Hydrogen On-Orbit Storage and Supply Spacecraft Custom Bus Design

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Abstract

The Hydrogen On-Orbit Storage and Supply (HOSS) mission is a proposed technology demonstration of the effects of liquid hydrogen storage and acquisition in microgravity. HOSS would test a liquid acquisition device and thermodynamic vent system (LAD/TVS) for a solar thermal thruster in an orbital environment. Previous cost estimates for a similar technology demonstrator have been estimated at as much as $300 million. Cost is the primary, driving requirement to all facets of this design and strongly influences launch vehicle and bus selection. To successfully implement this experiment, a low-cost spacecraft is proposed with a total expenditure of less than $10 million, including launch and mission operations. The spacecraft is designed using subsystems and structures sufficient to meet unique mission needs. Successful implementation of a "simple is better" philosophy plays a pivotal role in the design and analysis of the spacecraft bus.

1.0 Introduction

Spacecraft designers have two options when developing spacecraft for a particular mission. The bus, which is the primary structure and support system for the spacecraft, can be designed from an existing layout or a new design may be developed to meet specific mission needs. In recent years, the trend in small satellite design has been to employ a previously designed spacecraft bus to reduce cost and development time. However, these structures sometimes provide greater capability than is required for smaller, simple missions at a higher price. Therefore, it may be preferable to custom design a bus to meet the reduced needs of the mission.

The Hydrogen On-Orbit Storage and Supply mission has a limited set of design and mission requirements which allows the use of a simpler, more economical spacecraft design. Using minimal subsystems and off-the-shelf components, an adequate spacecraft could be designed for less than 20% of the original estimated mission cost.

This design provides the spacecraft structure for the HOSS mission. All necessary subsystems are identified and components selected to meet mission needs. However, the design will focus on the spacecraft bus and the assurance that it meets all structural, dynamic, and thermal requirements. Specific subsystem components may then be selected at a later time and integrated for manufacturing.

2.0 Background

For a number of years, the National Aeronautics and Space Administration (NASA) and the United States Air Force (USAF) have become increasingly interested in the concept of solar thermal propulsion. This technology uses concentrated solar energy to heat a low molecular weight fuel to sufficient temperatures and pressures to generate high specific impulse propulsion. Hydrogen has the lowest molecular weight, making it the ideal fuel to generate the highest level of specific impulse. However, it is difficult to handle and store due to its low boiling temperature (20 K). To be used effectively, hydrogen must be stored for relatively long periods and then delivered to a thruster in a liquid form. Recent solar thermal design concepts have combined this storage and supply of hydrogen into a single system which uses a thermodynamic vent system to provide propellant to the solar thermal collector. This technology can only be adequately tested in a microgravity environment.

On 21 January 1997, Utah State University's Space Dynamics Laboratory (SDL) received a contract from NASA Lewis Research Center (LeRC) to determine the feasibility of building and flying an experimental hydrogen dewar which would test cryogenic hydrogen storage and acquisition in space. SDL had worked previously with LeRC and NASA Marshall Space Flight Center (MSFC) on cryogenic storage systems and has extensive experience in this technology. The contract included a preliminary dewar design, launch vehicle and bus options, and systems requirements. The Hydrogen On-orbit Storage and Supply project resulted from this initial NASA proposal. As part of an effort to examine the various options for the spacecraft design, it became apparent that all commercially available spacecraft buses have adequate payload mass and volume capacity. However, they also have much more pointing capability than is necessary at a higher cost. Therefore, the decision was made to design a custom spacecraft for the hydrogen dewar. This reduces mission cost significantly while still meeting design requirements.

The spacecraft bus design, which provides support for the HOSS demonstration at minimal cost, has been completed. The main objectives of this design were to develop and analyze the spacecraft structure to provide a statically, dynamically, and thermally compatible structure for the experiment and all supporting hardware. Attitude control, communications, and data handling systems were examined only so far as was required to ensure a minimal impact on the HOSS payload. The bus was designed to be integrated with a selected launch vehicle and an 80-liter dewar designed by SDL.

3.0 Technical Issues

There are many technical issues involved in the development of the spacecraft bus. This section discusses briefly each point of concern for the custom bus design. The mission lifetime is defined to be 30 days when the hydrogen from the storage tank is expended. This is a primary and fundamental assumption.

