Composite Lattice Structures for Application in Airplane Components

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Composite Lattice Structures for Application in Aircraft Components

by

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4/22/2021

Date
Composite Lattice Structures for Application in Aircraft Components

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Table of Contents

I. Abstract ................................................................................................................................. 2
II. Nomenclature ......................................................................................................................... 3
III. Introduction .......................................................................................................................... 3
IV. Background ......................................................................................................................... 3
V. Aerospace Composite and Metallic Material Background .................................................. 4
VI. Classical Wing Structures .................................................................................................... 6
VII. Lattice Structures and Application in Light Attack Aircraft Wings ............................... 9
VIII. CAD Models of Standard Wings and Wings with Lattice Structures ............................ 13
IX. Structural Analysis ............................................................................................................. 18
X. Conclusion ............................................................................................................................ 34
XI. References ........................................................................................................................ 35

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I. Abstract

Creating a strong, rigid wing structure is integral to developing capable military aircraft that can carry out advanced maneuvers required by their missions. Wings have evolved to include two main inner structure components: the rib and spar. These components have had general formats that they have followed, being an I-beam-like structure, made out of aerospace-grade metals or light composite materials that can hold the loads applied to them during flight. Recently, there have been developments in metal lattice, 3D printed structures, that promise weight benefits, while maintaining structural characteristics. This project aims to examine these lattice structures in conjunction with rib and lattice models that will be applied to an AIAA conceptual light attack aircraft for their student design competition. With this, various CAD models were made of ribs and spars using cubic, triangular, and hexagonal lattice structures. Solid material ribs and spars were also modeled to give baseline comparisons. After the completion of the CAD models, these were then tested using Patran and Nastran, a finite element method software (FEM), to find the total deformations and stresses of the ribs and spars when a feasible load is applied. Overall, the lattice ribs performed admirably, having deformations and stresses only slightly higher than the solid material models, while actually saving weight. The spars, however, did not fair as well, showing deformations ten times the solid material spars, showing that this concept would require further optimization to ever compete with a standard I-beam. Originally, the secondary aim of this project was to also combine the metal lattice structures with the optimal structural characteristics of composites on the skins of the ribs and spars. Due to computational limitations, however, this was not fully achieved, but valuable data trends were still found, enabling the ability to assume that the combination of the composite and metal lattices would create an even more comparable structure to the solid material alternatives. Overall, this project achieved finding out how lattice structures would behave under load in a light attack aircraft, rendering them worth further consideration especially for application in the ribs.
II. Nomenclature

\[\begin{align*}
E &= \text{modulus of elasticity} \\
\sigma_y &= \text{yield stress} \\
\sigma_u &= \text{ultimate stress} \\
v &= \text{Poisson’s ratio} \\
\rho &= \text{density} \\
V &= \text{volume} \\
\sigma_{LU} &= \text{ultimate longitudinal stress} \\
\sigma_{TU} &= \text{ultimate transverse stress} \\
m &= \text{mass} \\
\delta &= \text{deformation} \\
w &= \text{load} \\
l &= \text{length} \\
I &= \text{moment of inertia} \\
M &= \text{bending moment} \\
y &= \text{distance from the centroid}
\end{align*}\]

III. Introduction

During the AY 2020-21 MAE 490/491 Aircraft Design course, students are developing a light attack aircraft design in response to the AIAA Undergraduate Team Aircraft Design Competition. The aircraft design and performance requirements are detailed in the RFP at:

https://www.aiaa.org/docs/default-source/education-and-careers/university-students/design-competitions/undergraduate-team-aircraft-design-competition/aiaa-2021-undergraduate-team-aircraft-design-rfp---light-attack-aircraft-2-.pdf?sfvrsn=b54e3cac_0

While the four student design teams will develop their respective aircraft concepts, the scope and timeline of this project limit the teams to being relatively general in their design approach. This may result in a less detailed design for several aspects.

Structural design is one detailed aspect that may not be fully realized in this project. Composite or metallic lattice structures are promising new technologies that could replace the traditional titanium or aluminum airframe components while providing structural strength at a reduced weight. The goal of this Capstone Project is to research and analyze this lattice structure technology in terms of relevant materials and design/optimization tools. One or more lattice structural components will be designed and the benefits of using these as replacements for the traditional components will be described. A general design tool for use by future aircraft design students will also be developed if time allows.

IV. Background

With the performance expected of aircraft in any market in this day in age increasing, so must the performance of the materials used to construct these aircraft, and further, the structures that are created with these updated materials. Composites have revolutionized the construction of aircraft, making them lighter, stronger, and in some instances, cheaper to produce, when compared to traditional materials, such as aluminum, titanium, and steel. The progressive movement of using composites in airframes has been slow, yet steady across all aircraft for about the past 30 years, with many aircraft in the fighter and passenger divisions being comprised of about 20 – 30% [1]. Sections of an aircraft most commonly composed of composites include the fuselage, tail, wings, and sections of the inner airframe. While the benefits of using composites in an aircraft could be evaluated for all sections of the airframe, this paper will focus on specifically the wings, and even more, the wings of the light attack aircraft being developed for the 2020-21 MAE 490/491 light attack aircraft.

Many current aircraft utilize composites as parts of their wings, with the skins, spars, ailerons, and wiglets all being possible production items, though not usually all together. To analyze a composite lattice structure in a wing, two main aspects of the wing can be combined for a potentially greater gain in structural performance: the wing spar and skin. The traditional configuration of a wing structure involves an aluminum skin wrapped around a central spar to
create a rigid, aerodynamic body. However, if the inside of the wing can be replaced with a lattice structure instead of a spar, and the skin is made entirely of composites, a sounder structure could be made all around. This has much of the same ideology as a honeycomb composite structure, but with voids created in 2 or more directions. Bear in mind that the possible structural benefits will be evaluated for composite lattice structures, and not necessarily the direct reality of their production.

V. Aerospace Composite and Metallic Material Background

To understand the possible application of composite lattice structures within airplane wings, a more detailed background of the current and former material solutions must first be provided. The earliest functional airplane wing, part of the 1903 Wright Flyer, was made of a wooden strut, a steel cable structure, lined with a canvas-like fabric [2]. Airplane wings however quickly evolved to be made primarily of high-grade materials, including aluminum, titanium, and steel, aluminum by far being the most prominent for wing skins and spars. Aluminum is and was the obvious choice for many wing compositions, as it offers fantastic performance for its relative weight and cost. The elastic modulus of most aluminum alloys hovers around $10 \times 10^6$ psi, with a tensile strength of 45,000 psi and a yield strength of 40,000 psi [3]. In fact, the Aluminum Alloy chosen for analysis later on, is Aluminum 6061, which is a commonly used aluminum in aircraft due to its relatively high grade and strength characteristics.

