DESIGN AND FABRICATION OF A LOW-COST COMPOSITE SPACEFRAME SPACECRAFT STRUCTURE

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Abstract

As part of an internally funded research program at Southwest Research Institute a low cost composite spaceframe structure, suitable for use in small satellite applications, has been designed and fabricated. For reasons of low mass, high strength, and dimensional stability composite material was investigated for use as the spacecraft load bearing structure. A design concept was developed that allowed for the use of composite material while minimizing the high cost and long lead times generally associated with fabrication. A large portion of the high cost of using composites is due to the investment in producing a mold. Typically, the high non-recurring engineering (NRE) cost of the mold would be offset by a large production run. When fabricating a single structure, as is common for spacecraft, most of the mold investment is lost. In order to overcome this difficulty, the design concept was an octagonal spaceframe consisting of eight individual wedges bonded together. This allows a single, small, simple mold for the individual wedges to be used repeatedly (at least eight times). In order to prove the concept Texas Composite, Inc. has fabricated a first prototype unit. A second unit has been ordered for a thorough structural and materials test program to be conducted by Marshall Space Flight Center.

I. Background

The continuing trend towards lighter weight spacecraft and pressure to increase the instrument mass fraction of satellites has made the use of composite material in spacecraft structures an attractive option. In addition to being lightweight, composite structures also offer dimensional stability with temperature. This stability is of particular value for optical instruments and for facilitating the use of GPS for attitude determination by separating receiving antennas by a known and constant distance.

The many benefits in using composite materials for spacecraft structures are, however, potentially offset by several factors. Among these drawbacks are the high fabrication costs, long lead time, and general unfamiliarity with design and fabrication processes. High fabrication costs are the result of a significant amount of non-recurring engineering (NRE) costs associated with the production of a mold and all necessary fixtures. Even after the mold has been produced, there are still some NRE costs in making adjustments to the fabrication process in order to account for the individual characteristics of each mold. These NRE costs are usually distributed over a large production run for composite parts in order to make it cost effective. However, this is not possible when producing only one or two items as is common with spacecraft.

Long lead times in receiving composite parts are also mainly attributed to the design and production of the mold and associated fixtures. The mold and fixture design and production is an intermediate step between the spacecraft design drawings and the finished part that is not present when fabricating metallic machined parts. This intermediate step is essentially another design and fabrication phase undertaken by the fabrication engineer and as such adds to the lead time in final part delivery.

Finally, there is a general unfamiliarity among many spacecraft design engineers, accustomed to metallic parts, with the fabrication of parts from composite material. This unfamiliarity can lead to designs that are not optimized to take advantage of composite material's unique properties. Also, the designs may inadvertently and unnecessarily complicate the fabrication process and therefore contribute more to the high cost and long lead times.

As part of an internal research program at SwRI, a low cost composite spacecraft structure has been designed and fabricated. This spacecraft structure needed to be low cost since the design was targeted towards new programs in the FASTSAT or USRA STEDI class of mission. The key to reducing the cost of this composite spacecraft structure was in reducing the NRE costs associated with the production and use of the mold. Also, the design was such that a single, simple mold could be used repeatedly thereby recovering or distributing the NRE costs over a larger production run. To validate the design concept and cost savings, a first prototype of the structure was actually fabricated by Texas Composite, Inc. A second prototype has been ordered from them by the Marshall Space Flight Center in order to undergo an extensive structural strength and materials test program.

II. Composite Structure Design

Design Goals and Requirements

The requirements placed upon a design dictate the physical configuration. Therefore, before the design concept is presented a listing and discussion of the imposed design goals and requirements are necessary. One goal of the SwRI internal research program was to design a composite spacecraft structure. A composite structure was baselined for reasons of (i) minimizing spacecraft structure mass, (ii) providing a dimensionally stable platform for GPS attitude determination, and (iii) gaining experience in an important emerging technology.

