Design, Fabrication, Structural Testing, and Numerical Analysis of a Small Scale Composite Wing

A Senior Project

presented to

the Faculty of the Aerospace Engineering Department

California Polytechnic State University, San Luis Obispo

In Partial Fulfillment

of the Requirements for the Degree

Bachelor of Science

by

Jacob David Gaunt
Juan Carlos Flores
And
Vincent Andrew Perry

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VII. Analysis
A. Comparison
B. Error Analysis and Future Work

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Nomenclature

E = modulus of elasticity (psi)

G = Shear Modulus

ε = strain

σ = stress (psi)

υ = Poisson’s Ratio

λ = Lame’s Constant
Abstract

A small scale composite wing based on a design found on an experimental aircraft was designed, constructed, and tested dynamically and statically. The wing was constructed similarly to an experimental aircraft wing. The performed static test was intended to produce pure bending. Strain gages were used to measure strains on the wing structure. The strains were converted to stresses to aid in analysis. The static test results suggested that the wing was actually under torsion. Four structural modes were found from the static test. A finite element analysis model was made to compare experimental results to numerical analytical results. The comparison showed a good correlation with spar stresses, but differences in the experimental and modeled load resulted in no comparison with rib and skin stresses.

I. Introduction

In the Spring quarter of 2009 a kit aircraft was donated to the Cal Poly Aerospace Engineering Structures Lab. The AERO 532 Composites course used the kit to allow students to fabricate one of the aircraft’s composite wings. The kit aircraft was a SeaHawk biplane donated by Gary LeGare. The kit came with various pre molded components as well as wing skins, and material to cut spars out of. The ribs were manufactured using Vacuum Resin Infusion techniques. A completed kit is shown below in Figure 1.

![Completed SeaHawk from(Aero Gare SeaHawk)](image)

Figure 1. Completed SeaHawk from(Aero Gare SeaHawk)

The kit included everything needed for assembly of a complete aircraft minus the engine. However, only one of the wings was constructed with the intent of using it as a senior project, and allowing the rest of the aircraft to be used for future projects. Wing building took a quarter to complete. The goal for the original wing was to be tested both statically and dynamically for a Senior Project, and compare results to a finite element analysis (FEA) model.
The project was started the next quarter. During the first week, testing methods needed to be established. Static loading was to be performed using the Instron machine, while the dynamic testing was to be done using the Structures Lab shake table. As plans progressed, some complications were found in the design for the test stand needed for wing support while testing. Some of the complications included stand geometry and positioning. The stand would have to support the wing while the wing was being loaded during static and dynamic tests. A large area would be needed for testing due to wing size. A reconfiguration of the lab was needed if the wing was to be tested. This was not feasible since AERO classes and other research needed to be performed in the same lab.

Another complication was retrieving accurate material property data from the wing components to use with the FEA. The wing did not come with test coupons to find material properties. In order to determine material properties, the remaining wing would have to be destroyed in order to obtain material samples. The components with different material properties were the skin, spar, ribs, and a section of the wing with a carbon fiber strip. This was deemed impossible due to the remaining wing being put to a better use as another project.

The solution to the problems found with the full scale wing was to make a scaled down version of the composite wing. The scaled down wing was made using the same exact manufacturing process as the full scale wing. The design of the structure would also be similar. The small model would allow for testing of all the components for material properties. The small wing would also allow the use of all testing equipment where they stood, without rearranging the laboratory.

A. Senior Project Description

The project involved construction, test, and analysis of a hollow composite wing. The goals of the project were to match experimental results with finite element analysis results. The results are expected to have some error. The error will come mostly from manufacturing the composites, joining the different components, and geometrical differences between the physical model and the FEA model, along with inherent limitation of the FEA model.

1. Wing Design

The wing was designed to resemble the design of the SeaHawk wing. The airfoil chosen was a NACA 2415 shown in Figure 2. Twenty percent of the airfoil was removed from the trailing edge for fabrication. The section was missing on the original wing to leave room for flight control surfaces. The wing span of the wing was decided to measure approximately two feet or smaller to fit in the aerospace department wind tunnel.
Figure 2. NACA 2415 Airfoils Used for Wing Root and Wing Tip

All aspects of the wing design were made to be similar to the SeaHawk.

2. **Wing Construction**
   The wing was made with the methods used in manufacturing the original wing. Materials and design were similar to find scaled down data.

