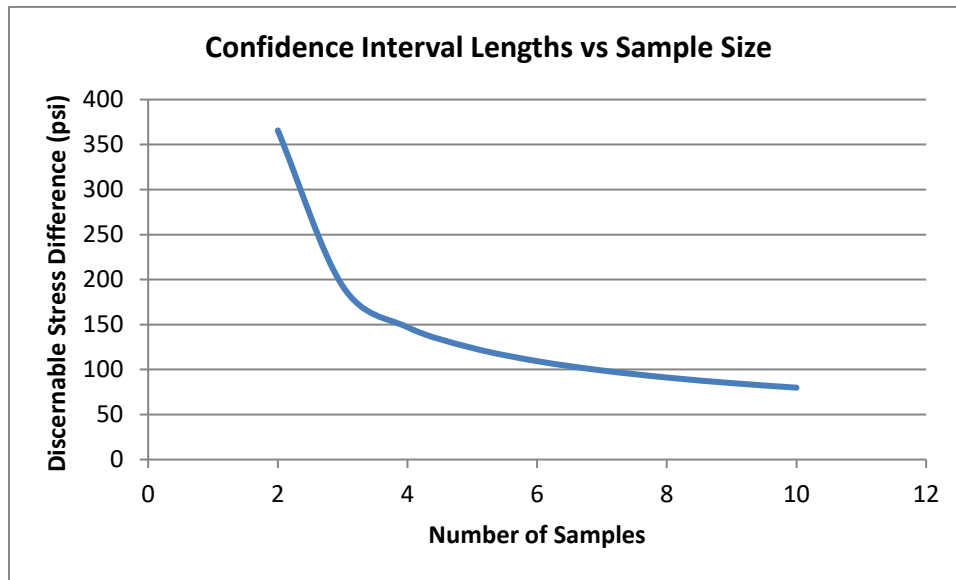


Figure 46: Confidence Interval Length vs Sample Size

IV. Results and Discussion

Engineers at AFRL/RQ have conducted double cantilevered beam tests and bearing tests on these hybrid composite structures. This testing recorded only load-displacement curves instead of the more desirable Mode-I delamination energy, but this still allowed engineers to verify the need for the grit-blast/sol-gel metal preparation in conjunction with the adhesive. This combination is depicted by the gray line (greatest value) in Figure 47. The lesser valued samples omitted certain parts of this process. A neat (untreated) foil was tried. This sample fully delaminated during machining.

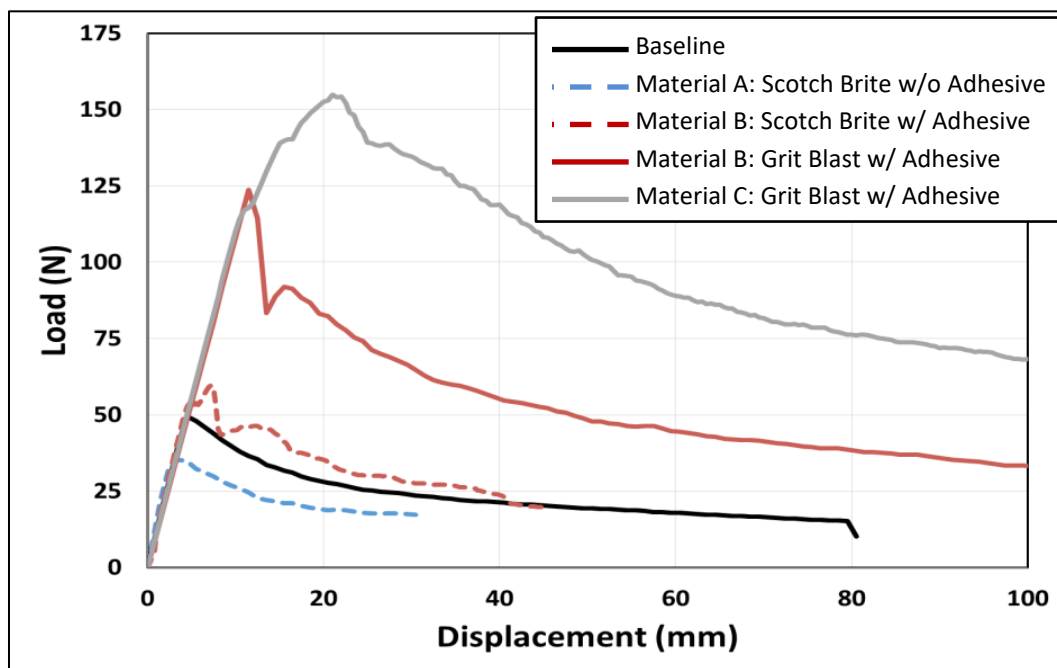


Figure 47: Double Cantilevered Beam Results

Twelve bearing samples were tested in slightly different configurations. Each contained six stainless steel foils in ten total composite plies. The average values are plotted in Figure 48 and Figure 49. When compared to control samples, these samples demonstrated

a 162% increase in ultimate bearing strength in the pinned loaded condition and an 84% increase in the bolted loaded condition.

