Minimizing Mass of a Spacecraft Structure

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Commercialization of the International Space Station (ISS) created the opportunity for a wider variety of minisatellites to be launched to and deployed from the ISS. By utilizing ISS resupply vehicles, these spacecraft are launched to the ISS in a soft stowed configuration and undergo much lower vibration loads than in a typical launch configuration. The FeatherCraft spacecraft is designed to fully exploit this opportunity by offering a 100-kilogram spacecraft with 45 kilograms available for science payload use. This leaves only 5 kilograms for the required side panels and internal mounting surfaces that constitute the spacecraft structure. structure components external side panels and internal surfaces for component mounting. Most spacecraft structures represent approximately 20% of the total spacecraft mass, so the reduction of the structure to 5% of the total mass requires innovative mass-relieving techniques. To solve this problem, undergraduate aerospace engineering students at the University of Colorado at Boulder created FISH, the FeatherCraft Integrated Structural Housing, which achieves the required mass reduction and integrates with other spacecraft components. This unprecedented mass reduction is accomplished by utilizing composite materials, minimizing structure area and thickness, and finally using adhesives for attachments on nearly every interface. Critical components of the structure design were preliminarily verified through bending tests, Finite Element Analysis (FEA), and adhesive tests. A complete structural full-scale model will be tested under the expected vibrational loads and acceleration measurements will be taken to verify expected performance. The success of this novel design creates a new cost-effective approach to Low-Earth-orbiting missions.

Nomenclature

C_1	=	Strength-Limited Material Index
C_2	=	Stiffness-Limited Material Index
CF	=	Carbon Fiber
E	=	Young's Modulus
F	=	Force
FEA	=	Finite Element Analysis
FISH	=	FeatherCraft Integrated Structural Housing
FOS	=	Factor of Safety
FR	=	Functional Requirement
g	=	Gravitational Acceleration, 9.81 m/s
grms	=	Root Mean Squared Gravitational Acceleration Vibration Measurement
ISS	=	International Space Station
m	=	Mass
SST-US	=	Surrey Satellite Technology – United States division
STM	=	structural test model
ρ	=	Density
σ	=	Tensile Strength
σ_f	=	Failure Strength

[1] Undergraduate student team members

I. Introduction

The FeatherCraft project, sponsored by Surrey Satellite Technology-US (SST-US), involves the design, assembly and testing of a lightweight satellite structure to serve as a platform for payloads deploying from the International Space Station (ISS) into Low-Earth orbit. Surrey Satellite Technology developed initial concept designs for the FeatherCraft structure in collaboration with Aerojet Rocketdyne to deploy from the Kaber launch service (developed by NanoRacks LLC), which is the next step in the progression from the currently available NanoRacks CubeSat Deployer[1]. The Kaber deployer can accommodate much larger satellites than the current deployment system[2], greatly increasing the variety of missions which can be performed from the ISS. The benefits of a FeatherCraft design that meets the described requirements are ease of construction and low mass, which will improve the minisatellite market and provide a cost effective platform for commercial customers to reach Low-Earth orbit.

Satellites launched from the ISS are transported to the Station in a pressurized volume onboard commercial cargo transportation vehicles in a soft-stowed configuration and therefore experience far lower launch loadings than traditional launch configuration. This means the structure is wrapped in foam ranging in thickness from 1.5 - 5 cm and secured with ratchet straps inside of the re-supply vessel for launch. This foam is expected to attenuate the vibration loads produced by launch from 9.47 grms to 1.29 grms. This attenuated load offers a unique opportunity to design a lightweight structure.

The largest challenge facing the design is the mass requirement for the structure. With a mass limit of 5 kg, the structure represents only 5% of the total mass of the 100 kg satellite, this is significantly lower than the traditional standard where 20% of the mass of the spacecraft is the mass of the structure. A team from Los Alamos National Laboratory and Composite Optics Inc. was successful in achieving a 33% mass savings from traditional designs by using flat stock panels of graphite/epoxy composite material over a more traditional aluminum alloy when designing the FORTE satellite[3]. FeatherCraft has achieved a much more drastic reduction of about 75% from this standard by utilizing carbon fiber aluminum honeycomb sandwich panels for most structural panels and utilizing adhesives rather than fasteners to hold the structure together.

The analysis presented here shows the methods used to determine the shape and material of the structure along with the fastening method.

II. Design Objectives

The FeatherCraft structure design is driven by a requirement of mass reduction, but it must also meet a series of other requirements to be fully successful. These requirements are summarized in Table 1.

Table 1. Functional Requirements of FeatherCraft Structure

FR 1	The FeatherCraft structure design shall have a mass of less than 5 kg.
FR 2	The FeatherCraft structure design shall reduce manufacturing time to 9 months and material cost to $20,000, which are 50% reductions from SST-US's typical spacecraft estimates.
FR 3	FeatherCraft Structure shall be designed to deploy from Kaber Deployment System on the ISS.
FR 4	FeatherCraft structure design shall interface with SST-US-provided spacecraft components and mission design.

