Structural Design of Micro-Satellite for ODIE

Hian Leng Chan
Student
Dept. of Aeronautics & Astronautics
Stanford University, Stanford, CA 94305
(650) 723-3524
hlchan@leland.stanford.edu

Professor Robert Twiggs
Director, Space System Development Laboratory
Dept. of Aeronautics & Astronautics
Stanford University, Stanford, CA 94305
(650) 723-8651
btwiggs@leland.stanford.edu

1. Abstract

The Orbiting Drag-free International Explorer (ODIE) is a technology demonstration of the proposed Orbiting Medium Explorer for Gravitational Astrophysics (OMEGA) mission in gravitational wave detection. The OMEGA mission requires the motion of the spacecraft to be stabilized around a proof mass reference such that spurious acceleration on the proof mass is reduced to below 10^{-15} m/s^2/Hz$^{1/2}$. ODIE will verify the feasibility of using the Capacitative and Electrostatic Sensitive Accelerometer Reference accelerometers (CAESAR) and the Field Effect Electric Propulsion (FEEP) thrusters to provide the desired drag-free performance necessary for gravitational wave experiments. This paper will discuss the structural aspects of the design of the micro-satellite, ODIE.

2. Introduction

The ODIE micro-satellite [1] was born out of a need to demonstrate the technology needed for the proposed Orbiting Medium Explorer for Gravitational Astrophysics (OMEGA), which is a pioneer in the field of gravitational wave experiments.

The OMEGA mission consists of 6 small satellites placed in orbit at an altitude of 600,000 kilometers. Each satellite functions as a large Michelson interferometer to detect gravitational waves. To differentiate these gravitational waves from the actual perturbation of the satellite motion, the position of each satellite has to be stabilized, such that all spurious accelerations on the proof mass is reduced to below 10^{-15} m/s^2/Hz$^{1/2}$.

As such, we need to create a drag-free environment for the accelerometer proof mass in which the only force acting on the proof mass is that due to the gravitational attraction between the proof mass and the accelerometer. All other residual forces, be it solar pressure, gravitational attraction between the proof mass and the accelerometer, magnetic, etc., has to be isolated from the accelerometer proof mass sitting at the center of the spacecraft, such that the proof mass is on a purely gravitational trajectory.

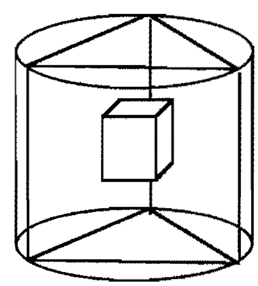

Figure 1: Schematic of ODIE structure and the proof mass

To protect the accelerometer proof mass from the external environment, the proof mass is housed in a cylindrical shell with a triangular baffle whose apex extends out to the inner circumference of the shell (Figure 1). The ODIE spacecraft will use CAESAR accelerometers and the FEEP thrusters to control the position of the spacecraft around the accelerometer proof mass to within 10^{-3} micrometers.

To have minimal interaction between the structure and the proof mass, the structure needs to be designed such that the it will not deform by more than that required for drag-free operation when under the action of external and internal loads. The structure also serves as a platform for solar panels and electronic equipment. Since these body-mounted solar panels work more efficiently at low temperatures, the cylindrical shell needs to have its outer surface temperatures lowered considerably. Lastly, the ASAP (Ariane Structure for Auxiliary Payload) V requires the spacecraft to have certain dynamic characteristics.

3. **Problem**

The constantly changing thermal load due to solar radiation acting on the spacecraft causes a temperature gradient around the circumference of the cylindrical shell. Under the action of these thermal loads, the dimensions of the structure will change. To keep these deformations to less than 0.1 micrometers, we would have to keep the coefficient of thermal expansion (CTE) to the order of 0.1 μm/m/K, in the longitudinal and radial directions.

To provide power for the spacecraft, it will have body-mounted solar panels. For an optimum operation of the solar panels, the temperature of the solar panels must not exceed 35 °C. In other words, the temperature on the outer surface of the cylindrical shell has to be kept to a maximum of 35 °C. To do so, the cylindrical structure has to provide a low-resistant heat path around its circumference in order to transfer all that thermal energy from one side of the shell to the other.

The structure of the spacecraft also has to withstand the high dynamic loads that occur during the launch phase. To do so, the structure must be stiff enough such that its first fundamental frequency in the longitudinal mode is at least 100 Hz and that due to other modes be above 50 Hz.