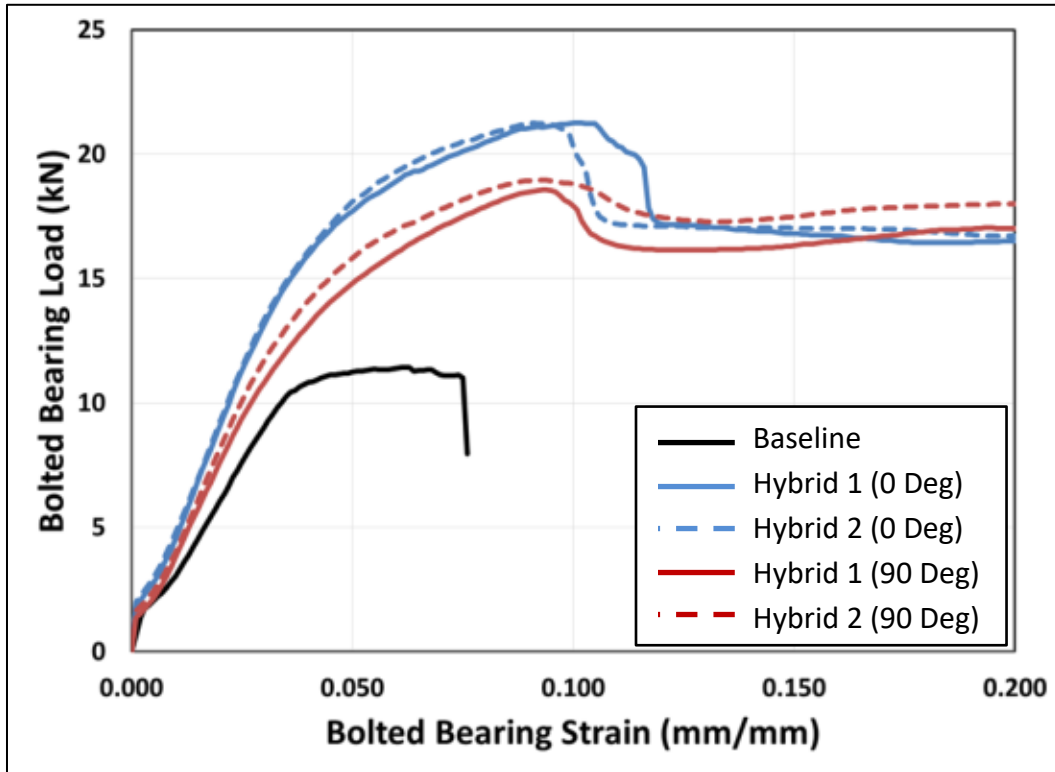


Figure 48: Bolted Bearing Results

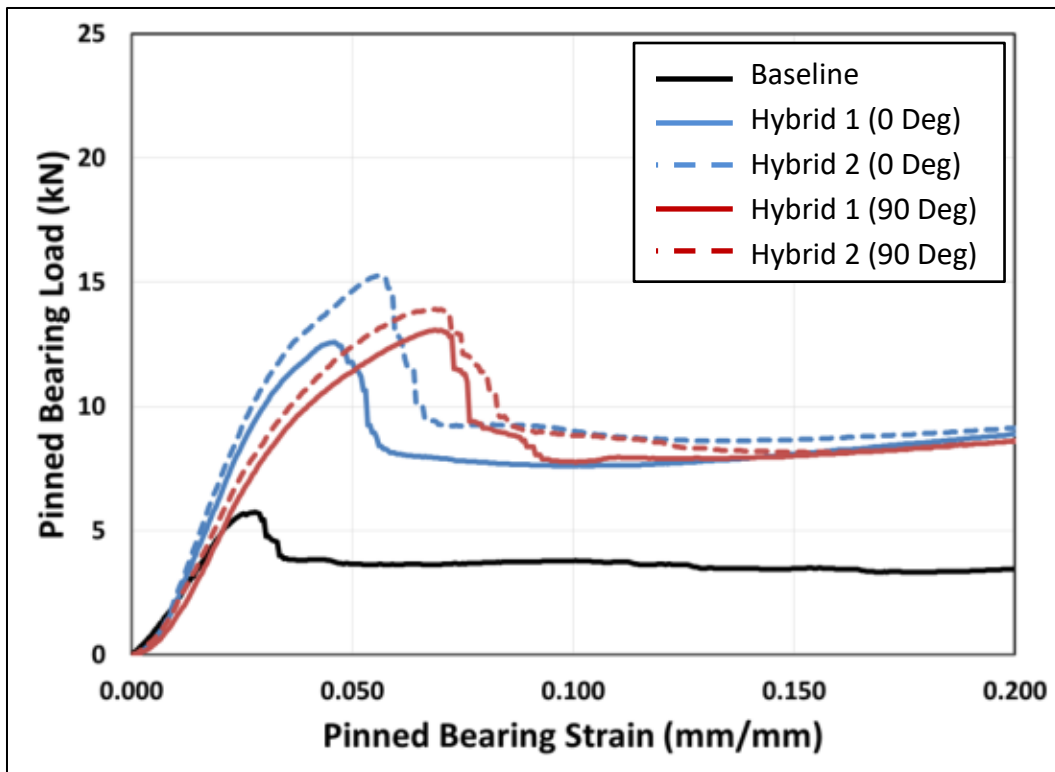


Figure 49: Pinned Bearing Results

An interesting phenomenon was noted during the metal prep procedure. Since these foils are manufactured by rolling, it was assumed that some residual stress was imparted in manufacturing. After one side was grit blasted, releasing the residual tensile stresses in the material, the material curled around the unprepared side (Figure 50).



Figure 50: Deformed foil after single side grit-blast

After the material was placed on the backer board and the other side was grit blasted, the material returned to a flat shape (Figure 51). The assertion is that the majority of the residual stresses from machining are released in the grit blasting process.

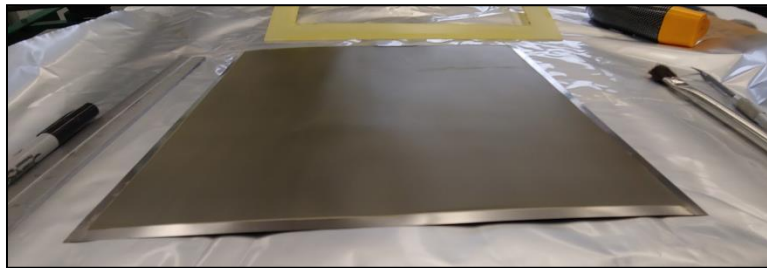


Figure 51: Flat foil after full grit-blast

Early modeling efforts have included the development of representative materials models in Abaqus and progress to tensile, open hole and bearing modeling.