FR 1 requires the most innovative improvement in this product, because a lower structural mass allows for greater mass available for avionics subsystems and science payloads. Because the spacecraft is designed to utilize the Kaber Deployment system, FR 3 includes surviving a launch to the ISS and fitting the entire spacecraft with all external components within the Kaber Deployment volume of 0.762 m x 0.762 m x 0.483 m (30 in x 30 in x 19 in).

The components of FeatherCraft that the structure must support are detailed in FR 4, and include 95 kg of masses on the external and internal sides of the spacecraft. Specifically, the sides of the structure are required to serve a set of pre-defined functions, and integrate with a propulsion unit (panel 4 in Figure 1 below). Three panels are required to have solar panel coverings (panels 1, 2, and 3), one is required to serve as a radiator for 100W of generated heat (panel 5), and the remaining face (side 6) is required to have an aperture of at least 0.305 m x 0.305 m (12 in x12 in) out of which payload instruments can operate. Finally, the internal volume of the structure is to be split into two volumes – an avionics bay (underneath panel 1) and a payload bay (above side 6). These component bays are required to be adaptable for different layouts and be able to mount currently unknown components.

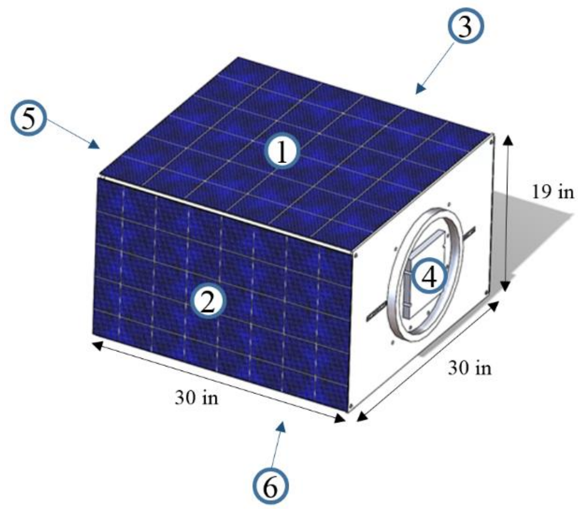


Figure 1. Overview of Structure Function and Baseline Geometry

Creating a structure design with all these components constitutes Level 1 design success, or the lowest success criteria for the project. For the higher Level 2 success, the design must also provide reduced manufacturing time and material cost, specifically a 50% reduction from SST-US's typical values.

A manufactured structural test model (STM) was created to the same specifications as the structure design, but with non-space-rated components to save time and cost. This STM has demonstrated the required weight reduction and support of SST-US's components, and will meet Level 1 success when it performs a full vibration test to a NASA-specified random vibration profile for spacecraft in the soft-stowed configuration. Level 2 success will be met with the survival of the structure through random vibration in all three axes, and Level 3 success will be indicated by demonstrated modes in the structure shifting in frequency by less than 10%. A change in modal properties below this level indicates that the structure has sufficiently maintained its structural integrity and has not sustained an invisible failure. Through this vibration test, the structure will demonstrate feasibility for ISS deployment and reveal possible improvement for future iterations.

The most critical project element from these requirements is the mass reduction of the structure while maintaining structural integrity through vibration. This balance initially presented a challenge in design, but the effort put into design simplicity became relevant in the construction of the STM, which was completed successful with cost and time much lower than required in FR 2. The steps taken to relieve mass in all components of the structure are detailed in the following sections.

III. Design Methodology

A. Composite Material

Several materials were considered for the structure: lightweight metal alloys (Mg, AlLi, etc), composites (carbon fiber), and combinations of the two. The main objective of material selection was minimizing mass. The main loads on the structure occur on the middle panel, where the structure supports 95 kg of payload and avionics components through the accelerations during launch. In this environment, loads act on the middle panel and are transmitted through the structure inside the transport vehicle. To simplify preliminary material selection, the structure was modeled as a single panel undergoing bending.

Material science demonstrates that material indices (a material property ratio that identifies which materials have properties that will result in a certain objective) provide great insight into potential materials. The material index for this strength limited case is shown as C_1 in Equation 1 and the stiffness-limited material index is shown as C_2 in Equation 2. This combination of strength and stiffness yields the desired material for the middle panel which is under the most extreme loads.

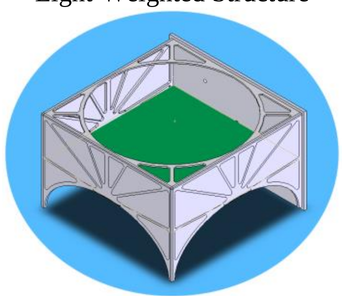
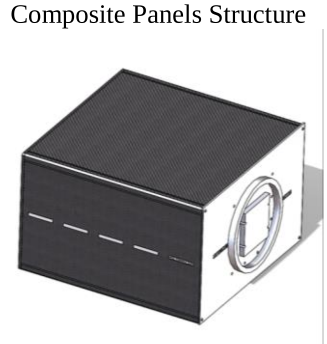
$$C_1 = \frac{\sqrt{\sigma_f}}{\rho} \qquad C_2 = \frac{E^{\frac{1}{3}}}{\rho} \qquad (1)\ \&\ (2)$$

Using a strength-density chart (see Figure 0.1[4] in the Appendix) and these two values, several materials were identified as potential candidates for the middle panel: Carbon fiber, Aluminum, Beryllium, and Magnesium.

