MECHANICAL PROPERTIES CHARACTERIZATION AND BUSINESS CASE ANALYSIS OF THE FIBER METAL LAMINATE GLARE-3 FOR USE AS A SECONDARY AIRCRAFT STRUCTURE

THESIS

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AFIT/GRD/ENV/10-M04

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Abstract

This effort explored the mechanical characteristics and economic feasibility of using the fiber metal laminate, GLARE-3, as a secondary aircraft structure; specifically, the cargo floor of a C-130. The mechanical properties were determined through static four-point bending and tensile testing and dynamic impact testing. Aggregate behavior of the constituent materials was predicted using a model which consisted of Mass Volume Fraction (MVF) and Classical Laminated Plate Theory (CLPT) methods using known values for the constituents.

Static testing was conducted on coupon-level specimens using standardized testing procedures. Static tensile tests were conducted on specimens with four different fiber orientations, 0°, 22.5°, 45°, and 90°, while static bending tests were conducted on fiber orientations of 0° and 90°. Two series of impact tests were performed on both GLARE-3 and 2024 T3 aluminum using 4-inch wide strips to show impact damage progression. Data for the economic analysis was gathered from existing literature and cost data was analyzed over a 30 year period for both GLARE and aluminum. Analysis of the data proved the use of GLARE-3 as a potential cargo floor material was both mechanically and economically feasible with the material paying for itself within the first year of its use.

To my Wife and Son

Acknowledgements

This thesis represents the culmination of my academic research work; none of this would have been possible without the expertise and support of all those who helped me along the way. First, I would like to thank Mr. Brian Smyers and Mr. Richard Wiggins, for without their time and technical expertise I would not have been able to perform the required experimentation in the laboratory. I would also like to give a special thanks to Dr. Ed Forster. Ed provided his time, technical and academic expertise, and professional guidance that helped me scope and perfect this thesis effort. Last but certainly not least, I would like to thank my thesis committee members, Dr. Al Thal, Dr. Som Soni, and Lt. Col Rob Wirthlin whose expertise and passion for teaching greatly strengthened my academic knowledge and formal writing abilities, both of which are invaluable skills I will use for the rest of my life.

Benjamin O. Elton

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MECHANICAL PROPERTIES CHARACTERIZATION AND BUSINESS CASE ANALYSIS OF THE FIBER METAL LAMINATE GLARE-3 FOR USE AS SECONDARY AIRCRAFT STRUCTURE

Chapter I. Introduction

Sustained high operations tempo due to ongoing combat operations in Iraq and Afghanistan have resulted in accelerating the aging process of U.S. Air Force aircraft. Many of these aircraft are already operating outside their respective intended service lives. The demand placed on these aircraft's structures combined with their respective age has resulted in increased structural inspections and repair activities, ultimately increasing the cost to maintain these aircraft. Replacing existing structural components with structures made from fiber metal laminates could alleviate the current inspection and repair workload and extend the service life of these aircraft. This research effort analyzes the material's mechanical properties and presents a business case analysis to explore the feasibility of using this material onboard U.S. Air Force aircraft as a secondary aircraft structure, specifically the cargo flooring.

General Background

One hundred years of aircraft evolution has resulted in the application of third generation materials to the primary structure of passenger aircraft (Vlot, 2001). First generation materials are exemplified by wooden primary structures, from the 1903 Wright Flyer built from spruce wood to de Haviland's Albatross built with plywood-balsa sandwich structures in 1938 (Paul & Pratt, 2004). Aluminum aircraft structures prevalent in the propeller-driven production aircraft of the 1930s to the jet-powered passenger aircraft of the 1990s mark the second generation of aircraft materials. Advanced composite materials have been increasingly applied to military aircraft structures starting in the 1960s, and have succeeded in transferring composites and hybrids to a significant percentage of commercial aircraft structural components in the early 21st century. Although unable to witness the first flight of hybrid and composite primary structures on commercial passenger aircraft, Vlot's (2001) realization of the potential uses and applications for these new materials of new aircraft production has been fulfilled in the Airbus 380 and the Boeing 787 Dreamliner.

The third generation of aircraft materials, composite and hybrid materials, provide advantages to standard monolithic aluminum, much in the same manner aluminum had provided advantages over wooden aircraft structures. Hybrid and composite structures have been tailored to achieve specific mechanical properties through the selection of fiber orientation and stacking sequence of materials. Tailoring these materials increases structural efficiency, while reducing weight, thereby enabling the aircraft to perform the desired function and meet the design requirement. The demanding structural properties required for the Grumman X-29, Bell/Boeing V-22 Osprey, and Scaled Composites Voyageur could not have been achieved through aluminum materials but were accomplished with composite structures (Forster, Clay, Holzwarth, Pratt, & Paul, 2008) Composite and hybrid materials have been used in advanced aircraft structural concepts specifically due to their increased stiffness and strength in comparison to aluminum structures, which could not achieve the same aircraft performance due to increased weight.

Hybrid materials can be tailored for structural performance such as stiffness, strength, and impact resistance. "Once an application is selected and requirements have been defined, the application of individual constituents can be tailored for that particular structure appropriately. This does not result in a new material concept, but in a new structural concept" (Alderliesten, 2009a, p. 1246). In addition to structural performance advantages, composite and hybrid structures exhibit good durability and damage tolerance characteristics, possibly providing a useful alternative to second generation aluminum. Fatigue can cause premature failure of aluminum structures. A fatigued aluminum structure is damaged due to cracks that have developed from cyclic loading. These cracks increase local stress levels, in particular near the crack tip, such that the cracks continue to grow. Historical data shows that once cracks form in an aluminum structure, they will grow in length exponentially until the structure ultimately fails. Inspection of entire aircraft structures for fatigue damage is tedious and expensive, in addition to the costs associated with repairing the cracks once detected.

Fiber Metal Laminates (FML) represent part of this third evolution of aircraft structure materials. More commonly known under the more general term of hybrid materials, the history of fiber metal laminates began at Technical University Delft, in the Netherlands. "Despite a long standing dominance in the field of aerospace technology, U.S. researchers were not the first to realize the benefits of metallic laminated structures" (Cox, 2009, p. 2). The need to create laminated structures began shortly after World War II in the Netherlands who, due to war damages, lacked the expensive metal working equipment required to fabricate intricate aircraft structures (Vermeerern, 2003). Thus, the need to research bonded structures and the resulting innovations spawned from necessity. They needed a new method of fabricating aircraft structures if they were to rebuild themselves.

Fiber Metal Laminates consist of monolithic sheets of aluminum and glass fiber reinforced polymer epoxy bonded together. Researchers at Delft University showed that these metal laminates displayed greater mechanical properties than those of traditional monolithic materials, and by adding glass fibers to the plastic adhesive material they were able to achieve even greater mechanical properties when compared to monolithic aluminum (Vlot, 2001). By adding fibers to these hybrid materials, the researchers at Delft University created Fiber Metal Laminates. One such Fiber Metal Laminate was GLARE, which is an acronym formed by combining the two terms GLAss and REinforced.

GLARE is produced from thin sheets of monolithic aluminum bonded together with an FM94K epoxy adhesive containing embedded S-2 glass fibers running in parallel. Researchers at Delft noticed how the addition of the glass fibers gave the material the added strength required for the material to achieve mechanical properties similar to that of monolithic aluminum (Vlot, 2001). Some of the remarkable properties this material provided were the ability to inhibit fatigue crack growth, increased damage tolerance, and lower weight per volume when compared to aluminum. “GLARE exhibits superior damage tolerance due to its crack bridging mechanism” (Vlot & Gunnink, 2001, p. 220). The material used in this research effort was GLARE-3 whose lay-up consisted of thin sheets of aluminum bonded together with pre impregnated (prepreg) glass fibers as shown in figure 1.

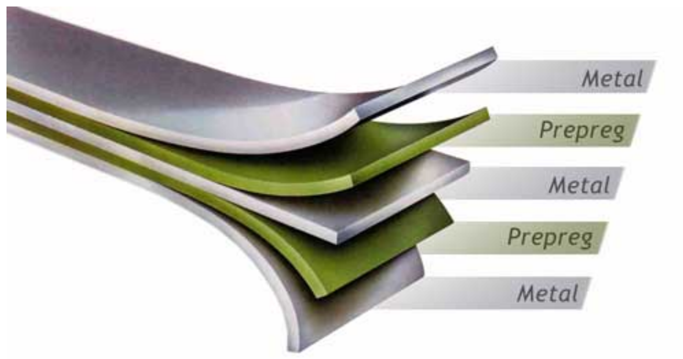


Figure 1. Illustration of Generic GLARE Lay-Up (CYTEC, 2009)

The unique properties of fiber metal laminates are what give the material its excellent fatigue characteristics and corresponding long inspection intervals. The time between fatigue initiation and the formation of a critical crack, called crack growth life, is significantly longer for fiber metal laminates compared to monolithic aluminum. “Initiation was defined as the fatigue life up until an initial crack of 1mm length has been formed. The crack growth life was then considered as the life from the initial length or greater up to critical crack lengths” (Alderliesten, 2009a, p. 1255).

Fiber metal laminates actually develop small cracks earlier in their lives than monolithic metallic structures. However, the crack growth rate in fiber metal laminates is much slower than the crack growth rates of aluminum, which allows fiber metal laminates to remain a structurally sound material for longer of periods of time. “…it is clear that the fiber bridging [in fiber metal laminates] induced slow crack growth and larger critical crack lengths…give a significant increase in life. This not only translates into longer inspection intervals, but because of the longer crack lengths that can be detected potentially during inspection, it will also shift the inspection threshold further in time" (Alderliesten, 2009a, p. 1256). This increased inspection interval could hold the key to finding additional Operations and Maintenance (O&M) savings.

Specific Background

Due to the high operations tempo seen in the early part of the 21st century, the aging process for many U.S. Air Force aircraft structures has accelerated, which in turn has caused increases in the frequency and duration of aircraft inspections. These increased inspections caused the O&M costs associated with sustaining these aircraft to increase. From 2001-2005, 37% of U.S. Air Force's maintenance man-hours were spent on inspecting and repairing cracks and corrosion associated with an aging fleet (Fredell, Gunnink, Bucci, & Hinrichsen, 2007). The structural issues just described gave birth to the need for a new material solution.

As of 2007, the average age of U.S. Air Force's aircraft was approximately 26 years (Fredell, Gunnink, Bucci, & Hinrichsen, 2007). Table 1 depicts the actual averages for specific airframes in the U.S. Air Force fleet inventory. The age of an aircraft is irrelevant if the issues and symptoms associated with the aging process can be effectively managed. The viability of these aircraft becomes a more accurate measure to gage the health of an aircraft fleet. An aircraft which must be grounded more frequently for tedious inspections or an aircraft that has weight restrictions placed on it is considered less viable than an aircraft without these maintenance burdens. Therefore, this research examines one specific method of managing these structural issues by researching the feasibility of using 3rd generation materials for 2nd generation aircraft material structures in an effort to extend the service life of the overall military weapon system and lower the O&M costs associated with sustaining these aircraft.

Table 1. Average Age of Specific U.S. Air Force Aircraft

Aircraft	Average Age(Years)
KC-135	48
C-130E	40
B-52	46.6
F-15	25.5

The aircraft in Table 1 have structures built from monolithic aluminum due to the fact that these aircraft were designed and built during the time when monolithic aluminum was the prevalent aircraft structure material being used. These materials are beginning to fatigue resulting in increased maintenance costs. “Current trends in aircraft operations are showing an increasing demand for lower operational and maintenance costs. Practically, this translates into aircraft with longer design lives, longer inspection intervals, and shorter inspection downtimes” (Alderliesten & Benedictus, 2008, p. 1184). Aircraft availability is crucial during high operational environments much like that experienced during the Iraq and Afghanistan wars.

With the average age of the C-130 fleet approaching 40 years, these second generation materials have been required to operate well beyond their initial design lifetimes. The number of critical cracks that have developed in the aircraft structures has increased significantly. The costs associated with inspecting the entire aircraft structure and the costs associated with repairing those cracks have increased. The cargo floor on the C-130 is repeatedly subjected to impacts. Over time, these repeated impacts cause cracks to form and eventually grow in length sufficient to warrant the replacement of an entire floor panel. One of the added benefits associated with fiber metal laminates is the reduction of inspection and repair frequencies typically associated with aluminum. "The need for FML structures originates from the desire to develop damage tolerant structures that are carefree (i.e. low inspection and maintenance burden)" (Alderliesten, 2009a, p. 1258). This provided an additional area where an investment made in using this material could potentially pay for itself over the material's life-cycle. GLARE was developed with the purpose of providing a feasible replacement for aluminum that lowered O&M costs.

Problem Statement

The cargo floor of the C-130 is a secondary structure that warrants research into the feasibility of using a fiber metal laminate material. The high operation tempo in Iraq and Afghanistan has resulted in increases in the number of impacts experienced due to increases in the numbers and types of missions performed. AC-130 gunships have subsequently had to increase the thickness of the cargo floor in sections, and thus adding weight, to accommodate the added stresses associated with the use of the onboard guns. "Efforts have been made to reinforce the standard cargo flooring, undoubtedly adding weight. Hybrid structures may enable this type of loading history with minimal degradation of performance" (dagsi.org, 2009). This situation more than warrants the study for a new feasible replacement material.

Research and Development management within the Air Force exists to facilitate the process of mating feasible new technology solutions to existing problems. The process consists of stating and defining a known problem and then determining what research should be accomplished to focus on feasible solutions. Lastly, it involves marketing the new technology to possible users who might be able to use the new advantages provided by this new technology.

Research Objectives

The aim of the current research is to characterize the mechanical behavior and impact resistance properties of GLARE, coupled with an economic business case analysis, to compare with published material requirements for a cargo floor and the properties 0.08” monolithic aluminum. Additionally, this effort consisted of examining the process used for mating a known material problem or issue with potential replacement materials. This process included testing the materials and analyzing the experimental data to determine if the GLARE material would be a feasible candidate for a cargo floor replacement material. To test whether or not this process was valid, an available fiber metal laminate specimen was used to obtain experimental test data for comparison to data obtained from theoretical models. Another objective of this research effort is to be able to generalize the numerical findings from the theoretical model to be able to accurately predict how other GLARE lay-ups would perform under the same given loading and bending conditions.