3. **Testing**
   Static loading will be performed using an apparatus designed by the group. The wing was rigidly secured in the middle section while the wing tips were loaded. The wing was loaded upside down to help with stability and to simulate a real maneuver. Strain and stress versus load is compared using the experimental data. The experimental data is compared to an FEA modal. Dial indicators were used to measure wingtip deflection. These measurements were also compared with FEA results. Dynamic testing was also performed using a shaker table. The purpose of the dynamic testing is to find the dynamic modes of the structure. Again the experimental results are compared with the FEA model.

**II. Wing Design**

The airfoil chosen was a NACA 2415. The root airfoil measured 9.2 inches chord length while the wing tip airfoil measured 7.2 in chord length. After twenty percent of the airfoil was removed the root chord was 7.4 inches while the wing tip chord was 5.75 inches. Figure 3 shows the wing tip airfoil and a side view of the wing design. The wing was designed with taper and dihedral, so that the structure would resemble that of the SeaHawk’s structure.
The wing was originally designed to have wing tip caps. The caps were left off to leave room for strain gage wires and avoid excessive reinforcement. Figure 4 shows the dihedral. Two wings were built with a dihedral of 9.5 degrees.

The length of each side was 10.9 inches making the wing span just less than 22 inches. Figure 5 shows a top view of the wing design. An isometric view of the wing is shown in Figure 6.
III.  Wing Construction

A. Material Types  
The materials used in the construction were fiberglass, carbon fiber, and foam. The materials were chosen to be similar to those used in the construction of the SeaHawk wing. The SeaHawk wing used multiple layers of fiberglass along with one layer of carbon for spar reinforcement. An example of these materials is shown in Figure 7.

![Figure 7. Foam, Fiberglass, and Carbon Fiber as Used for Construction](image1)

The spars and ribs of the wing are made from a fiberglass, foam sandwich composite plate. The foam used for this is shown in Figure 8.

![Figure 8. Foam Used to Create Sandwich Panels](image2)

B. Mold Experimentation  
The first step needed for construction was a mold to form the wing skin. This step in the project took a significant amount of time. Several ideas were traded, and the group thought a plaster mold might work the best. A
container to hold the plaster was made out of wood, and a machined foam wing as, wrapped in saran wrap was used to shape the plaster as the plaster was setting. Once the plaster dried, the wing was removed. In the end, the mold was too rough and cumbersome to work with. There were also curing issues with the plaster which prevented the mold from being usable in a reasonable amount of time.

![Figure 9. Machined Foam Wing](image)

The next idea was to use a CNC hotwire cut wing as a male plug instead of using the wing to make a female mold. The foam was cut using the Cal Poly UAV Lab’s CNC hotwire as shown in Figure 11.

![Figure 10. Example of Plaster Mold](image)
Figure 11. CNC Hotwire

Two halves of a wing were made due to dihedral and taper, and later glued together to make a solid, single wing. One half of the wing mold is shown in Figure 12.

Figure 12. Foam Wing

The layup procedure involved vacuum bagging the wing and laying up fiberglass over the wing using a vacuum resin infusion (VRI) process. Half of the wing was done at a time. After the first try, adjustments were made, and a useable wing skins were produced. This method proved to be successful, and suitable for the project.

B. Construction Techniques

1. UV Resin

An early method for skin construction used ultraviolet (UV) curing resin. This would have significantly shortened the time for skin construction due to UV resin curing in about 10 minutes sitting in the sun, while
conventional resin needs to cure overnight. However the UV resin produces a more intense exothermic reaction and caused extreme deformation to the foam mold. This deformation is shown in Figure 13.

![Damaged Mold Due to UV Resin](image13)

### Figure 13. Damaged Mold Due to UV Resin

2. **Hand layup**
   Another method considered was to use the hand layup method. This method involves saturating fiberglass cloth in epoxy, and draping the cloth over a mold, then wrapping a vacuum sealed bag around the whole part to let the fiberglass cure in the correct shape. This is a very simple method, but parts made this way are overly saturated with resin, resulting in more brittle, heavier parts. There is also more variability in material properties, which would make an accurate analysis much less accurate.

3. **VRI Process**
   Vacuum resin infusion, or VRI, is a process where a vacuum is used to pull resin through dry fabric. In this application, the fabric is laid over a mold. Peel ply, followed by a material known as flow media is placed on top of the fabric to allow the resin to flow over the entire part. A vacuum bag is then sealed over the part, with tubes installed to allow epoxy to be added when the vacuum has formed. When ready, the epoxy is added by letting the vacuum draw the resin into the tubes and through the part. Vacuum resin infusion allows an optimum strength to weight ratio, in addition to an infinite set up time since the resin isn’t added until the fabric and vacuum bag is situated as desired. Some disadvantages of VRI is the process may not be suitable for large parts due to the resin starting to cure before the part is fully saturated, and there are problems with the flow media curving around sharp corners. A set of the wing skins made using the VRI method is shown in Figure 14.
C. Wing Assembly

The three sets of components built for the wing assembly were the wing skins, ribs, and spar. All components were manufactured using VRI techniques. The same foam mold was used for the wing skin and rib manufacturing.