Beryllium was eliminated because of its toxicity and cost, leaving aluminum, magnesium and carbon fiber. Since, material data charts do not account for combinations such as sandwich panels, this unique material was considered next. Composite sandwich panels have become prominent in the aerospace industry for their attractive mass and stiffness properties. These sandwich materials are generally composed of stiff face sheets on either side of a thicker, lightweight material, imitating the properties of I-beams, in which the mass is distributed where it is most needed to provide stiffness. Due to the low mass and high stiffness properties, carbon fiber sandwich panels were added to the list of potential materials. Common cores for carbon fiber sandwich panels include Nomex (aramid) and aluminum. Aluminum was chosen for the honeycomb middle layer due to its availability and strength. Because of the high strength and stiffness of this composite for the middle panel, it was chosen for use in the rest of the structure to simplify manufacturing and streamline assembly. The shape of the structure was developed next.

Several potential shapes were considered for the full structure: a concept for a light-weight aluminum or magnesium structure, a skeletal frame of aluminum or carbon panels and beams, and a plain design of sandwich panels. First principles analysis proved each of these designs were strong enough to survive launch. Table 2 compares each of these designs created with only aluminum and only carbon fiber sandwich panels for a mass comparison.

Table 2. Comparison of Structural Mass Using Aluminum vs. Carbon Fiber

	Light-Weighted Structure	Skeleton Structure	Composite Panels Structure
Design:			
Aluminum T6061-T6 Mass:	14 kg	5.7 kg	N/A
Carbon Fiber Mass:	7 kg	4 kg	4 kg

As shown above, the use of carbon fiber significantly decreased the mass of the first two structure options; this led to the decision to use primarily carbon fiber and aluminum sandwich panels despite the increase in complexity in manufacturing and modeling.

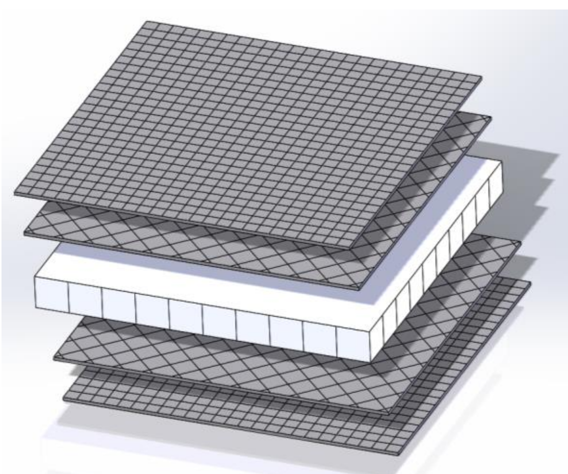


Figure 2. Middle Panel Layup. Carbon fiber is shown in grey and aluminum honeycomb in white.

Since most of the weight of the structure is applied to the middle panel, the core of this sandwich panel must be the stiffest component. To achieve the necessary stiffness, this panel was created of 1.27 cm (0.5 in) aluminum core layered between two plies of 0.3048 mm (0.012 in) prepreg carbon fiber, shown in Figure 2. To maximize strength in all directions, the weave of the carbon fiber was alternated between a 0º-90º and a 45º-45º directions.

In order to save cost and mass, the four external panels were constructed of 9.5 mm (0.375 in) thick core with one ply of 0.3048 mm (0.012 in) thick prepreg carbon fiber on each side.

The choice of carbon fiber-aluminum honeycomb composite structure limited the choices for attaching structure joints and mounting components, excluding welding or brazing. Two suitable fastening techniques were analyzed: traditional (threaded) fastening and adhesive fastening. Through trade studies discussed in the later sections, adhesive fastening was chosen as the primary adhesion method and was incorporated into finalizing the structure design.

B. Structure Shape and Weight Relief

A detailed structural design was developed after determining the material and fastening technique. However, the structural joints between panels (the points where the edge of a sandwich panel meets the side of another) were still a point of concern. The edge of a sandwich panel provides very little adhesive area as only the face sheets act as a glue interface. To improve this joint, a portion of the Skeleton Structure was added to the design. Carbon fiber square tubes with 2.54 cm x 2.54 cm dimensions were used to add a stiff frame that connects the radiator and propulsion panels. The middle panel of the structure is supported through a tab interface where it intrudes into slots in the four sides of the structure (sides 2-5). Figure 3 illustrates how these tubes form the skeleton of the structure, how the tabs interface with the side panels, and indicates adhesive locations in red. The structural mass was estimated to be 5.6kg, violating the requirement of 5kg.