Other metals could be considered for the construction of these lattice structures, such as titanium or steel: two commonly used aerospace metals. For applications in wings, however, steel proves to have an excessive weight that is not justified by its additional strength. Titanium on the other hand has excellent structural properties for its weight, however, it is vastly more expensive to use than aluminum, and seeing as this analysis is being done in conjunction with a light attack aircraft that is meant to be relatively inexpensive, it also does not prove to be worth the cost. In Table 1 below, some commonly used aircraft metals can be seen, along with their strength characteristics.

<table>
<thead>
<tr>
<th>Material</th>
<th>$E \ [\text{psi}]$</th>
<th>$\sigma_y \ [\text{psi}]$</th>
<th>$\sigma_u \ [\text{psi}]$</th>
<th>$\nu$</th>
<th>$\rho \ \left[\frac{\text{lb}}{\text{in}^3}\right]$</th>
</tr>
</thead>
<tbody>
<tr>
<td>Aluminum 6061</td>
<td>$10 \times 10^6$</td>
<td>40,000</td>
<td>45,000</td>
<td>0.33</td>
<td>0.0975</td>
</tr>
<tr>
<td>Stainless Steel 18-8</td>
<td>$28 \times 10^6$</td>
<td>31,200</td>
<td>73,200</td>
<td>0.29</td>
<td>0.289</td>
</tr>
<tr>
<td>Titanium</td>
<td>$16.5 \times 10^6$</td>
<td>160,000</td>
<td>170,000</td>
<td>0.33</td>
<td>0.160</td>
</tr>
</tbody>
</table>

Metals, however, are not the only material used to construct aircraft. Composites are making up a larger and larger percentage of the total construction of aircraft and for good reason. Composites tend to maintain much of the same properties as metal while being less dense, and overall, lighter. Composites combine two or more materials into one to create new, stronger materials than either of the original ones would be. For example, one of the most commonly used composites is a fiberglass-epoxy combination, where the fiberglass strands represent the fiber in the composite, and the epoxy represents the matrix material. Matrix materials are the “filler” material that eliminates any open space in between the individual fiber strands in a composite. Epoxies allow composite materials to maintain their rigid shape under any load, however, they are not the material in a composite that carries the load [4]. Below is a diagram of how the fiber and matrix materials interact in a composite.
A unidirectional composite represented schematically

Figure 1: Fiber and Matrix Material Diagram [5]

This diagram represents a unidirectional composite, meaning that all of the fibers in the composite are running in the same direction. This phenomenon of the matrix material filling in the open gaps remains true for the woven fiber composites that most composites use. This woven fiber mat is much like a fabric, where the load can be carried efficiently in more than one direction to further increase the strength of a composite.

For analysis, some common composites used in aircraft include fiberglass-epoxy, carbon fiber-epoxy, and Kevlar-epoxy combinations. The properties of these can be seen below in Table 2 [4]. In the case of the light attack aircraft being developed, the main composite used in the wings was carbon fiber-epoxy, due to its high strength to weight ratio, while also maintain a relatively middle ground cost, between fiberglass and Kevlar.

<table>
<thead>
<tr>
<th>Material</th>
<th>$E$ [psi]</th>
<th>$\sigma_{LU}$ [psi]</th>
<th>$\sigma_{TU}$ [psi]</th>
<th>$\nu$</th>
<th>$\rho$ [lb/in^3]</th>
</tr>
</thead>
<tbody>
<tr>
<td>Fiberglass</td>
<td>$5.6 \times 10^6$</td>
<td>154,030</td>
<td>4497</td>
<td>0.26</td>
<td>0.0578</td>
</tr>
<tr>
<td>Carbon Fiber</td>
<td>$26.3 \times 10^6$</td>
<td>217,560</td>
<td>5802</td>
<td>0.28</td>
<td>0.065</td>
</tr>
<tr>
<td>Kevlar</td>
<td>$11 \times 10^6$</td>
<td>203,053</td>
<td>1740</td>
<td>0.34</td>
<td>0.0527</td>
</tr>
</tbody>
</table>

In Table 2, there are some slightly different values for strength than with the metal Table 1. The ultimate strength of composites is measured in the longitudinal and transverse directions. The longitudinal strength is along the length of the fibers, with the transverse is measured perpendicular to the fiber direction. This again is why a woven fiber fabric has far superior strength to a unidirectional composite. As can be seen in Table 2, the carbon fiber-epoxy composite has the highest modulus and strengths, while maintaining a much lower density than any of the metals. This is why it was chosen to be used in the wings of the light attack aircraft and would be a good candidate for creating lattice structures in a wing. A carbon fiber composite could create the planar sections that are required to bound the lattice structures within a wing. This combined with an aluminum lattice in the center could create a very strong, resilient structure for an area such as a wing, that sees large amounts of flex and stress during a flight. More discussion of the possible creation of a composite lattice structure will occur in section VII on the different models tested.
VI. Classical Wing Structures

Though there have been many different styles and purposes for aircraft since their invention, one general design was made early on and has been used in one-way shape or form in most aircraft wings since. This design provides both longitudinal and lateral stability through the use of a spar and multiple ribs to form the airfoil shape of the wing. When required, a double spar can be employed as well to increase structural integrity along the length of the wing. An example of this common design can be found below in Figure 2.

![Wing Internals of A-10 Warthog](image)

**Figure 2: Wing Internals of A-10 Warthog [6]**

In Figure 2 above, a cutaway of a double-spar structural wing design can be seen, which is used in the high aspect ratio wing of the A-10 Warthog. Due to its high aspect ratio wing, there will naturally be greater deflection observed through the length of the wing, thus the need to ensure that its structural characteristics are sound. In the context of the light attack aircraft for MAE-491, this same issue presents itself. To achieve the payload and range requirements of the AIAA competition, a high aspect ratio wing was chosen to promote lift and endurance. One stark difference between the A-10 and The Golden Egg is that the Warthog weighs close to three times the amount of the other, with similar wing dimensions. Due to this, a single-spar design was chosen to compose the internals of the wing for the prospects of the competition that it has been entered in. However, to support the stability of the wing, three smaller stringers have also been added parallel to the main spar. These are not as large as the spar, nor can support the same load, but act as braces within the wing to promote rigidity against the ever-changing air resistance as the aircraft flies.

To support the focus of this analysis, stringers will not be analyzed using finite element analysis later, where the two main internal structures of the wing will be. The first, as explained before is the spar: a beam-like structure that runs longitudinally through the length of the wing, essentially carrying the load of the aircraft as it flies. A spar could be imagined as a bridge spanning a valley, where the supporting land is found at the wingtips. The bridge must support the bodies that cross it, in this case, the fuselage, crew, weapons, engine, and wings. If not properly braced, or not given the proper rigidity, the spar will fail, and the wings will collapse. To support the spar and maintain a consistent airfoil shape down the length of the wing, ribs are employed. These run perpendicular to the spar, parallel to the
fuselage. The ribs do not see quite the load that the spars do, however, they are still integral to the flight of an aircraft. The ribs help the metallic or composite skin of the wings maintain its airfoil shape against the air pressures exerted on it. These ribs, much like the stringers act to further stabilized the wing itself to promote rigidity within the wing, decreasing some of the stress on the spar. Because the spar and the ribs see a fair amount of stress during a flight, these are the two aspects that will be analyzed in section VIII.