Another goal was to maintain fabrication costs of the composite structure at a level consistent with metallic fabrication techniques (honeycomb and machined parts). The achievement of this goal is difficult to measure since a composite design is inherently different than a metallic design. That is to say one would not take a composite part design and, without altering it, machine it from a metallic material. For example, the higher stiffness of the composite material allows for thinner cross sections. Another inherent difference is in methods of joining. Bolted joints in metallic structures can become bonded joints or even made seamless with the use of composite material.

To realistically constrain the design, it was assumed that the finished satellite would be a secondary payload on an expendable launch vehicle (ELV) similar to a Pegasus. This requirement limits the height of the structure so as to minimize consumption of ELV resources (i.e. available payload envelope) and thereby maximizing the number of potential primary payload candidates. Furthermore, since this design was meant to be applicable to

FASTSAT or USRA STEDI class missions, the diameter was limited to less than 30 inches. Provisions for adapting to the ELV were also to be considered and in this instance a 23 inch Marmon clamp arrangement was baselined.

The final goal of the design was versatility. Here, versatility is the ability to use the basic design concept in a wide variety of specific applications with minimum modifications. These modifications would mainly take the form of changes to the height and diameter, which do not alter the design concept. Design versatility can be used to reduce the cost of producing a spacecraft structure by minimizing the NRE associated with the development of completely new design concepts and in the production of detailed part drawings necessary for fabrication.

Design Concept

To reduce the cost of fabricating a composite structure a modular approach was used. This approach relies on the repetitive use of a small and simple "building block" that can be assembled to create a larger more complicated final spacecraft structure. The mold cost for the small and simple "building block" is significantly less than the cost for a mold needed to produce the entire spacecraft structure as a single piece. Also, refining the composite fabrication process on a simple mold is less costly in terms money, time, and material.

An octagonal spaceframe design was selected as the best way to implement the modular fabrication approach. A trapezoidal wedge, shown in Figure 1, is the elemental "building block" of the spacecraft. The part is 6 inches tall with inner and outer parallel flats being 10 inches and 3 inches, respectively. The material thickness is .040 inches which enables 8 plys of composite tape to be used in a quasi-isotropic pattern. There is some local thickening of the two vertical members on the 10 inch flat since they will bear the thrust loads during launch. The total mass for a single wedge has been measured to be 89 grams.

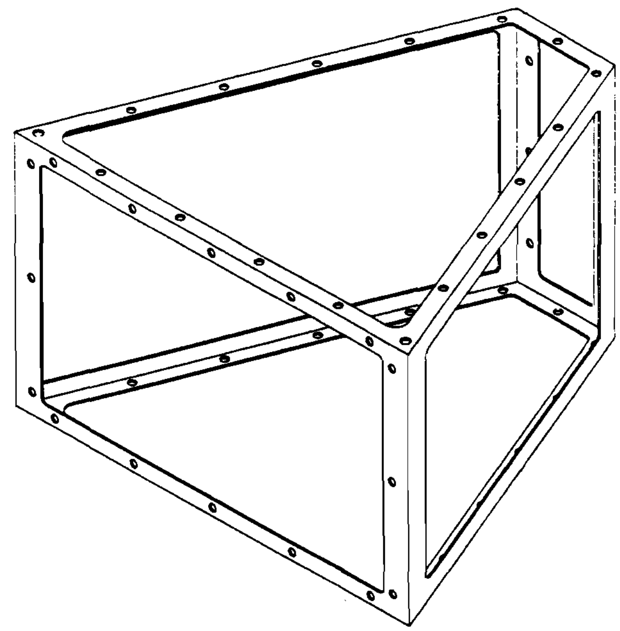

Figure 1. Elemental Component of Spacecraft Structure

The modular approach was not only instrumental in achieving the cost reduction design objective, but was also a key factor in creating a flexible design, which was another stated objective. Using parametric 3-D (solid) computer software, it is easy to modify the existing design to accommodate a wide range of mission/instrument requirements. For instance, it would be a relatively minor alteration to create a wedge that was 12 inches tall rather than the current 6 inches. Of course, this would incur new mold costs, but the reduced NRE in the design phase would be significant. Another straightforward modification would be to increase the overall diameter of the finished octagon which would increase the 10 inch flat dimension and the length of the overall wedge. It is true that there is a limit as to how much this current wedge design can be scaled. Geometric instabilities will emerge when lengths of truss members become too long which would require the addition of intermediate supports.