3.1 Cost Requirements

Maintaining a low program cost was the driving design requirement for this mission. Therefore, spacecraft components were selected from off-the-shelf sources, and the launch vehicle selection was made from lower-cost launch providers.

3.2 Launch Vehicle Selection

Launch vehicle selection is one of the more critical choices for a low-cost project. Roughly half of the total cost of a spacecraft is absorbed in the cost of the launch vehicle. Because of the smaller mass and volume requirements of HOSS, the standard launch vehicle choices would have been Orbital Sciences' Pegasus or Taurus, or Lockheed Martin's Athena. The costs of each of these rockets, even as a secondary payload, ranges from $5-20 million. However, a unique possibility presented itself early in the design. Jim French of Rocket Development Corporation offered an inexpensive launch on a test flight of its Intrepid rocket. The constrained budget of the program, coupled with the high mass and volume margins this vehicle would provide, led to the selection of the Intrepid as the baseline launch vehicle, launching in mid-2000. Although this selection presents a higher risk to mission success due to an unproven vehicle, the opportunity to launch at such a low cost outweighs the risks incurred.

3.3 Vertical or Horizontal Loading Constraints

Early in the design process it was necessary to define the orientation of the spacecraft during launch and deployment. The primary load path is a major factor in vehicle and bus layout, and is almost totally dependent on the environment in the launch vehicle payload shroud. For obvious reasons, an axially loaded condition was desired so that the dewar could be incorporated as the primary load-bearing member. The selection of the Intrepid launch vehicle facilitated this option. A side, or transverse, loading was a less desirable option and was one of the reasons why a secondary payload opportunity on the Taurus was not selected. Once the Intrepid was selected as the primary launch vehicle, all designs focused on an axial configuration.

3.4 Structural Material and Component Selection

Due to the cost constraints of the spacecraft, structural material and component selection needed to remain as low-cost as possible. Several off-the-shelf components were examined to facilitate this need. Many different materials were available for the construction of the spacecraft structure, ranging from basic aluminum to advanced composite materials. Because of the extra mass margin provided by an Intrepid launch vehicle, the decision was made to use aluminum as the primary structural material rather than composites. After reviewing several types of aluminum, it was concluded that 6061-T6 aluminum would be sufficient for the structural requirements of the mission. This metal would accept adequate thermal, vibration, and static loading, and is readily available on the open market. The spacecraft design included structural parts which would be fairly easy to manufacture and would provide a relatively straight-forward construction and integration scheme. All structural members could be machined, bent, and/or drilled to meet the final structural requirements.

The proper component selection was an issue which needed examination to provide an understanding of the dynamic qualities of the spacecraft. Understanding the mass, volume, and placement of these components, along with their power requirements, would drive such issues as component placement and solar panel and battery size.

3.5 Component Placement

Proper placement of spacecraft components is driven by the need to place the center of mass near the center of the spacecraft and to create favorable moments of inertia for the attitude control system. To accomplish this, a symmetric dispersion of mass is required.

3.6 Power Analysis

Due to the low cost requirements of this mission, a simple silicon solar array and nickel-cadmium battery system were selected. Battery system size is dependent on eclipse time, which in turn is dependent on orbital selection; batteries tend to be one of the highest mass non-structural components on a spacecraft. In this design, the solar array serves an additional function than that of providing power for the spacecraft; it acts as a thermal shield to the hydrogen dewar from direct sunlight. Analysis of the required surface area of this solar array has shown that it is adequate to shield the dewar and provide appropriate power.

3.7 Thermal Isolation of the Dewar

Due to previous thermal analysis of the dewar system by engineers at SDL, it was determined that to ensure a 30-day lifetime for the spacecraft's hydrogen supply, a dewar surface temperature lower than 250 Kelvin must be maintained. The spacecraft bus, components, and solar panel were consequently designed to minimize heat leak into the dewar and provide a shield against direct solar radiation.

3.8 Attitude Control System (ACS)

After preliminary design of the spacecraft, the only remaining requirement for the HOSS spacecraft, with respect to attitude determination and control, was to maintain a sun-pointing attitude. This provides for the thermal isolation of the hydrogen dewar while yielding necessary power input for all spacecraft functions. Thus, hardware selected for the ACS system simply needs to produce this sun-pointing spacecraft attitude.