The material used in this research effort was GLARE-3 in a 6/5 stack, meaning the material is composed of six 0.3mm thick layers of 2024-T3 aluminum bonded together with five layers of prepreg. In this instance, each layer of bonding consisted of two separate layers: one with fibers running in a 0 degree fiber orientation and the other with fibers running in a 90 degree orientation. The GLARE-3 in a 6/5 stack was the only GLARE material available for experimental testing. Figure 2 shows the lay-up in greater detail.

0.0118 inch 2024 T3 Aluminum
0.010 inch Prepreg (0 & 90 degree fiber orientations)
0.0118 inch 2024 T3 Aluminum
0.010 inch Prepreg (0 & 90 degree fiber orientations)
0.0118 inch 2024 T3 Aluminum
0.010 inch Prepreg (0 & 90 degree fiber orientations)
0.0118 inch 2024 T3 Aluminum
0.010 inch Prepreg (0 & 90 degree fiber orientations)
0.0118 inch 2024 T3 Aluminum
0.010 inch Prepreg (0 & 90 degree fiber orientations)
0.0118 inch 2024 T3 Aluminum

Figure 2. GLARE-3 6/5 Lay-Up Illustration

Research Questions

Do the mechanical properties of this material make it feasible to serve as a suitable and qualified replacement for use as a cargo panel on C-130 aircraft? Does the material meet the minimum bending and impact resistance requirements to perform the cargo floor function? What are the advantages of using this material compared to monolithic aluminum? The main material properties being researched are impact resistance, elastic modulus, bending deflection as a function of bending moment, and possible weight savings. The importance of these properties will be discussed later. What are the life-cycle costs associated with the use of this material? Is there any potential for O&M savings? Data from designed experimentation and modeling will be analyzed in an effort to answer each research question.

Methodology Overview

The process in this research effort consists of developing a theoretical model, based on classical laminated plate theory and validated by experimental data, to predict the materials behavior under loading, bending, and impact energy in an effort to determine if this third generation material is a feasible candidate for use as a replacement material for use on an aircraft produced from second generation materials. Data related to the mechanical properties for this research effort will be generated through theoretical modeling and validated by testing the material. The experimentation will include testing to determine the material's elastic modulus, deflection properties, and impact resistance. Experiments to determine elastic modulus and bending profiles will be performed solely on GLARE as the specific mechanical properties of 2024 T3 aluminum are widely known and testing is not necessary. The data obtained from the tensile and bending tests will be compared against the known properties of aluminum and also against the design requirements for cargo floors on large wide-bodied, fixed-wing aircraft. Impact resistance testing will be performed on both GLARE and 2024 T3 aluminum to generate data to analyze GLARE's impact resistance compared to aluminum. The impacts will be examined for dimple depth and material failure. Analysis of this experimental data should provide the impact energy the material is able to withstand.

Data for use in the business case analysis will be gathered from the existing literature. Specifically, data relating to government depot maintenance labor rates, material repair frequencies and repair duration, and material cost will be obtained on

2024 T3 aluminum and GLARE for a life-cycle cost comparison. This data will then be used in a life-cycle cost analysis to predict the cost of using GLARE versus aluminum.

Assumptions and Limitations

This research effort is limited. It will not include qualification or certification of this material for use onboard a C-130 cargo aircraft. Some of the experimental limitations included the machinery available for testing. For the impact testing a 5/8" spherical impactor was not available and testing was performed using a 1/2" spherical impactor. The weights available for the impact tower were limited to the weight scheme provided by the dynatup impact tower. The closest available masses were utilized to obtain impact energies as close to the required impact energy as possible.

The type of material available for testing was another limiting factor. The material available for use in various experimentation was a 24" x 24" GLARE-3 panel in a 6/5 stack with 0 and 90 degree prepreg fiber orientations. This was the only available stacking sequence of GLARE-3 available for testing.

Significance of Study

Currently, there is not any literature on use of a fiber metal laminate onboard a military aircraft for use as a cargo floor to replace the existing monolithic aluminum or plywood. Additionally, in the research and development field, there is not a specific process that takes the material needs of a customer and produces hybrid material candidates as possible solutions for suitable material replacement. This research effort will build such a process to enable research and development personnel to determine if feasible fiber metal laminate solutions exist. That is, do these new hybrid materials meet or exceed the stated mechanical properties required from monolithic metallic structures?

Additionally, none of the cargo or tanker aircraft in the Active, Reserve, or National Guard fleet inventory currently use hybrid materials or fiber metal laminates as their cargo floor panels.

Currently, the only cargo aircraft in the United States Air Force fleet that utilized a Fiber Metal Laminate material in production was the C-17. The outer skin on rear cargo ramp doors of 40 C-17 aircraft was made from a fiber metal laminate similar to GLARE called Arall in an effort to reduce the weight in that section of the aircraft. “After qualification, the Arall-3 material could be applied in the C-17 cargo aircraft of the US Air Force. The tail section of this aircraft was too heavy and therefore a desperate search for possible weight savings in the back made the application of Arall possible” (Vlot, 2001, p. 91).

The following chapter will focus on what other researchers have accomplished in this field of study as it relates to this effort including validation of the proposed methods and testing matrix by comparing those methods to methods of several other researchers and sources in the existing literature. It will also discuss classical laminated plate theory (CLPT) and mass volume fraction; these two estimation tools or techniques were used to determine the theoretical mechanical properties of GLARE-3 and construct a theoretical model for material behavior. Chapter III will delve into the methodology behind the model development and experimentation processes in greater detail. Chapters IV and V will discuss the analysis of the experimental data and compare those results to the theoretical model. This will be followed by some discussion on the findings and suggestions for future work.

Chapter II. Literature Review

The accepted process repeated throughout literature for determining the mechanical properties of hybrid and composite materials consists of two steps. They used mathematical theory to estimate the properties of hybrid materials and various experimentation methods to test the material in an effort to gather data for comparison and eventually validation of their theoretical model. This chapter will discuss what has been accomplished in this research area. Specifically, this chapter will present the experiments and numerical theories that have been used to evaluate the tensile, bending, and impact properties of an unknown hybrid material and the resulting expected benefits of using this material.

Mathematics and Theory for Model Formulation

The mechanical properties of GLARE-3 in a 6/5 stack are not as widely published or known compared to the properties of 2024 T3 aluminum. This is due to the versatility of the GLARE material, its relative "newness" in the industry, and some proprietary manufacturing restrictions. The fact that a seemingly infinite number of "stacks" can be produced with different layering thicknesses and prepreg fiber orientation combinations make the task of cataloging the mechanical properties of each unique stack nearly impossible. However, the mechanical properties of each individual component in GLARE are known. The elastic modulus, density, and other properties are shown in Table 2 for both the 2024 T3 aluminum sheets and the prepreg material containing the S-2 glass fibers.

Table 2. Mechanical Properties of Aluminum and Prepreg (Alderliesten, 2009a)

	Aluminum	Prepreg
E1(GPa)	72.4	48.9
E2(GPa)	72.4	5.5
G12(GPa)	27.6	5.55
ν12	0.33	0.33
ν21	0.33	0.0371

The process in this research effort for determining potential candidates for a substitute material includes determining the mechanical properties of the material and comparing those properties with values provided by the manufacturer. The published figures, if any, must be checked as precaution to ensure the Air Force is receiving the specified material properties. Making the assumption that the manufacturer's numbers are correct adds risk to the process that can be easily mitigated with a few simple bench-level experiments.

While the mechanical properties of each individual component (2024-T3 aluminum, FM94K adhesive, and S-2 glass fibers) are widely known, their combined properties in a 6/5 stack are not as widely known or available, especially for fiber matrix directions other than in the 0 or 90 degree orientation. To determine the mechanical properties, the first step was to establish the materials Modulus of Elasticity (E) in the material's 1 and 2 directions for the 0, 22.5, 45, and 90 degree fiber orientations. Testing in the 67.5 degree orientation was omitted due to its symmetry with the 22.5 degree fiber orientation. Figure 3 illustrates the material with respective fiber orientations; as shown in the figure, the 1 and 2 directions are parallel and perpendicular, respectively, to the fiber orientation.

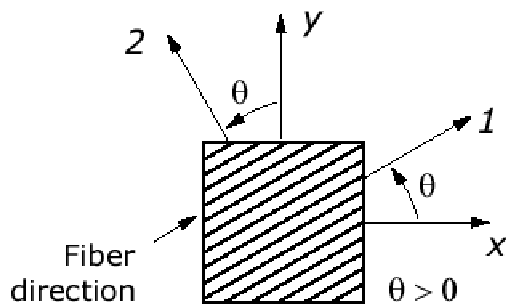


Figure 3. Material and Fiber Orientation (eFunda, 2009)

The second part of this theoretical modeling effort included using the model to predict the deflection of the material for a given bending moment. The bending experimentation thus provided the required data to validate the theoretical model. The third experimentation effort consisted of observing how the material reacted to various energy impacts and comparing those findings to the same energy impacts on a monolithic aluminum the same thickness as the C-130 cargo floor.

Mass Volume Fraction

This effort uses two of the primary material property estimation techniques or theories to validate the proposed model. The first technique used for calculating the Elastic modulus for GLARE was Mass Volume Fraction (MVF) as shown in Equation 1

$$E_1 = E_{1a}\frac{V_a}{V_t} + E_{1p}\frac{V_p}{V_t} + E_{2p}\frac{V_p}{V_t} \tag{1}$$

where… E_{1a} = The elastic modulus of aluminum in the material 1 direction

$\frac{V_a}{V_t}$ = The ratio of the volume of aluminum to the total material volume

E_{1p} = The elastic modulus of prepreg in the material 1 direction

$\frac{V_p}{V_t}$ = The ratio of the volume of prepreg to the total material volume

E_{2p} = The elastic modulus of prepreg in the material 2 direction

This formula takes into account the volume based percentage of each component: the aluminum, the 0 degree prepreg and the 90 degree prepreg with respect to the material 1 direction. Given that E_{1a}= 72,400 MPa, E_{1p}= 48,500 MPa, and E_{2p}=5,500 MPa, Equation 1 can be solved to produce 53.7 GPa or 7,788 ksi for E_1. Figure 4 depicts the linear approximation used for calculating the properties of hybrid materials and where GLARE-3 in a 6/5 stack occurred as a function of constituent material composition.

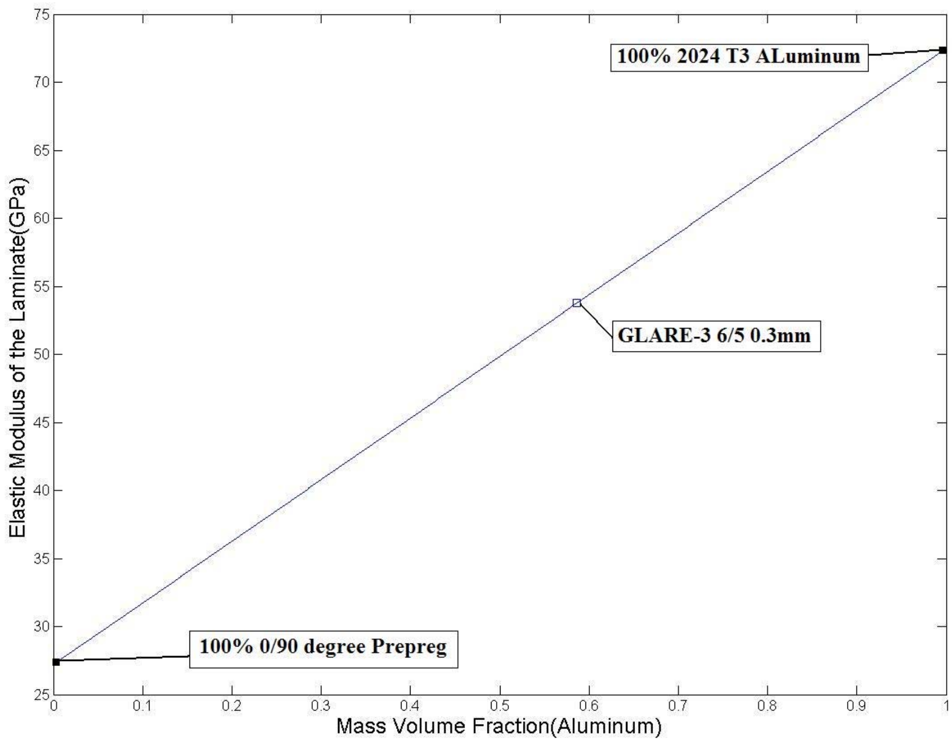


Figure 4. Elastic Modulus of the Laminate as a Function of MVF percentage Aluminum

MVF is the simpler of the two techniques used as it is simply the sum of each material's respective elastic modulus multiplied by its respective volume percentage. However, a limitation to MVF is that it is only valid for estimating the elastic modulus of the material and not intended for determining bending deflection or other needed properties. Its purpose is a correctness check for the initial development of the theoretical material model to validate the theoretical numbers produced by the proposed model. MVF is heavily cited in the literature as a means to quickly and accurately ascertain a material's elastic modulus in orientations parallel and perpendicular to the fiber directions.

Classical Laminated Plate Theory

The second theoretical method used was Classical Laminated Plate Theory (CLPT). This method is more rigorous both mathematically and conceptually. It has been used heavily in academia and among researchers in numerous areas involving hybrid materials, specifically fiber metal laminates, as a tool to estimate the mechanical properties of laminate materials when the properties of each component are known. CLPT is also useful in determining the bending profile for a given material specimen for any given load or loading situation. For this research, the calculated properties were then compared to published data for maximum allowable loading for a C-130.

One of the advantages of fiber metal laminates is their strength in the direction of the glass fibers; however, a disadvantage is their relative lack of strength in the off-axis directions. "Unidirectional reinforced fiber composites have superior properties only in the fiber-direction. In practical applications, laminae with various fiber orientations are cured together to form laminated composites which are capable of carrying loads of multiple directions. Due to the lamination, the material properties of a laminate become heterogeneous over the thickness" (Sun, 1993, p. 38). Therefore, testing the material in the off-axis directions is a must if the mechanical properties of the material as whole are to be known. This reveals just how weak the material was in the off-axis directions.

> One could also follow an engineering approach and determine the behavior of the whole glare laminate for each property and configuration. From this data the correlation between the property values of each configuration can then be established. An example of this is the Metal Volume Fraction (MVF) method. Though this method reduces the elaborate testing of laminate lay-ups, still many tests on possible laminate lay-ups with inherently complex failure modes (which are determined by both aluminum and the glass fibre epoxy layers in several orientations) remain necessary. (Hagenbeek, 2005, p. 24)

“The classical laminated plate theory is an extension of the classical plate theory to composite laminates” (Reddy, 2002, p. 112). Since it is based on classical plate theory, the main assumption for laminated plate theory is that the equations only hold true for very thin plates. A very thin plate is defined as one whose thickness is many times smaller than its width (t<<<w). Additionally, the normal directions remain in the normal direction, remain the same length, do not stretch, and do not bend. A normal is defined as direction perpendicular in two or three dimensional space.