There are a few different classical designs of both spars and ribs that have been used in aircraft. Though these designs have evolved to improve structural characteristics and decrease weight in the wings, the same overall shapes can be observed. A spar, as seen below in Figure 3, generally has an I-beam-type structure. These classically exhibit a high amount of strength for the amount of material used. An alternative to the I-beam is a box section type spar. If made out of metal, these weigh more than their skinnier counterparts, possibly not justifying their extra strength, due to their added material. However, if a composite is used, where the densities and therefore overall weights of the spars are lower, a box-style spar can certainly be justified. This style of spar can be seen below in Figure 4. Note that I-beam spars can be made out of a variety of materials, including metals and composites as well.

![Figure 3: Classic I-Beam Wing Spar Cross-Sections](image1)

![Figure 4: Box-Style Composite Wing Spar](image2)
The ribs within a wing, as stated before, are positioned perpendicular to the main spar. These ribs have a teardrop shape to match the airfoil. Generally, ribs also have a pseudo-I-Beam style design, with an inner web of the shape of the airfoil, and then an outer flange running the perimeter of the piece. An example of this can be seen below, in Figure 5. The web of a rib can be solid, for added strength, or can have sections removed to reduce the overall weight of the wings. Usually, this is a good measure to take, as the ribs do not see the same stresses that the spars do, throughout a flight. Older aircraft used ribs that had smaller beams in a truss formation to make up the webs. For newer airplanes, that would see higher stress values due to tighter, more extreme maneuvers, much like the light attack aircraft might, circular or oblong sections can be taken out, distributing the loads better through the ribs. These circular cutouts also provide ample room for cables and fuel lines to pass through, as an added benefit. The rib in Figure 5 shows an example of circular sections cut out of the web, formally called lightening holes.

Figure 5: Ribs with Circular Cutouts [9]

VII. Lattice Structures and Application in Light Attack Aircraft Wings

With the understanding of materials that are used in military aircraft and also the conventional designs that make up the structures of a wing, an analysis of the potential applications of lattice structures in wing structures needs to be completed. The theory behind this thesis is to replace key components in wing structures with lattice structures to see a possible improvement in weight, strength, and cost. Also, the goal is to see if these lattice structures can be made of composite materials, again to promote weight savings and increased strength, though it is very unlikely that the overall cost of these would initially be lower than current methods. To begin, lattice structures can be found in nature primarily in crystalline formations on the molecular level in materials such as quartz and organic materials like wood and honeycombs. Honeycombs are the most visible example of a uniform structure in nature, though they are only a 2D formation, where full lattices do require a 3D formation. The reason that the honeycomb is such a vivid example in nature is that it lies on the macro scale, allowing us to visualize what is occurring on the micro-level within many materials. The reason that many crystalline materials are so strong is due to their molecular lattice structures. The goal,
therefore, is to exploit these benefits for use on the macro scale within the structure of the wing on the light attack aircraft.

Though there are many different designs for a lattice, only certain basic ones can efficiently be created at this time. These include triangular, cubic, hexagonal, and octagonal. Examples of these structures can be seen below in Figures 6 – 9.
Examining these structures, some stark differences correlate directly with their names. These lattices are referred to as ball and strut lattices, due to spherical-shaped, “ball”, hubs that are connected by the cylindrical
“struts” in between them. The triangular lattice in Figure 6 creates a 3D triangular and hexagonal pattern that creates a strong bond between the different sections of the lattice. The cubic lattice in Figure 7 is possibly the most recognizable structure, creating small cubes as the main pattern. The hexagonal lattice is generally regarded as the strongest lattice, as it has the highest relative volume packing. This is due to the staggering ball pattern, where the balls in one plane lie in between the balls of the next, not only giving the highest volume packing, but also the strongest lattice. The strongest lattice is not necessarily required, however, for the object is to optimize the structures of the wing on the light attack aircraft in the most efficient way. Though it is the strongest and will exhibit this character in later tests, it will also weigh the most and therefore require more material and time to produce, raising the cost. Finally, the octagonal lattice can be found in Figure 9. Though it is quite organic in stature, the octagonal lattice leaves a fair amount of open space in between the balls and structs, creating large voids within the centers of the cells. Also, due to its increased voids and complex nature, the octagonal lattice is the most labor-intensive structure to create, making it not ideal for application in an aircraft. Because of its open structure and high amount of material required, the octagonal lattice is not ideal for application in the relatively affordable light attack aircraft.

To further understand the basic structures of these lattices, the renderings in Figures 6 – 9 all maintain similar characteristics. The overall cube structure of each lattice network is 10in x 10in x 10in, with the “cell” size of each lattice set to 5in x 5in x 5in. The cell refers to the inner, individual structures made up of a singular ball and its related struts. This is true except for the octagonal lattice, where the cells can be easily defined by the small 3D octagons made up of 16 balls and struts. The triangle lattice makes up a triangular prism, the cubic makes a cube, and the hexagonal makes a hexagonal prism. The balls on each of these are 1.5” in diameter, and the struts are 0.5” in diameter. This feature is not so much an optimization technique, but to be able to consistently test the structures, with little alteration apart from their overall structure. Given these sizing characteristics, it is easy to not only calculate the volume of the void in the structures, but also the weight and volume of the material used for each. These sample calculations can be seen below as:

\[ V_{\text{cube}} = 10in \times 10in \times 10in = 1000in^3 \]  \hspace{2cm} [1]

\[ V_{\text{void}} = V_{\text{cube}} - V_{\text{material}} \]  \hspace{2cm} [2]

\[ m_{\text{material}} = (V_{\text{material}})(\rho_{\text{material}}) \]  \hspace{2cm} [3]

Though it is possible to calculate these volumes by hand, CAD software proves to ease this process, by conveniently providing the volume of material used for us. These can be seen below in Table 3.

<table>
<thead>
<tr>
<th>Table 3: Material Volumes of Lattice Structures</th>
</tr>
</thead>
<tbody>
<tr>
<td>( V_{\text{material}} \ [in^3] )</td>
</tr>
<tr>
<td>Triangular</td>
</tr>
<tr>
<td>-----------</td>
</tr>
<tr>
<td>143.620</td>
</tr>
</tbody>
</table>

Using these values and EQ. 2, it is easy to calculate the volumes of the voids and the weights of the material used. Because aluminum is a commonly used metal in aircraft, its density will be used to calculate the lattice weights, though this process would be the same for any other material. The aluminum used is an alloy called aluminum 6061, with its density being 0.0984 lb/in\(^3\) [10]. These values can be seen below in Table 4.