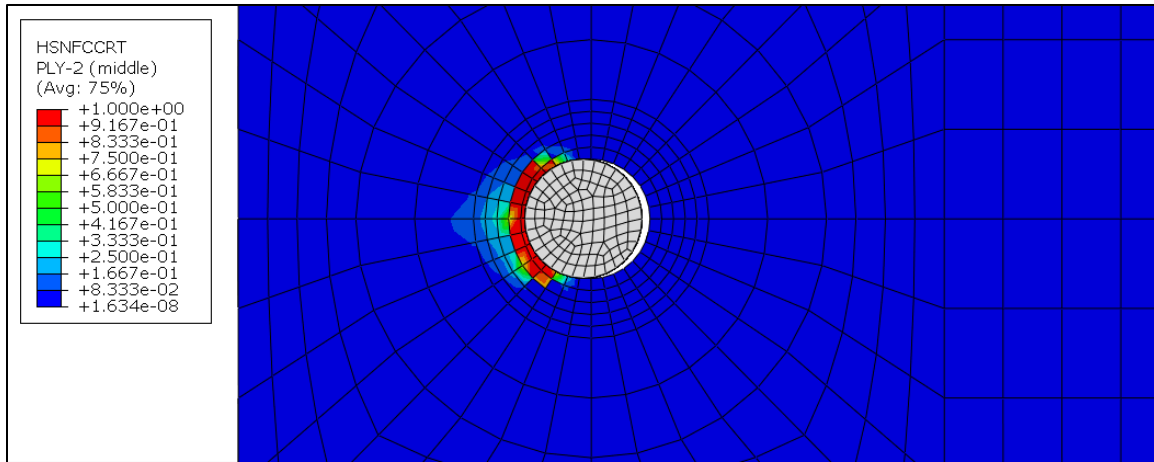


Figure 52: Composite Layup Tool Bearing Model

Much of the modeling has been done using the Abaqus Composite Layup tool (Figure 52). While this tool is effective at modeling composite shell structures, bearing modeling has proven the need to transition to a layerwise modeling approach. This is most evident when element deletion is employed. The Abaqus Composite Layup tool is able to model effects in each layer but constructs the mesh with one element through the thickness of the material. Thus, when any layer exhibits failure, the entire element is deleted. This proves extremely problematic around fasteners when the slightest amount of bending is present.

In order to study strain rate dependence in this study, a basic model of a foil was developed. A bilinear elastic-plastic model was defined, and a bearing strain of 0.5 was applied over a unit analytical time. Strains were plotted over time at three nodes deemed to have the most critical strain values as depicted in Figure 53. Von Mises stress and equivalent plastic strain are seen in Figure 54 and Figure 55. These contour plots correspond well to the behavior of the foil observed in preliminary pinned hybrid samples

(Figure 56). Finally, the strain at each of the critical nodes is plotted with respect to time (Figure 57). As time progresses, the strain at the critical elements develops at a rate very similar to the commanded rate. The only departure is due to the foil sliding along the pin in the axial direction. This enabled the decision to exclude strain rate dependence from this study and reinforces the validity of a bilinear elastic-plastic model.