“For a laminated composite plate, the material properties can be quite different from layer by layer, and the stress distribution may be significantly different between layers. As a result, it is desirable to express the moment and force resultants in terms of the normal and the shear stresses in each layer” (eFunda, 2009). This explains how CLPT uses the properties of each individual lamina in conjunction with the properties of all other components, to project the laminated material’s mechanical properties.

The first step using CLPT was to populate the constitutive equations which related the stress and strain in each lamina by means of the stiffness matrix [Q] based on the material properties of each independent material component as shown in Equation 2,

$$Q = \begin{bmatrix} \frac{E_{11}}{1-\upsilon_{12}\upsilon_{21}} & \frac{\upsilon_{21}E_{11}}{1-\upsilon_{21}\upsilon_{12}} & 0 \\ \frac{\upsilon_{12}E_{22}}{1-\upsilon_{12}\upsilon_{21}} & \frac{E_{22}}{1-\upsilon_{12}\upsilon_{21}} & 0 \\ 0 & 0 & G_{12} \end{bmatrix} \tag{2}$$

“Since the laminate is made of several orthotropic layers with their material axis oriented arbitrarily with respect to the laminate coordinates, the constitutive equations of each layer must be transformed to the laminate coordinates” (Kim, Thai, & Lee, 2009, p. 199). This ensures that each layer in the overall material is contributing its mechanical properties in its respective material 1 or 2 direction relative to the overall laminate material's 1 or 2 direction. The equations for the transformation matrix and its coefficients can be shown as (Reddy, 2002),

$$\begin{Bmatrix}\sigma_1\\ \sigma_2\\ \sigma_6\end{Bmatrix}=\begin{bmatrix}Q_{11} & Q_{12} & Q_{16}\\ Q_{21} & Q_{22} & Q_{26}\\ Q_{61} & Q_{62} & Q_{66}\end{bmatrix}\begin{Bmatrix}\varepsilon_1\\ \varepsilon_2\\ \varepsilon_6\end{Bmatrix} \tag{3}$$

where σ_1 = a given stress; ε_1 = the resulting strain; Q_{ij}= are determined from Equation 2

The first equation shows the laminate relation to stress and strain via the stiffness matrix. An equation like this exists for each material type within the overall laminate. These equations will need to be transformed to relate their relative material orientations to the orientation of the overall laminate. The coefficients for the transformed stiffness matrix can be shown as:

$$\begin{Bmatrix}\sigma_1\\ \sigma_2\\ \sigma_6\end{Bmatrix}=\begin{bmatrix}\bar{Q}_{11} & \bar{Q}_{12} & \bar{Q}_{16}\\ \bar{Q}_{21} & \bar{Q}_{22} & \bar{Q}_{26}\\ \bar{Q}_{61} & \bar{Q}_{62} & \bar{Q}_{66}\end{bmatrix}\begin{Bmatrix}\varepsilon_1\\ \varepsilon_2\\ \varepsilon_6\end{Bmatrix} \tag{4}$$

where $\bar{Q}_{11} = Q_{11}cos^4\theta + 2(Q_{12} + 2Q_{66})sin^2\theta cos^2\theta + Q_{22}sin^4\theta$

$$\bar{Q}_{12} = (Q_{11} + Q_{22} - 4Q_{66})sin^2\theta cos^2\theta + Q_{12}(sin^4\theta + cos^{4)}\theta = \bar{Q}_{21}$$

$$\bar{Q}_{16} = (2Q_{11} - 2Q_{12} - Q_{66})sin\theta cos^3\theta + (2Q_{12} - 2Q_{22} + Q_{66})sin^3\theta + cos\theta$$

$$\bar{Q}_{61} = \bar{Q}_{16}$$

$$\bar{Q}_{22} = Q_{11}sin^4\theta + (2Q_{12} + Q_{66})sin^2\theta cos^2\theta + Q_{22}cos^4\theta$$

$$\bar{Q}_{26} = (2Q_{11} - 2Q_{12} - Q_{66})cos\theta sin^3\theta + (2Q_{12} - 2Q_{22} + Q_{66})cos^3\theta + sin\theta$$

$$\bar{Q}_{62} = \bar{Q}_{26}$$

$$\bar{Q}_{66} = 2(2Q_{11} + 2Q_{22} - 4Q_{12} - Q_{66})sin^2\theta cos^2\theta + Q_{66}(cos^4\theta + sin^4\theta)$$

Once the matrix has been transformed for material orientation the coefficients for the plate constitutive equation can be computed. These coefficients are determined by the following formulae,

$$A_{ij} = \sum_{k=1}^{L} \bar{Q}_{ij}^{k} (z_{k+1} - z_k) \tag{5}$$

$$B_{ij} = \frac{1}{2}\sum_{k=1}^{L} \bar{Q}_{ij}^{k} (z_{k+1}^2 - z_k^2) \tag{6}$$

$$D_{ij} = \frac{1}{3}\sum_{k=1}^{L} \bar{Q}_{ij}^{k} (z_{k+1}^3 - z_k^3) \tag{7}$$

where $z_{k+1} =$ the distance from the midpoint of the laminate to the centroid of the k-th layer.

The plate constitutive equation is then used to produce material strains and bending curvatures for any given load *N* or bending moment *M*. This equation is commonly referred to as the *ABBD* matrix and is normally represented in the simpler form.

$$\begin{Bmatrix} N \\ M \end{Bmatrix} = \begin{bmatrix} A & B \\ B & D \end{bmatrix} \begin{Bmatrix} \varepsilon \\ \kappa \end{Bmatrix} \tag{8}$$

where $N = \begin{Bmatrix} N_x \\ N_y \\ N_{xy} \end{Bmatrix} \quad M = \begin{Bmatrix} M_x \\ M_y \\ M_{xy} \end{Bmatrix} \quad \varepsilon = \begin{Bmatrix} \varepsilon_x \\ \varepsilon_y \\ \varepsilon_{xy} \end{Bmatrix} \quad \kappa = \begin{Bmatrix} \kappa_x \\ \kappa_y \\ \kappa_{xy} \end{Bmatrix}$

$$A = \begin{bmatrix} A_{11} & A_{12} & A_{16} \\ & A_{22} & A_{26} \\ sym & & A_{66} \end{bmatrix} B = \begin{bmatrix} B_{11} & B_{12} & B_{16} \\ & B_{22} & B_{26} \\ sym & & B_{66} \end{bmatrix} D = \begin{bmatrix} D_{11} & D_{12} & D_{16} \\ & D_{22} & D_{26} \\ sym & & D_{66} \end{bmatrix}$$

The next section of this chapter will shift from theoretical model development to review experiments other researchers have accomplished on hybrid materials in an effort to gain a better understanding of the testing procedures.

Tensile Testing

The uni-axial tensile test is a common experiment used for producing stress strain curves. The slope of the stress strain curve in the elastic region of deformation is then used to determine the elastic modulus in a material. This type of test has been reported numerous times throughout the literature. Specifically, Cook and Donnellan (1991) performed tensile testing on specimens of GLARE similar to the material lay-up used in this effort.

"Flat, 1/2 inch wide specimens were machined per ASTM B557 by AKZO. Tests were run on an MTS closed loop servo hydraulic test machine, operated in load control" (Cook & Donnellan, 1991, p. 2). The testing machinery available for this research effort is identical to the machinery used in their experiment. This provided a solid benchmark to be used in the current experiment. Additionally, it provided numerical data values that could be used for comparison purposes to ensure the data was accurate for use in validating the theoretical model.

Hagenbeek (2005) performed tensile tests on GLARE-2, GLARE-3, and GLARE-4 variants, each in a 3/2 lay-up. The tests were performed with dog bone specimens, which allowed the experimenter to control where the point of failure would occur, which is the thinnest cross section of the specimen. Resulting stress strain curves were obtained and used to determine the experimental elastic modulus of each GLARE variant. Hagenbeek (2005) then used these values to validate his theoretical model.

Bending

Another important mechanical property to consider when searching for potential cargo floor replacement materials is the material's bending profile with respect to loading. The bending profile must be analyzed against the maximum allowable loads to predict how the GLARE material will respond. Cook and Donnellan (1991) performed similar bending tests on different lay-ups of GLARE to determine the maximum allowable shear stress. They performed a 3-point bending test using the same ASTM standards used in this research effort. Within CLPT, this effort's model should be able to predict the bending profile for a specific laminate composition for a given load or moment. Once the bending profile is determined, the maximum deflection can be found and compared to published maximum allowable displacement values for current cargo floor panels. Concurrently, the bending data from GLARE will be compared to the bending profile of 2024 T3 aluminum to see how GLARE performs compared to aluminum. The data obtained from these bending experiments will be used to compare with the theoretical values produced by the model and then validate or invalidate the model.

GLARE, due to its S-2 glass fiber composition, has been categorized as having high strain rate sensitivity, meaning that the material's stress-strain properties are dependent on the rate of loading. The material is predicted to behave differently when loaded slowly versus quickly for a given load. Therefore, one of the critical properties this material would need to possess is a high tolerance of, or the ability to withstand, repeated exposure to high strain loading conditions. This effect might possibly surface during the impact loading experimentation.

Impact

Since cargo floors are considered secondary aircraft structures, the main mechanical property of interest it its resistance to impact damage and its residual strength after impact. "…the good impact resistance of GLARE is partly due to the strain rate sensitivity of the glass fibers in the material" (Vlot & Krull, 1997, p. 1049). Moving away from the theoretical side of the literature review, there are some practical applications and instances where this material has been used in industry. The Boeing Dreamliner has installed GLARE for use as its bulk cargo floor material. "In 1990, the excellent impact properties of GLARE were put to work in a cargo floor of the Boeing 777…because floor structures are particularly prone to impact damage. This was the first commercial application of a GLARE product" (Vlot, 2001, p. 101). Although the cargo floor is not a primary structure in that it does not carry any loads required to maintain flight, it does experience repeated impact loads from day-to-day use.

Vlot and Krull (1997) performed impact testing on several GLARE variants and lay-ups, carbon composites, and monolithic aluminum using a 5/8" spherical impactor. The objective of their research was to analyze the differences between low velocity and high velocity impacts holding the impact energy constant and thus varying the mass of each impactor to achieve constant impact energies. They performed several series of experiments trying to find the minimum energy required to cause cracking on the reverse impact dimple.

Liaw, Liu, and Villars (2001) performed a series of impact tests on unidirectional and multidirectional, to include off-axis fiber orientation matricides, in ARALL and GLARE to see which properties or fiber matrix directions made the material more impact resistant than the others and to see what effect temperature had on the impact properties. They conducted some experiments at room temperature and found a range of impact energies required for a given GLARE-5 material thickness. GLARE-5 is a cousin of

GLARE-3 and is composed of the same materials, aluminum and prepreg, with a slight modification to the stacking sequence and fiber matrix orientation. In addition to the 0 and 90 degree fiber matrix, GLARE-5 contains an additional third layer of prepreg with a 45 degree fiber orientation in between each of the aluminum layers.

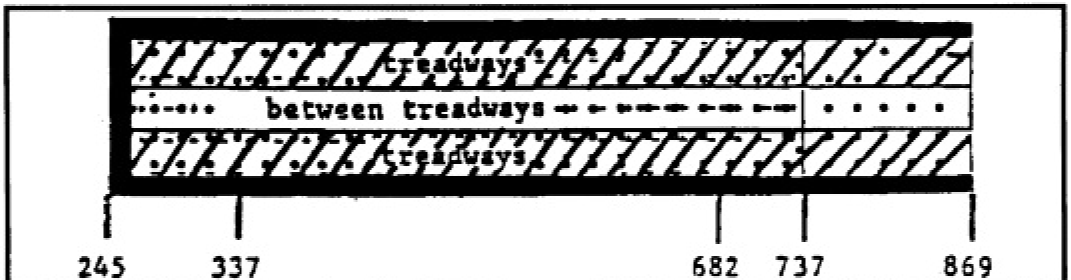
"An important feature shown after impact is that the damage in GLARE, permanent deformation and denting of the aluminum layers can be easily found by visual inspection. Delamination, matrix cracking, or fibre breakage was also found to be limited to the dent area" (Hagenbeek, 2005, p. 17). This finding illustrates the impact resistance of the material and its ability to limit damage from impact energy events to the location of the damage. Also noted in this work was the material's ability to absorb the energy from a 2.25 inch ball of ice traveling at speeds over 300 miles per hour, during a simulated hail storm strike, without denting the material or delaminating the material.

The size and shape of the impactor used will affect the results of the impact test; a large spherical impactor will be able to dissipate greater amounts of energy, thereby allowing the material to absorb larger amounts of energy before cracking occurs (Liu & Liaw, 2007). This fact is important for making comparisons to other energy values found in other research efforts. Liu and Liaw (2007) conducted several experiments on GLARE-2 and GLARE-3 in 0.05 inch thicknesses to obtain data on the energy each variant was able to absorb as a function of impactor shape and size.

Currently, the only Air Force aircraft application of a fiber metal laminate has been on the outer skin section of the aft cargo ramp on 40 C-17 transport aircraft. While the GLARE material has been applied to the bulk cargo floors on a commercial airliner, the idea of using a GLARE variant as the cargo floor material onboard a military cargo aircraft has not been published in the literature.

Cargo Floor and Material Requirements

Currently most if not all wide-body Air Force cargo floors are designed to carry cargo loads and withstand low velocity impacts experienced during routine day-to-day loading and unloading activities. Figure 5 depicts the maximum allowable cargo floor loads for an Air Mobility Command (AMC) C-130 aircraft.



FUSELAGE STATION	*257-337		337-682		682-737		RAMP	
	TREAD-WAYS	BETWEEN TREAD-WAYS	TREAD-WAYS	BETWEEN TREAD-WAYS	TREAD-WAYS	BETWEEN TREAD-WAYS	TREAD-WAYS	BETWEEN TREAD-WAYS
PSI	50		50		50		50	
PLF	1400		3000	1800	1400		500	
TONGUE[1] LOAD		2000		2000		2000		450
MAX AXLE LOAD	6000	5000	13000	5000	6000	5000	2500[2]	1200
MAX WHEEL LOAD	3000	2500	6500	2500	3000	2500	1250	600

[1] Do not exceed 50 pounds per square inch (psi).