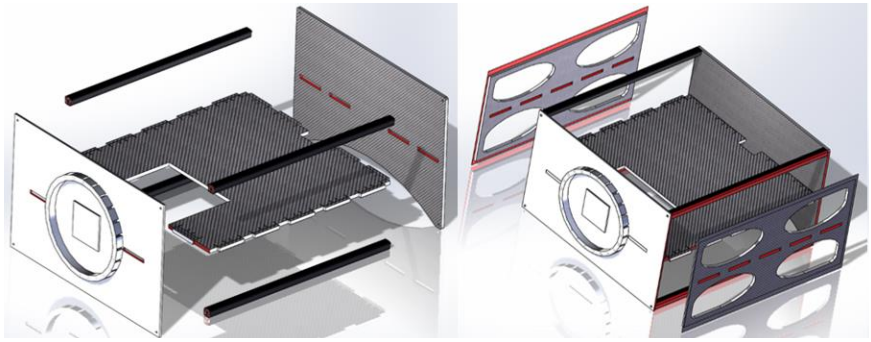


Figure 3. Tab and Column Interfaces of FeatherCraft Structure

To save more mass, weight reliefs were incorporated into all of the structural panels not directly supporting internal components. Figure 4 shows the progression of the weight relief design from an initial first principles reduction (left) to the final design, which was reached after integrating the relief design into a detailed Finite Element Analysis (FEA) model.

The first relief was added to the radiator panel. Area was removed from the bottom center of the panel (in the locations of the smallest loads) until the remaining area was equal to the required emissive area, 0.303 m^2. Next, triangular reliefs were added to the side and top panels of the structure as an initial design. Area was removed until a 5% mass margin was achieved (a total estimated structural mass of 4.75 kg).

Figure 4. Initial weight relief (left) and final relief (right).

After the relief was incorporated into the FEA model, it was found that although the triangular relief was mass efficient, the sharp corners introduced high stress concentrations on the side panels. In addition, because the load was supported by the frame, loads from the middle panel could be better transmitted through the center of the side panel rather than diagonally to the corners.

In the launch environment, the experienced loads are purely translational; the structure undergoes no torsion and the top panel does not support large loads. This means that the top panel experiences low stresses despite the triangular relief. An iterative approach was then applied to the side panel relief to optimize mass savings while satisfying three constraints: preserving a load path through the center of the panel, providing large radii near that load path, and maintaining an aesthetic design.

Figure 5 on the next page shows a visual result from the FEA simulations. A quasi-static acceleration was simulated to mimic the launch vibration environment, and several stresses were analyzed: equivalent stress in the face sheets, shear in the core, and delamination stresses between the face sheets and the core. The geometry of the relief proved to have a large influence on the stresses seen in the face sheets and the stress concentration factor from this relationship was used to size the radii of the relief. All analyzed regions were designed to have a 1.9 factor of safety, as recommended for composite structures in NASA GEVS.

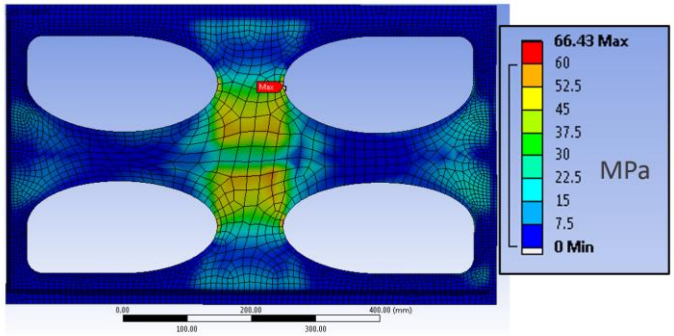


Figure 5. Stress concentrations in side panel face sheets. *Generated in the student edition of ANSYS 15*

The structure's strength will be verified for the launch environment with a 3-axis random vibration test. Before and after each test axis, a modal survey will be conducted on the structure. Changes in the detected modal frequencies will be used to identify non-visible structural failures. These modal surveys will also be used as a validation for the FEA model. The detected modal frequencies will be compared with a table of predicted natural modes and used to correlate the structural test model to the numerical model. This validation technique will provide accurate data to compare to the numerical model, because the locations and frequencies of natural modes are tied to the mass stiffness distribution in the structure. Thus if the modal surveys match the predicted modes, the numerical model has accurately captured the most important aspects of the structure.

C. Adhesives

One of SST-US's initial requirements on the structure design was a low part count (around 12 parts) and implied simplicity in the design. This requirement was later removed and changed to the reduction in manufacturing time, as this was the underlying goal of the part reduction, but the desire for low part count remains integrated in the design. After eliminating welding and brazing because of the chosen composite material, three options were investigated: adhesives, "snap-together" pieces, and traditional fasteners.

Traditional fasteners are more difficult to implement in a laminate than in a homogeneous panel. The honeycomb core material cannot be threaded, so every fastener requires a threaded insert to be placed in the panel to be joined. This adds to the necessary mass of every joint, specifically an estimated 20 g for each fastener. Traditional fasteners also add to the total part count, adding complexity and increasing assembly time. However, fasteners are widely available, inexpensive, well documented, and relatively easy to analyze.