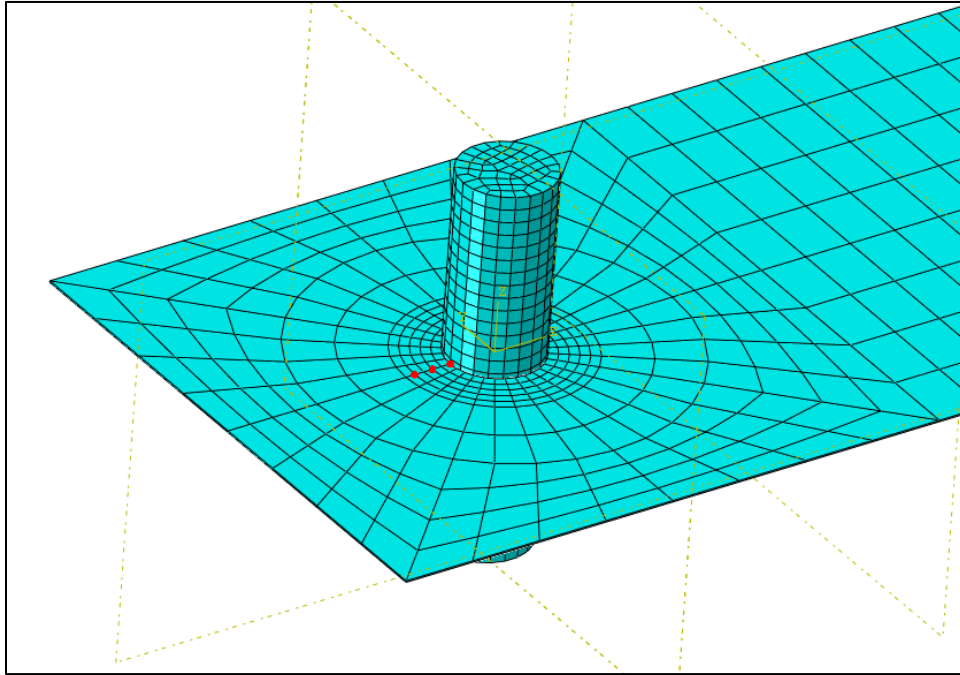


Figure 53: Critical Nodes

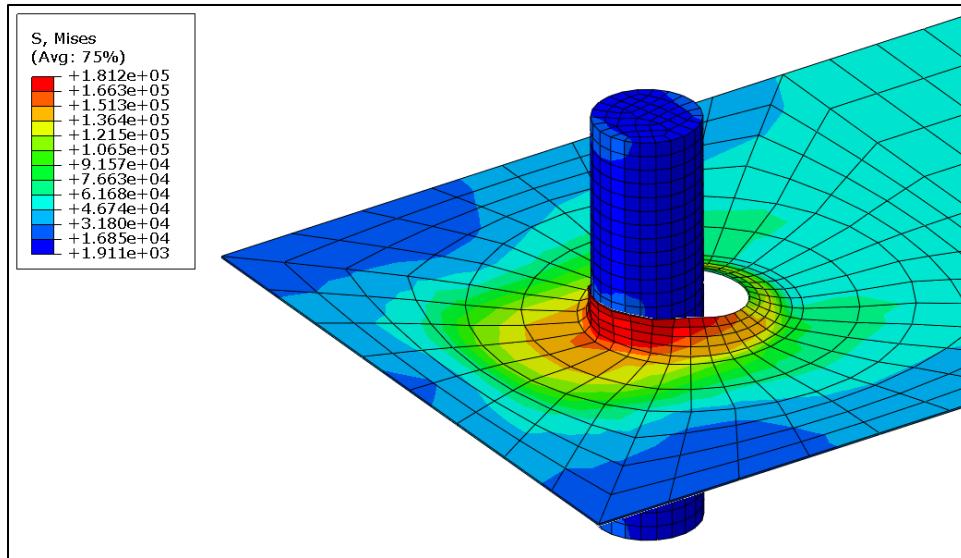


Figure 54: Foil Von Mises Stress

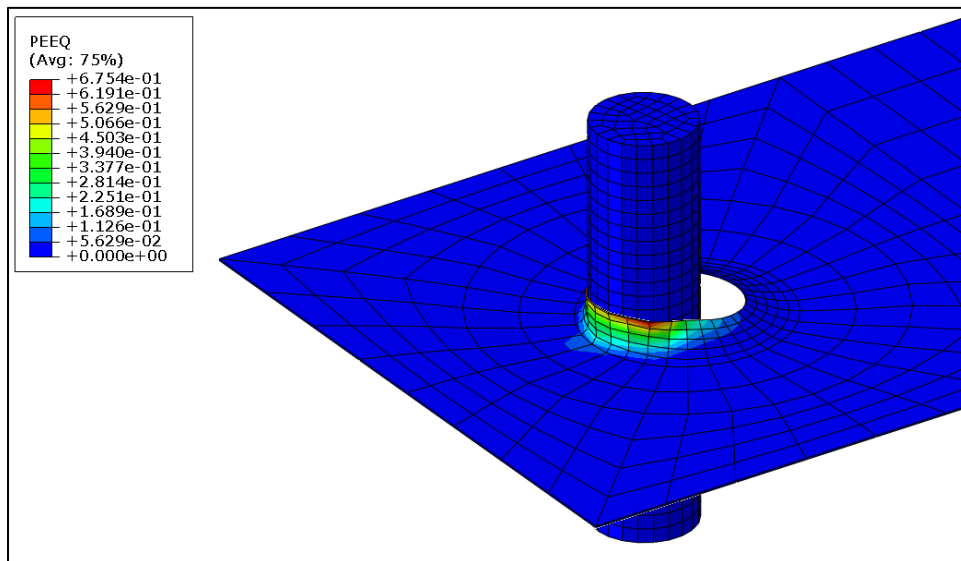


Figure 55: Foil Equivalent Plastic Strain



Figure 56: Foil deformation in preliminary pinned sample

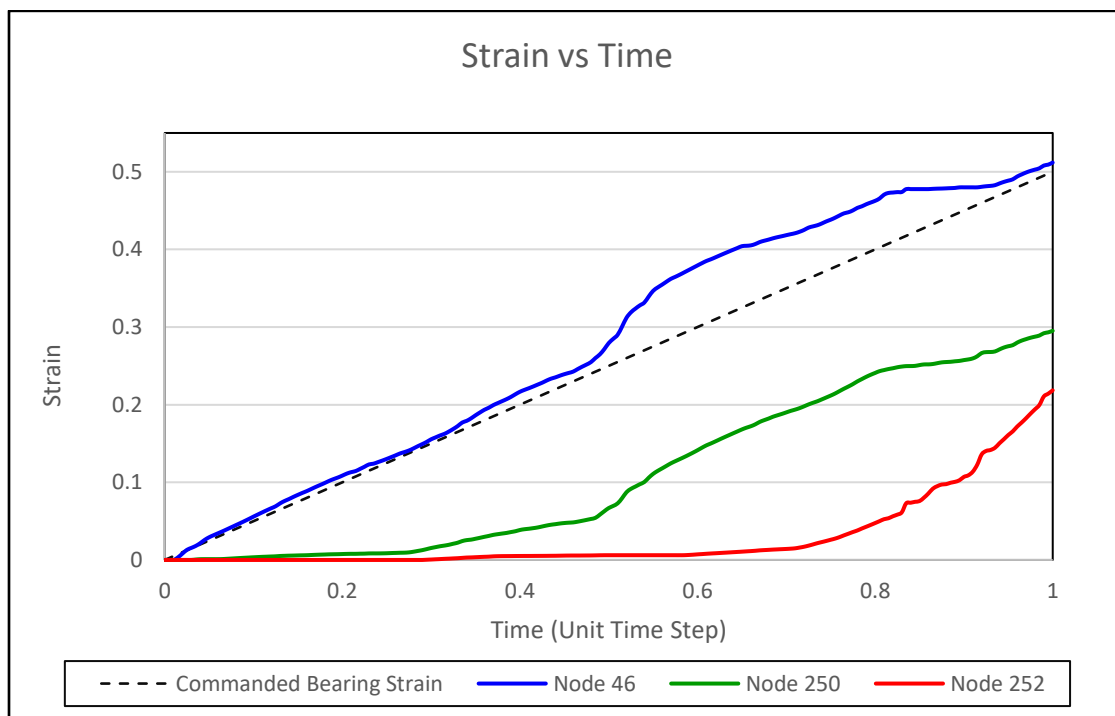


Figure 57: Strain at Critical Nodes

V. Summary

Thus far, the majority of the work on this area of study exists on the horizon. However, conclusions can be made based on preliminary testing and modeling, literature reviews, and preparation of this document. From related literature, modeling efforts should begin with a layerwise approach that included the 90%/10% layup where the foils are modeled using bilinear plasticity and the composite layers are modelled using Hashin Criteria. Early double cantilevered beam testing has proven the requirement for metal preparation and the use of adhesive. Metal preparation and characterization has proven that the process produces consistent results and that residual stress is released during the process. Finally, bearing testing has proven an increase in bearing strength. Most importantly, it has proven the efficacy of this material to include its ability to be manufactured properly and consistently. This provides confidence that this study may now proceed with characterizing the failure modes in this material system.

Appendix 1: Acronyms and Abbreviations

- 3D – Three Dimensional
- AFRL/RX – Air Force Research Laboratory, Materials and Manufacturing Directorate
- AFRL/RQ – Air Force Research Laboratory, Aerospace Systems Directorate
- ASTM – American Society for Testing and Materials
- CALE – Composite Airframe Life Extension
- ESL – Equivalent Single Layer
- FIRST – Facility for Innovative Research in Structures Technology
- FML – Fiber Metal Laminates
- in – Inch
- ksi – One Thousand Pounds Per Square Inch
- Lbf – Pounds Forced
- LW – Layerwise
- m – Meter
- mil – milli-inch – one thousandth of an inch
- MPa – Megapascal – One Million Newtons Per Square Meter
- msi – One Million Pounds Per Square Inch
- N – Newton
- psi – Pounds Per Square Inch
- SEM – Scanning Electron Microscopy/Microscope

Appendix 2: Variables and Operators

f_n – normal force

f_t – tangential force

g_n – the normal penetration