1. Wing Skin

The wing skins were made by vacuum bagging the foam airfoil cutout and using the bagged airfoil as a male plug. Peel ply was applied to the plug first, next, two layers of fiberglass followed by a strip of carbon fiber for added strength. The original scale wing had a carbon strip on each wing directly over the spar. The wing manufactured for the project was designed to use a wider carbon strip due to symmetry shown in Figure 15. A thinner carbon strip could move significantly more during VRI, leaving the wing unsymmetrical about the root chord. Figure 16 shows the wing skin with peel ply still attached. The peel ply separated the foam wing wrapped in vacuum bagging from the fiberglass used for the wing skin.
Four wing skins were made in total. Two upper and two lower. The wing skins were manufactured the exact same way using the same foam mold and amount of fiberglass with carbon. Fibers were carefully placed in matching directions to ensure consistency between the wings. Figure 17 shows a top view of a wing upper skin. This picture illustrates the movement in the carbon strips during VRI. West Systems epoxy was used for every step in the wing construction process requiring epoxy.
2. **Wing Ribs**
   The ribs were also made using VRI, only they were cut out of a sandwich panel. To get precisely shaped airfoils to cut out of the panel, the actual foam mold was cut. Figure 18 shows the mold inside the wing skin.

![Figure 18. Cut Foam Mold Inside Wing Skin](image)

After cutting the mold, the airfoils were traced on the sandwich panel. These airfoils shaped ribs were cut out and sanded to fit flush in the wing skins. Figure 19 shows the cut out pieces of the mold used for stencils.

![Figure 19. Cut Pieces Of Foam Mold Used As Stencils For Ribs](image)

3. **Assembly Process**
   The first step for assembly was to attach the spar to the upper wing skin. Figure 20 shows the spar secured in place with tape while epoxy mixed with foam dust was applied. Packing tape was applied to the bottom of the wing
skin to avoid permanent attachment to the spar. The wing skin was permanently placed on the upper wing skin at a later time.

![Figure 20. Spar Attachment To Wing Skin](image.jpg)

Once the resin cured shown in Figure 20, a strip of fiberglass saturated with resin was hand layed at the base of the spar and wing skin for added strength. Peel ply was then applied to the strip to add support while curing, and absorb some excess epoxy. shown in Figure 22.

![Figure 21. Spar Attached To Wing Skin](image.jpg)

![Figure 22. Fiberglass and Peel Ply Added To Spar](image.jpg)
After the spar was in place, the ribs were cut and sanded shown in Figure 23 to fit in the wing skin at designated distances from the root chord. The ribs were attached the same way the spar was, using micro-balloons (foam dust) in the resin shown in Figure 24.

![Figure 23. Ribs Sized To Fit With Spar And Wing Skin](image)

The upper wing skin now had the spar and ribs in place. The lower skin was ready to be attached. To attach the lower skin, the ribs and spar were routed to allow a volume for resin. The channel made between the fiberglass shown in Figure 25 allowed the application of resin without having the resin spill over onto the wing surface.
Figure 25. Routing Ribs And Spars

The channels made by routing were filled with epoxy shown in Figure 26. Cotton flox was added to the resin to add volume and to make it the consistency of peanut butter. This allowed for the resin to remain in place while the lower skin was attached. Figure 26 shows the application of the thick resin.

Figure 26. Resin With Flox To Attach Lower Wing Skin

The lower skin was placed on the ribs and spar. Weights were placed on the top skin to apply pressure while the resin cured. Figure 27 shows the two wings supported underneath by wood while being compressed.

Figure 27. Weights Compressed Wing During Curing
The wing was now one piece. The top wing skin required removal to route a channel in the ribs and spar. This was done to add strength to the upper wing skin bond. A wedge tool was placed between the wing skin and spar shown in Figure 28 to carefully pry the wing skin off of the ribs and spar. Figure 29 shows the wing assembly with the upper skin removed.

![Wedge Used To Remove Top Wing Skin From Ribs And Spar](image)

Figure 28. Wedge Used To Remove Top Wing Skin From Ribs And Spar

Channels were routed in the ribs and spars as done before. Strain gages were then applied to the spar and rib. Figure 30 shows the wing assembly with routed ribs and spar with attached strain gages.