Adhesive fastening is well suited for sandwich panels because joints can be directly connected to the surface of the panel, and loads can be distributed over large areas, avoiding stress concentrations. Adhesive joints are lightweight and do not add to the total part count. However, adhesive strength is less predictable than that of traditional fasteners, and is more difficult to analyze.

The concept of "snap-together" pieces, much like a puzzle, was desirable to the customer because of the simplicity in assembly, but proved difficult in design. The tab inserts in the final design show an element of fitting structure components together without additional fastening, but a complete snap-together structure was unrealistic for all aspects of the structure.

A trade study between adhesives and traditional fasteners was performed, and adhesives were chosen as the preferred method. Despite the complexity of joint analysis, the adhesive's low mass, simplicity, and diverse applicability made it the favorable choice.

The requirements for the chosen adhesive are to support the weight of all SST-US's component masses, hold the structure together under vibrational loads, and have space heritage. Because of these requirements, the high-strength, aerospace grade epoxy ScotchWeld EC2216 was chosen. It has desirable strength data from a broad temperature range (-253°C to 82 °C) and cures to form a flexible bond, which is critical for a bond under vibration loadings to avoid fracture.

For the structural interfaces such as the tab inserts, the adhesive is not undergoing direct loads and is mainly counteracting any imperfections in the machining of the two objects and holding two fitting components together. However, the epoxy also adheres mass analogs simulating SST-US's avionics and payload to the mid-panel, the most critical of these being the communications box mass. This box, made of solid aluminum for the vibration test, has a surface area of 0.0257 m^2 and a mass of 10 kg. For vibrational loading, the maximum force observed on the components is a quasi-static load shown in Equation 3, assuming a combination of the static load and the maximum vibrational load multiplied by four to capture all instantaneous loads.

$$F = mg(1 + 4 * 1.29) \tag{3}$$

For the known mass, the resulting maximum force is 604.3 N. Dividing the force by an effective area of 25% yields a pressure on the epoxy of 94.2 kPa. Because this epoxy is flexible, it is stronger in shear than in tension, so the tensile strength was the focus of later analysis. All components would experience similar loads in pull and shear because vibration loads are applied in all three axes. This method of margin calculation used in Section IV.C. is shown in Equation 4.

$$margin = \frac{\sigma - FOS * \sigma_{required}}{\sigma} \tag{4}$$

Despite the large computed margin on the epoxy joints, adhesive performance is not as repeatable nor as reliable as the bond with a fastener. Thus, a detailed procedure was created, involving sanding both surfaces to be bonded, cleaning both with acetone and alcohol, and using a bond line controller for an optimal 0.13 mm bond thickness. The results of components tests will be discussed in the next section.

IV. Design Results

A. Composites Performance

To verify initial assumptions on composite sheet material properties, several types of material testing were performed on carbon fiber sheets as well as on honeycomb structural panels. The overview of conducted tests is presented in Table 3.

Table 3. Summary of Structural tests on Carbon Fiber

Specimen:	**Tensile Material Properties**	**Bending Material properties**
Carbon-fiber woven fabric in epoxy('Dragon plate' outer layer), 0^{o}-90^{o}	Elongation/Applied force	N/A
'Dragon plate', two layers of fabric with foam core (6.35 mm total height)	N/A	Deflection/Applied force
Flight configuration of 9.53 mm Al honeycomb and 2 layers of 0^{o}-90^{o} fabric (side panels)	Failure point (N)	Deflection/Applied force
Flight configuration of 12.7 mm Al honeycomb: 2 layers of 0^{o}-90^{o} fabric and 2 layers of 45^{o}-45^{o} fabric (mid panel)	Failure point (N)	Deflection/Applied force

The performance of the test samples suggested several conclusions about the design. First of all, the tensile properties of the carbon fiber closely correlated to the accepted values given by the manufacturer. The repeatable carbon-fiber to aluminum honeycomb failure indicated the most likely mode of failure for the middle panel. Finally, the bending properties of the mid-panel were analyzed and incorporated into the FEA model to predict vibrational stresses more accurately.

The most significant of these tests were the bending tests, because these were performed on both professionally manufactured composite panels and the in-house manufactured composite panels. The bending test was performed in a calibrated INSTRON® machine by applying relatively small force (less than 300 N) and recording deflection. The test configuration is shown in Figure 6.

The bending test data for the professional panels and in-house manufactured test panels is shown in Figure 7.