The final octagonal spacecraft structure is created by bonding together eight of the trapezoidal wedges. Figure 2 shows the fully assembled spacecraft structure with equipment tray inserted. The dimensions are 26 inches point-to-point, 24 inches across the flats with the outer flat length of 10 inches, and 6 inches high. The measured weight of the spaceframe including nutplates (for mounting panels) and adhesive is 750 grams.

The geometry of this particular implementation of the flexible modular design concept was driven by the objective of minimizing the consumed ELV payload envelope. A 23 inch Marmon clamp launch vehicle adapter ring can be mounted to the bottom face of the structure. In this way the vertical launch loads will be transmitted directly to the strengthened outside vertical members. Configuring this spacecraft for particular missions might mean solar cells on the eight flat panels operating as a spinner. The current implementation has a single solar cell panel on the top octagonal face and operates as a sun pointing spinner. Internal electronic

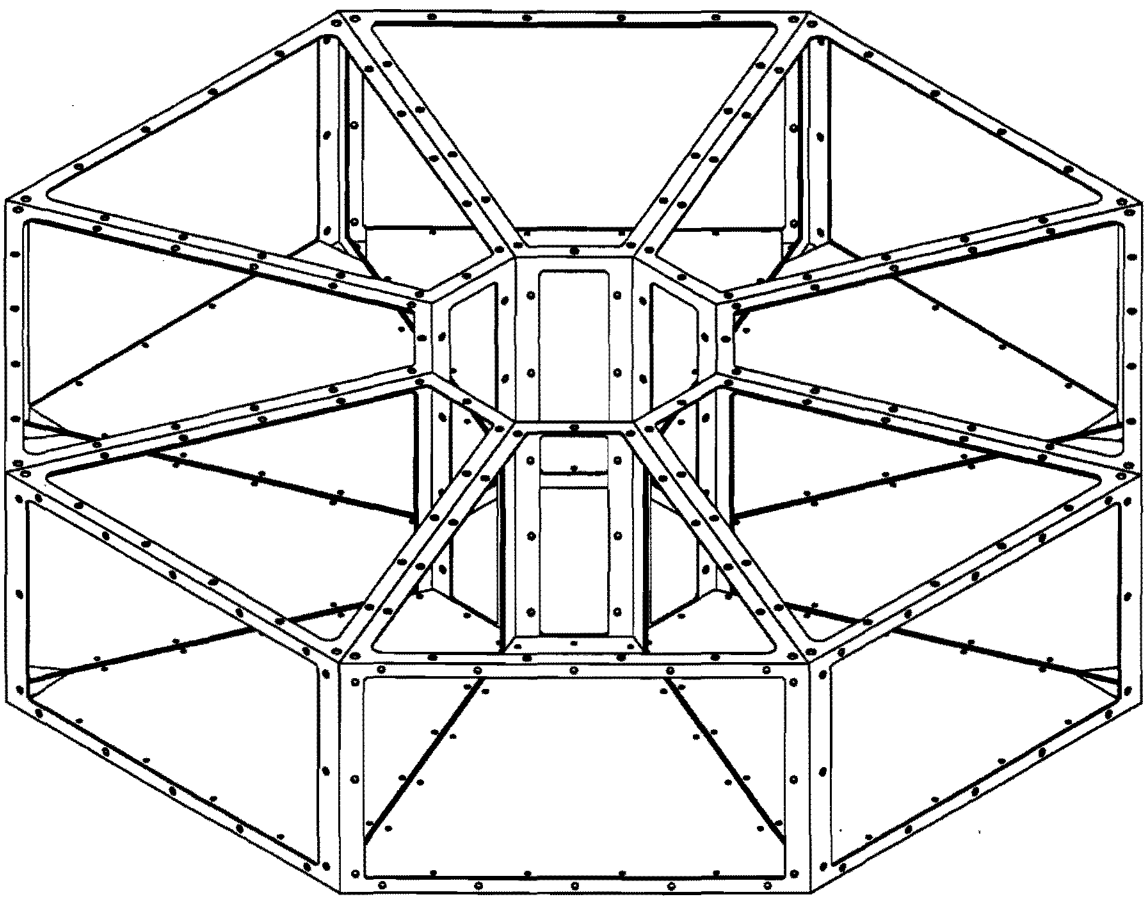

Figure 2. Assembled Structure

equipment/instruments can either be mounted to the eight side panels or, as is currently the case, to equipment trays inserted into the structure. This is another way in which the design provides flexibility for a wide range of mission and instrument requirements.

III. Composite Fabrication Process

The fabrication approach for the structure was based on two factors - material selection and construction techniques.

Material Selection

Based upon the goals of this program, i.e. light weight and low cost, the material selection criteria had to be balanced between raw material cost, manufacturability, performance, and weight. As such, after determining that the target weight of the completed satellite predicated the use of a carbon fiber composite structure, and the target cost of the satellite would not support the justification of an exotic high modulus carbon fiber, and since the design loads did not require the use of an excessively high modulus carbon fiber, a medium modulus (approximately 30 MSI) T-300 or AS-4 carbon fiber was selected.

