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Title: DEVELOPMENT OF A NEW CLASS OF SATELLITE: A COMPOSITE LIGHTWEIGHT AFFORDABLE SPACECRAFT STRUCTURE

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DEVELOPMENT OF A NEW CLASS OF SATELLITE: A COMPOSITE LIGHTWEIGHT AFFORDABLE SPACECRAFT STRUCTURE

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INTRODUCTION

The Los Alamos National Laboratory (LANL) in partnership with Composite Optics Incorporated (COI) has succesfully designed and tested a Composite Lightweight Affordable Structure (CLASS). The use of advanced composites in space applications is well developed, but the application of an all-composite satellite structure has not been achieved until now. The development of this low-cost, lightweight, composite technology for use in small satellite structures, in this case, for the Fast On-Orbit Recording of Transient Events (FORTÉ) satellite mission will make an considerable contribution to the advanced spacecraft community.

A common practice for constructing small spacecraft structures is to use an all-aluminum spacecraft bus. Compared to a composite structure, this reduces the payload capacity significantly, however the cost of the aluminum structure has historically been lower than one that uses advanced composites. LANL mission requirements dictate the need for a long term solution that substantially increased the ratio of payload to structural mass while maintaining a low-risk low-cost approach. LANL intends to use the concept developed for FORTÉ on future missions requiring similar enhanced payload capacities.

SPACECRAFT DESIGN

Design Background

Staying close to known designs and well-known materials can go a long way in reducing risk and cost of the spacecraft. The original proposed design was an all-aluminum bolted structure that did not meet the weight target.

Several factors influenced the FORTÉ design process. The approach used by LANL was to do a sufficient amount of analysis to validate the concept and to thoroughly test the design through rigorous testing of the spacecraft. The schedule permitted two design iterations that allowed the Engineering Model (EM) to be thoroughly tested and subsequent changes to be fed back into the final flight hardware.

Spacecraft configuration

The FORTÉ spacecraft, shown in Fig. 1, consists of 3 octagonal decks connected by 3 structural trusses (cages). Twenty four solar array substrate (SAS) panels are attached to the cages with 10 bolted fasteners each. The 3 decks and cages are designated as lower, mid and upper decks and cages. All solar panels and decks are made of a sandwich of aluminum honeycomb bonded to quasi-isotropic graphite epoxy skins (T50/ERL1962). The lower and mid cages are identical and made of 8 rectangular frame subassemblies bonded together. Eight trapezoidal frame subassemblies form the upper cage. Diagonal members are added to 4 out of 8 frame subassemblies of the lower and mid cages to increase the bending and torsional stiffnesses of the spacecraft (see Fig. 1).

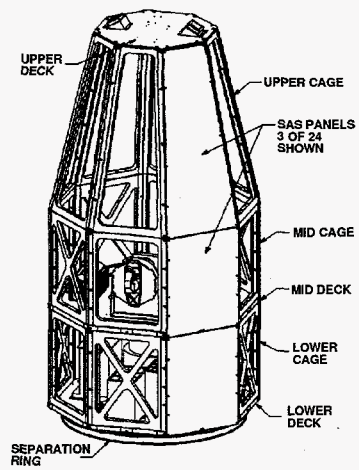

Fig. 1: Structural components of the FORTÉ spacecraft.

For all cages, graphite/epoxy corner clips are used to join the frame subassemblies together. Aluminum corner fittings are bonded into the cages and decks, these aluminum fittings receive the bolts that attach cages and decks together .

The total launch weight of the FORTÉ satellite is about 400 lbs. The spacecraft structural bus (including solar panels, spacecraft support, and mounting hardware) weights 140 lbs, which corresponds to 35% of the total weight.

SPACECRAFT MODEL VALIDATION

A finite element model of the FORTÉ structure was created using the finite element analysis software COSMOS/ M1.71.

One of the most severe environments for the structure will be during the short time duration when the Pegasus-XL launch vehicle is dropped from the carrier aircraft (drop transient). The spacecraft structure should be able to sustain lateral accelerations varying linearly from 5.7 g at the bottom deck to 9.2 g at the top deck. Static tests have

been conducted to simulate these accelerations and two model parameters have been adjusted to predict the deflections to within 10% of error with respect to experimental data. The finite element model is also capable of accurately predicting the first 8 natural frequencies of the spacecraft.