[2] A single axle up to 3,500 pounds in weight may be transported on the aircraft ramp, provided it is the only item on the ramp.

Figure 5. C-130 Flight Limitations Chart (Field Manual (FM) 55-9)

Analyzing Vlot and Krull's (1997) experiments with various GLARE stacks and thicknesses, it was determined that the minimum energy required to cause cracking for the GLARE-3 6/5 stack would be in the neighborhood of 40 Joules using a 1/2” spherical impactor. Cracking is the term used to describe the crack produced by a failed material associated with the resulting dimple caused by the impact event. “Damage mechanisms due to low energy impact onto composite panels have been studied extensively. In general they are characterized by incomplete penetration with damage consisting of delamination, matrix cracking, and fiber failure. However, the exact failure mode and damage evolution sequence are greatly affected by the constituent properties, lay-up, configuration, thickness, and bending rigidity. “For instance, experimental studies consistently report that very often delamination occurs at the interface between plies with different fiber orientation” (Liaw, Liu, & Villars, 2001, p. 536).

Economic Advantages

The weight savings GLARE has to offer, due to the lower density of the material, and the advantageous mechanical properties seen in many fiber metal laminates provides an opportunity to provide a replacement material that lowers the overall weight of the aircraft. “…the glass fibre epoxy layers [in GLARE] have a considerably lower weight than monolithic aluminum, 1.96g/cm^3 vs. 2.77 g/cm^3, and can offer approximately 10% material and 20-30% structural weight reduction even for ‘cross-plied’ laminates” (Hagenbeek, 2005, p. 16). Weight reduction for war-fighting aircraft makes their aircraft more versatile. Every pound of aluminum they are able to shed translates into one pound more of fuel or cargo, which helps aircraft fly further and accomplish more. Aircraft with serious weight restrictions, such as the AC-130 gunships which constantly push their maximum allowable wartime takeoff weights, could benefit greatly from this new material.

"AFSOC's [Air Force Special Operations Command] challenges require listing to keep them in view such as the fleet-wide problems in the C-130 aircraft with the discoveries of the center-wing box cracking problems due to extended combat operations at high gross weights. This will result in the grounding for repair of approximately one third of all AFSOC aircraft over at least the next two to three years" (Murdock, Grant, Comer, & Ehrhard, 2007, p. 28). This research effort will look into whether or not the GLARE material in its current lay-up would meet the published requirements for maximum loading and impact resistance, and then examine the amount of weight savings that could be realized if the material becomes certified for use on military aircraft.

The potential for weight savings along with reduced maintenance costs and aircraft downtime, are critical if this technology is to be adopted. Unless a new technology meets a critical flight safety issue or saves a human life, without some form of "payback" or potential long-term life-cycle cost savings, no agency in the Air Force would unlikely be willing to accept the use of GLARE material. Additionally, the technology transfer community would probably not be willing to take on the additional workload of further testing the material without some sort of added benefit associated with using the material. Although "payback" is typically monetary, the potential weight savings may also bring added capability to current aircraft which would also be considered some form of payback.

Experimentation Phases

The research and development career field manages only part of a new technology's journey as it climbs the testing and validation pyramid towards actual use in the field. The scope of the process mentioned in this research effort would be to take known user requirements and mate them with potential material solutions. Perhaps those material solutions are at a relatively low technology readiness levels such as level two or level three. The job of this overall process is to obtain a greater understanding of these new technologies, in this instance their mechanical properties, to achieve a greater understanding of how the technology behaves in early small-scale experiments, then component testing, and finally full-scale testing and certification

The Aircraft Structural Integrity Program (ASIP) exists to facilitate the safe transfer of new technology toward potential uses or applications. The entire ASIP process is subdivided into five different areas or tasks. Each of these tasks refer to five distinct phases the structure of an aircraft passes through along its life-cycle from material and structure conception all the way to management and sustainment activities. The research effort explained in this thesis would represent part of the activities associated with Task One, the design information phase. In this phase small scale material testing occurs to determine the material's feasibility for use as part of an aircraft structure. Once all the parts in the first phase or task of the ASIP process have been completed and agreed upon, the program passes into Task Two, where larger scale testing, perhaps testing of assemblies, may begin. Eventually, this process will deliver a certified material or even an entire aircraft to the warfighter as it enters the delivery and sustainment phase.

The next chapter will review the methodology used in greater detail and describe the various experiments that were performed, along with the related standards and theory behind the tests and the analysis used to interpret the data. Additionally, the formulation of the economic business case will be discussed in detail, with the results and implications being discussed in Chapter IV.

Chapter III. Methodology

This chapter will discuss the specifics regarding data collection and analysis for both the mechanical properties investigation of GLARE and the economic business case. The mechanical properties were first determined through modeling and validated by experimental testing. Tensile, four-point bending, and impact testing will provide the data necessary to answer the question as to whether or not GLARE's mechanical properties will meet the requirements of a cargo floor. To determine if GLARE possesses the required properties, the experimental data will be compared to the published properties of 2024 T3 aluminum. The adherence to credible experimental standards was paramount in this thesis effort to ensure the data collected was of the highest quality for use in validating the theoretical model. The more credible the experimental data, the more credible the theoretical model. The business case consisted of using cost data obtained from literature to predict the life-cycle costs for both aluminum and GLARE over a 30-year period. Additionally, a sensitivity analysis was performed to add confidence to the business case.

Tensile Testing

In the uni-axial tension experiments, ASTM standard D3552 was used to determine the specimen geometry and procedures required to perform the tensile testing. However, the scope of the tensile test was found to be outside the region specified by this standard. "This test method covers the determination of the tensile properties of metal-matrix composites reinforced by continuous and discontinuous high-modulus fibers. This method only applies to specimens tested in the direction of reinforcement or perpendicular to the direction of reinforcement" (American Society of Test and Measurement, 1976).

Since the tensile testing matrix included tests on materials with fiber orientations of 0, 22.5, 45, and 90 degrees, the experimental test would be operating outside the scope of this technical standard. This issue is addressed later in the chapter and discussed in greater detail in Chapter IV. The tensile testing utilized a 5 kip MTS machine and an extensometer. Four specimens of each fiber orientation mentioned above were tested under tension until failure and the data was recorded by the MTS machine and the extensometer. The data collected included the load, displacement, strain, and time. Sixteen dog bone test specimens, each measuring four inches in length and half an inch in width with a quarter inch neck, were excised from a two foot square panel of GLARE-3 according to specifications stated in the ASTM standard mentioned above.

Bending

The bending portion of the experiment was performed according to ASTM Standards D7264, Standard Test Method for Flexural Properties of Polymer Matrix Composite Materials, and D790 Standard Test Methods for Flexural Properties of Unreinforced and Reinforced Plastics and Electrical Insulating Materials. This combination of standards was used to best approximate the standards required for testing a Fiber Metal Laminate since a standard for testing the flexural properties of a hybrid material currently does not exist. There were minor differences in each of the testing standards and no issues were discovered that would have caused the invalidation of this test.

Six specimens total, three each with 0 and three with 90 degree orientations, were excised and prepared according to the ASTM standards. The bending specimens were rectangular in shape measuring five inches in length and half an inch in width. The same 5 kip MTS machine was utilized in this experiment; however, the upper and lower tension-tension jaws were replaced by a 4-point bending apparatus. Figure 6 illustrates the experiment set-up. The displacement of each GLARE-3 specimen was measured with a laser calibrated and aimed at the midpoint of the specimen where maximum deflection would occur. Data was collected on the MTS machine from the load cell and the laser. The data collected was load, maximum displacement, and time.

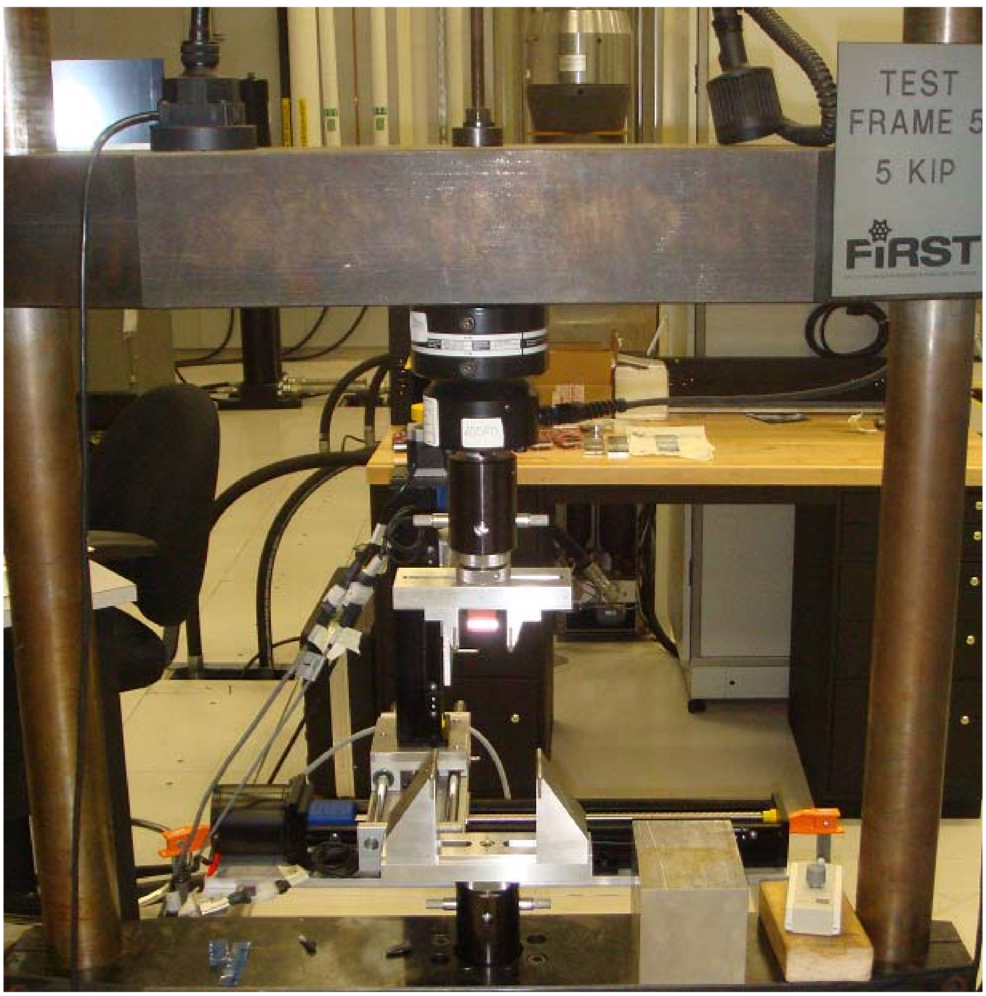


Figure 6. Four Point Bending Test Apparatus

Impact

The impact testing was accomplished in a manner similar to experiments conducted by Vlot and Krull (1997) and Liu and Liaw (2007). The objective of the impact tests was to conduct a series of impact tests at increasing energy ranges on both GLARE-3 and the material currently used for the C-130 cargo floor in its required thickness. This was done to have a direct comparison between the 0.08” thick monolithic 2024 T3 aluminum, the current material used as the C-130 cargo flooring, and the GLARE material.

The first series of impact tests ranged from 20 Joules to 100 Joules, with the velocity held constant 4.2 m/s and the mass of the impactor varied to reach the desired energy event. To determine the mass required to achieve the required energy events, the following widely known formula was used:

$$Energy = \frac{1}{2} m V^2 \tag{9}$$

where m=the mass of the tup and drop carriage and V = the velocity of the tup the moment before impact.

The second series of impact tests were conducted holding the mass constant at 2.5 kg and varying the velocity to achieve the desired range of 20 to 100 Joules. These two series of tests were performed to discover the differences, if any of varying mass or velocity on the impact crater. Tests were performed from the low energy 20 Joules and increased until either penetration occurred or the 100 Joule upper range was reached. The impact testing was performed on a Dynatup impact tower. The tests in which the velocity was held constant were performed using gravity and height to achieve the desired velocity. The tests performed holding the mass constant utilized preloaded springs controlled by compressed air to achieve the desired impact velocities. Figure 7 shows the Dynatup impact tower and experimental test set-up. Table 3 depicts the test matrix used for both the 2020 T3 0.08" aluminum and the GLARE impact strips.

Figure 7. Dynatup Impact Testing Apparatus

Table 3. Impact Testing Numerical Test Matrix for GLARE-3 and 0.08 inch 2024 T3 Aluminum respectively

Test #	Mass(kg)	Velocity (m/s)	Energy (J)		Test #	Mass(kg)	Velocity(m/s)	Energy (J)
1	2.77	4.2	20		1	2.5	4	20
2	4.54	4.2	40		2	2.5	5.65	40
3	6.81	4.2	60		3	2.5	6.92	60
4	9.08	4.2	80		4	2.5	8	80
5	11.24	4.2	100		5	2.5	8.95	100
	Mass(lb)	Velocity(ft/s)	E(ft-lbs)			Mass (lb)	Velocity(ft/s)	E(ft-lbs)
1	6.10	13.77	14.76		1	5.5	13.12	14.76
2	10	13.77	29.52		2	5.5	18.55	29.52
3	15	13.77	44.29		3	5.5	22.69	44.29
4	20	13.77	59.05		4	5.5	26.24	59.05
5	24.77	13.77	73.75		5	5.5	29.35	73.75

Data Analysis

Once data was gathered from each of the three experimental tests, it was compared to values generated by the theoretical model for validation. The two main properties of interest were bending deflection and impact resistance. Data obtained from the model and experiments were used to generate a bending profile for GLARE that displayed the maximum deflection of the material at its midpoint for a given load and respective bending moment. That bending profile was then compared to a bending profile of 0.08 inch 2040 T3 aluminum to see how the two materials compared.

Impact data was collected on both GLARE and 0.08 inch 2024 T3 aluminum. The damage occurring from each impact energy event was recorded along with the minimum energy required to cause reverse side cracking. Both GLARE and aluminum were analyzed for impact resistance and the minimum energy required to cause cracking.

Business Case

A business case analysis was developed by comparing the life-cycle costs associated with the use of GLARE or 2024 T3 aluminum over a 30-year period. The life-cycle costs for each material were determined by adding all material and labor costs associated with using each respective material. A sensitivity analysis was performed on the GLARE life-cycle cost to add a level of confidence to the figures and to see how sensitive the life-cycle costs would be to a ±10% margin of error. The results of the business case and the experiments can be seen in the following chapter.