<table>
<thead>
<tr>
<th>Table 4: Volume of Voids and Material Weights</th>
</tr>
</thead>
<tbody>
<tr>
<td>( V_{\text{void}} \ [in^3] )</td>
</tr>
<tr>
<td>---------------------------------</td>
</tr>
<tr>
<td>Triangular</td>
</tr>
<tr>
<td>856.380</td>
</tr>
<tr>
<td>14.132</td>
</tr>
</tbody>
</table>

As can be seen in Table 4, the weight savings from using a lattice structure with reasonable ball and strut dimensions can be enormous. Over the solid aluminum cube weight, there is an average weight savings of 85.62%, even with the heavier octagonal lattice excluded. Weight however cannot simply justify the use of lattice structures in wings, as strength is the other main governing factor. The combination of these lattice structures with ribs and spars will be explored and tested further in the later sections using FEM software, however, some governing equations can be used to find the stress and deformation of a beam under a load. The beam in this case would be a lattice spar, with
the weight of the aircraft pressing into it. Specifically, a lattice spar would be closely related to the simple cantilever beam example. The excellent graphic below in Figure 10 shows how to calculate the deformation of a cantilever beam under different load scenarios. The scenarios for this project will be the one with the load distribution along the length of the beam. This is the third row of the table.

<table>
<thead>
<tr>
<th>ART. 59</th>
<th>Slope at free end</th>
<th>Deflection at any section in terms of $x$: $\delta$ is positive downward</th>
<th>Maximum deflection</th>
</tr>
</thead>
<tbody>
<tr>
<td>1. Cantilever Beam—Concentrated load $P$ at the free end.</td>
<td>$\theta = \frac{P l^2}{2 E I}$</td>
<td>$\delta = \frac{P x^2}{6 E I} (31-x)$</td>
<td>$\delta_{\text{max}} = \frac{P l^3}{3 E I}$</td>
</tr>
<tr>
<td>2. Cantilever Beam—Concentrated load $P$ at any point.</td>
<td>$\theta = \frac{P a^2}{2 E I}$</td>
<td>$\delta = \frac{P x^2}{6 E I} (3x-a)$ for $0 &lt; x &lt; a$, $\delta = \frac{P a^2}{6 E I} (3x-a)$ for $a &lt; x &lt; l$</td>
<td>$\delta_{\text{max}} = \frac{P a^2}{6 E I} (31-a)$</td>
</tr>
<tr>
<td>3. Cantilever Beam—Uniformly distributed load of $w$ lbs. per unit length.</td>
<td>$\theta = \frac{w l^3}{6 E I}$</td>
<td>$\delta = \frac{w x^2}{6 E I} (x^2 - 4x)$</td>
<td>$\delta_{\text{max}} = \frac{w l^4}{8 E I}$</td>
</tr>
<tr>
<td>4. Cantilever Beam—Uniformly varying load; maximum intensity $w$ lbs. per unit length.</td>
<td>$\theta = \frac{w l^3}{24 E I}$</td>
<td>$\delta = \frac{w x^2}{3 E I} (105l^2 - 105l^2 + 51x^2 - x^2)$</td>
<td>$\delta_{\text{max}} = \frac{w l^4}{30 E I}$</td>
</tr>
<tr>
<td>5. Cantilever Beam—Couple $M$ applied at the free end.</td>
<td>$\theta = \frac{M l^2}{4 E I}$</td>
<td>$\delta = \frac{M x^2}{2 E I}$</td>
<td>$\delta_{\text{max}} = \frac{M l^2}{2 E I}$</td>
</tr>
<tr>
<td>6. Beam freely supported at ends—Concentrated load $P$ at the center.</td>
<td>$\theta = \frac{P x^2}{16 E I}$</td>
<td>$\delta = \frac{P x^2}{4 E I} (31 - x^2)$ for $0 &lt; x &lt; \frac{1}{2}$</td>
<td>$\delta_{\text{max}} = \frac{P l^3}{4 E I}$</td>
</tr>
</tbody>
</table>

Figure 10: Cantilever Beam Deflections [11]

From this table, we see that the deflection in a cantilevered beam is equal to:

$$\delta_{\text{max}} = \frac{w l^4}{8 E I}$$

[4]

This is where $w$ is the load, $l$ is the length, $E$ is the modulus of elasticity, and $I$ is the moment of inertia of the beam. Using Equation 4, a very basic estimation of the deflection of the spar could be calculated, however, the moment of inertia remains in question. This value relates directly to the geometry of the spar, particularly the lattice section. With so many different members present in even just a single cell of a lattice structure, calculating the deflection becomes quite the task. This is exactly why FEM software was used, as it minimizes the hand calculations, and gives an excellent estimate of the deflection values. Calculating the stress of a lattice spar has much of the same limitations. Equation 5 below could be used to calculate the stress in a standard beam; however, this value is related to the deflection, which once again depends much on the geometry of the spar itself [12].

$$\sigma_x = -\frac{M y}{I}$$

[5]

This is where $M$ is the bending moment and $y$ is the distance from the centroid of the spar. With these two governing equations, the stress and deformation of the lattice spars can be defined. These equations form the basis for the values found in the results section of this paper.

VIII. CAD Models of Standard Wings and Wings with Lattice Structures

Tabulating all the different concepts correlated with lattice structures, materials, and classical wing design, models of potential lattice structure wing components could be made. These models were then used to conduct FEM testing to find deformation and stress values associated with each design. Using the three lattice structures deemed sufficient
and efficient to use in any practical application, as well as a baseline solid material structure, four different sets of ribs were made, and five different spars were created. To maximize unity between all of the models and tests, so just the structure form itself was being tested, each of the cell sizes of the lattices were created similarly. For the ribs, each of the cells measured 0.75” x 0.75” x 0.75”. For reasons discussed later, this was a scaled-down version of what would be the full-size model. In fact, this is scaled by a factor of 0.25, which had an original cell size of 3” x 3” x 3”. This downsizing was done to assist the computers used to run FEM tests by decreasing the overall volume of the models created. To maintain consistency with the AIAA competition, the same airfoil was used for creating the models, as was used in the project. This airfoil is a NACA 2418, which was most famously used in the Cessna Dragonfly. This airfoil is fairly symmetrical, which also made it ideal for testing, as the lattice structures would not have to create so many tight bends at the leading and trailing edges. A representation of the airfoil can be seen below in Figure 11, with the unscaled dimensions used for the rib.

![Figure 11: NACA 2418 Airfoil [13]](image)

The ribs in the wing of an airplane tend to match the airfoil of the wing, to add stability to the skin, as discussed earlier, and for this reason, the dimensions of the wing airfoil were assumed to be the same for the interior rib. However, again, the rib was scaled down to 0.25 of its size, so the rib actually measured 27.5 in long and 4.95 in thick.