g_t – the tangential penetration

I – Internal Forces

k_n – normal penalty term

k_t – the tangential penalty term

P – External Forces

ε – Strain

ε^e – Elastic Strain

ε^p – Plastic Strain

$\bar{\varepsilon}^{pl}$ – Equivalent Plastic Strain

$\dot{\bar{\varepsilon}}^{pl}$ – Equivalent Plastic Strain Rate

δ_{ij} – Kronecker Delta. When $i = j$, $\delta = 1$. When $i \neq j$, $\delta = 0$.

σ – Cauchy, true stress

σ_{hyd} – Hydrostatic Stress

σ' – Deviatoric Stress

μ_d – the coefficient of friction

sgn – the sign operator which takes the sign of the parenthetical quantity

$\text{tr}()$ – Trace of a matrix. Sums the diagonal values of a matrix.

Appendix 3: Schedule

Task Name	Start	Finish
Procurement	Mon 1/28/19	Fri 3/29/19
Material Prep	Mon 2/25/19	Fri 5/10/19
SS Surface Prep	Mon 2/25/19	Fri 5/10/19
Cut IM7 Pre-Preg	Mon 4/1/19	Tue 4/30/19
Prep-AF-191U	Wed 5/1/19	Wed 5/1/19
Cut Fiberglass Tabs	Wed 5/1/19	Mon 5/6/19
Submit Prospectus	Thu 4/25/19	Thu 4/25/19
Defend Prospectus	Thu 5/9/19	Thu 5/9/19
Panel Fabrication	Mon 5/13/19	Mon 5/13/19
Autoclave Cure	Tue 5/14/19	Tue 5/14/19
Machine/Prep Specimens	Wed 5/15/19	Mon 5/20/19
Cut Bearing Specimens	Wed 5/15/19	Thu 5/16/19
Drill Bearing Specimens	Thu 5/16/19	Fri 5/17/19
Label samples	Fri 5/17/19	Mon 5/20/19
Specimen Test Prep	Mon 5/20/19	Tue 5/28/19
Conduct Ultrasonic/CT Scans	Mon 5/20/19	Mon 5/27/19
Glue Fiberglass Grip Tabs	Mon 5/27/19	Tue 5/28/19
Phase 1 Testing (Double Shear)	Mon 6/17/19	Wed 6/26/19
Test 3 Primary Samples (Control and Hybrid - 6 tot)	Mon 6/17/19	Mon 6/17/19
Decide on Points of Interest	Mon 6/17/19	Mon 6/17/19
Test 3 Progressive Samples (Control and Hybrid - 6 tot)	Tue 6/18/19	Tue 6/18/19
Post-Testing Analysis	Tue 6/18/19	Wed 6/26/19
Conduct Ultrasonic/CT Scans	Tue 6/18/19	Tue 6/25/19
Section/Polish Specimens	Tue 6/25/19	Wed 6/26/19
Phase 2 (Phase 1 Initial Analysis)	Mon 7/8/19	Fri 7/12/19
Phase 3 Testing (Single Shear Bolted)	Mon 7/15/19	Wed 7/24/19
Test 3 Primary Samples (Control and Hybrid - 6 tot)	Mon 7/15/19	Mon 7/15/19
Decide on Points of Interest	Mon 7/15/19	Mon 7/15/19
Test 3 Progressive Samples (Control and Hybrid - 6 tot)	Tue 7/16/19	Tue 7/16/19
Post-Testing Analysis	Tue 7/16/19	Wed 7/24/19
Conduct Ultrasonic/CT Scans	Tue 7/16/19	Tue 7/23/19
Section/Polish Specimens	Tue 7/23/19	Wed 7/24/19