4. **Structural Design Objectives**

The requirements on the structure of the micro-satellite of ODIE are as follows:

(i) It cannot deform by more than 0.1 microns while the spacecraft is in orbit

(ii) The outer surface temperature of the cylindrical shell has to be below 35 °C

(iii) The first longitudinal mode must be above 100 Hz and any other modes must be above 50 Hz.

This paper will focus on the first 2 objectives of the design.

5. **Design Approach**

Material and the geometry are the 2 key factors that will affect the engineering performance of the structure.

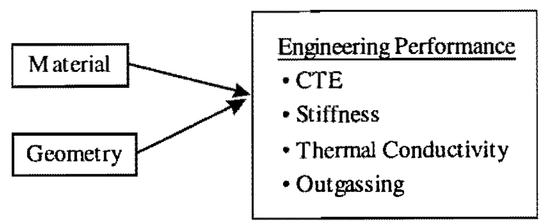

Figure 2: factors affecting the mechanical properties of ODIE structure

The parameters that will be crucial to this design are coefficient of thermal expansion (CTE), thermal conductivity, stiffness and outgassing. This is shown in Figure 2.

6. **Material Selection**

The key challenge to this project is in designing and building a structure that has an extremely low CTE, high thermal conductivity and a high stiffness-to-weight ratio. In order to achieve all this, we cannot use traditional metals. We would have to turn to composite materials. Not only do they have high stiffness-to-weight ratio, they also have a low CTE (in the fiber direction normally) which can be used to tailor our design to meet the zero CTE requirement.

However, composites tend to outgas or release trapped moisture, especially in space. This is due to the presence of trapped vapors in the matrix of the composite material. The typical resin material used for GFRP is epoxy, which traps a significant amount of moisture and can cause a lot of outgassing, which causes cause a lot of outgassing, which causes dimensional changes and may contribute as propulsive forces on the structure. Both of these effects are undesirable to the drag-free environment we are trying to create. Lately, a new resin system using a cyanate ester called RS-3 has been found to trap significantly less moisture as compared to epoxy [2]. To further reduce the moisture content in the resin, we will have to bake the composite structure in a vacuumed-oven. This can be done just before launch to get rid of the moisture trapped in the resin.

7. Physical Geometry

Figure 3 shows a scaled-down model of the cylindrical shell. It consists of two key structural components, vis-à-vis, longi's and circ's. The longi's represent longitudinal members of the shell that are made using pultrusion method. The circ's are made using filament winding method. Both methods are low-cost manufacturing methods as compared to the autoclave.

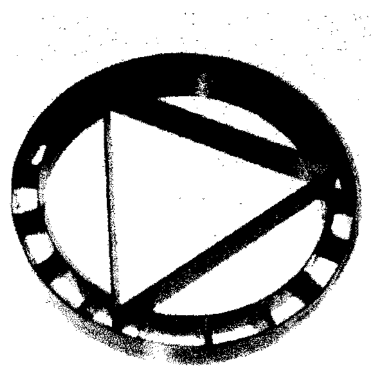

Figure 3: Picture of ODIE shell with baffle

A triangular baffle made out of a grid structure has its apex extended to meet the inner circ's. The unique feature of the grid structure is that its CTE in the 2 orthogonal directions are independent of each other. This is the key advantage of grid structures over traditional laminates. Figure 4 shows a typical grid structure.

By integrating the baffle with the laminated cylindrical shell, we are able to build a spacecraft that has zero CTE in 2 orthogonal directions. In other words, the spacecraft would not distort in the radial and axial directions when a thermal load of any magnitude is applied to it.

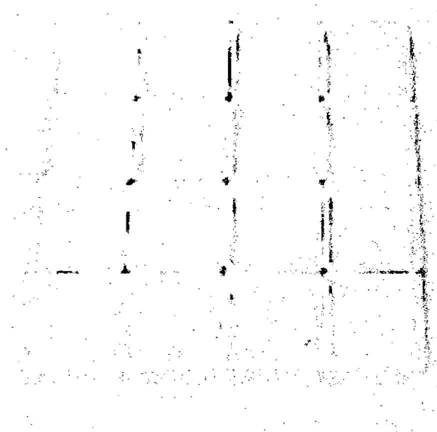

Figure 4: Picture of grid structure

8. Design for CTE

The CTE of aluminum and copper is 24 μm/m/K and 16 μm/m/K respectively, which are approximately 2 orders of magnitude larger than the required CTE. Notice that the CTE of composite materials are very low in the direction of the fibers as compared to the CTE of metals.