SPACECRAFT MANUFACTURING

The 24 SAS panels for the upper, mid, and lower cages were fabricated and machined from 8 large panels that could produce 16 lower or mid panels and 8 upper panels. The large panels were 0.020 inch thick precured panel assemblies of Gr/E T-50/ERL1962, [0/45/90/135] with either co-cured 0.0002 inch copper one one side and co-cured 0.002 inch Kapton. These precured skins were then bonded using FM-300-2U film adhesive to 0.25 inch aluminum honeycomb core (1/8 inch cell; 3.1 lb/ft^3). All aluminum inserts were post potted in Corefil 615 and bonded using room temperature epoxy adhesive, Hysol EA9394.

Insert locations were machined into the various panels at the time the sandwich subassemblies were cut from the larger panels. Then, using master bond plates that are common to those used for the corresponding frame subassemblies, all inserts were located into the SAS panel.

The space frame assemblies and equipment decks that make up the spacecraft structure differ in construction. The decks are manufactured similarly to the SAS panels, except that copper was co-cured on both sides of each deck. The space frame is made from flat laminates. The upper deck is the same thickness as the SAS panels but the mid and lower decks have a one inch thick aluminum core (1/8 inch cell, 4.5 lb/ft^3). The skin thickness on all decks is 0.030 inch with an orientation of $[0/60/120]_S$. The frame subassemblies are made from flat 0.048 inch thick laminates of T50/ERL1962 with a $[0/45/90/135]_S$ orientation. As is typical of flat laminate construction, all details can be "nested" tightly on larger cured laminates and machined out with a waterjet machining head mounted to a programmable router. Utilizing COI's concept for a self-fixturing fabrication process (the Short Notice Accelerated production Satellite or SNAPSAT™)[1], all details are removed from a completely processed panel (prepped and bonding) and snapped together. The snapping together feature is mortise and tenon joints that are precision machined into the details.

SPACECRAFT STRUCTURAL OPTIMIZATION

Our goal is to minimize the cost, i.e. maximize the ratio of payload over structural weight while keeping the manufacturing cost as low as possible. In order to accomplish this task, we first need to identify design parameters with the largest potential weight savings. Among the potential design parameters, only a few account for significant fractions of the total weight of the spacecraft. The only parameters of interest are the cage laminate thickness and the skin thickness of the solar panels. Together, they account for about 44% of the structural weight.

We then need to isolate the structural design constraints as well as those resulting from manufacturing concerns. Material characterization efforts were initiated to define design allowables in critical areas of the structure and isolate the structural constraints. As a general guideline for the manufacturing requirements, we want to keep the FORTÉ structure homogeneous, i.e. the thickness of the beams constituting the 3 cages have to be identical, as well as the skin thickness of the SAS for all cages. This is done to minimize the complexity of manufacturing. Also, diagonal members have to be present on both lower and mid cages or not at all so that these cages are identical and interchangeable. The lay-up of the graphite/epoxy prepregs is another concern. It is very difficult to handle plies of less than 0.0025 inch thick. We therefore limit the thickness of the cage beams to a minimum of 0.02 inch and the skin thickness of the solar panels to 0.01 inch.

We then vary the candidate parameters for weight savings and predict the FORTÉ response using the finite element model previously described. The resulting designs will be acceptable if the model response satisfies the predefined design constraints with reasonable safety margins. Sensitivity studies showed that the thickness of the cages beams and the solar panel skins could be reduced from 0.048 to 0.02 inch and 0.02 to 0.01 inch respectively. This represents a weight savings of 31 lbs from the initial design, i.e a 20% reduction in structural weight. Since it is expected that manufacturing cost remains about the same, this weight reduction would save $930K in launching cost (based on $30K/lb).

CONCLUSION

LANL and COI have successfully designed, built and tested a new CLASS of satellite. The FORTÉ experiment has provided the test bed and space validation for this type of structure. This major technology development will make a significant contribution to the nation's many industrial pursuits that involve advanced spacecraft.

ACKNOWLEDGMENTS

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[1] SNAPSAT™ is a patent-pending trademark of COI.

DISCLAIMER