Once the reinforcement fiber was selected, the concern in material selection was centered on the base material resin system selection and the adhesive system to be used in the assembly of the space frame. Selection was based upon the following criteria:

- Vacuum Stability - Must meet minimum requirements as specified by NASA.
- Temperature Range from -60° F. to 250° F. (-51° C to 121° C).

Due to the planned orbital life for the spacecraft, environmental concerns such as degradation due to atomic oxygen erosion or radiation was not considered, although considerable work has been done in the areas of protective coatings for AO rich environments and as such the materials selected would not preclude the use of this spacecraft in an orbit that would subject the structure to AO bombardment.

Based upon the above factors, RS-3M, a space qualified cyanate ester resin from YLA was chosen for its excellent vacuum properties, relatively low cost and ready availability of mechanical, thermal, and vacuum stability test data.

Also, based on the same factors, FM-300 a space qualified epoxy based film adhesive from Cytec was chosen.

Technique

The structure consists of two different components - the frame and the equipment tray. One complete SmallSat Spaceframe consists of eight frames and eight equipment trays. The eight frames are manufactured individually and subsequently assembled to form an octagonal space frame to which the eight equipment trays are mechanically fastened. To minimize the nonrecurring cost in the development of tooling and manufacturing processes, a hand lay-up autoclave cure technique was chosen for the manufacture of the structure components. This is a low risk method of manufacture commonly used in the fabrication of advanced composite structures to produce low cost structures with consistent mechanical properties. Both components utilize conventional hand lay-up autoclave cure techniques and are fabricated using the carbon fiber/cyanate ester materials described above.

Frame Manufacture

The hand lay-up technique utilized in the manufacture of the Frame involves the placement of eight plies of carbon fiber/cyanate ester preimpregnated tape laid up in a quasi-isotropic (+45/-45/0/90/90/0/-45/+45) orientation into a closed mold as shown in Figure 3. The 0° axis is defined as shown in Figure 4.