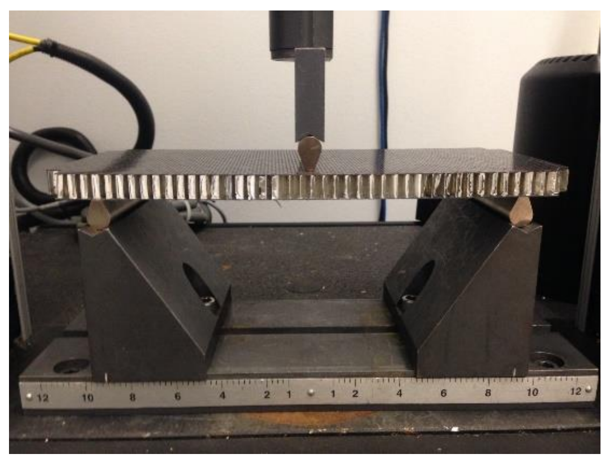


Figure 6. Bending Test Configuration in INSTRON® Machine

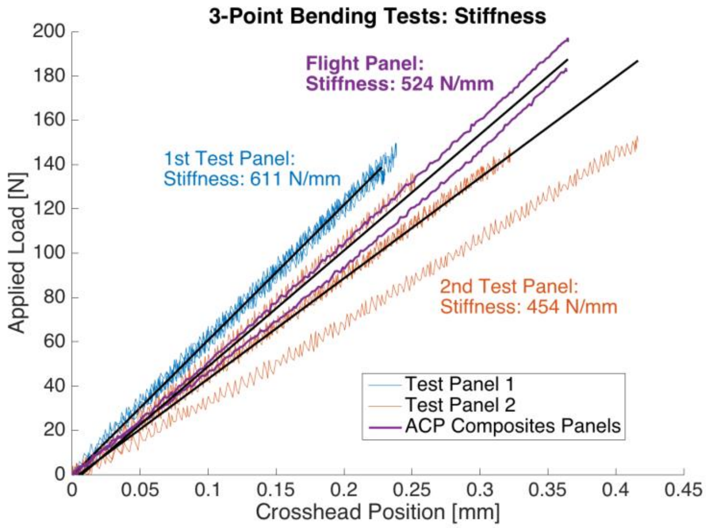


Figure 7. Composite Panel Bending Test Results

The results of the test show consistent stiffness on both sides of the 1st test panel, and two stiffness results relatively close to the professionally-manufactured panels. However, the primary motivation of the bending test was to feed the test data back into the FEA model and improve model predictions for full structure tests.

All the performed structural tests yielded results instilling confidence in the structure design. The analysis of failure mechanisms in vibration will provide definitive information of the relevance of these structural tests.

B. Panel Mass Savings and Performance

The final mass-relieved design removed enough area on the side and top panels to save 0.73 kg of structural mass or 23% of the original mass, and put the final total panel mass at just over 2.5 kg. This left the remaining 2.5 kg for the rest of the structure: carbon fiber tubes for the frame, adhesives, and various inserts and brackets in physical structural joints. To avoid exposed core in areas of high stress, an edge closeout material was added (ScotchWeld 3550) to the mass estimates.

The carbon fiber tubes added a total of 0.7 kg, inserts and brackets added 0.55 kg. The most significant additional mass besides the panels came from adhesives, structural epoxy and edge closeout material. Initial estimates put the epoxy at 0.22 kg and the closeout at 0.87 kg for a total of 1.09 kg.

After final estimates, the total structural mass was 4.84 kg, leaving a 3% mass margin. After constructing the STM, the estimated mass of the structure was 4.01 kg. The difference in these two values is due primarily to conservative estimates of adhesive mass, both in the structural epoxy and edge closeout material. The final STM is shown in Figure 8.

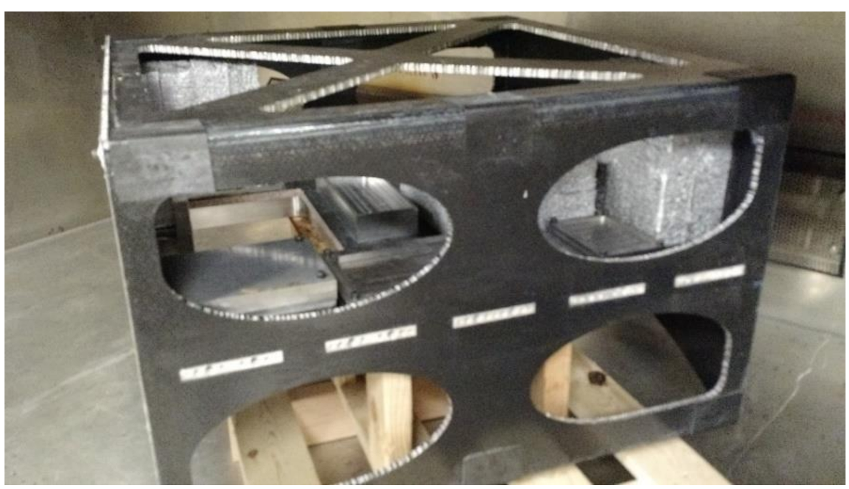


Figure 8. STM of FeatherCraft Structure with Mass Analogs

C. Adhesive Performance

1. Tension Testing

To determine the performance of ScotchWeld EC2216 in tension, test pieces were adhered and configured as in Figure 9, where the red line represents the adhesive bond line of a 0.13 mm. This thickness was controlled using wires of the desired 0.13 mm thickness and applying pressure to the bond during its twelve-hour cure time to handling strength. The adhesive was then baked for two hours to achieve full strength.