Phase 4 (Phase 3 Initial Analysis)	Wed 7/24/19	Wed 7/31/19
Phase 5 Testing (Single Shear CSK)	Wed 7/31/19	Mon 8/12/19
Test 3 Primary Samples (Control, Hybrid, Shank-only Hybrid - 9 tot)	Wed 7/31/19	Thu 8/1/19
Decide on Points of Interest	Thu 8/1/19	Thu 8/1/19
Test 3 Progressive Samples (Control, Hybrid, Shank-only Hybrid - 9 tot)	Fri 8/2/19	Fri 8/2/19
Post-Testing Analysis	Mon 8/5/19	Mon 8/12/19
Conduct Ultrasonic/CT Scans	Mon 8/5/19	Fri 8/9/19
Section/Polish Specimens	Mon 8/12/19	Mon 8/12/19
Phase 6 (Phase 5 Initial Analysis)	Tue 8/13/19	Mon 8/19/19
Phase 7 Testing (Fatigue)	Tue 8/20/19	Wed 8/28/19
Test 6 Control Samples- Double Shear	Tue 8/20/19	Tue 8/20/19
Test 6 Hybrid Samples - Double Shear	Tue 8/20/19	Tue 8/20/19
Test 6 Control - Single Shear	Tue 8/20/19	Tue 8/20/19
Test 6 Hybrid Samples - Single Shear	Wed 8/21/19	Wed 8/21/19
Post-Testing Analysis	Wed 8/21/19	Wed 8/28/19
Conduct Ultrasonic/CT Scans	Wed 8/21/19	Tue 8/27/19
Section/Polish Specimens	Wed 8/28/19	Wed 8/28/19
Phase 8 (Phase 7 Initial Analysis)	Thu 8/29/19	Wed 9/4/19
1st Dissertation Draft to Committee (est)	Thu 4/23/20	Thu 4/23/20
Draft Dissertation to Committee (est)	Thu 7/9/20	Thu 7/9/20
Defend Dissertation	Thu 8/6/20	Thu 8/6/20
Graduation (est)	Thu 9/17/20	Thu 9/17/20

Appendix 4: Foil Preparation Procedure

Degreasing

All adherend surfaces of the adherends were wiped with clean, acetone-moistened, lint-free wipes (Chicopee® Duralace® 9404). Wiping continued until the surface of a new, clean wipe did not present discoloration from contamination.

Grit-Blast Process

The bonding surface of each adherend was grit-blasted with 50 µm diameter (nominal) [240 grit #] aluminum oxide grit using a 0.22-inch blast nozzle and 70 psig of clean, dry nitrogen propellant. The surface was grit-blasted using a 90 ± 15 -degree angle of incidence, and a stand-off distance of 6 to 9 inches was maintained throughout the process. Grit-blasting continued until a uniform finish was achieved on the treated surface. Residual grit was removed from the treated surface using ~35 psig of clean, dry nitrogen.

Sol-Gel Application

Sol-gel application was initiated within 60 minutes after completion of the grit-blast process. AC-130-2 sol-gel was mixed per the manufacturer's recommendations and used within 8 hours after the initial mixing of the solution. The mixed solution was applied to the bonding surface of each adherend using a clean acid brush, and the surface remained wetted for 3-4 minutes before being oriented vertically to allow excess solution to drain. The surface was dried at ambient conditions (75 ± 5 °F and $40 \pm 5\%$ relative humidity (RH)) for 60 minutes in a clean laboratory environment.

Primer Application

Cytec Solvay BR 6747-1 primer was sprayed on using a horizontal and vertical box pattern to achieve a thickness of 0.2 mils. The adhesive thickness was measured using a “traveler panel” that was prepped and sprayed alongside the main panel. This prevents contamination of the main panel from over handling.