Chapter IV. Results

This chapter will first discuss the results obtained from each of the three experiments and the business case. Based on these results this material's potential as a replacement material for the C-130 aircraft cargo floor will be discussed.

Experimental Results

The results of the tensile testing are shown graphically in Figures 8, 9, 10, and 11. Each of the four respective figures display the experimental stress versus strain curves obtained from each respective fiber orientation. The slopes of each curve were calculated and averaged to show the experimental elastic modulus obtained from the testing. The values are shown in Table 4 with the theoretical values for easy comparison.

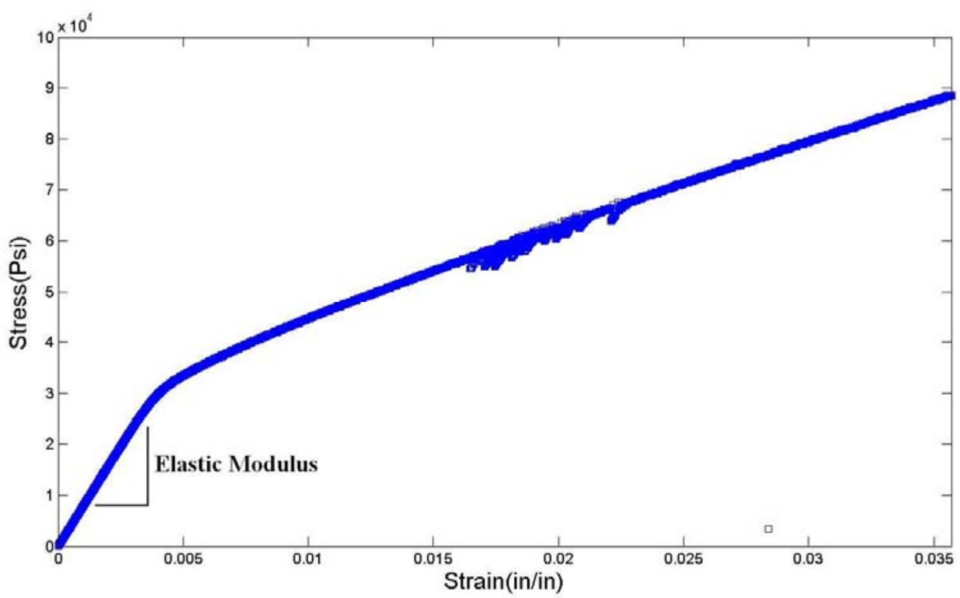


Figure 8. Stress vs. Strain Curve in the 0 Degree Fiber Orientation

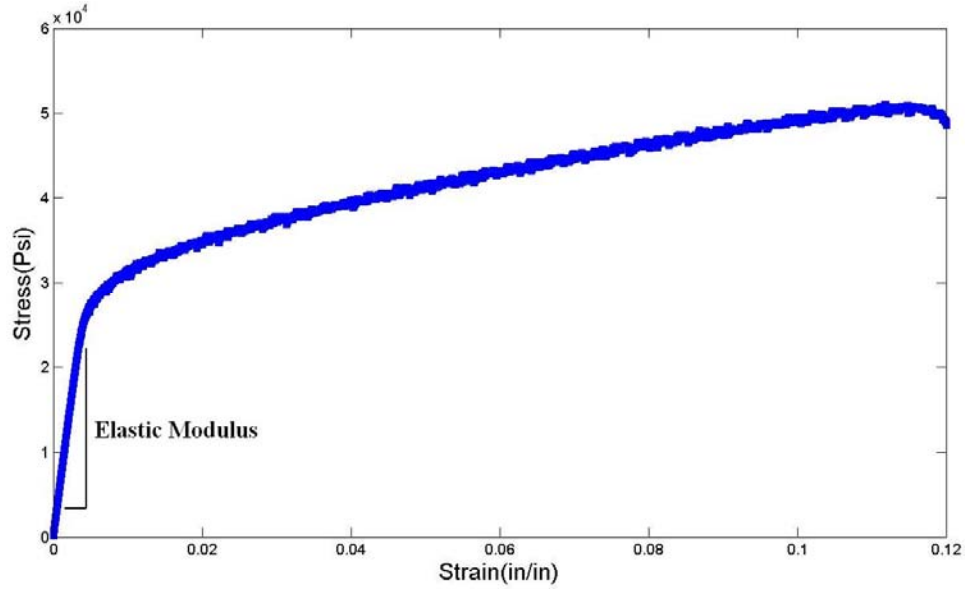


Figure 9. Stress vs. Strain Curve in the 22.5 Degree Fiber Orientation

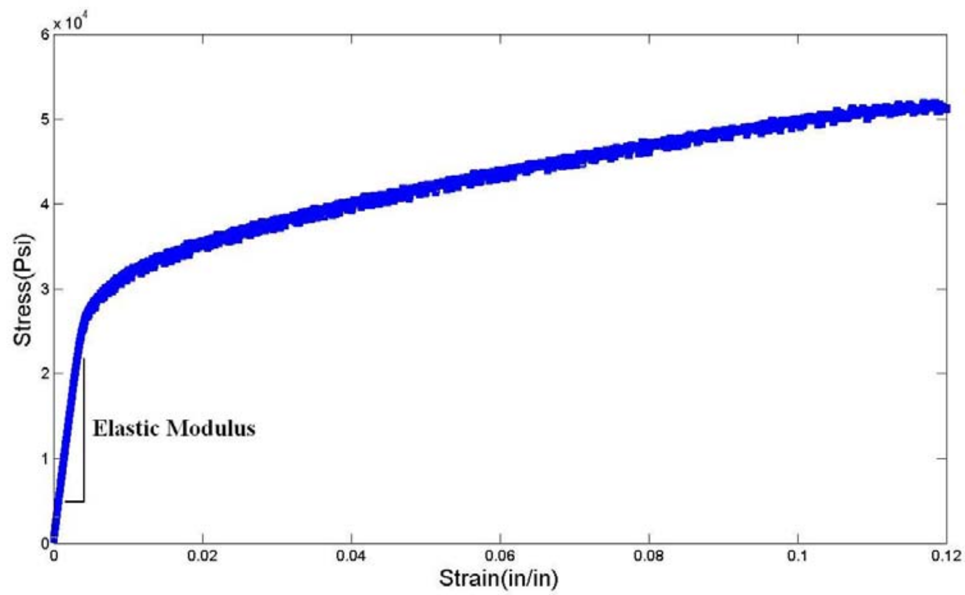


Figure 10. Stress vs. Strain Curve in the 45 Degree Fiber Orientation

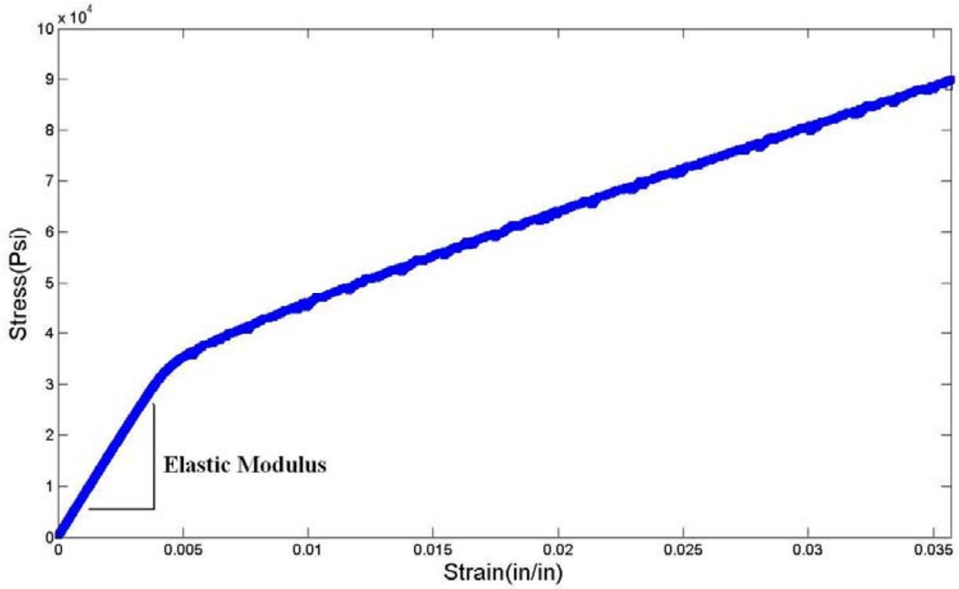


Figure 11. Stress vs. Strain Curve in the 90 Degree Fiber Orientation

Table 4. Theoretical and Average Experimental Elastic Modulus Results

Degrees	Theoretical		Experimental	
	ksi	GPa	ksi	GPa
0	7804	53.7	7755	53.4
22.5	7463	51.2	6558	45.2
45	7046	48.5	6385	44
90	7804	53.7	7765	53.5

The experimental results and the theoretical values for the Elastic modulus were extremely close for the 0 and 90 degree fiber orientations; however, severe differences were discovered between the experimental and theoretical values for the 22.5 and 45 degree fiber orientations. The average elastic modulus for the four zero degree tensile tests was 7,755 ksi with a standard deviation of 24.06. The average elastic modulus for the four 22.5 degree tests was 6,558 ksi with a standard deviation of 15.55. The average elastic modulus for the four 45 degree tests yielded 6,385 ksi with a standard deviation of 10.82. The average elastic modulus for the four 90 degree tests yielded 7,765 ksi with a standard deviation of 18.16. This indicates the mechanical property consistency that can be expected from this material.

This difference in the off axis specimens was explained through simple geometry. The tensile test coupon geometry limited the number of fibers, and in this case completely eliminated the available fibers, which were available to handle a given load in tension. With a fiber orientation of 22.5 and 45 degrees, the coupon geometry does not allow for any of the fibers to completely transverse the length of the tensile coupon, which essentially eliminates the fiber's load carrying contribution.

When the contribution of the fibers in the mass volume fraction equations were removed, the experimental values more closely matched the expected elastic modulus values. In GLARE-3 with a 6/5 stack, aluminum occupies a 0.5863 mass volume fraction. When multiplied by the elastic modulus of aluminum (10,600 ksi), the resulting value of 6,240 ksi was close to the experimental data. If the minor contributions of the FM-94K adhesive are included, the adhesives MVF of 0.4137 is multiplied by its respective elastic modulus of 377 ksi to yield a value of 156 ksi this results in a combined elastic modulus of 6,395 ksi, which was extremely close to the experimental results obtained from testing. The geometry of the test coupon thus removed any load carrying contribution of the fibers; therefore the aluminum and epoxy resin were the only materials available to handle the load.

In future research efforts when conducting tensile tests of fiber metal laminates, it is important to ensure that the test coupon geometry is created in such a manner that the fibers are allowed to run from end-to-end to give the fibers a chance to handle the load imposed upon the coupon as a whole. Figure 12 depicts the theoretical values and respective elastic modulus curve as a function of the fiber orientation for the experimental data. Additionally, Figure 12 depicts the elastic moduli of 100% MVF 2024-T3 aluminum and 58% MVF to bracket the results and illustrate the contributions of the fibers in the respective orientations.

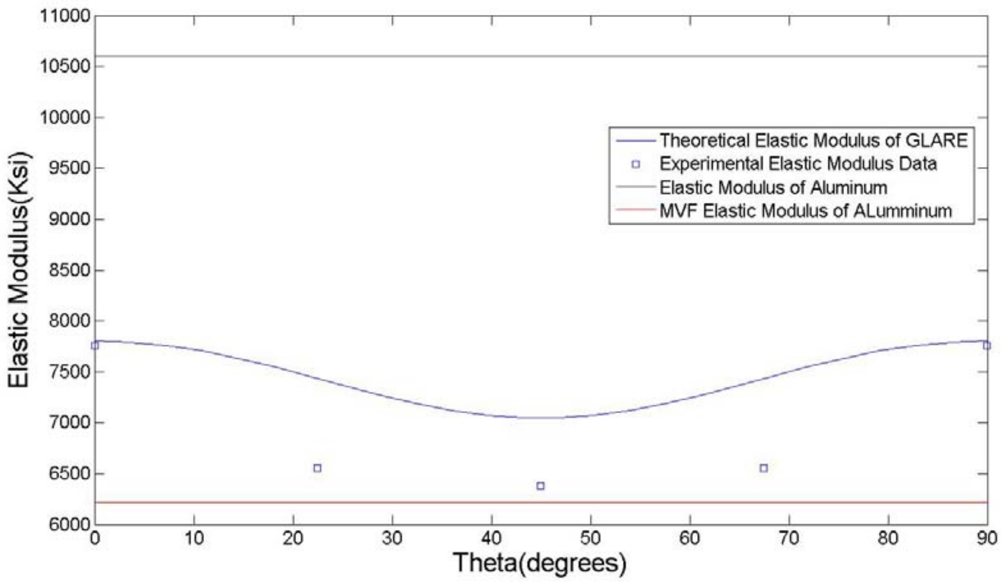


Figure 12. Graph of Theoretical and Experimental Elastic Modulus Values versus Fiber Orientation

As shown in Figure 12, the theoretical values obtained from the model displayed a flattened s-curve. The values obtained from the experimental data appeared to be significantly lower than the predicted values. This exposes a limitation associated with small coupon testing at the lower level of material testing. It is believed that the GLARE material, in the 22.5 and 45 degree fiber orientation, would exhibit mechanical properties closer to the theoretical values for a full-sized panel with a geometry such that the fibers would be present to transfer the load all the way through the material.

The experimental results of the bending can be seen in Figure 13, along with the theoretical bending slope. The slopes of both the theoretical values for bending displacement and the experimental results are quite similar with the exception of the first portion at the onset of material loading. This is due partly to the test methodology and the material characteristics. The test was performed in load control with a loading rate of 50 pounds force per minute instead of being performed in displacement control, which would have produced a more linear displacement curve at the origin. The material itself, due to the numerous layers, should also have a certain amount of internal “wiggle room” in which the bending load is absorbed by the material without showing significant overall displacement. Similar to a locomotive at its onset, the main engine may move 50ft before the caboose begins to move due to all the slack associated with each individual train car. The same phenomenon may have occurred with the GLARE material as the individual fibers needed to adjust to the loading before the whole beam began to bend as one rigid bar.