3.9 Vibration Requirements

Vibrational requirements are driven by the dynamic envelope of the launch vehicle. Typically, a company publishes data defining the random vibrations encountered and the minimum natural frequency required for the spacecraft. Because the Intrepid launch vehicle did not have a published payload users guide, a rough estimate for the dynamic envelope was used. The design was constrained for this analysis by the dynamic envelope for an Orbital Sciences Taurus launch vehicle. This limited natural frequency in the longitudinal direction to ≥ 30 Hz and ≥ 20 Hz in the transverse direction. Although a variety of different vibrations are encountered during launch, sufficient margin is obtained by designing to these values.

3.10 Mass Requirements

Because the Intrepid launch vehicle will have the capability to lift large masses to polar orbit, maintaining a low mass was not as critical as it is for most spacecraft designs. This extra margin allowed for the use of heavier components and instruments. Because the structure and other subsystems are not as constrained by mass requirements, overall mission cost and manufacturing time are reduced.

3.11 Orbital Selection

The HOSS spacecraft was intentionally designed to perform its mission in any low-earth orbit that might be selected. This resulted from the need to test the liquid hydrogen dewar in a weightless environment without any specific pointing or orbital requirements. Having this broad mission parameter also allowed greater flexibility in choosing a launch vehicle and attitude control system.

4.0 Structural Design

4.1 Constraints

The selection of the Intrepid rocket as the primary launch vehicle allowed for a flexible spacecraft structure design. Had a smaller launch vehicle been selected, such as a Taurus, a smaller volume and mass margin would have caused significant design restrictions. Preliminary analysis of the Taurus payload shroud indicated that the hydrogen dewar would have been mounted transverse to the major axis of the launch vehicle, necessitating a significant redesign of the spacecraft bus. Also, mass constraints would have required the design of a more complicated structure. These mass constraints are not a serious issue with the Intrepid launch vehicle. The addition of solid panels, where traditionally a lattice structure would have been required, is an acceptable design feature. Without the mass and volume constraints, a liberal design approach was taken.

4.2 Primary Concept

The preliminary concept for this spacecraft included a hexagonal polyhedron. A circular cross-section complicates the integration of solar panels on the side faces and also increases the difficulty of structural analysis. The hexagonal shape was chosen because it tends to provide a relatively simple structural architecture while having a large internal volume. Upon further investigation, it was concluded that the hexagonal shape would not provide a side panel with sufficient surface area for a solar array that would meet the needs of the spacecraft without including an extending member. It was decided upon analysis that a predominantly triangular cross section would be used, with bays at the vertices for instrumentation and other equipment. (See Figures 1 and 2.) Despite the larger payload radius required for this design as compared to a circular or hexagonal shape, the large volume margin of the Intrepid launch vehicle made this an acceptable solution.

4.3 Preliminary Design

It was concluded that to minimize costs, simple structural members that would be fairly easy to manufacture from readily available materials would be a high priority. Aluminum 6061-T6 was chosen as the material of choice since it is supplied by a variety of providers at relatively low costs. Also, this material has been used many times in spacecraft structures and is a proven material for the space environment. All structural members would be 1/4" thickness, which is a material dimension that can be ordered directly from producers.

4.4 Final Design

A modular approach to the design was a major design philosophy inasmuch as each structural part would be used in several locations around the spacecraft. As can be seen from Figure 1, there is the inclusion of several different members, which are each used multiple times. There are six fins, nine bays, and two rings, included in the preliminary design. After further study, three trusses, 18 truss connectors, 24 brackets, and 12 bracket plates were added to give stability, strength, and stiffness to the structure.

Each of these parts can be manufactured fairly easily by cutting, bending, and drilling. Complicated geometries were avoided to facilitate construction. All parts were designed to be bolted together for a basic final structural assembly. These were design guidelines intended to minimize manufacturing costs.