Appendix 5: Layup Sequences

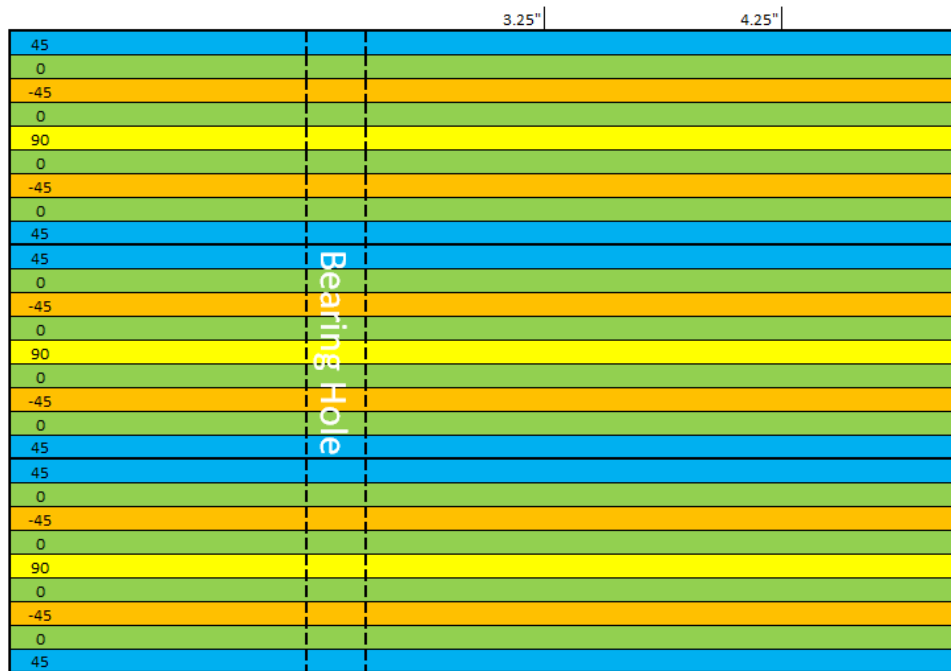


Figure 58: Control Layup

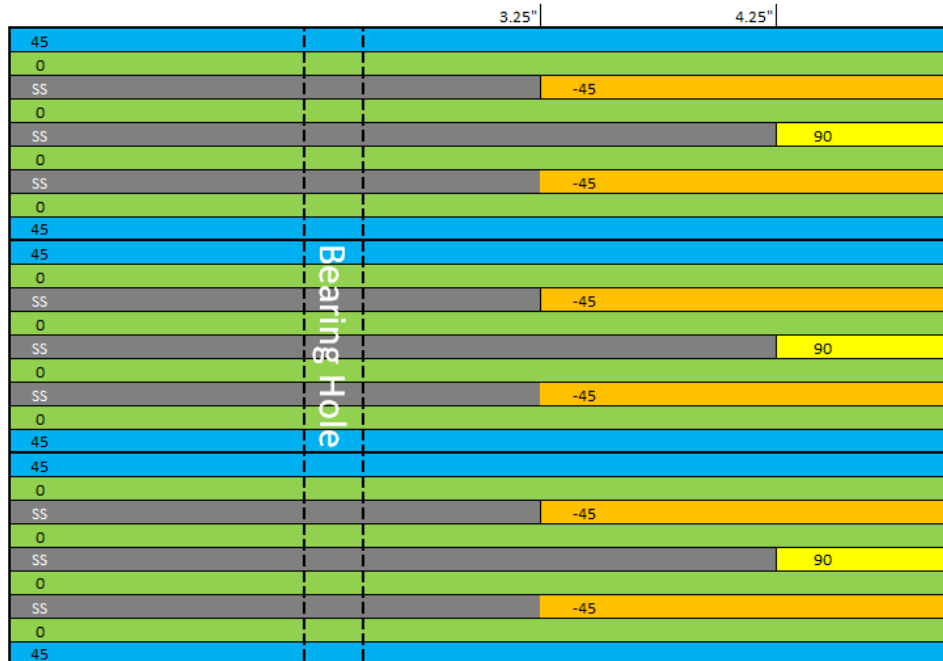


Figure 59: Hybrid Layup

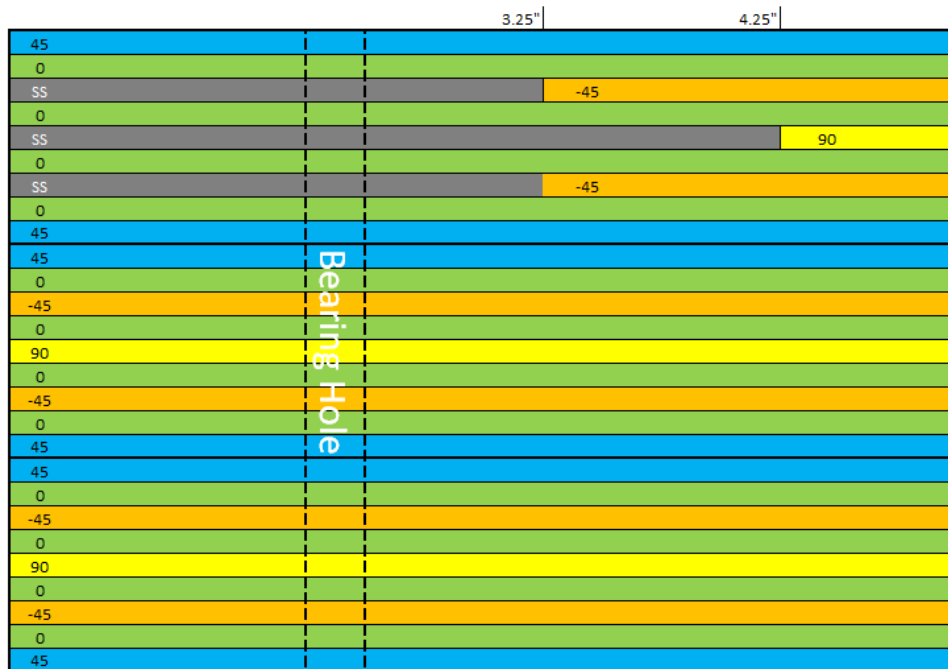


Figure 60: Shank-Only Hybrid Layup

Appendix 6: Layup and Bagging Process

- AFRL/RQV will be responsible for the fabrication of the test specimens listed in Table 1.
- Cut out all pre-preg plies, and AF-191 adhesive layers
- Prepare all stainless-steel plies by treating all surfaces accordingly. Surface treatment of the SS material will be performed by RXSA
 - Grit Blast
 - Sol Gel treatment
 - Primer
- Collect surface roughness measurements of SS with profilometer
- Perform layup and debulking
 - All SS plies should be oriented with the rolling direction of the steel parallel with the 0-degree ply orientation
- Place panels on caul plate and perform bagging procedure according to Cycom recommendations provided in appendix
- Cure panels according to Cycom recommendations provided in appendix
- Cut panels into coupons & drill/ream holes for bearing coupons using panel cutout images and coupon dimensions provided in appendix.
 - Take care not to create burrs / smearing of stainless steel when drilling and cutting.
 - Label each specimen with panel name and number according to cutout schedule provided in appendix
- Attach fiberglass grip tabs to bearing coupons
- Obtain ultrasonic or CT images of all bearing coupons at end of fabrication process