For all of the models, a rib was created with lightening holes, as well as one without. These holes decrease the overall amount of material required for the spar, while in some cases still meeting the stress and design criteria of the aircraft. Because this is a common occurrence in aircraft, it proved to be a good comparison for this test to compare to one that was consistent all of the way through. The lightening holes in each of the ribs measures as the following, from left to right: 10 in, 14 in, 12 in, 8 in. The aim of using varying size holes was to not only mimic trends seen in conventional ribs but also allow another level of examination where the sections of lattice reduce due to the voids.

The first of the ribs created can be seen below in Figure 12. These ribs were created with a cubic lattice as shown back in Figure 7, using the same consistent dimensions previously stated. The second set of ribs was created using a triangular lattice, as shown in Figure 13, again with the standardized dimensions. The last of the lattice ribs is shown in Figure 14, using a hexagonal lattice. Finally, a solid version of the rib with and without lightening holes can be found in Figure 15. This model was made and analyzed to give a comparison point from what ribs are currently used to the characteristics of the lattice ribs. Note that all of these ribs were drawn in the CAD modeling software creo, using the lattice generator to fill in the outer flange/skin material that would be solid metal or a composite plate in a real application. The use of this software facilitated creating consistent, even lattice cells, with the correct geometry for each style throughout the extents of the ribs.
Figure 12: Cubic Lattice Rib Models

Figure 13: Triangular Lattice Rib Models
The volume and weight of all of these ribs can be seen below in Table 5. Note that the volume of material used for the outlining skin of each rib is the same, and the only varying factor is the structure within, whether that be a lattice or solid material. Some interesting volume values were found regarding these ribs. Normally, it would be expected for the ribs with lightening holes to use a lower volume of material, however, due to the extra flange sections required to contain the lattice structures surrounding the holes, the volume was actually higher. This led to an overall higher mass value accordingly. These models were created in Creo using only an aluminum 6061 material. This once again allowed for a consistent comparison between the ribs regarding their mass and configuration. As will be seen later, the triangular lattice begins to show its disappointing behavior here, as it not only increased on the mass of the solid ribs but also the volume. The hexagonal and cubic lattice ribs, however,
have volumes and masses lower than standard and begin to start the case for the application of these structures in more aircraft, due to their reduced mass and volume.

Table 5: Volumes and Masses of Rib Models

<table>
<thead>
<tr>
<th>Design</th>
<th>Cubic w/o Holes</th>
<th>Cubic with Holes</th>
<th>Triangular w/o Holes</th>
<th>Triangular with Holes</th>
<th>Hexagonal w/o Holes</th>
<th>Hexagonal with Holes</th>
<th>Solid w/o Holes</th>
<th>Solid with Holes</th>
</tr>
</thead>
<tbody>
<tr>
<td>Volume (in³)</td>
<td>26.532</td>
<td>28.208</td>
<td>34.28</td>
<td>32.964</td>
<td>30.62</td>
<td>30.968</td>
<td>31.572</td>
<td>31.708</td>
</tr>
</tbody>
</table>

To represent the other main aspect of a wing structure, lattice spar models were also created and tested using FEM software. The same set of lattice patterns were used for the creation of these models, except for the solid tests. To add an extra case to the tests, a box section spar was created along with the i-beam spar, to show two different spar designs that are currently used. The overall dimensions of the spars measure as the following: 6 in tall, 2 in wide, and 239 in long. This may seem like a very narrow setup for the overall length, however, the purpose of this test was to show more the deformation and stress characteristics of different spar designs, rather than optimizing them to find the best dimensions for a test. For this reason, the spars were left fairly skinny, to maximize the potential differences in the measured values, making the best lattice design easier to distinguish. In Figures 16 – 20, the models go as the following: Cubic, Triangular, Hexagonal, Solid Box, and Solid I-Beam. These spars were also created in Creo, using the same lattice generator that the ribs used. The cell size, however, differed, to make the orientation of the lattice structures more even within the skin of the spar. For this reason, the cells of each lattice section measure 3 in x 3 in x 1 in. This design made the most of the 6 in x 2in sizing, giving a 2 x 2 layout with height and width in mind. For the purposes of eased analysis, these models were once again decreased by a factor of 0.25. The rations within each spar, however, remained the same for analysis.
Much the same as the rib models, the volumes and masses of the spars can be seen below in Table 6. Once again, the same trends exist with these models, as did with the ribs. The only exception is the box spar, which has a far higher material volume and mass than any of the other configurations. Similar to the ribs, the triangular lattice continues to not prove its worth, having a volume and mass higher than the solid I-beam. Overall, these values once again do not show any astounding decreases in volume or mass of the spars, though they do begin to. With further optimization of the lattice sizing, it is very possible that the volumes could be reduced even further to produce similar results as what will be seen in the next section.

**Table 6: Volumes and Masses of Spar Models**

<table>
<thead>
<tr>
<th>Design</th>
<th>Cubic</th>
<th>Triangular</th>
<th>Hexagonal</th>
<th>Solid Box</th>
<th>Solid I-Beam</th>
</tr>
</thead>
<tbody>
<tr>
<td>Volume (in$^3$)</td>
<td>37.872</td>
<td>49.272</td>
<td>42.024</td>
<td>67.734</td>
<td>43.42</td>
</tr>
</tbody>
</table>

**IX. Structural Analysis**

Using the CAD models discussed in the previous section, FEM tests could be conducted to analyze the stress and deformation characteristics of the lattice structures, when a simulated load is applied. Due to the discussion earlier, the tests for the ribs only used a pressure load applied to the outer surface area of the skin. The maximum ambient air pressure was chosen to represent this pressure at 14.7 lb/in$^2$. Though the aircraft would most definitely experience higher forces on the ribs during flight, this was a consistent value that once again allowed for a fair comparison between the designs. In order to receive deformation stress data from a FEM test, a fixed point must be defined to
base values from. In the case of the ribs, there was a small, rectangular section created at the trailing edge, where each corner of this was constrained to have zero translation in the x, y, and z directions, as well as zero rotation in these directions as well.

Originally, the plan for this analysis was to be able to test composite and metallic versions of these ribs, as well as a combination of the two, with the outer skin of the ribs and spars being made of a composite material, and the inner lattice being a metallic structure. Unfortunately, some compromises had to be made to obtain any results with these structures. Due to the sheer size of these models, they first were scaled to 0.25 of their original size so that the testing computers could even open the files within Patran, the FEM software. The idea within the software is to be able to simulate material properties by applying the Modulus of Elasticity and Poisson’s ratio values to different surfaces or elements on a model. When the complexity of a model becomes as high as one of these ribs, however, the bounds between two different surfaces becomes less defined and creates a problem for the machine attempting to calculate what is going on. Essentially, a much more powerful computer would be required to fully run a test such as this with two different materials applied.