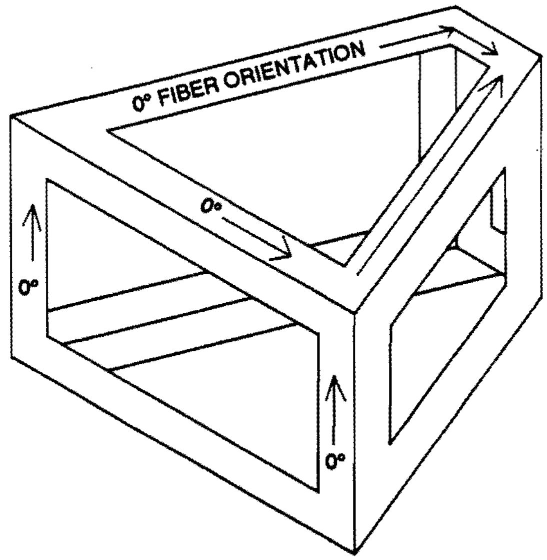

Figure 4. Frame Fiber Orientation

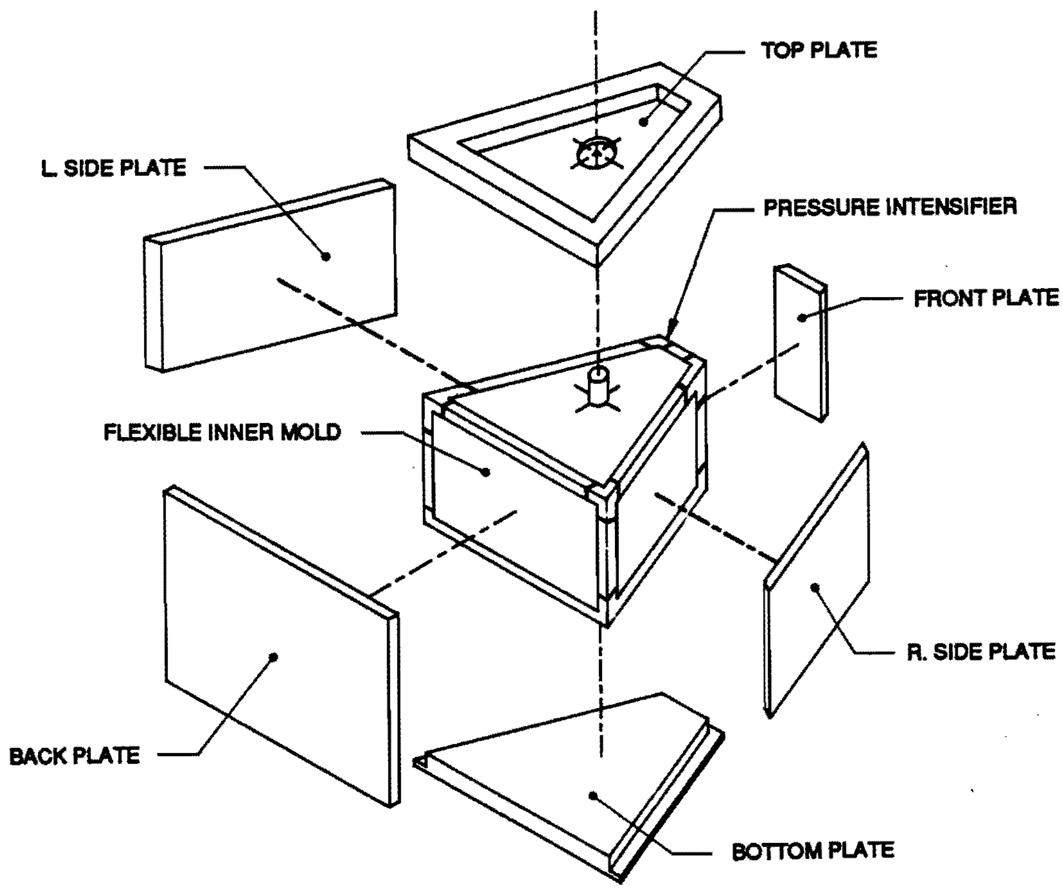

Figure 3. Frame Mold

To ensure that the frames would assemble properly, and to minimize secondary finishing, mold control on the outside frame surfaces was necessary; but to provide the necessary ply compaction during the cure cycle, a flexible, inflatable inner mold was required. This inner mandrel is vented to autoclave pressure during the cure cycle while maintaining vacuum on the exterior surfaces as shown in Figure 5.