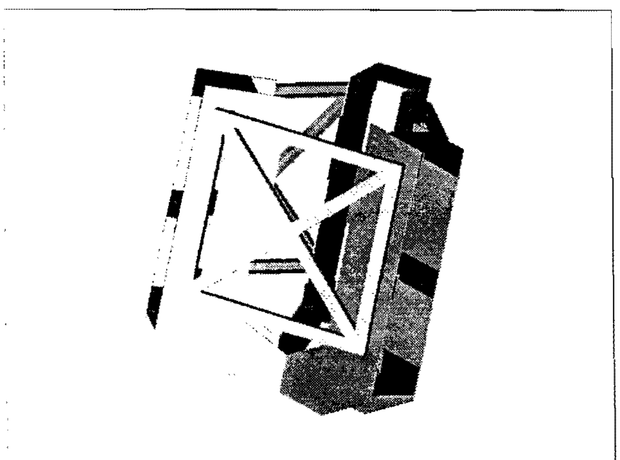


Figure 1: HOSS triangular bus design.

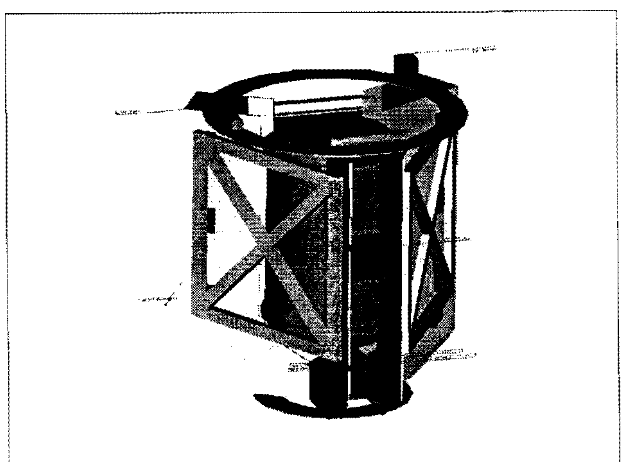


Figure 2: HOSS structure with instrumentation.

Further design of the spacecraft structure included a center of gravity (CG) analysis. Placement of components was intended to place the center of gravity of the spacecraft as near to the geometric center of the space vehicle as possible. This geometric center was also designed to be the exact geometric center of the hydrogen dewar. Knowledge of the position of the spacecraft center of gravity is essential for proper attitude control requirements.

The center of gravity analysis required the inclusion of all spacecraft components, instruments, and other equipment, such as batteries and transmitters. Preliminary numbers for the mass and volume of each component were gained from actual measured values from the Skipper spacecraft, developed by SDL. These values were used as place holders since they are actual values that have been space qualified and would provide the support systems for the mission. A complete mass, power, and cost analysis has been included as Figure 3. Lacking in the CG analysis is placement and distribution of any wires, brackets, mounts, or plumbing. Also, the mass of each included component was considered to act at the geometric center of the mass.

Following these assumptions, the placement of components was adjusted until the center of gravity approached the geometric center of the dewar. According to analysis, the CG was less than 3/4 of an inch from the dewar center. Rough placement of components can be seen in Figure 2.

HOSS

TOTAL BREAKDOWNS

Total Mass Breakdown

	Current Mass (lb)	Current %	Est. Mass (lb)	Est. %
Structures and Mechanisms	174.1	39.33	120.0	40.00
Payload	213.5	48.23	150.0	50.00
Thermal Control	2.2	0.51	3.0	1.00
Electrical Power	24.8	5.61	9.0	3.00
Command and Data Handling	9.7	2.20	6.0	2.00
Telemetry, Tracking, and Control	3.7	0.83	3.0	1.00
Guidance. Navigation, and Control	14.6	3.29	9.0	3.00
Propulsion	0.0	0.00	0.0	0.00
TOTALS	442.7	100.00	300.0	100.00

Total Power Breakdown

	Current Power (W)	Current %	Est. Power (W)	Est. %
Structures and Mechanisms	0.0	0.00	0.0	0.00
Payload	0.0	0.00	0.0	0.00
Thermal Control	0.4	0.56	4.0	5.00
Electrical Power	1.2	1.52	4.0	5.00
Command and Data Handling	9.6	12.20	8.0	10.00
Telemetry. Tracking, and Control	24.3	30.82	20.0	25.00
Guidance. Navigation, and Control	43.2	54.90	44.0	55.00
Propulsion	0.0	0.00	0.0	0.00
TOTALS	78.7	78.69	80.0	100.00