![Wing Assemblies With Top Wing Skins Removed](image)

Figure 29. Wing Assemblies With Top Wing Skins Removed
The upper wing skin was then glued back on to the assembly. The last step of the construction was hand laying one strip of fiberglass to the leading and trailing edge, the trailing edge is shown in Figure 31. Figure 32 shows the wing fully assembled with the leading edge in place.
IV. Experimental Procedure

Two type of experimental test were performed, dynamic and static loading. The purpose of the dynamic test was to determine the dynamic modes of the structure. The purpose of the static loading test was to apply a distributed load over the wing tip to simulate bending to determine strains, and thus stresses, and to determine wing deflection. All of this data is then compared to an FEA prediction to evaluate the accuracy of the modeling methodology.

A. Preparation

1. Material Properties

In order to evaluate a model of the wing in FEA, material properties had to be found to be input to the software. Sample plates were made using the exact same procedures and materials as were done with the wing skins in order to create test specimens that had the same properties as the skins. Using the VRI method, a plate of two layers of fiberglass, and a plate of two layers of fiberglass and one layer of carbon were made. These amounts of layers were the same as were used for the skins. Ten 2” x 5” test coupons were made for each laminate type. Each test coupon had metal tabs glued on its ends that it could be secured in an Instron tensile test machine. The test machine used was an Instron 8801 tensile test machine and is shown in Figure 33.

![Figure 33. Instron 8801 Machine](image)
A 1 mm/min loading rate was used and each of the pieces was tested to failure. A failed sample in the machine is shown in Figure 34.

![Failed Carbon, Fiberglass Laminate Sample](image)

**Figure 34. Failed Carbon, Fiberglass Laminate Sample**

All of the data was transferred to an Excel file and the average values were found. The results of the tensile test are as follows in Table 1.

<table>
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<tr>
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<th>Carbon, fiberglass</th>
<th>Fiberglass</th>
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<tr>
<td>Maximum Load (lbf)</td>
<td>2373.49</td>
<td>802.12</td>
</tr>
<tr>
<td>Young’s Modulus, E (ksi)</td>
<td>2579.67</td>
<td>1533.4</td>
</tr>
<tr>
<td>Rel. Std. Dev. (%)</td>
<td>8.94</td>
<td>5.96</td>
</tr>
</tbody>
</table>

As expected to carbon, fiberglass laminate was significantly stronger than the fiberglass only. This is why the SeaHawk had a strip of carbon down the wing skin, to significantly increase strength where it was needed. The samples were all very consistent as well. The standard deviation on young’s modulus was below 5% for each laminate, and the ultimate load’s standard deviation was below 10% for each as well.

2. **Fixture Design and Construction**

In order to safely and accurately load test the wing, a fixture was designed and constructed to secure the wing. The fixture was designed to be used for both static and dynamic tests. The fixture was machined from a piece of wood to match the contour of the middle section of the wing. Weather stripping was placed inside the fixture to securely hold the wing in place during the static load test. The weather stripping was removed during the dynamic
test so that no added damping occurred. A hard wood was chose so that the fixture could be assumed to be rigid. The fixture is shown in Figure 35.

![Figure 35. Fixture, Top View](image)

The fixture was a clam shell design so that it could be clamped around the wing and securely fastened. There are recessed holes which allow the fixture to be attached to a shaker table and other testing apparatus. These are shown in Figure 36. The fixture on the wing is show in Figure 37.

![Figure 36. Fixture](image)

![Figure 37. Fixture on Wing](image)
3. Strain Gage Placement

Strain gages were placed at several points along the wing. This was done so that multiple experimental data points could be compared to FEA results. All strain gages were $45^\circ$, $90^\circ$, $45^\circ$ rosettes. This allowed for the simple calculation of principle strains. The gages were applied using manufacture suggested procedures, which included being attached with manufacturer provided glue, and protected with a polyurethane coating. A terminal was placed close to the gage as well. Wires and lead wires were soldered on so the gages could be connected to a strain indicator. A finished strain gage with lead wires is shown in Figure 38.

![Attached Strain Gage](image1)

**Figure 38. Attached Strain Gage**

Strain gages were placed on the left and right skins, the outermost rib on one side, and the spar on one side the locations are shown in Figure 39. The gage factors of the gages on the skins were 2.11. The gage factors for the gages on the rib and spar are 2.05.

![Strain Gage locations](image2)

**Figure 39. Strain Gage locations**
The squares represent the location of the skin gages, while the “x’s” represent the gages on the spar and the rib. The locations of the spar and rib gages can also be seen in Figure 40.