The difficulty in using this method results from having to provide consistent, evenly distributed pressure on the interior of the surface of the laminate to ensure consistent laminate properties throughout the frame structure, and to minimize the variation between the various components utilized in the construction of the SmallSat Spaceframe Assembly. The laminate consistency between individual frames is important because variations in mechanical and/or thermal properties between frames would induce thermal stresses into the structure due to differences in thermal properties.

To improve laminate consistency TCI developed tailored pressure intensifier's which, when used in conjunction with the inflatable inner mold, greatly improves laminate consistency and enhances compaction in the corners of the structure.

After the material lay-up is complete, the mold assembly is encapsulated by a nylon vacuum bag, as shown in Figure 5, and the air is evacuated from the assembly. The part is then placed into an autoclave and processed under approximately 80 PSI of pressure at 350° F for approximately two hours. Tight controls for the processing parameters provide the necessary thermal and mechanical property consistency. After the part cure is complete, the frame is then removed from the mold and the inflatable tool extracted. The frame is then machined and drilled before final assembly.

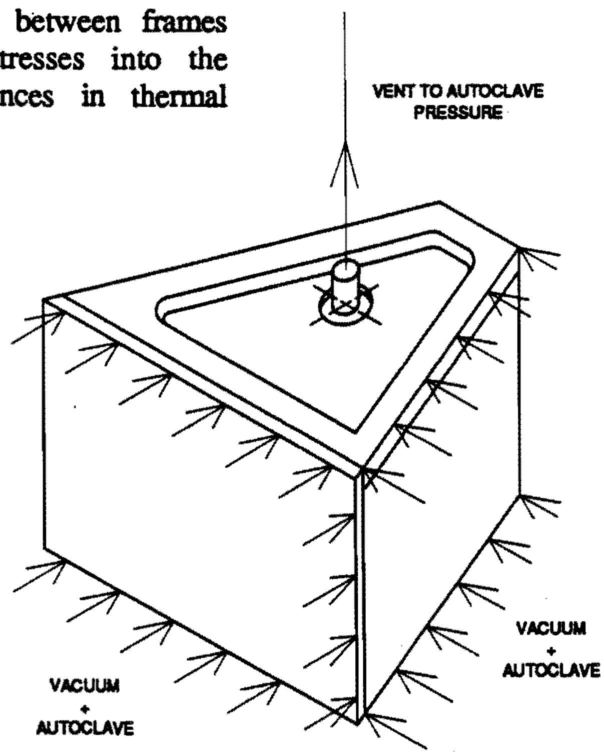

Figure 5. Frame Pressure Schematic

Equipment Tray Manufacture

The Equipment Tray is manufactured using the same materials as the Frame and is processed in a similar fashion. The mold for the Equipment Tray consists of three individual segments as shown in Figure 6 which provides mold control on all faces of the tray and is designed to provide a net shape molded part that will require minimal secondary finishing. The Equipment Tray is a quasi-isotropic lay-up with the 0° axis as shown on Figure 7. A stiffened edge consisting of unidirectional carbon fiber/cyanate ester tape orientated as shown on Figure 7 is incorporated into the laminate and co-cured using the same cure cycle as the Frame. After the cure cycle is complete, the Equipment Tray is removed from the mold, drilled and tapped (future applications may require the installation of thread inserts) and is now ready for installation into the Space Frame Assembly.

Space Frame Assembly

The space frame assembly is accomplished using two different operations - bonding and finishing.