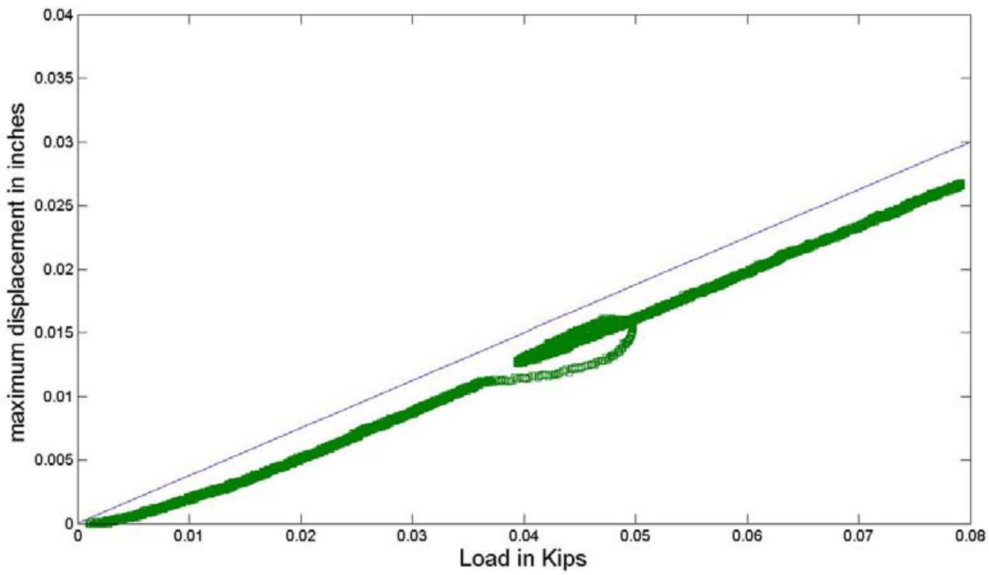


Figure 13. Displacement of GLARE versus Load for 4-point bending in 0 Degree Fiber Orientation

Data on the requirements for a C-130 cargo floor and maximum allowable deflection could not be found. Therefore, the GLARE results were compared against the mechanical properties of 2024 T3 aluminum, which are readily known. Figure 14, thus shows the bending slopes for GLARE and 2024 T3 under identical loading conditions and the associated moments ranging from a 0 to 90 lb force. The loop in the experimental graph was thought to be the result of a slight sticking of the rollers on the 4-point bending apparatus as the experiment was performed. The results seem to indicate that GLARE-3 in a 6/5 stack deflects less when compared to the current 0.08" aluminum floor for the same given load producing the same resulting bending moment. The stiffness of GLARE in a 6/5 stack could possibly exceed requirements and add unnecessary capability and perhaps weight.

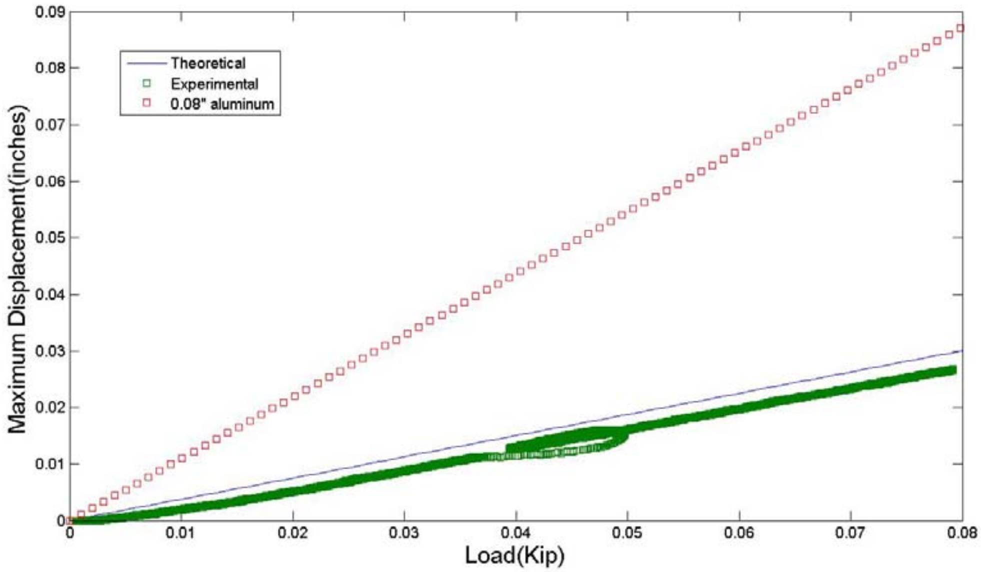


Figure 14. Comparison of Maximum Bending Displacement between GLARE and 0.08" 2024-T3 Aluminum

The results of the impact properties of GLARE are perhaps the most important factor since cargo floors built for ruggedness and their ability to withstand the impacts associated with day-to-day operations. Figure 15 depicts the results of the first series of impact testing in which both the GLARE-3 in a 6/5 lay-up and 0.08” 2024 T3 aluminum specimens were subjected to the test matrix shown in Table 3 of Chapter III.

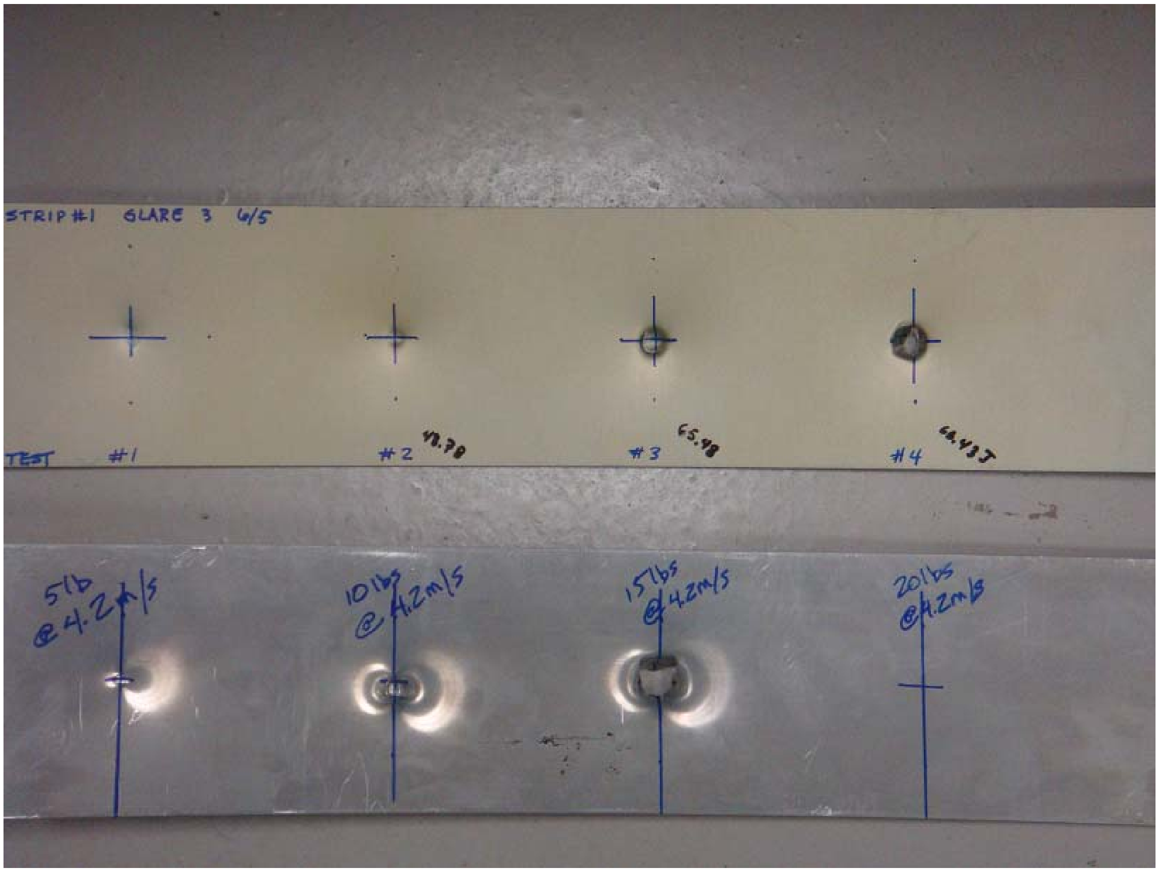


Figure 15. Impact Test Results for GLARE and 0.08" 2024-T3 Aluminum for 20J, 40J, 60J and 80J Energy Events

In the first series of impact tests, the velocity at impact was held constant and the mass was varied to obtain the desired impact energy. The impact energies ranged from 20 Joules to 100 Joules and testing was aborted once the tup penetrated the material. Obviously there was no need to perform the 20 lb mass impact test on the aluminum sheet since the penetration was achieved on the previous test. For these tests the desired metric was the minimum energy required to cause cracking on the reverse side dimple. Figures 16, 17, 18, and 19 display the progression of impact energy events and the energy event required to cause the dimple on the reverse side of the respective materials to crack.

One major finding was that the minimum energy required to cause cracking on the reverse side dimple was 20 Joules higher for the GLARE than for the 2024 T3 aluminum. The other finding was that varying mass or velocity, while holding the other constant, did not seem to have an effect on the outcome with respect to the minimum energy required to cause cracking or puncturing for all velocities under 10 m/s. Both series of impact tests indicate that the GLARE material performs better compared to the 0.08" aluminum under controlled low velocity impact testing. The aluminum consistently punctured at the 60 Joule impact energy and cracked at the 40 Joule impact energy, while the GLARE punctured at the 80 Joule impact energy and cracked at the 60 Joule impact energy.

Figure 16. GLARE Reverse Side Dimpling and Cracking as a Function of Impact Energy Progression (20, 40, 60 and 80 Joules) with Constant Velocity

Figure 17. 0.08" 2024-T3 Aluminum Reverse Side Dimpling and Cracking as a Function of Impact Energy Progression (20, 40, and 60 Joules) with Constant Velocity

Figure 18. Reverse Side Dimpling and Cracking on GLARE as a Function of Impact Energy Progression (20, 40, 60, 80, and 100 Joules) with Constant Mass

Figure 19. Reverse Side Dimpling of 2024-T3 Aluminum as a Function of Increasing Impact Energy Progression (20, 40, and 60 Joules) with Constant Mass

Engineering Analysis

GLARE-3 can exist in a variety of thicknesses depending on the combinations of thin layers of aluminum and prepreg. In this effort, the mechanical properties of GLARE-3 in a 6/5 lay-up were determined both theoretically and experimentally. Discrepancies in the values were easily explained through the analysis of the coupon geometry such that the model was validated by the experimental data results. Further analysis of the data showed that the mechanical properties of GLARE-3 with the 6/5 lay-up met or exceeded the mechanical abilities of 0.08” 2024 T3 aluminum, since the GLARE-3 material in a 6/5 lay-up is heavier per unit volume than the 2024 T3 aluminum, different configurations of the GLARE material were considered. The goal was to determine if different lay-ups of the GLARE-3 material could be produced to allow the thickness to remain equal to or less than the current 0.08 inch thick aluminum yet retain properties similar to the 6/5 stack. Figures 20 and 21 show the relationship between the elastic modulus multiplied by the second moment of inertia (EI_{yy}) and cross sectional area, respectively, for other available GLARE-3 variants. These figures were based on the Euler-Bernoulli equations governing displacement in very thin plates.

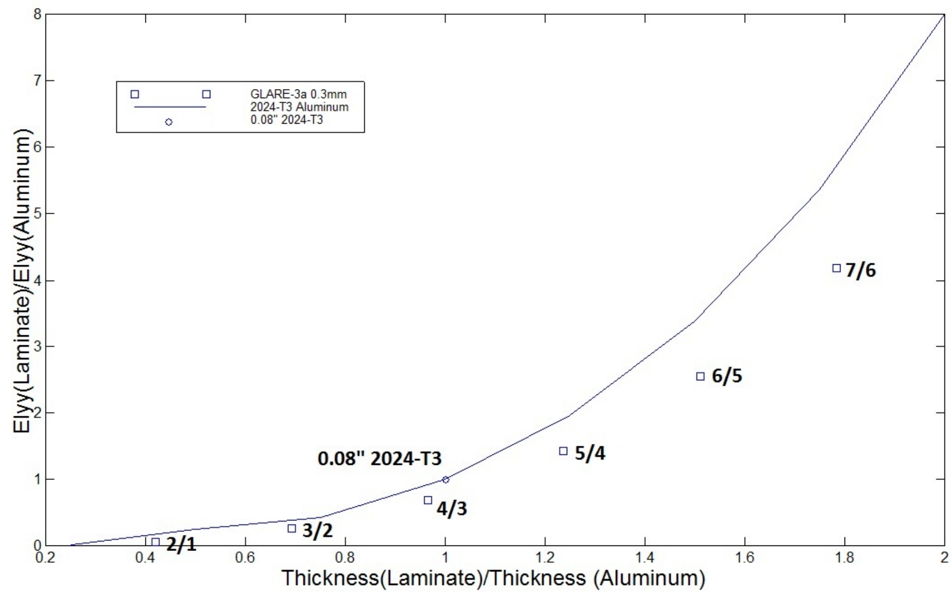


Figure 20. Comparison of EI_{yy} between 2024-T3 Aluminum and Several GLARE-3 Variants

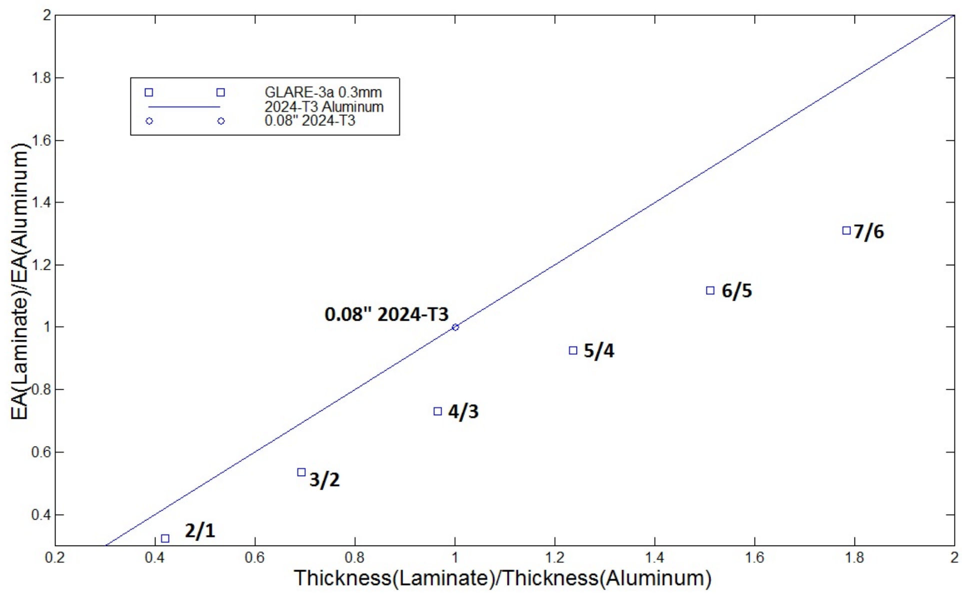


Figure 21. Comparison of EA between 2024-T3 Aluminum and Several GLARE Variants

Two GLARE-3 variants, the 4/3 and 5/4 lay-ups, had the smallest vertical distances from the EI_{yy} value of the current 0.08 inch thick aluminum cargo floor, meaning that these two lay-ups were predicted to behave most similarly to the current cargo floor paneling in terms of material deflection for a given force applied over the plate area. Moving farther away from the EI_{yy} of aluminum, in either direction, the GLARE variants would either act too stiff, or even worse, too soft when compared to the current aluminum floor. Figure 21 illustrates how close each GLARE variant is relative to the product of its respective thickness and elastic modulus.