![Figure 40. Strain Gages Inside Wing](image)

**B. Static Testing**

1. *Apparatus Design and Construction*
   
   A composite beam bolted to a heavy steel table and further supported by a wooden post was used for static testing. This composite beam is shown in Figure 41.

![Figure 41. Composite Beam used for static testing.](image)

The purpose of the beam extending out from the table was so that the wing could be loaded on its tips. The wing was mounted upside down, with weights on the end, so that the wing would be seeing a load similar to a “positive g” maneuver similar to an actual aircraft wing. The loaded wing in this configuration is shown in Figure 42.
Containers were placed under the weights in case of slippage of the clamps; the weights remained secure during the entire test. C-clamps were secured to the strips of metal which were attached with weather stripping. This was done to approximate a distributed load. This connection is seen in detail in Figure 43. Weight carriages were hooked to the clamps to hold weights to load the wing.
2. **Procedure**

The strain indicator used to measure strains in the strain gage was a Vishay P-3500 strain indicator. Three of these were used so that each gage could be read on a rosette at a single time. The strain indicator is shown in Figure 44.

![Vishay P-3500 Strain Indicator](image)

Figure 44. Vishay P-3500 Strain Indicator

For each strain indicator, a half bridge circuit was used to insure variances in temperature and lead wire resistance were minimized. The temperature compensating circuit was made from a second wing which was unloaded. The strain gages were attached to the same material as the loaded wing, so temperature compensation is possible. A diagram of a half bridge circuit is shown in Figure 45.

![Half Bridge Circuit](image)

Figure 45. Half Bridge Circuit from (How Sensors Work - Strain Gage)

The load test was performed using the following procedure. The wing was attached to the apparatus, and a single strain rosette was attached to the three strain indicators using the half bridge circuit described previously. The gage factor was entered as the appropriate value. The indicators were balanced to a 0 strain at an unloaded state to make
data reduction easier. The weight carriages were then attached. These weighed 4.5 pounds, which is counted as a load. Strain measurements were taken. A 10 pound weight was attached to the weight carriages, and the strain was recorded. Another 10 pound load was added and strain was recorded. This process repeated until 40 pounds of weight were attached (in addition to the load from the carriages). Each weight was removed carefully and the strain was rechecked to insure consistency. This test was repeated for each strain gage. One of the authors is seen attaching lead wires to the strain indicator in Figure 46. For each gage, the correct gage factor and temperature compensating circuit was connected.

Figure 46. Experimental Setup

Wing deflection was also measured on each wingtip for each loading condition; this was done with a dial indicator secured to a weight. This is seen in Figure 47. Deflection was recorded for each condition. The wing was removed and replaced with the second wing. The Spar and left wing skin strain gage were damaged, so only measurements on the right wing skin and rib were taken.

Figure 47. Dial Indicator to Measure Deflection
3. Data Reduction

After all strain data was taken, the data was transferred into an excel worksheet. The strains were transformed into normal and shear plane strains using the transformation equation:

\[ \varepsilon_\theta = \frac{1}{2} (\varepsilon_x + \varepsilon_y) + \frac{1}{2} (\varepsilon_x - \varepsilon_y) \cos(2\theta) + \frac{\gamma_{xy}}{2} \sin(2\theta) \]

The equation was used for each strain gage direction (45°, 90°, -45°), and allowed normal strain in the x and y directions and shear strain to be found. These directions are defined in Figure 48.

![Figure 48. Directions of Strain](image)

The strains in the x and y directions were then converted into stresses using the following equations

\[ \sigma_x = \lambda (\varepsilon_x + \varepsilon_y) + 2G \varepsilon_x \]
\[ \sigma_y = \lambda (\varepsilon_x + \varepsilon_y) + 2G \varepsilon_y \]
\[ \tau_{xy} = G \gamma_{xy} \]

Where

\[ \lambda = \text{Lame's Constant} = \frac{vE}{(1 + v)(1 - 2v)} \]
\[ G = \text{Shear Modulus} = \frac{E}{2(1 + v)} \]

and

E= Modulus of Elasticity
The modulus of elasticity was found on page 21, and Poisson’s ratio was assumed to be 0.3.

C. Dynamic Testing

1. Apparatus

The structures lab Bendix shake table was used for dynamic testing shown in Figure 49. The shaker allows for various types of wave, amplitude, and frequency inputs.

Figure 49. Bendix Shaker

Amps were controlled by an MB Electronics Power Amplifier Model 2250 MB shown below in Figure 50.