The individual frames are bonded together in a fixture using a film type adhesive. Due to the extremely low coefficient of thermal expansion of the space frame, and the elevated cure temperature requirement for the film adhesive, the thermal properties of the assembly fixture would have to be similar to the thermal properties of the space frame. To accomplish this, a flat assembly plate using carbon fiber/epoxy face skins with a honey-comb core was constructed. To provide positive alignment of the individual seg-ments, bushings were installed in locations

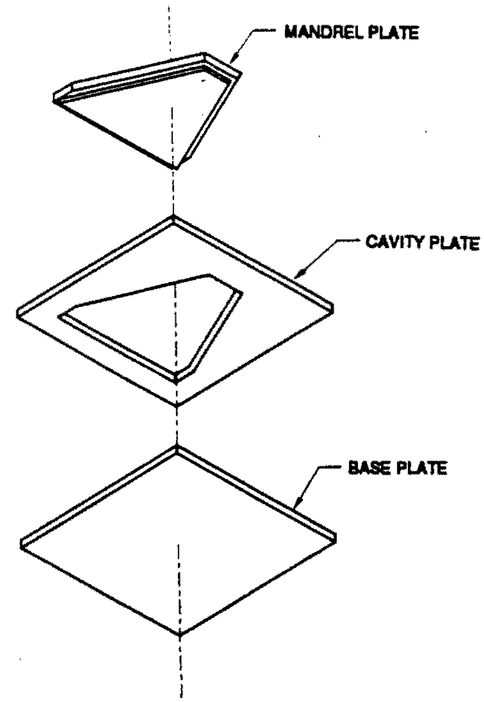

Figure 6. Equipment Tray Mold

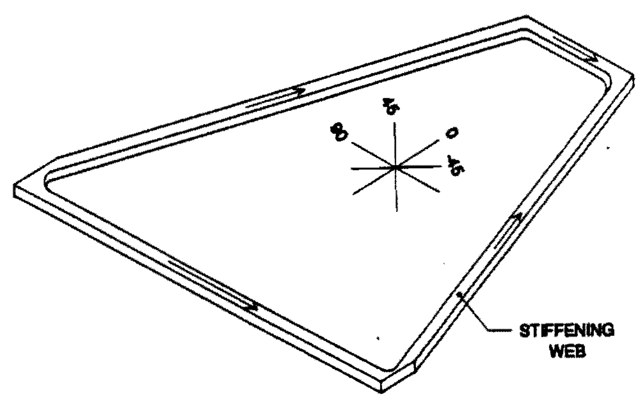

Figure 7. Equipment Tray Fiber Orientation

locations corresponding to the pilot hole locations that were drilled in the individual frames.

To bond the Space Frame Assembly together, each individual frame was masked with a Teflon based tape in the areas where adhesive was not required, mating surfaces are prepared with 180 grit sandpaper and cleaned with solvent. Film adhesive was then applied to the bond surfaces and the frame was then pinned to the assembly plate. After all of the individual frames were placed on the assembly plate, each individual frame was clamped to the adjoining frame in several locations to provide additional clamping force,. The assembly was then oven cured at 250° F. for two hours. After the cure cycle was completed, the part was removed from the assembly plate and excess adhesive was removed.

The finishing requirements for the space frame assembly consisted of drilling the various attachment holes in the space frame, bonding nutplates onto the exterior and interior periphery of the space frame and installing the eight Equipment Trays with mechanical fasteners.

IV. Current and Future Activity

Test Program

Marshall Space Flight Center (MSFC) has ordered a second prototype structure from Texas Composite, Inc. for the purpose of conducting a comprehensive structural strength and materials test program. The structural tests will include (i) a static load test to verify the shear capacity of the bonded joints, (ii) vibration tests in all axes to determine the dynamic characteristics, and (iii) non-destructive evaluation (NDE) of the fabricated composite structure.

The materials test will evaluate (i) the performance of the material under vacuum and expected temperature extremes, (ii) dissimilar material compatibility, and (iii) atomic oxygen and radiation exposure tolerance. The results of these tests will be used to further refine the design concept with the eventual goal of obtaining an environmentally (vibration and thermal vacuum) flight qualified structure.