Total Component Cost Breakdown

	Current Cost ($k)	Current %	Est. Cost ($k)	Est. %
Structures and Mechanisms	$55	8.12	$100	10.00
Payload	$55	8.12	$150	15.00
Thermal Control	$40	5.91	$50	5.00
Electrical Power	$302	44.61	$400	40.00
Command and Data Handling	$95	14.03	$150	15.00
Telemetry, Tracking, and Control	$45	6.65	$50	5.00
Guidance. Navigation, and Control	$85	12.56	$100	10.00
Propulsion	$0	0.00	$0	0.00
TOTALS	$677	100.00	$1,000	100.00

Figure 3: HOSS mass, power, and cost budgets, current estimates versus initial estimates.

5.0 Power Analysis

5.1 Preliminary Design

The only requirements for the power subsystem was ensuring that the solar panel would provide enough power to supply the hydrogen dewar and other spacecraft equipment. Since the instruments for the spacecraft were baselined as Skipper components, power needs were readily available. The hydrogen dewar power need was supplied by SDL. As shown in Figure 3, the total power needed for the duration of the mission was approximately 80 watts.

5.2 Final Design

It was decided to use standard solar cells and batteries to minimize costs. Silicon solar cells, which are the most widely used cells in the industry, would be the baseline for this analysis. Also, nickel-cadmium batteries are the least expensive of the commercially available, rechargeable, space-qualified batteries. The analysis for the solar panel is included in Figure 4.

It is demonstrated that a solar panel of dimensions 32.3 inches by 34.8 inches would be sufficient for the provision of the needed power. Certain assumptions were made in this analysis that are considered to be worst-case in nature. These assumptions decreased the theoretical efficiency of the solar array, and are listed, as follows:

- Temperature coefficient of -0.5%/°C
- Solar cell efficiency of 11.5% at 28°C
- Solar cell degradation of 30% over 10 years pro-rated to mission duration

POWER ANALYSIS

Constants:	Temperature Coefficient	-0.5	%/Deg_C		
	Needed Bus Voltage	33	V	(for charging)	
	Solar Cell Voltage	0.45	V		
	Solar Cell Efficiency	11.5	%	at	28 Deg_C
	Solar Cell Degradation	30	%	over	10 Years
	Solar Cell Packing Factor	0.9			
	Solar Intensity	1369.8	W/m^2		
Variables:	Wanted EOL Power	78.690	W (includes charging and all effects)		
	Operating Temperature	50	Deg_C (worst case)		
	Sun Incidence Angle	5	Deg (worst case)		
	Cell Length - Side A	2	cm		
	Cell Length - Side B	4	cm		
	Duration of Mission	1	mo		
Outputs:	Cells per String for Proper V	74	cells		
	Temperature Effect	-4.11	% (loss of efficiency)		
	Assumed Power at BOL	88.976	W (with no effects)		
	Needed Total Cell Area	0.565	m^2		
	Cell Area	8	cm^2		
	Total Cells	707	cells		
	Array Size	0.628	m^2	or	972.964 in^2
	Array Size (per side square)	0.792	m	or	31.189 in
	No. Strings for Proper Current	10	strings		
	Actual Number of Cells	740	cells (for proper current)		
	Actual Required Array Size	0.658	m^2	or	1019.558 in^2
	Potential Arrangement	37	cells		
	x	20	cells		
	Potential Total Cells	740	cells (Identical to Theoretical Area)		
	Potential Area	82.000	cm	or	32.283 in
	x	88.444	cm	or	34.821 in
	Potential Total Area	0.725	m^2		1124.131 in^2
	Actual Power at BOL	103.618	W (with no effects)		
	Actual Power at EOL	91.639	W		

Figure 4: Solar cell sizing analysis.

- Solar cell packing factor of 0.9
- Solar intensity of 1370 W/m^2
- Worst case operating temperature of 50°C
- Worst case sun incidence angle of 5°
- Cell dimensions of 2 cm by 4 cm

These dimensions yielded a maximum solar panel size which was similar to that designed for the model. This panel area would produce a best case beginning of mission power of 103.6 W and a worst case end of mission power of 91.6 W, which is enough for given needs with acceptable margin for error.