Appendix 7: Autoclave Cure Cycle

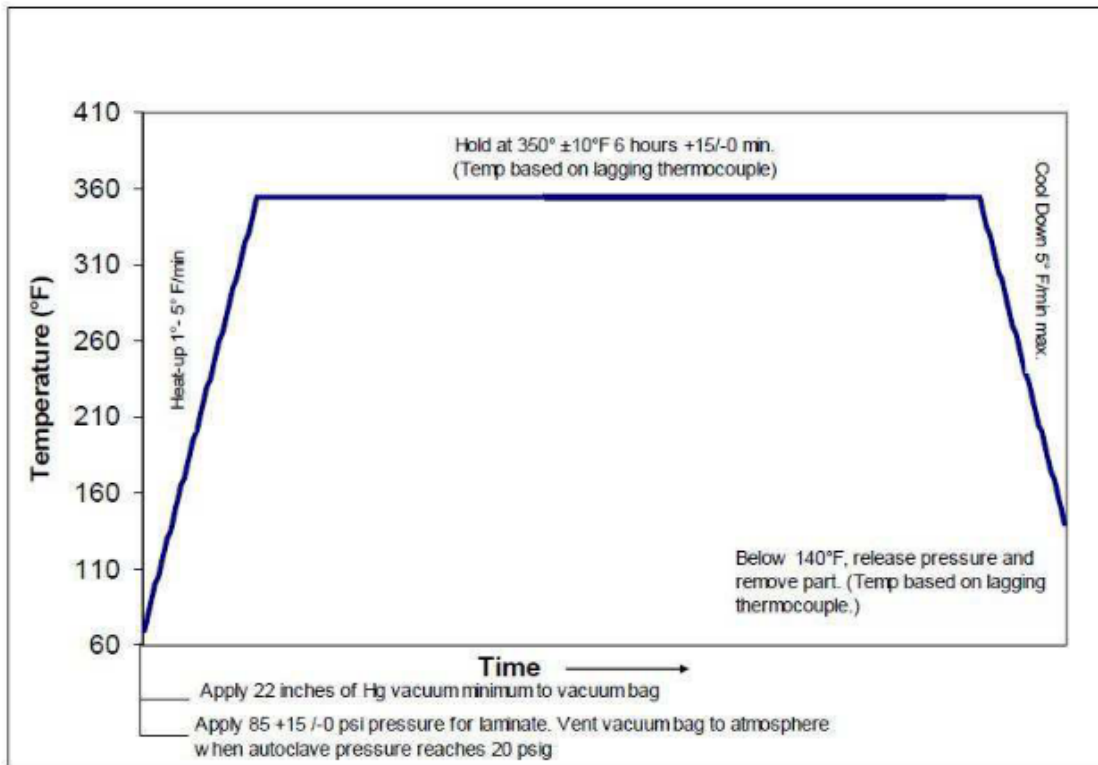


Figure 61: 977-3 Cure Cycle [78]

[illegible]

Notes:

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Appendix 9: Test Procedure

Testing Procedure

Bearing Testing – ASTM D5961M Standards

Bearing Test Equipment

- Bearing Test Fixture A and C (ASTM D5961)
- LWO-04 Load Washer
- Extensometer
- Ultrasonic Scanner (C-scan)
- 0-1" Micrometer (thickness measurements) (accuracy ± 0.0001 in.)
- 10" + Calipers (all other measurements) (accuracy ± 0.001 in.)

Pre-test Measurements

- w – specimen width (near hole)
- h – specimen thickness (near hole)
- D – hole diameter
- f – distance to closest specimen side
- g – distance to specimen end
- d – pin diameter

Loading procedure

- Place extensometer to measure between fixture and closest point on specimen to hole not obstructed by the fixture. This distance should be measured, notated and marked on the specimens for reference during modeling.
- Preload fastener to 500 lbf.
- Calibration test speed – 2mm/min
- Test speed goal – failure within 1-10 minutes (adjust accordingly)
- Data recording – 2-3 recordings per second, minimum 300 data points per test
 - Cross-head displacement
 - Bearing load
 - Extensometer displacement
- Stop test at 30% hole elongation

Post test data processing

- Calculate bearing strain $\epsilon_{br} = \frac{\text{Extensometer displacement}}{\text{Hole diameter}}$
- Calculate bearing stress $\sigma_{br} = \frac{\text{bearing load}}{\text{hole diameter} \times \text{specimen thickness}}$

- Plot bearing stress-strain curve

Post-Processing

- Obtain C-scans after bearing tests to capture delamination
- Section samples through center of hole along 0° direction to enable microscopy
- Ensure that all sectioned parts are labeled with original specimen number

Appendix 10: Materials Properties

Table 9: IM7/977-3 Properties [28]

Lamina Properties		Strength Properties		Toughness Properties	
E11 [msi]	18.91	F11tu [ksi]	392.45	G_FT (psi-in)	600
E22 (E33) [msi]	1.26	F11cu [ksi]	256.06	G_FC (psi-in)	85.65
nu12 (nu13)	0.32	F22tu [ksi]	13.88	GIC_MT (psi-in)	1.69
nu23	0.461	F22cu [ksi]	34.33	GIIC_MC (psi-in)	3.538
G12(G13) [msi]	0.824	F33tu [ksi]	13.88		
G23 [msi]	0.432	F12su [ksi]	17.11		
CTE [μ in/in-°F]	-0.36	F23tu [ksi]	17.11		
		F13su [ksi]	17.11		

Table 10: 304 Stainless Steel Full Hard Properties [86]

Mechanical Properties		Thermal Properties	
Ultimate Tensile Strength [ksi]	186	CTE [μ in/in-°F]	9.61
Yield Strength [ksi]	140	Specific Heat Capacity [BTU/lb-°F]	0.120
Modulus of Elasticity [ksi]	28000	Thermal Conductivity [BTU-in/hr-ft ² -°F]	112
Poissons Ratio	0.29		
Elongation at break [strain]	0.21		
Thickness [in]	0.004		

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