Since GLARE-3 is available with 2024 T3 aluminum thicknesses of 0.3mm, 0.4mm, and 0.5mm, respectively, further analysis was completed on each lay-up (2/1, 3/2, 4/3, 5/4, 6/5, and 7/6) with the three different thickness possibilities. The analysis performed in Figure 20 was expanded to include all three available aluminum thicknesses for each respective stacking sequence. A close up of Figure 22, shown as Figure 23 is also presented to better illustrate the variants closest to the 0.08" 2024 T3 Aluminum.

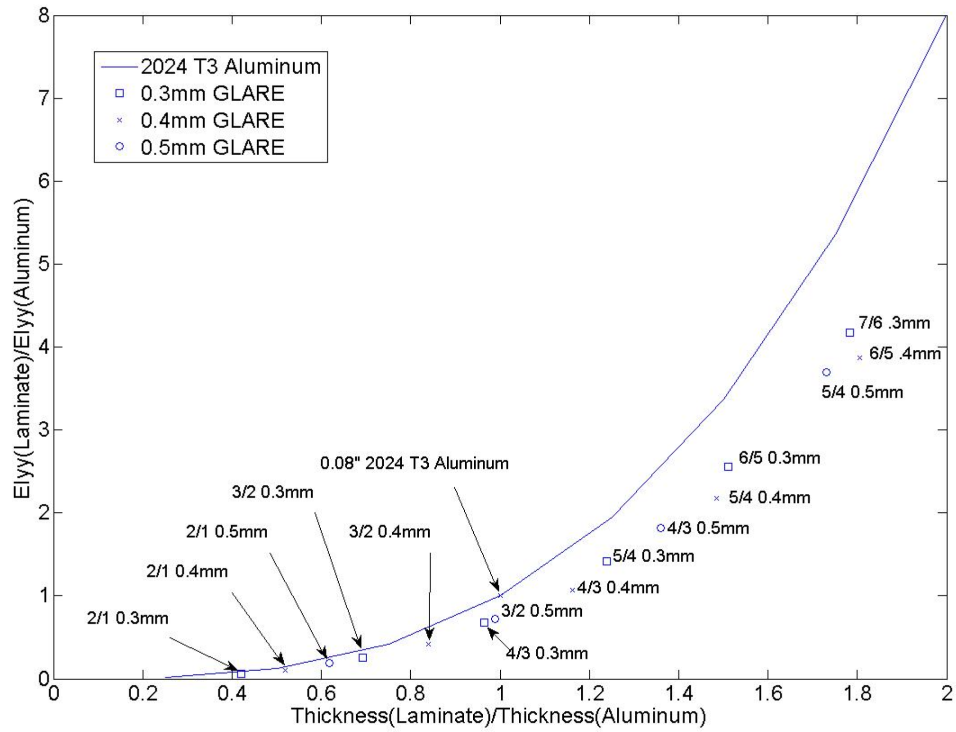


Figure 22. Comparison of EI_{yy} Between 2024-T3 Aluminum and Several GLARE Variants for Aluminum Thicknesses of 0.3mm, 0.4mm, and 0.5mm

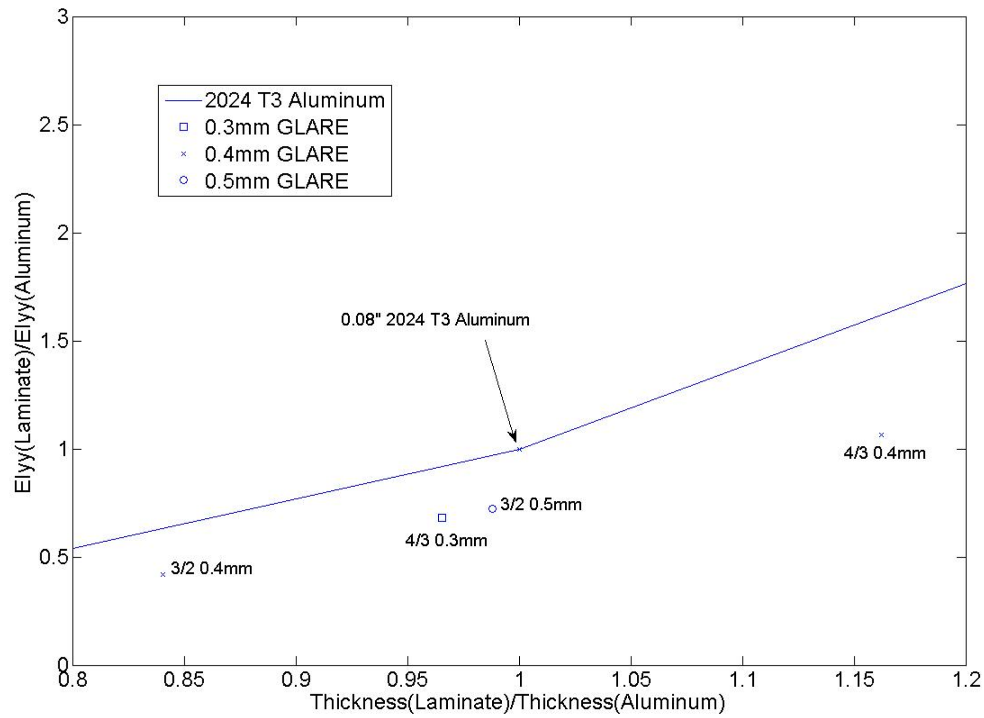


Figure 23. Close Up of Figure 22

These figures indicate that GLARE-3 in a 0.4mm 4/3 stacking sequence would expect to behave nearly identical to the current cargo floor in terms of its bending deflection as a function of imposed load. Additionally, GLARE-3 in a 0.4mm 4/3 stacking sequence would increase the thickness of the current cargo floor 0.1 inches, but would not increase the overall weight of the cargo floor as the 6/5 lay-up. Further analysis and testing of each of the three closest variants depicted above would be beneficial in the effort to determine the optimal stacking sequence and aluminum layer thickness for the cargo floor material application.

Based on the results from Figures 20, 21, 22, and 23 and impact testing results, GLARE-3 in the 4/3 0.4mm stack would provide a close match for material thickness while providing a superior impact resistant material. This is consistent with results reported in the literature. "Glass fibre reinforced Fibre Metal Laminates (GLARE FML) show an approximately equal or 15% better specific minimum cracking energy at low velocity impact (10 m/s) compared to monolithic aluminum" (Vlot & Krull, 1997, p. 1050). Therefore, this would indicate that a fiber metal laminate of equal or slightly less thickness as its aluminum counterpart would still possess approximately 15% better damage tolerance capabilities.

This led to the analysis seen in figures 20, 21, 22, and 23, which show numerous GLARE-3 stacks ranging from a 2/1 stack to a 7/6 stack. The independent variable used in each graph was the ratio of the laminate thickness to the 0.08 inch thickness of the current cargo floor. The dependent variable in Figure 22 is the elastic modulus of the laminate multiplied by the cross sectional area of the laminate as a ratio against the elastic modulus of the cargo floor material multiplied by its respective cross sectional area. The dependent variable in Figure 20 is the elastic modulus of the material multiplied by its moment of inertia compared to the elastic modulus of the aluminum multiplied by its respective moment of inertia. It was observed that GLARE-3 in a 4/3 0.4mm stack was the closest variant of GLARE 3 to the 0.08 inch aluminum point on both graphs without sacrificing any rigidity.

Additionally, GLARE-3 in the 4/3 stack would have an inherent weight savings of approximately 15 %. GLARE-3 in a 4/3 stack has a density of 2471 kg/cubic meter while 0.08" 2024 T3 aluminum has a density of 2780 kg/cubic meter. A simple glance at the impact data from Vlot's (1997) research would indicate that for low velocity impacts, GLARE-3 in a 4/3 stacking sequence would still maintain the "ability to withstand ~15%

or better specific minimum cracking energy" (Vlot & Krull, 1997, p. 1051). This material in a 4/3 stacking sequence would still retain the desired impact resistance without adding to the overall thickness of the current material or adding any additional weight. In fact, this material in a 4/3 stacking sequence would reduce the overall weight of the cargo floor as its density is lower than 2024 T3 aluminum. Figure 24 depicts other minimum energy values required to cause cracking in various GLARE-5 thicknesses (inches) taken from Law, Liew, and Villars (2001). The GLARE material tested with a thickness similar to the 0.08" cargo floor performed similarly to the aluminum tested in this research effort. Additionally, these figures show that decreasing thickness has a logarithmic decrease in the minimum energy required to crack the opposite outer layer of aluminum. This further illustrates the ability to use less of this material without sacrificing capability.

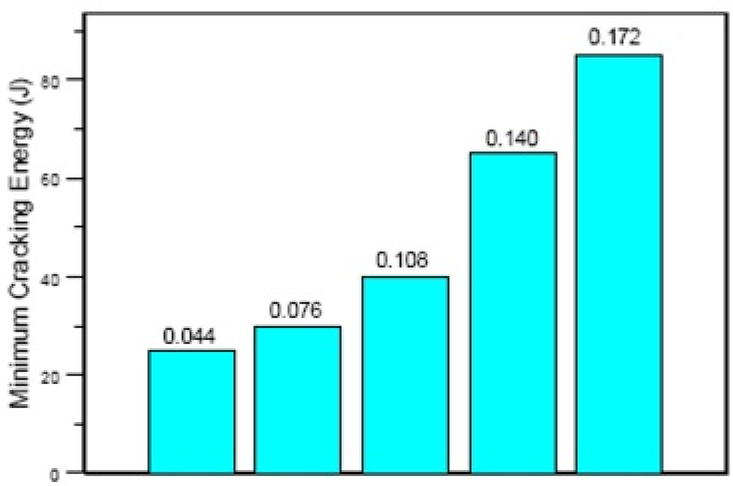


Figure 24. Graph of Varying Thicknesses of GLARE-5 Showing Respective Minimum Cracking Energies

Economic Analysis: Business Case

To replace aluminum, a material that has performed well for over 40 years as an aircraft cargo floor, with a fiber metal laminate requires an advantageous business case. The extended inspection threshold and decreased inspection frequency associated with the damage tolerance and corrosion resistance properties of the material decrease the life-cycle costs by eliminating expensive inspections, repairs, and replacements. Figure 25 illustrates GLARE's ability to resist crack growth and corresponding residual strength as a function of crack length. As the figure shows, GLARE has a higher residual strength after damage has occurred and maintains that strength much longer than monolithic aluminum.

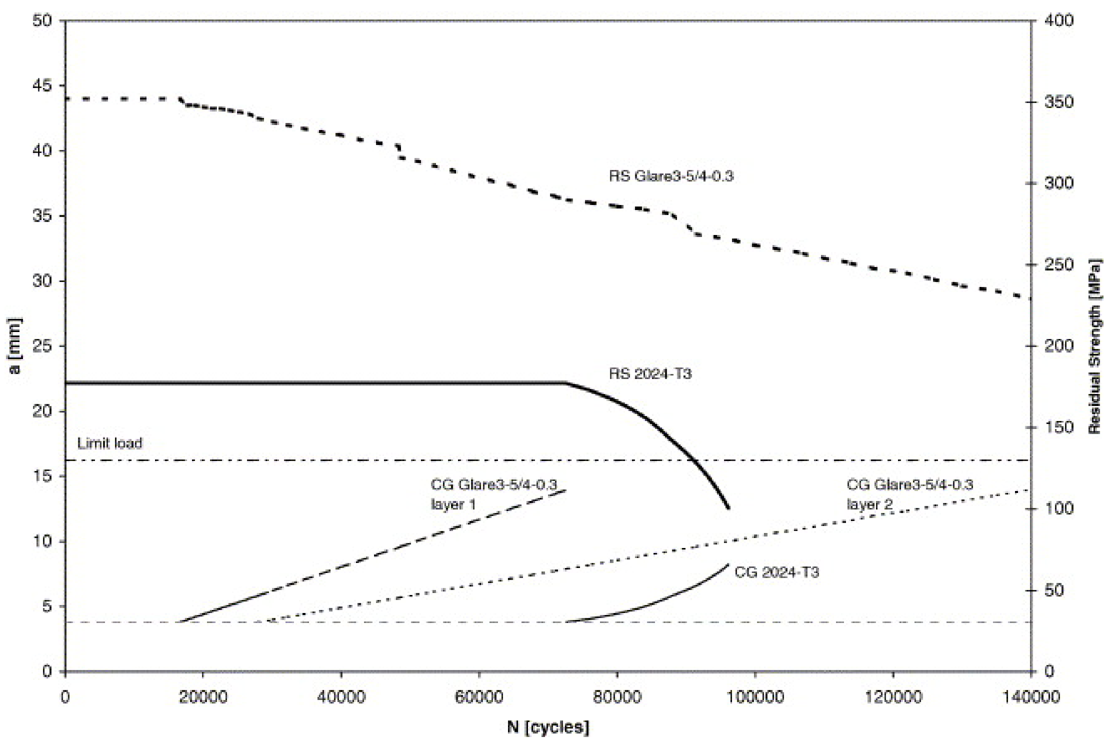


Figure 25. Residual Strength of GLARE and Aluminum as a Function of Crack Length and Cyclic Loading (Alderliesten, 2006)

When compared to monolithic aluminum on mass basis, GLARE is five to ten times more expensive. However, the advantageous properties of GLARE allow for less material to be used to achieve the same material properties as aluminum; but the weight savings alone does not recoup enough of the cost. (Vermeerern, 2003). Additional savings are realized from reductions in maintenance and inspection hours. As of 2004, the average Air Force depot maintenance labor rate with supplies neared $240 per hour (United States General Accounting Office (GAO), 2004).