Figure 50. Power Amplifier

Frequency and wave type was controlled using a Hewlett 33311A Function Generator shown below in Figure 51.

Figure 51. Function Generator
2. Procedure

Amplitude was first set at one and using the range selection buttons on the function generator, the frequency was increased until the wing modes were reached. This was found by the distinct tone the wing made when frequency was on these modes. The amplitude was increased to two and three, increasing the frequency as before. Figure 52 shows the wing attached to the apparatus.

![Figure 52. Wing Mounted To Shaker Apparatus](image)

As amplitude changed, the same modes were found for each of the amplitudes. The results were recorded and plotted. Accelerometers were not available for the experiment, only sound was used to find modes. The inclination of the wing did not allow for a visual analysis, salt or other visual aids would slide to the wing root.

V. Experimental Results

Results were recorded and plotted. Each of the experiments has a specific method for interpreting results.

1. Static Test Results

The data created from the strains is presented in the following section. It should be noted that the load in the figures is the distributed load seen on each wingtip. Figure 53 shows the stresses of the right skin. The large stress in the y (span wise) direction is expected, but the large amount of shear suggests the wing in under a torsion load.
Figure 53. Right Skin Stresses

Figure 54 shows the stresses on the right skin of the second wing tested. There is a high amount of stress in both the horizontal and vertical directions, suggesting that the wing was not loaded under an equal load on both wingtips. There is much less shear in the second wing tested, compared to the first. This is likely a difference in loading conditions.

Figure 54. Alternate Wing Right Skin Stresses

The left skin’s stresses are shown in Figure 55. These results are similar to that of the second wing’s right side skin, though with more shear. Again, this suggests the wing was not loaded equally on both sides. The weights
themselves were equal, so the imbalance was probably due to the “distributed” loads being centered at slightly different locations, causing torsion.

Figure 55. Left Skin Stresses

Figure 56 shows the stresses of the spar. The results indicate that there is little load in horizontal direction of the spar and most load is in the vertical direction. This suggests that the load is mainly being taken in the vertical direction or in other words the spar is taking mainly bending loads, and the spar does not take much tension.

Figure 56. Spar Stresses
Figure 57 shows stress on the rib. There is a significant amount of load in the horizontal direction, suggesting there is a torsion load. As the wing twists, the rib starts to take loads in its horizontal direction. Again, this suggests that the wing is not under pure bending. The negative (compressive) stress in the rib suggests the wing skin is “squeezing” the rib as it bends down.

Figure 57. Rib Stresses

Figure 58 shows that the rib of the second wing only sees loads in the vertical direction, which makes sense as there should be no torsion giving a load in the horizontal direction. This data suggests the second wing may have been loaded in a more pure bending orientation.

Figure 58. Alternate Wing Rib Stresses
The experimental data suggests that the wing was not under pure bending as intended. The torsion was likely caused by unaligned wingtip loads. This will likely cause a disagreement with a FEA prediction due to FEA boundary conditions applying a precise bending load. In future work, the fixture should be created that insures the loading on the wingtips are balanced to eliminate torsion. This fixture was beyond the scope of this project, so it was not attempted.

Figure 59 shows the wingtip deflection averaged over several tests. There were difficulties obtaining accurate measurements due to instability in the bases on which the dial indicators were mounted. The dial indicators would move slightly, thus slightly changing the results. The results lack accuracy, but do mostly show the expected trend of increased deflection at higher loads.

![Figure 59. Average Wingtip Deflection](image)

The results show a decrease in deflection at the highest load. Obviously this is erroneous and is caused by the inaccuracy of the dial indicators measurements due to unstable bases.

2. Dynamic Test Results

Four nodes were found at three different amplitudes. These results were compared to plate theory for any similarities. For modal analysis, a four mode pattern is common. Many sources researched used an accelerometer plot to determine location and magnitude of modes. If the experiment used an accelerometer, the plot would have looked like the example in Figure 60. For our experiment, location was found without the magnitude.
Results show a good indication that at least 4 modes exist. Table 2 and the plot shown in Figure 61 illustrate the frequency location of the modes. The y axis is to bring attention at which amplitude setting these frequencies were found.

Table 2. Modal Response

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VI. FEA

1. Background

Finite Element Analysis was used to calculate stresses, displacements, and modes to compare with experimental data. Finite Element Analysis was chosen due to the complexity in the wing’s geometry. FEA software lets the user input the geometry, material, boundary conditions, and loads. This method could be used to find stresses in a component or assembly without over simplifying the geometry.