This being said, the compromise that had to be made for any test to run, was to assign only one material to the ribs and spars. While this does in a way defeat the point of the discussion on composites, some valuable data was acquired that can be compared to composite properties to give at least an estimate of the deformation and stresses that would be exhibited by a composite version of this. To at least allow for comparison to current ribs and spars, the material selected was aluminum 6061 due to its low density and wide range of uses throughout the rest of the plane. Keep in mind that this had an E of 10 x 10⁶, a Poisson’s ratio of 0.33, and a density of 0.0975. Also to note is the sheer number of elements and nodes required to make up the mesh that encompassed the model. In FEM software, analysis cannot be conducted simply on a CAD model that is imported. Rather, a model is brought in, and a mesh of lines, called elements, and nodes, the connection points between nodes, is layered over the model like a blanket. Using these elements and nodes, the FEM software can create a very detailed estimate of the deformation and stress, based on how much these move from a simulated load. In the first plot below in Figure 21, the cubic rib with lightening holes can be seen. With this configuration, 2,088,483 nodes and 1,114,554 elements were used to create the mesh. Without reference, this may seem an acceptable number, however, the solid rib with lightening holes used 13,314 nodes and 6,834 elements. This is a much more reasonable number for any kind of efficient testing, as a mesh did not have to be created around each individual rod and ball of each lattice cell. The resolution of the meshes can be altered, though making the resolution any less than it was created large faults that would not let the tests run. For these reasons, compromises were made to at least show the overall characteristics of lattice structures within ribs and spars.

With all of the parameters of the tests set, the analysis on all of the rib and spar models was conducted and the results can be seen below. To begin, the cubic lattice spars are shown with a view of the deformation, stress, and an isometric view of stress in Figures 21, 22, and 23, respectively. The maximum deformation and stress values for both the cubic rib with lightening holes and the one without can be found in Table 7. The first plot, Figure 21, shows the overall deformation exhibited during the test of the cubic lattice structure with lightening holes. Shown in this figure is not only the original, unaltered structure, but also an exaggerated version, showing where the maximum deformations would be found. The maximum deformation occurred on the lower edge of this rib, in the left section of red under the first lightening hole. The scale on the right of the figure can be used to estimate the deformation in any region based on its color. Keep in mind also, that this scale is in inches, and most deformations are going to be in between 0.0005” and 0.0003” for this test. Regardless of the results for any of these tests, these deformations are incredibly tiny, which is great for the prospects of implementing these structures in future aircraft.
The stresses found during the test of the cubic lattice rib with lightening holes can be seen below in Figure 22. When creating a plot of results for these tests, two different factors can be applied to show. The first is the deformation output of the model, meaning what the resultant structure looks like. In this case, it is the mangled shape of the rib, however, this can be changed to not show this. The other aspect is called the fringe result, which changes the colors on the model and the scale to the right to be able to show where variation in a measured result takes place. In the previous figure, the fringe was set to show deformation. In Figure 22 below, the fringe was set to the stress, where the colors now highlight the areas with the highest stress calculated. The maximum stress in this test was found to be $2.48 \times 10^3$ psi. This may seem high, though the majority of the rib remained within the lowest region of the scale with a stress of $7.58 \times 10^{-9}$ to $6.62 \times 10^2$ psi. Again, a fantastic result for the overall dimensions used for the lattice cells and the load applied.

The final plot for the test of the cubic lattice rib with lightening holes is simply an isometric view of the model to show a little more of the complex variations of stress calculated throughout the model. It appears that the lattice elements on the exterior of the model tended to be experiencing higher stress than the interior, meaning that not only the structure of the lattice but also the orientation of it could affect the quality of the structural characteristics when under load. This variation once again is valuable data, however, as future testing could take place with an altered orientation to see if this would beneficially change the results of the stress tests.
The same test parameters and mesh resolution were used to run the calculations for the cubic lattice rib without lightening holes. As will be seen for the other configurations, the ribs without lightening holes produced much more consistent deformation plots than the ones with lightening holes, though the maximum deformation was always higher. This is a valuable finding though, as this type of deformation would be much easier to control, rather than a completely inconsistent version, such as in the last example. Overall, however, the maximum deformation still only reached 0.000589", which is still extremely small.

The stress plot also provided consistency, with the majority of the higher stresses taking place on the vertical members of the lattice. It is worth noting that the maximum stress found in all tests occurred at, or near the nodes that were set to be fixed during the test. This naturally is a point of conversion for the load that is applied to the rest of the model, and would likely not be seen to this degree in the field, as these points are not actually fixed. Even with its uniformity, the lattice structure without lightening holes did still see slightly higher stress than the latter test, though still only at 2.79 x 10³ psi.
The maximum deformation and stress values for the cubic lattice ribs can be seen below in Table 7. These are the most extreme values, meaning that the vast majority of the rest of the values calculated for the rest of the ribs were much lower for both criteria, again showing good results overall, especially for having material volumes and masses lower than the solid material alternatives.

<table>
<thead>
<tr>
<th>Rib Variation</th>
<th>Maximum Deformation [in]</th>
<th>Maximum Stress [lb/in²]</th>
</tr>
</thead>
<tbody>
<tr>
<td>Without Lightening Holes</td>
<td>$5.89 \times 10^{-4}$</td>
<td>$2.79 \times 10^{3}$</td>
</tr>
<tr>
<td>With Lightening Holes</td>
<td>$4.99 \times 10^{-4}$</td>
<td>$2.48 \times 10^{3}$</td>
</tr>
</tbody>
</table>

The next configuration to be run was the triangular lattice ribs. Before, a remark was made that the triangular lattices provided less than satisfactory results, and through these tests, this trend continued. First of all, no matter the amount of tweaking of the model without the lightening holes, the stress plot could not be generated. The cause for this remains unknown, however, the stress values were most likely only slightly lower than that of the cubic lattice without lightening holes. This is due to the minute difference in maximum deflection at $5.05 \times 10^{-4}$ in, so the maximum stress could most likely end up around $2.2 \times 10^{3}$ psi. This result does not seem like a good enough justification for the use of this configuration, however, due to the higher volume of material and mass used. The triangular lattice rib with lightening holes did not compare to the cubic ribs quite as well either, displaying not only a higher maximum...
deformation but also higher stress. The one benefit of this design was the uniformity, which simply allows for easier calculation and comparison later on.

Figure 27: Deformation of Triangular Lattice Rib

Figure 28: Deformation of Triangular Lattice Rib with Lightening Holes

Figure 29: Stress of Triangular Lattice Rib with Lightening Holes
Once again, the maximum deformation and stress values for the triangular lattice ribs can be found below in Table 8. Note the missing stress value for the rib without the lightening holes as described earlier.