The cargo floor of the C-130 occupies approximately 645.75 square feet and 0.08 inch aluminum costs anywhere from $4 to $5 per square foot. Using the higher value, the total unfinished material cost for a cargo floor would equate to about $3,228. GLARE panels cost approximately $26/sq ft, which equates to $16,800 in material costs alone. The densities of aluminum and GLARE-3 in a 4/3 stack are shown in Table 5 along with the material requirements for each respective material to occupy the C-130 cargo floor.

Table 5. Density, Volume of Material Required, and Respective Weight of 2024-T3 Aluminum and GLARE-3 in a 4/3 Lay-up

Material	Density(kg/cu m)	Vol. Req'd (cu m)	Weight kg(lbs)
Aluminum	2780	0.1232	342.5(753.5)
GLARE-3 4/3	2471	.117	289(636)

A weight savings of 117.5 pounds (~15% reduction) could be achieved if the GLARE-3 in a 4/3 stack were to replace the current 0.08 inch 2024-T3 aluminum. With each pound of weight shed from an aircraft, less fuel is required to power the aircraft to its destination. For every pound of weight shed from all aircraft in the AMC fleet, there would be an annual savings of $8,828 in fuel costs alone (McAndrews, 2009). While this is not significant to justify a material replacement, additional savings result from reduced maintenance inspections that cost $240 per hour.

Evancho (2001) performed a cost analysis using 48 square feet of 0.06 inch thick GLARE-5 panel to replace a similar aluminum panel. These dimensions are consistent with the bulk cargo floor panels in use on the Boeing 777. He based the analysis on data stating that heavy-use aluminum panels experience a repair rate of one per month and need to be replaced annually, while GLARE panels would require replacing every five years with an additional repair rate of one per year. Using commercial airline data from Evancho's (2001) analysis and the Air Force depot maintenance labor rate provided from the GAO report, Tables 6 and 7 summarize the cost analysis. Based on the data in Table 6, each GLARE panel would essentially pay for itself within the first year of service. As the tables show, the majority of the cost savings is associated with the reduction of labor and the corresponding costly labor rates.

Table 6. Cost Data on Aluminum and GLARE

	Aluminum	GLARE
Depot Labor Rate	$240/hr	$240/hr
$/panel (48sq ft)	$240	$1,248
Heavy Use Panel Life	1 year	2 years
Heavy Use Repair Freq	1/mo	3/yr
Hours to replace/repair floor	8/1	8/1

Table 7. Maintenance Cost Estimate for Aluminum and GLARE Panel over a 30-year Period

	Aluminum	Aluminum Cost($)	GLARE	GLARE Cost($)
Panels Replaced	30	$7,200	15	$18,720
Labor Hours	240	$57,600	120	$28,800
Repair Actions	360	---	120	---
Repair Hours	360	$86,400	120	$28,800
Total 30 yr cost per 48 sq ft Heavy Use panel		$151,200		$76,320

One 48 square foot panel over a 30-year period would save $74,880 in current year dollars. If this material were to be used for the entire 645.75 square foot cargo flooring, a 30-year lifetime savings of $1,007,370 in current year dollars could be realized per aircraft. A sensitivity analysis was performed on the economic analysis to illustrate how sensitive the findings would be to slight fluctuations in any of the above variables. A 10% increase and a 10% decrease in all variables associated with the life-cycle cost was analyzed and the results are depicted along with the cost comparison in Figure 26. Results of the sensitivity analysis show the figures used in this cost analysis are not sensitive to changes above or below 10%, thus adding a level of confidence to the overall results.

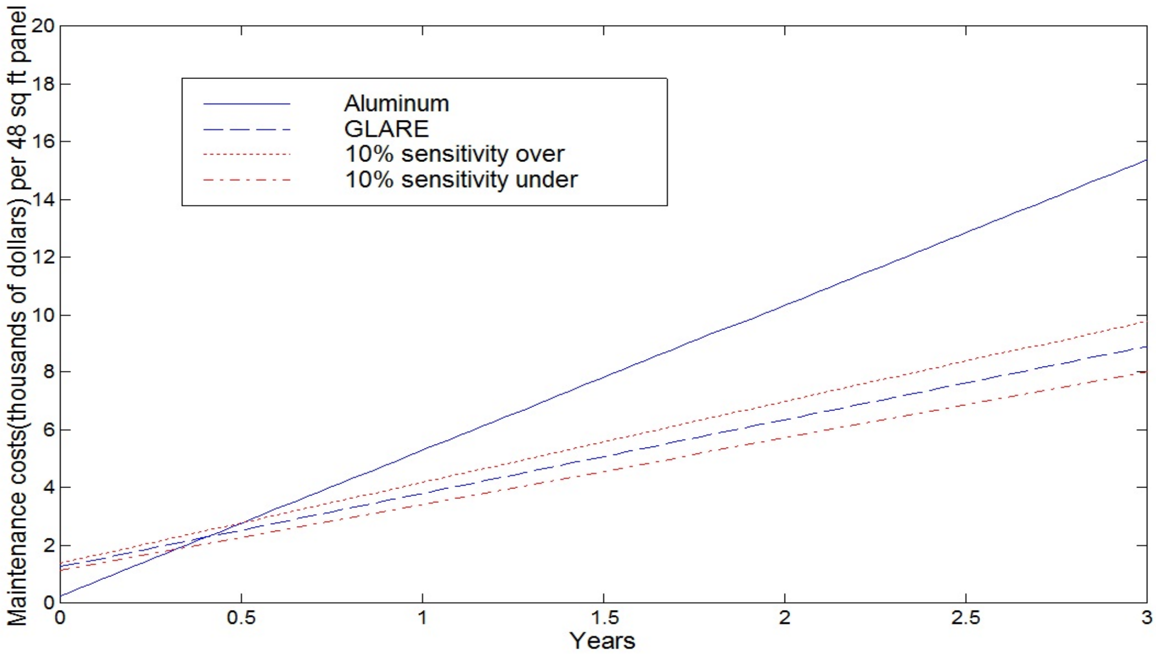


Figure 26. Graph of Maintenance Costs for Aluminum and GLARE

This business case analysis makes several assumptions over the 30-year length of the life-cycle costs. The costs of each material remain constant over the entire life-cycle and do not take into consideration economies of scale for material purchase. Additionally, different equipment necessary to excise and prepare each material was not taken into account. GLARE does require particular excision equipment that does not cause delamination of the material making it slightly more delicate to handle than its aluminum counterpart. Lastly, the time value of money was not considered in this life-cycle cost analysis and all figures are in current year dollars.

Chapter V. Conclusion

This chapter provides a summary of the results from the experimentation, engineering and business case analyses, corresponding conclusions, and recommendations for follow on research to improve the overall understanding of the behavior and performance of composite materials as a secondary aircraft structure.

Summary

The fiber metal laminate GLARE-3, in a 6/5 lay-up, meets or exceeds the mechanical material requirements for bending and impact resistance exhibited by 2024 T3 aluminum. Discrepancies in the modulus testing of off-axis specimens were explained by specimen geometry; coupon level tensile testing of off-axis dog bone specimens will not yield results conducive to how larger specimens of the material will behave. Through further analysis, removing the contributions of the S-2 glass fibers for the off-axis tensile coupons yielded theoretical results that matched the testing data for the tensile tests.

The theoretical model was validated by the experimental data and proven to be a useful tool in conjunction with the Mass Volume Fraction (MVF) technique for estimating the mechanical properties of GLARE. The only discrepancies were found in off-axis tensile testing and those were explained through coupon geometry limitations and MVF proved useful in that analysis. Additionally, it was noted that testing this material above the coupon level should alleviate the geometry constraints for tensile testing specimens and the experimental values should match the theoretical values. The model accurately predicted the expected bending profile of the GLARE material which provided accurate results to use for comparison against monolithic aluminum to determine that GLARE in a 6/5 configuration met the maximum deflection and bending requirements.

Conclusions

The process of taking a known user problem and mating it with a feasible material solution successfully identified a feasible material, a fiber metal laminate, which satisfied the requirements of being both mechanically and economically viable for use as a replacement material. The experiments accurately captured the required data for determining the material's elastic modulus, bending profile, and impact resistance. These specific mechanical properties were found to be the main properties of interest as cargo flooring is primarily designed to carry loads and be rugged enough to resist impact damage from daily use.

This particular variant of GLARE-3 in a 6/5 lay-up, because it is 0.045 inches thicker than 0.08 inch aluminum and slightly heavier than the current cargo floor material, is not considered a true form, fit, function replacement material. However, the engineering analysis performed in this research effort showed that other GLARE-3 variants could meet the physical requirements while maintaining the mechanical properties necessary to meet the requirements of a cargo floor. The two variants with the most potential would be GLARE-3 in a 4/3 0.4mm and 3/2 0.5mm lay-up.

GLARE-3 in a 4/3 lay-up is 0.077 inches thick and weighs less than the current cargo floor. The bending and impact properties should be similar to the properties of the 6/5 lay-up. As noted earlier, the change in the material's impact resistance from a 6/5 lay-up to a 4/3 lay-up would be minimal; however, this property would need to be investigated further to determine exactly how much the impact resistance might change.

The GLARE-3 material was proven, by answering the research questions, to be a good potential candidate for further testing of the different material configurations.

The mechanical advantages of using GLARE, compared to monolithic aluminum, were its higher impact resistance, greater ability to resist bending, and greater residual strength. Based on the assumptions made in the business case, GLARE provides a source for O&M savings in that the added upfront costs associated with use of the material as a replacement for aluminum would pay for itself within the first year of its life by reducing the repair frequency and repair costs associated with the use of 2024 T3 aluminum. Sensitivity results provided a certain confidence to the life-cycle costs and showed that the material would provide almost immediate O&M savings under the stated assumptions. Finally, O&M costs from structural repair and replacement have been continually increasing each year due to accelerated aging of the U.S. Air Force’s fleet from sustained high operations tempos in Iraq and Afghanistan; this material provides a feasible opportunity to combat those rising costs.

The residual strength of the material further extends its service life by requiring less repair and maintenance than monolithic aluminum meaning the material could continue to perform as effective cargo flooring after being damaged while the current aluminum flooring would require repair or replacement. The impact testing revealed that GLARE in a 6/5 stack is more damage-tolerant than 2024 T3 aluminum; in fact on average it is able to absorb an additional 20 Joules of impact energy before reverse side cracking occurs for low velocity impacts. Analysis reported in the literature also noted than once GLARE is damaged its residual strength is higher than aluminum. This ability allows for lengthening the replacement intervals and repair intervals, which significantly was found to be one of the main factors included the business case.

Concluding Remarks

This research effort makes a positive case for the use of GLARE for use on a military cargo aircraft as its cargo floor. The optimal solution of material thickness, fiber orientation, and stacking sequence, however, was not investigated in this effort. A follow-on research effort might include testing of the GLARE-3 or GLARE-5 material in a 4/3 or 5/4 stacking sequence in the various thickness available and then test larger specimens to obtain a greater understanding of performance, specifically in terms of tension and bending.

A greater understanding of the impact properties of GLARE would also be useful for presenting this material as a feasible candidate to replace aluminum. The impact testing in this effort consisted of only testing one impactor size and shape while numerous other sizes and shapes exist. A full impact testing profile consisting of low velocity and high velocity impacts with impactors of various sizes and shapes would create a much fuller understanding of the impact behavior of this material. Smaller diameter spherical impactors, for example, would be able to demonstrate the material's ability to withstand piercing impacts. A full impact property profile would help further champion this material as a feasible replacement for monolithic aluminum.

In addition to understanding the mechanical properties of each variant further research is needed to strengthen the business case for this material. Data relating to the repair frequency and replacement intervals would provide greater assurance in the values presented in this business case. One possibility should include installing a single GLARE panel onboard a C-130 aircraft and allowing the material to perform for a set duration of time, followed by removal and further analysis against its aluminum counterpart. This actual data would further refine the assumptions included in presenting a case of making this material a feasible replacement material.

With a full cargo panel specimen, the geometry issues associated with the fibers not traveling the length of the specimen would not arise. Follow-on efforts might also include testing different GLARE variants such as GLARE-5 which is built in a similar stacking sequence as GLARE-3 with the addition of a 45 degree fiber orientation prepreg layer with each 0 and 90 degree layer. While the 15% weight savings was minimal on the C-130, a possible follow-on research effort could examine the possibility of using a Fiber Metal Laminate on a C-17 or C-5 aircraft where the square footage of each cargo compartment is much larger than the C-130, thus the potential for weight savings would be greater.

Additional follow-on research could investigate the feasibility of using a GLARE variant for other secondary or primary structural applications. In fact, several instances of fleet-wide groundings due to aircraft structure fatigue have been witnessed due to the demands of recent wartime operations. Numerous fatigue issues are currently plaguing the aging military cargo aircraft community therefore, each instance could provide an opportunity where a third generation aircraft structural material could be used to replace a second generation aircraft structure.

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14. ABSTRACT

This effort explored the mechanical characteristics and economic feasibility of using the fiber metal laminate, GLARE-3, as a secondary aircraft structure; specifically the cargo floor of a C-130. The mechanical properties were determined through static four point bending and tensile testing and dynamic impact testing. Aggregate behavior of the constituent materials was predicted using a model which consisted of Mass Volume Fraction (MVF) and Classical Laminated Plate Theory (CLPT) methods using known values for the constituents. Static testing was conducted on coupon-level specimens using standardized testing procedures. Static tensile tests were conducted on specimens with four different fiber orientations, 0°, 22.5°, 45°, and 90°, while static bending tests were conducted on fiber orientations of 0° and 90°. Two series of impact tests were performed on both GLARE-3 and 2024 T3 aluminum using 4 inch wide strips to show impact damage progression. Data for the economic analysis was gathered from existing literature and cost data was analyzed over a 30 year period for both GLARE and Aluminum. Analysis of the data proved the use of GLARE-3 as a potential cargo floor material was mechanically and economically feasible with the material paying for itself within the first year of its use.

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