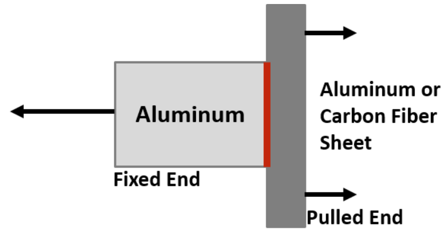


Figure 9. Adhesive Tension Testing Configuration

All provided surface preparation instructions and resulting strength for ScotchWeld EC2216 is given for bonds between aluminum pieces. In order to confirm the predicted strength of the adhesive, aluminum blocks were bonded together and pulled apart using an INSTRON® machine. These results were compared to the bond strength predicted by data sheets and the bond strength between aluminum and carbon fiber, which represents the interface between the mid-panel and avionics or payload components. The results are shown below in Figure 10.

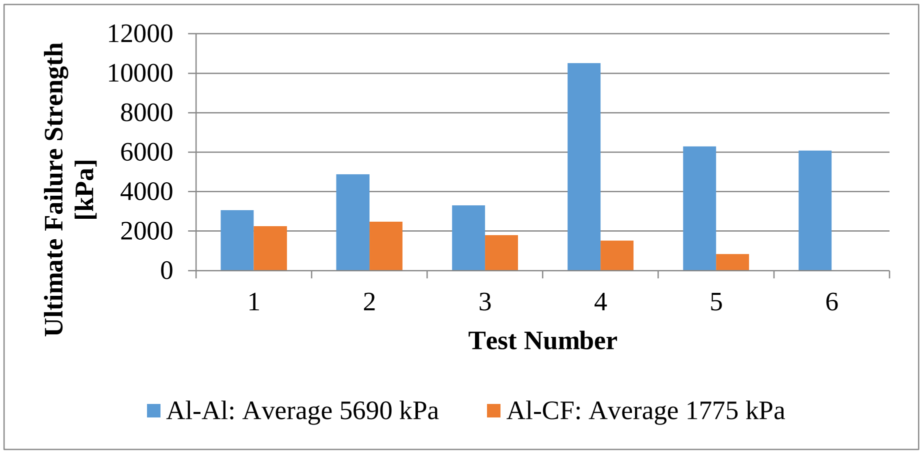


Figure 10. Tension Testing Results

The demonstrated aluminum bond strength was 50% of that predicted by the ScotchWeld EC2216 data sheet. Also, as shown, the bond between two aluminum pieces is significantly stronger than that of a bond between aluminum and carbon fiber. In order to ensure a conservative design, the lowest experimental failure strength of 980 kPa was used to demonstrate the feasibility of using ScotchWeld EC2216 to adhere the avionics components to the middle panel. Using the method explained in Section III.C, the margin on the largest avionics component was 82%. This testing also revealed how challenging it is to achieve a 100% bond area. Based on the visual findings shown in Figure 11, an effective area of 25% was used for all adhesive calculations which translates to an additional factor of safety of 4.

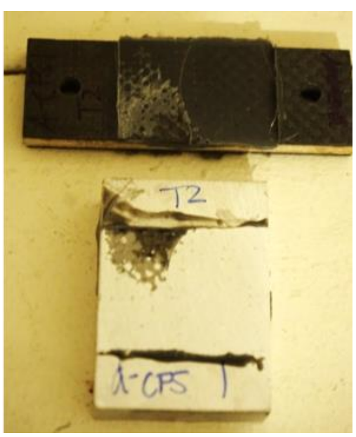


Figure 11. Aluminum to Carbon Fiber Tension Effective Area

Adhesive was also included in tensile tests of the composite panels, where two aluminum test pieces were adhered to each side of a manufactured sandwich panel. This testing showed the adhesive bond line between the aluminum and the carbon fiber face sheet consistently out-performed the bond between the carbon fiber face sheet and the aluminum honeycomb core in the manufactured composite material. Therefore, adhesive failure was deemed a lower risk than the delamination of the middle panel.

2. Shear Testing

Shear testing of ScotchWeld EC2216 utilized the set-up shown below in Figure 12, where the bond line is represented in red. The same surface preparation and bond line control methods described in the tension testing section were used for the shear adhesive tests.

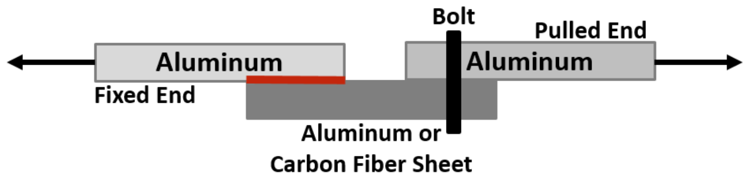


Figure 12. Adhesive Shear Testing Configuration

The same procedure from the previous experiment was followed to compare the results of shear testing between aluminum-to-aluminum bonds and aluminum to carbon fiber bonds. These results are shown in Figure 13.