FEA uses the idea of breaking geometry into smaller elements to simplify a problem. The elements used can be of various shapes, the most commonly used are elements with three or four nodes. Physical representations of nodes and elements are shown in Figure 62. The nodes and elements are connected, making a mesh. This mesh is what connects the overall geometry and provides the way for element interaction. More nodes make a finer mesh, and more accurate results. A finer mesh also results in longer calculation time.

![Figure 62. Example Airfoil with Meshed Ribs and Skin](image)

Boundary conditions and material properties are normally set before a mesh is applied, however a correct mesh is what ultimately drives a good analysis. If all the nodes align, and a proper mesh is created, boundary conditions and an applied load can be used to find stresses and displacements at any node. Figure 63 shows an example wing similar to the manufactured wing, with a mesh, boundary conditions, and loads applied.
A solid wing was used as an example to illustrate what a typical FEA program displays. Figure 64 shows the model once the numerical analysis has been performed.

2. Methodology

One of the biggest challenges faced with running FEA for the manufactured wing geometry was getting a solid model to correctly input into FEA software. The problem was due to the way the wing components interacted. The ribs and spar were made as one component, while the wing skin was made as another. The ribs and spar meshed as a part, and the wing skin meshed as a separate part. The top surfaces of the rib and spar assembly would not touch perfectly due to software geometry.

The biggest challenge associated with the finite element analysis was the construction of the model to be analyzed. We have had experience with modeling wings, however, the addition of wing sweep and wing dihedral significantly increased the complexity of the wing. The final wing model with mesh is shown in Figure 65 below.
The first model was designed in Pro-E, with the assumption that we would be able to import the model into a finite element analysis program. After many unsuccessful attempts we soon learned that we were trying to send volumes using an .IGS file. The problem was that .IGS or .iges files can be used to transfer cad models using shells, in order to send a volume we would need to use a “.STEP” file. Next we looked at the height to width ratio of the cross sectional areas, which was less than the 5:1 ratio we used as the cut off for thick shell methods. When we were taught FEA we were told repeatedly that while it is possible to import models from other cad programs there can be translation errors between the programs. From our experiences building models the more complex the model is, the more likely translation errors will occur. We had not had enough experience building importable models so we decided to build the model in Geostar 2.0 so that we knew there would be no error.

Once the decision to build the model in Geostar 2.0, it was then important to control the accuracy of the datum points. Simply calculating the points defining the airfoil is a poor use of time and ultimately results in a poor model due to accuracy errors in the calculations defining the curves of the wing. After some thought a point file for the
NACA 2415 airfoil was imported into the Pro-Engineer software. The more points used, the better the curve will match the desired airfoil curvature. The curvature and the points are shown in Figure 66 below.

Once the data points were imported a curve was fit to the defining airfoil points. Our physical model was cut off at 80% chord to model the flat surface where the flap and aileron systems would be mounted. In order to model this, the point files are imported with a chord length of 1. The actual chord length of $7\frac{3}{8}$ inches was then multiplied by 1.2 and then cut off at 80% to give us a match to our actual physical test piece. Once the 2-D airfoil was created we then used the ratio of wing root to wing chord to scale the original airfoil down to the match the end cap of our physical test piece. This accomplished several things. First we saved a considerable amount of time importing and defining the wing tip and wing root 2-D airfoils. Second this approach allowed us to create a skin spanning the wing by blending the two wing profiles. Once the skin was defined we could then use the wing skin to define the outline of the ribs as well as the spar.

The Pro-E model was not used directly for the FEA analysis. Once the wing model was complete in Pro-E, the wing was then sliced at every surface location and points were placed along the cut wing surface at every 10% chord.
location with one additional point at 95% chord which is very close to the leading edge of the wing. The lead edge is the location of greatest curvature for our airfoil. The extra data point on the lead edge was used to allow better matching of the curvature of the wing. The Cartesian coordinate system X, Y, Z point locations were stored using Excel.

Using the point locations developed in Pro-E we were able to then simply develop the model starting with the points for all of the surfaces of our wing as shown below in Figure 67.

![Figure 67. Airfoil Points](image)

After the point locations are defined the curves and surfaces were then created as shown below in Figure 68.
Once the model was built the element groups were created. Two different element groups were used. Group one was thick shelled elements and this was used for all of the skin surfaces including the surfaces on the ribs and spars surrounding the foam. The second element group was a volume defined by the two fiberglass shells surrounding the rib/spar foam. The composite materials were not in the Cosmos data base so material properties defined in the material testing phase was used for the glass, carbon/glass, and glass/foam respectively. The real constants were set to .025 inches thick and .035 inches thick, for the fiberglass and fiberglass/carbon parts respectively.