<table>
<thead>
<tr>
<th>Rib Variation</th>
<th>Maximum Deformation [in]</th>
<th>Maximum Stress [lb/in²]</th>
</tr>
</thead>
<tbody>
<tr>
<td>Without Lightening Holes</td>
<td>$5.05 \times 10^{-4}$</td>
<td>NA</td>
</tr>
<tr>
<td>With Lightening Holes</td>
<td>$6.83 \times 10^{-4}$</td>
<td>$3.85 \times 10^3$</td>
</tr>
</tbody>
</table>

The final lattice structure rib tested was the hexagonal configuration. Overall, this lattice had very middle ground results, displaying slightly higher deformations with and without holes, and slightly higher stresses as well. The main difference was the way that the hexagonal ribs deformed. In this configuration, the ribs deformed downwards and crumpled into a thin structure. This gave a slightly different pattern of deformation overall, though, with all of these tests, these numbers remained low. Furthermore, the stress values were once again highest on the exterior elements, though under examination, the elements appear to fold under load, giving reason to why the structure compacted so much. Because the volume and mass values were the lowest for this configuration, and the stress and deformations were still very good, this would be the most efficient configuration for use in ribs.
Figure 32: Stress of Hexagonal Lattice Rib

Figure 33: Isometric View of Stress of Hexagonal Lattice Rib

Figure 34: Deformation of Hexagonal Lattice Rib with Lightening Holes
Figure 35: Stress of Hexagonal Lattice Rib with Lightening Holes

Figure 36: Isometric View of Stress of Hexagonal Lattice Rib with Lightening Holes

Table 9: Hexagonal Lattice Deformation and Stress Values

<table>
<thead>
<tr>
<th>Rib Variation</th>
<th>Maximum Deformation [in]</th>
<th>Maximum Stress [lb/in²]</th>
</tr>
</thead>
<tbody>
<tr>
<td>Without Lightening Holes</td>
<td>$5.71 \times 10^{-4}$</td>
<td>$2.72 \times 10^{3}$</td>
</tr>
<tr>
<td>With Lightening Holes</td>
<td>$6.06 \times 10^{-4}$</td>
<td>$3.75 \times 10^{3}$</td>
</tr>
</tbody>
</table>

With all of the lattice tests completed for the ribs, the final test calculated the deformation and stress of solid material ribs with and without lightening holes. These were I-beam-style ribs, having a much wider “flange” around the perimeter, with a narrow flange. In fact, the flange was 3 in wide, with all sections having a thickness of 0.25 in. Overall, the solid tests yielded the lowest stresses and deformations. The main difference with these tests, however, is that the maximum stresses were exhibited on a much larger area of the ribs. In these figures, the color red represents high stress, and it can be abundantly seen on both configurations of the solid ribs. This is in comparison to the lattice ribs, where the maximum stress is very hard to find in the plot, as it appears to only occur on a small area where the rib was fixed. So even though, this max stress was lower, the solid ribs will ultimately not perform as well over the lifespan of the aircraft that they are used in. It is worth noting that for all tests, the stresses exhibited were far below the yield stress of aluminum 6061 from the materials definition section, usually by about 37,000 psi. Seeing a small area of high stress is preferable to a large area, as the design can simply be supported in that area to reduce stress, rather than having to redesign the whole rib, as would be required with these solid ribs.
An interesting note about these solid ribs is that the rib with lightening holes experienced a little higher deformation, but a much lower maximum stress, dipping down to 860 psi. This again comes at a volume and mass penalty overall, but creates a very exciting stress plot, showing the stress dips around the lightening holes.
Figure 40: Deformation of Solid Rib with Lightening Holes

Figure 41: Stress of Solid Rib with Lightening Holes

Figure 42: Isometric View of Stress of Solid Rib with Lightening Holes

Table 10: Solid Rib Deformation and Stress Values

<table>
<thead>
<tr>
<th>Rib Variation</th>
<th>Maximum Deformation [in]</th>
<th>Maximum Stress [lb] [in²]</th>
</tr>
</thead>
<tbody>
<tr>
<td>Without Lightening Holes</td>
<td>$3.28 \times 10^{-4}$</td>
<td>$1.05 \times 10^3$</td>
</tr>
<tr>
<td>With Lightening Holes</td>
<td>$3.45 \times 10^{-4}$</td>
<td>$8.60 \times 10^2$</td>
</tr>
</tbody>
</table>
The second portion of the analysis dealt with the lattice spars created, as well as the solid baseline configurations. Testing these spars led to the same difficulties that plagued the rib tests, so once again, the spars were tested on a scale of 0.25, with an all-aluminum 6061 material. The same plots for each test were made as in the rib analysis, having one for deformation, stress, and an isometric view of the stress. Instead of applying pressure to the exterior of the spars, a load was instead applied, of 2,337.5 lbs. At the time of testing, the light attack aircraft that these spars were meant to be used in weighed 18,700 lbs. Because the scale of the spars was decreased by a factor of 0.25, so was the load applied to the spars. 2,337.5 lbs is equal to ¼ of half of the weight of the aircraft since a single spar should theoretically experience half of the load for a plane, with a configuration that has two spars per plane. What will be found for all tests of the spars is that extremely high deformation and strain values were found. This is for a few reasons, mainly that the spar dimensions were far too small to carry this sort of load, and also, the load applied is still high for a light attack aircraft, further exaggerating the effect. The saving grace of this, however, once again is that the trends exhibited by these spars could easily be scaled to any dimension and load.

The first test for the spars was once again the cubic lattice configuration. There were no lightening holes used for this test, as it is less common to see those in any current setup. The spar was also shown to be fixed on the respective four corners of the left end. This is actually a true setup, as this would be a mounting point in the fuselage for the wings, where the spars should not move. Though the load is applied in a downward direction, the same deformations and stresses would be seen if it were pointed in the direction that the load is technically applied on an aircraft, upwards. This is a small technicality that would not change the results, as the z-orientation of the lattices is consistent and equal. Overall, the cubic lattice performed the best in the deformation category for the spars, yielding a value of 4.38 x 10^8 in. Of course, this again is not a true value but is still good for evaluation. The stress plot also is promising, with the majority of the spar seeing very little stress, and the maximum once again being at the fixed points. This is a theme that occurred on all of the tested spars.

![Figure 43: Deformation of Cubic Lattice Spar](image-url)
The values for the maximum deformation of the cubic lattice spar can be seen in Table 11. With the correct dimensioning of the overall spar, this could be a very promising solution for future use. Its low material usage and relatively uncomplicated lattice pattern would make it ideal for production out of the three lattice patterns, making it truly a viable choice.