6.0 Guidance, Navigation, and Control Analysis

The primary Guidance, Navigation, and Control requirement is to point the solar panel normal to the sun. This also keeps the dewar thermally shielded. Because of mass and budget constraints, the most simple GN&C solution was a series of three torque rods placed orthogonal to one another. A thesis presented by John D. Haskell of Utah State University has shown that torque rods can indeed provide the necessary attitude control at the cost of a higher power requirements.

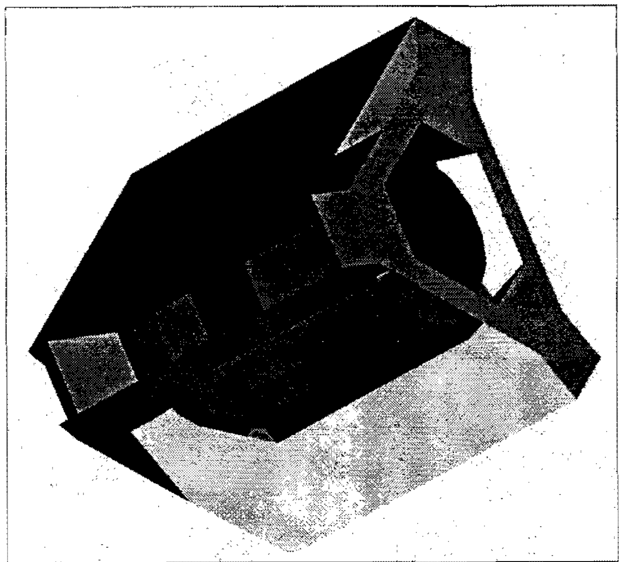


Figure 5: Thermal dewar and structure model.

It has been assumed that the launch vehicle would place the spacecraft in the proper orbit and orientation, which predicates that only minor attitude adjustments will be necessary. The torque rods are capable of providing this control. In addition, the use of a sun sensor and a three-axis magnetometer will provide attitude knowledge, which will be used by the on-board computer to control the rods.

7.0 Thermal Analysis

7.1 Constraints

The basic constraint on the thermal design, as mentioned previously, was the need to keep the dewar shell surface temperature at 250 K during the nominal 30-day mission. Additional constraints were to keep instruments from overheating and attempting to hold the solar panel at the lowest possible temperature for maximum efficiency.

7.2 Tools

The basic tool employed in the thermal analysis of the HOSS spacecraft was I-DEAS Thermal Modeling Guide (TMG). TMG is a finite-element modeler that can model the effects of solar radiation and Earth albedo on an orbiting spacecraft. This capability proved to be extremely helpful in modeling radiation effects on the spacecraft.

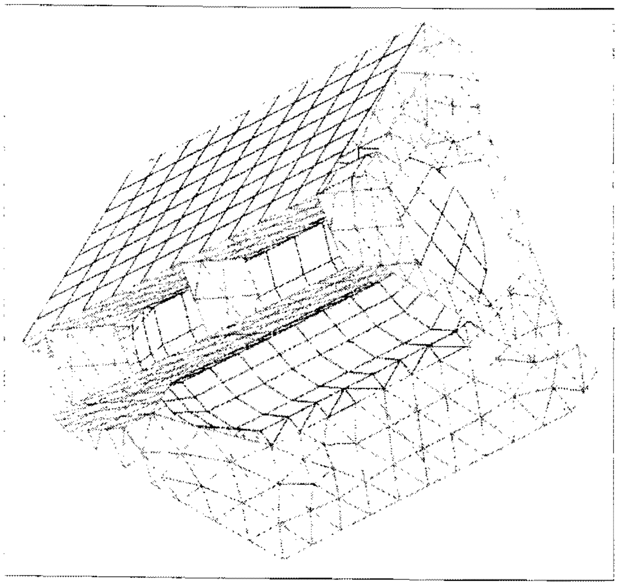


Figure 6: Thermal finite element model (FEM).

7.3 Method

The HOSS spacecraft is composed of many thin surfaces, examples of which are the fins, the solar panel, and the dewar shell. For simplicity and ease of analysis, the decision was made to thermally model the HOSS as a series of thin shells. This necessitated the construction of a new model, which was simpler than the ones used in structural and vibrational analysis.