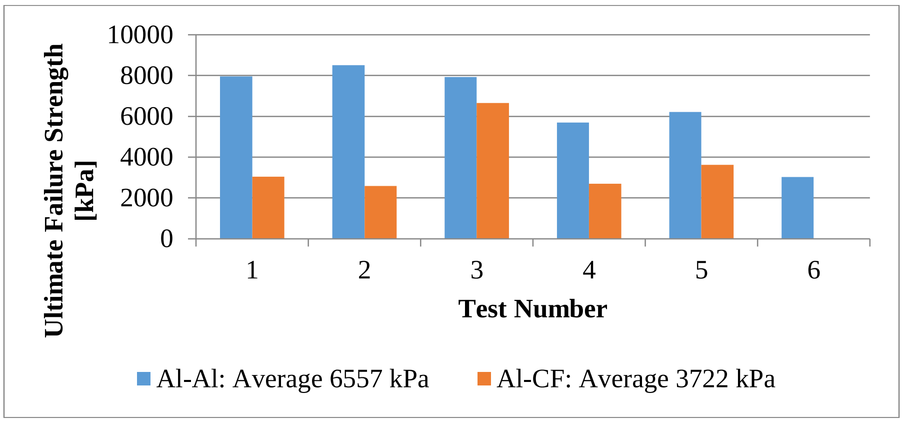


Figure 13. Shear Testing Results

This testing once again revealed that the bond between two aluminum pieces is stronger than that of aluminum to carbon fiber. This testing also confirmed that ScotchWeld EC2216 has higher strength in shear than it does in tension. The same principles from Section III.C. were used again to obtain a margin of 79%.

While all adhesive tests confirmed that while the carbon fiber bond with aluminum did not perform as well as the aluminum-aluminum bond, it still provided a large margin for supporting all the mass analogs on the structure for vibration testing. The combined strength and simplicity of application of adhesives allowed them to be utilized at every interface for reinforcement, and contribute to a customizable design.

V. Conclusion

Although spacecraft are subject to stringent requirements to survive both the launch environment and the space environment, there is still room for innovation and incorporation of new materials and designs. Composite materials are becoming increasing relevant in the aerospace industry, especially in aircraft, and they yield great benefits for spacecraft producers looking to reduce the cost of a flight to low-earth-orbit. The use of composites combined with a creative mass-relieving panel design greatly reduced the possible mass of the FeatherCraft structure from a traditional aluminum-box design. The use of adhesives is especially crucial not only for mass reduction but also maintaining the required simplicity and modularity of the design. Upon successful vibration testing, the structure design will be fabricated in a second iteration, improving on any observed weaknesses of the structure as well as using all professionally-manufactured parts. The structure, integrated with SST-US' avionics components, can then be offered to science payloads and provide a unique and cost-effective opportunity for Earth-related missions.

Appendix

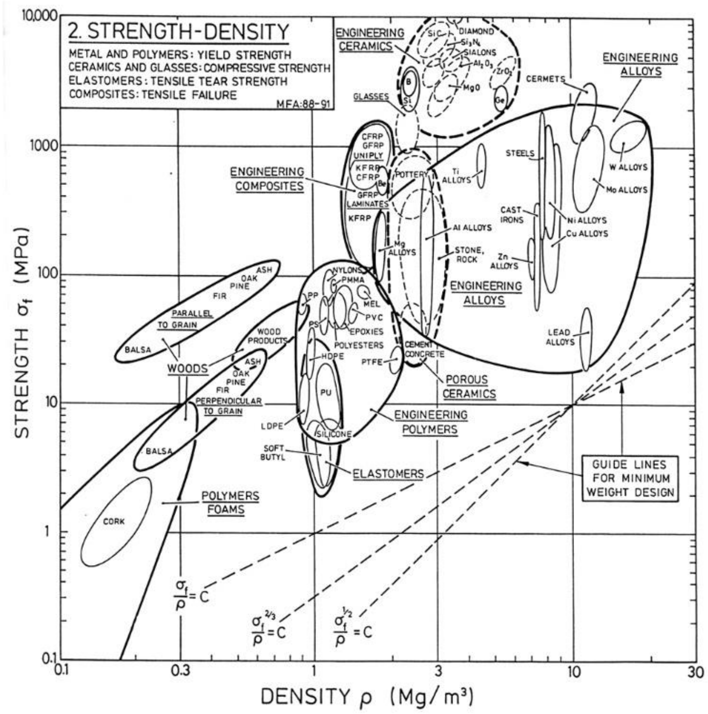


Figure 0.1 Material data chart[4]

References

[1] "NanoRacks Successfully Deploys After On-Orbit Repairs." *NanoRacks*. N.p., 27 Feb. 2015. Web. 30 Aug. 2015.

[2] Foust, Jeff. "Companies Offer Larger Satellite System For ISS Deployment - SpaceNews.com." *SpaceNews.com*. N.p., 08 Apr. 2015. Web. 30 Aug. 2015.

[3] Grastataro, C.I. et al. "Development of a Composite Satellite Structure for FORTE." United States: N. p., 1995. Web. 2 Sep 2015.

[4] Ashby, Michael F. "Materials Selection in Mechanical Design – Strength vs. Density." *Lehigh University – Department of Mechanical Engineering and Mechanics*. 5 Oct 2010. Web. 30 Aug 2015.