	CTE-x (μm/m/K)	CTE-y (μm/m/K)
Aluminum	24.0	24.0
T300/5208	0.02	22.5
AS/3501	-0.3	28.1
YS90A/RS-3	-1.3	33.0

Table 1: CTE values of various materials

By stacking plies on top of one another, with each ply in a different orientation, we can tailor the overall CTE of the laminate to that which we need.

However, the disadvantage of using laminates is that the CTE of the laminate in one direction is dependent upon the CTE in the orthogonal direction. As such, we will not be able to control the CTE of the laminates in 2 directions independently. To circumspect this problem, we can use grid structures like the one shown in Figure 4. The CTE in one direction unique feature of composite grid structures. Much of this work was done by Tsai, S. [3] and a Stanford patented is pending on this design. Figure 5 shows the possible range of CTE with grid structures.

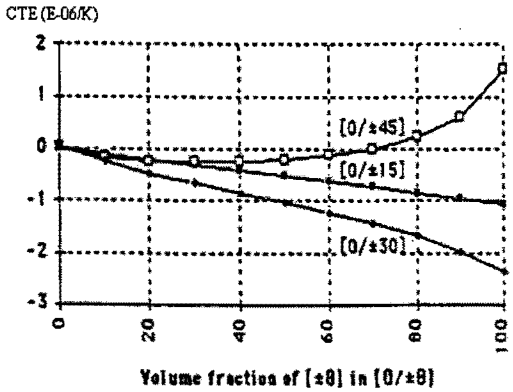

Figure 5: Range of CTE possible with grid structures

Since the cylindrical shell is more of a laminate structure, the CTE of the shell could only be designed with zero CTE in the axial direction rather than the radial direction. To keep the structure from deforming in the radial direction, the baffle made out of grid structure has to be designed such that it compensates for the radial displacement of the cylindrical shell. By having the CTE of the baffle zero in the longitudinal direction and the CTE in the radial direction which is equal but opposite in magnitude to that of the shell, the structure of the spacecraft can have zero CTE in both axial and radial directions.

9. Design for Thermal Conductivity

To meet the requirements for the optimum operation of the solar cells, the cylindrical shell has to provide a low-resistant heat path around its circumference. The total amount of heat flux that is coming in from the sun is about 1360W/m^2 (Figure 6). Based on the projected area of the cylinder, this amounts to about 570 Watts of energy that needs to be moved around the cylinder in such a manner that the maximum temperature on the outer surface of the cylindrical shell is not more than 35 ^{0}C.

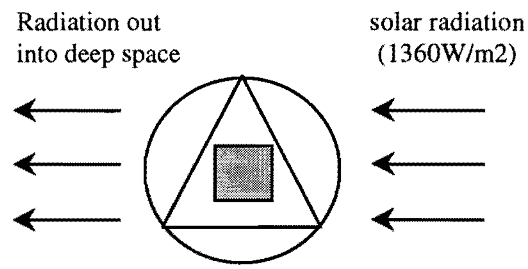

Figure 6: Radiation on ODIE

Based on a simple one dimensional analysis, this would require an equivalent thermal conductivity of more than 10,000W/m/K for a 1 millimeter thick wall and a temperature difference of 20 ^{0}C. Copper has a thermal conductivity of 400 W/m/K while graphite alone has a thermal conductivity of 1100 W/m/K. To cool the outer surface down quickly, we need to either increase the thickness of the cylindrical shell by 10 times or use a more effective form of heat transfer.

This brings us to the use of heat pipes as a heat transfer tool. Many computers and space applications use heat pipes to transfer the heat load from one point to another. Heat pipes make use of the processes of evaporation and condensation to perform the heat transfer. It does not rely on the mechanism of heat conduction, which is far less effective than condensation or evaporation. The fact is that the latent heat of vaporisation is usually very large for fluids. The presence of heat pipes can reduce the temperature gradient from 80 ^{0}C to 20 ^{0}C (see Figure 7).