The next test run was on the triangular spar. This test yielded the highest deformation of any spar, and also refused to produce a stress plot for comparison, but it clear that the highest stress would have been found on this test as well. For this reason, the triangular lattice structure would not be ideal to use in any application on an aircraft. There are better, stronger configurations that use less material and therefore weigh less, making the triangular obsolete.
The last lattice test run for this project was of the hexagonal lattice spar. Overall, this spar performed the best of the lattice structures, with the lowest deformation and lowest stress. Most likely, this is due to the more complex nature of the hexagonal lattice, where the different rod elements are aligned in larger angles, allowing for the overall structure to have more room to shift and spread the load over the majority of the elements. This can best be seen in Figure 48, where almost the entirety of the spar is at its lowest stress. Once again, as indicated in the figure, the highest stress is seen at the fixated points. If the peak performance is desired for a spar, and lattice structures can be produced in a format such as this, the hexagonal lattice would be the one to use. Its compromising factor is that the structure is much more complicated than the cubic structure, which would make it harder to produce a clean, consistent structure overall. 3D printing would be the method to produce any metal lattice structure at the moment, and this can be created using this method, however, so it is feasible in a broad sense, with the right resources.
The final test for this project was that of the I-beam and box section spars, created to compare to the lattice structures. Plainly, with comparison to the lattice configurations, these should most definitely still be used for the time being. These spars might weigh a bit more, but the deformation and stress seen were decreased but quite a large factor, dropping from 5.07 x 10^8 in on the hexagonal lattice spar to 2.92 x 10^7 in on the I-beam spar. The box section spar would also not be the spar design to use, as it uses more material for worse results, much like the triangular lattice spar does. Finally, the solid spars showed the lowest stress overall, with the I-beam showing a maximum stress of 1.34 x 10^{13} psi. If the lattice spars could be further optimized, it is entirely possible for the hexagonal or cubic spars to meet or surpass the I-beam as far as results go, though, under this configuration, the I-beam would most definitely be the safest choice for any aircraft. For a light attack aircraft, where advanced, high-stress maneuvers could be seen at any time, a spar with the best stress characteristics would be desirable to minimize failure risk. Below are the final deformation and stress plots for the solid material spars. Their result values can be found in Table 12.
Figure 50: Deformation of Solid Box Spar

Figure 51: Stress of Solid Box Spar

Figure 52: Isometric View of Stress of Solid Box Spar
Figure 53: Deformation of Solid I-Beam Spar

Figure 54: Stress of Solid I-Beam Spar

Figure 55: Isometric View of Stress of Solid I-Beam Spar
The creation of these ribs and spars in a real application would be nothing short of a challenge. Currently, the best method to produce a lattice structure such as what is being suggested is to use metal 3D printing. Aluminum 6061 is a metal that can be printed; however, the quality of 3D printed metals can be inconsistent. This process uses metal powders and a high-powered laser, which fuses the individual molecules of metal together in a layering process, creating the structures desired. For the fairly compact lattice structures that will be discussed later, this process would be ideal. Using composites for the lattice section would be extremely hard, and most likely composed primarily of the matrix material, rendering the structure very weak. Though it is not impossible, the cost would likely be too high to ever consider for production. A mold would have to be created to line with composite materials for production, but this would be extremely tedious work. This is possible, but inefficient at best. The outer flange/skin portion of the ribs and spars could be created of composites, however, as these are flat, plate-like sections that are ideal for composite production. It is likely that if the inner, lattice structure of a wing component was made out of 3D printed metal, this could be combined with a non-cured composite to essentially bond the two together. The extra rods that stick out of the lattice could be embedded into the voids of the fiber fabric material, and the matrix material could then essentially glue the two together, while also making the fiber material rigid. If the computer used to conduct the FEM tests could handle it, this would be a very valuable test to conduct. It would be interesting to see if points of high stress would develop where the two materials meet, or possibly yield. It is fair to assume from its material properties that a carbon fiber composite lattice structure would outperform an aluminum one. This is especially true in a completely simulated environment such as Patran. Though the tests performed were purely conceptual, the tests were designed to reflect structures that had the potential to be actually used and produced. This is another reason why the solely aluminum structures were tested instead of an entirely carbon fiber composite structure.

X. Conclusion

This project aimed to test and evaluate composite lattice structures for use in critical wing components. Through the use of advanced CAD and FEM software, this was achieved to the highest degree possible, with the materials given. Though dual-material ribs and spars could not be tested efficiently to find their deformation and stresses, their relative trends were found. Three different lattice structures were tested overall: triangular, cubic, and hexagonal. A solid material version of both the rib and spar was also tested to give a fair baseline for what is currently used within aircraft. Having the lowest stress and deformation for the spars, the hexagonal lattice appeared to be the superior of the three patterns. The cubic lattice pattern, however, is a close contender, as its overall mass and volume of material were lower, and its stress and deformation for the ribs were better. Also, due to the slightly less complex structure of the cubic lattice, it would be easier for 3D printers to replicate consistently at a high quality, as it features many more straight runs than the hexagonal. If lattice structures were to be used in an aircraft, it would be best to use the cubic lattice in the ribs and the hexagonal lattice in the spars. When designing an aircraft, top performance is required, so exploiting every structure to its fullest would make sense in this scenario.

With the combination of a material, such as a carbon fiber composite, the performance of all of these spars would ultimately rise. Though the manufacturing of these ribs and spars would most definitely be challenging, this would be a fantastic way to save weight on the aircraft, while sacrificing little on the overall stress design. This is true especially for the ribs, where they are not fully carrying any load, but rather supporting the airfoil under the pressure of the air around it. The use of lattice structures here could be justified much more easily than in the spars, simply because the performance was very similar to the solid material ribs. The cost is higher, but again, weight is saved on the aircraft, therefore also decreasing the power and fuel required. This project was aimed at using the findings in a light attack aircraft for an AIAA paper competition, and the lattice ribs would most definitely add to decreasing the weight of the light attack airplane. Even more, this shows a new design, not previously used in aircraft, opening the door for more optimization testing, to most certainly improve upon the findings in this paper. Using a composite lattice spar in any application soon could be hard to justify. The cubic and hexagonal designs saved very little weight over the I-beam spar and performed much worse. This could definitely be worsened by the orientation lattices were set in the spar, but all the same, their performance was ideal. For now, it seems best to continue using I-beam spars, with composite materials to increase the overall rigidity of the wing for its lifespan.

### Table 12: Solid Spar Deformation and Stress Values

<table>
<thead>
<tr>
<th>Spar Configuration</th>
<th>Maximum Deformation [in]</th>
<th>Maximum Stress [lb/in²]</th>
</tr>
</thead>
<tbody>
<tr>
<td>Solid Box</td>
<td>3.27 × 10^7</td>
<td>1.34 × 10^13</td>
</tr>
<tr>
<td>Solid I-beam</td>
<td>2.92 × 10^7</td>
<td>1.24 × 10^13</td>
</tr>
</tbody>
</table>
XI. References