Meshing was done using an auto mesh. Prior wings we have experience were geometrically simple. Symmetric wings with evenly spaced ribs, and no sweep or dihedral, were meshed manually with the user controlling the locations and number of nodes used for analysis. Due to the complex geometry we used an auto mesh function. The automesh does not allow the user the same control over mesh density variations as a function of position as the parametric mesh. The distinct advantage in this case was the auto mesh tool allowed us to mesh the surfaces while still allowing us to merge the nodes and bond the surfaces together. Merging the nodes effectively takes two surfaces and combines the nodes the two surfaces share in common thus creating a structural interaction between the surfaces.

Figure 68. Points of Model
Prior to comparing our experimental data with our finite element analysis results we inspected the stress profile as a contour plot to look for obvious errors. The stress contour plot for one loading case is shown below in Figure 69.

The stresses are at a maximum furthest from the boundary condition near wing tip, which is also the location of the maximum displacement. The spar is in tension on the lower surface and in compression on the upper surface. This is what we expect if we think of the spar as a cantilever beam. Note the red “high stress” marks on the top of the spar near the wing tip. These stress concentrations are due to the placement of the forces at these nodes, and not a stress concentration due to geometry. What are interesting are the low stress concentrations near the rib locations. While in theory we model ribs as acting in torsion, there is no doubt the ribs are increasing the stiffness when deflected about the X axis. Since the loading is only in the normal direction we do not see large shear stresses forming in the skin, as the skin takes shear loads. The model looked consistent so we ran the other load cases. The strain contour plot looks similar to the stress contour plot and this is what we expected since stress and strain are related through Hook’s law. The strain contour plot is shown below in Figure 70.
A. Comparison

After the experimental data was collected and analyzed, and the FEA analysis was performed and analyzed, the data from each was compared to each other. The boundary conditions in the FEA modeled a wing under pure bending with a point load on the wing tip. This essentially caused the model to only stress on the spar. Since only the spar saw stresses, the wing skin and ribs are shown has having no stress. Obviously this was not the case for the tested wing. The tested wing was subjected to torsion due to misplacement of the wingtip point loads and some deficiencies in the fixture. Because of the previously mentioned issues, only spar stress can be compared. A dynamic simulation was attempted, but due to inadequate processing power and memory, a solution was not able to be converged upon.

Experimental and FEA predicted stresses are compared in the following graph, Figure 71.
Stress in the “x” direction shows a good comparison, as does stress in the “y” direction. Despite the fact that the experimental wing was loaded under torsion, the majority of the load was still concentrated in the spar, leading to the favorable comparison with the FEA model. There was not favorable comparison with the shear stress. This was likely caused by the torsion the experimental wing was subjected to.

B. Error Analysis and Future Work

The factor most responsible for the inconstancies between experimental and FEA results is a proper modeling of the boundary conditions. As shown in the experimental results section, the intended pure bending load was not achieved and torsion loads were present. In future work concerning this project, a more rigid fixture may be built to try to eliminate the torsion. A more precise method to apply the loads may also be developed. If a fixture can apply an equal, pure bending load to the wing, results can be compared to the FEA results. Another possible approach to improve results would be to extend the FEA model to a whole wing, and apply a wingtip point load in two slightly off of opposite locations. This would create torsion and more closely model the wing that was tested.
VIII. Conclusion

A small scale composite wing based on a design found on an experimental aircraft was designed, constructed, and tested dynamically and statically. The wing was constructed using similar materials and construction techniques as an experimental aircraft wing. The structure was based off of the design of an experimental wing. Material properties were found using test coupons made of the same material as the wing. The static test was intended to be a pure bending load test, but the applied loads resulted in the wing being in torsion. Strain gages were used to measure strains on the wing structure. The strains were converted to stresses to aid in analysis. The static test results showed that a torsion load was indeed present. The dynamic test resulted in a total of four dynamic modes being found. A finite element analysis model was made to compare experimental results to numerical analytical results. The comparison showed a good comparison in normal stresses on the spar. However, because the FEA model was subjected to the intended bending load, no stresses on the skin and rib were present and no comparison could be made to experimental results. Due to computational inadequacies, a dynamic simulation could not be solved. There is potential future work to improve results through fixture design, and refined FEA modeling.
### Appendix

1. Excel data

   All strains in $10^{-6}$  
   All Stresses in psi

#### Left Wing

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#### Alternate Wing, Right Skin

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**Alternate Wing, Rib**

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2.

**References**


Microsoft. (n.d.). Excel. Redmond, WA.