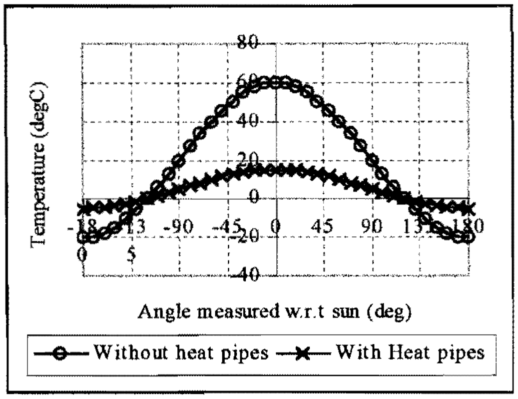

Figure 7: Temperature gradient around cylindrical shell

By integrating these heat pipes with the cylindrical shell, we would be able cool the outer surface of the shell and hence cool the solar panels to a temperature that would not degrade the performance of the solar panels. The current design of the cylindrical shell has the component called longi's that has a gap of 1 inch. By cutting up the longi's and plugging an insert into the gap, we are able to accommodate the heat pipes (which has an outer diameter of not more than half an inch) into the cylindrical shell without seriously compromising the other mechanical properties of the structure. Figure 9 shows the ODIE shell with the grooved insert.

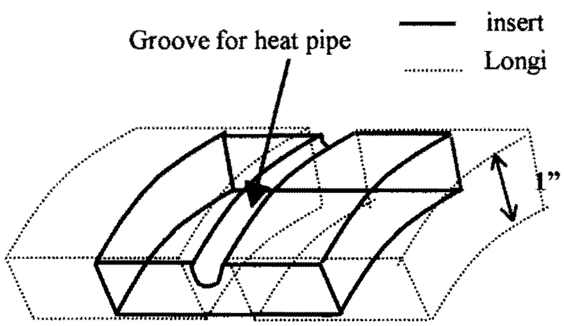

Figure 8: Schematic of insert

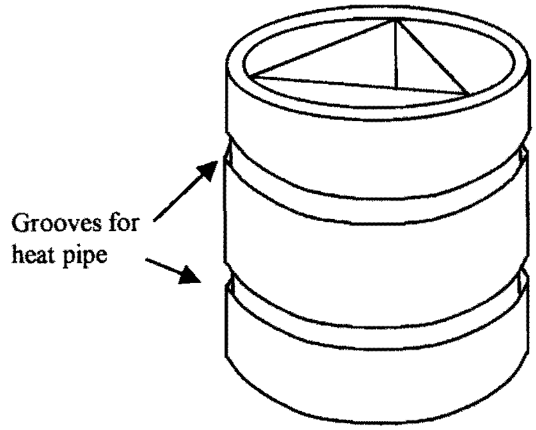

Figure 9: ODIE shell with insert

10. Future Work

A full-scale model using the graphite-epoxy system will be scheduled for a ground vibration test. CTE measurement tests are being scheduled for the graphite-cyanate system. The first prototype of the micro-satellite structure using the actual graphite-cyanate system will be ready in April 1999.

11. Conclusion

The stringent requirements for drag-free operation of the micro-satellite renders the structural design a difficult task. Our analysis showed that by using a cyanate ester, RS-3, as the resin material of a high modulus GFRP and by employing a hybrid laminate/grid structure, structural disturbances on the proof mass accelerometer can be minimized to the level necessary for drag-free operation. By integrating the heat pipes with the cylindrical shell, the body-mounted solar panels will be allowed to operate at the desired temperature. Hence, not only is this design able to control the deformations in the axial and radial directions, it is also able to control the temperature gradient on the outer surface of the spacecraft.

12. Acknowledgements

I would like to thank Professors Robert Twiggs and Stephen Tsai for their invaluable advice, encouragement, coaching and ideas. I would also like to thank Jet Propulsion Laboratory for their support on this project. Last but not least, I would like to thank ITRI and TPI for making the graphite/epoxy models.

13. References

1. Floyd, Allison, "Conceptual Design of an Orbiting Drag-free International Explorer (ODIE) Small Satellite Mission", Proceedings of the 11th AIAA/Utah State University Conference on Small Satellites, August 1997.

2. Matsumoto, T., Mohri, M. and Oshima, A., "Moisture Absorbing Properties of Thermosetting Matrix Resins for Composites", Nippon Oil Co. Ltd., Yokohama, Japan.

3. Tsai, S. W., " Theory of Composites Design", Think Composites, 1992.