The first item to model was the dewar shell. Because of the difficulty of modeling radiation interaction between curved surfaces, the dewar was modeled as a 15-sided polyhedron with faceted top and bottom surfaces. This led to a total of 47 surfaces on the dewar model.

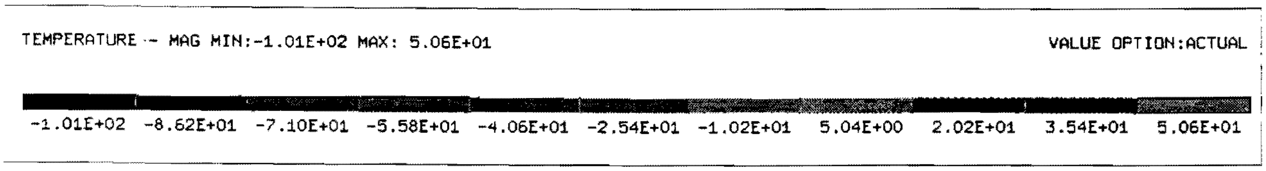
Next the fins were modeled with dimensions approximately the same as the spacecraft. Slight liberties were taken where geometries merited it, such as small dimensions in brackets and so forth. Items that were not significant thermal paths were omitted. Because of the effects needed for modeling, such simplifications were appropriate. The fins were "joined" with the dewar shell by modeling the surface between them as G10 fiberglass. This insured that during meshing, the thermal solver would employ a continuous path from the dewar to the fins.

The final step involved placing the solar panel between the two fins. The solar panel in the thermal model was not identical to that used in the other models, but similarity of dimensions provided for the thermal model. The solar panel was sized to fully shield the dewar shell, as seen in Figure 5.

7.4 Assumptions

The first assumption of the model was that only four materials were present in the spacecraft: 6061-T6 aluminum with a silver Teflon coating, which was used for the structure of the spacecraft; 6061-T6 aluminum coated with emissive white paint, which was used for the dewar shell; a silicon solar array; and G10 fiberglass thermal insulation, which was used to isolate the dewar from the bus structure. These materials are indicated in Table 1.

Table 1: Material Summary	
6061-T6 aluminum with a silver Teflon coating	
Density	2710 kg/m^3
Specific Heat	860 J/(kg·K)
Conductivity	175 W/(m·K)
Emissivity	0.78
Absorptivity	0.08
6061-T6 aluminum with emissive white paint	
Density	2710 kg/m^3
Specific Heat	860 J/(kg·K)
Conductivity	175 W/(m·K)
Emissivity	0.88
Absorptivity	0.25
Silicon solar array	
Density	2100 kg/m^3
Specific Heat	1600 J/(kg·K)
Conductivity	200 W/(m·K)
Emissivity	0.88
Absorptivity	0.67
G-10 fiberglass thermal insulation	
Density	n/a
Specific Heat	n/a
Conductivity	0.05 W/(m·K)
Emissivity	n/a
Absorptivity	n/a



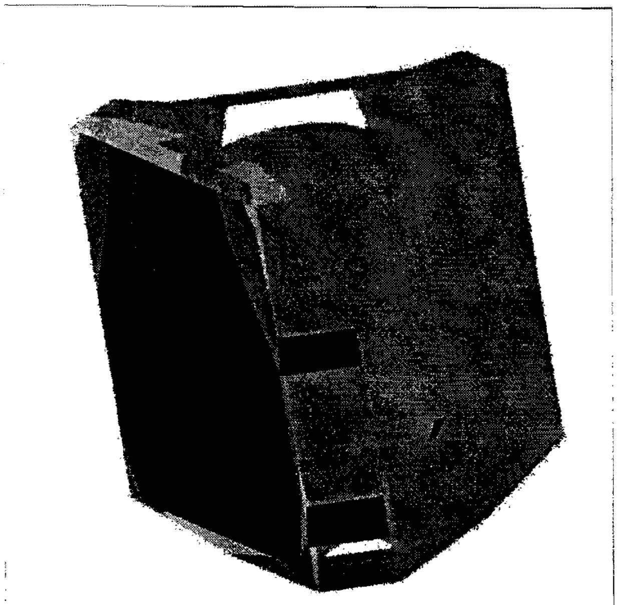


Figure 7: Clear view of the solar panel and two bays at orbital noon.

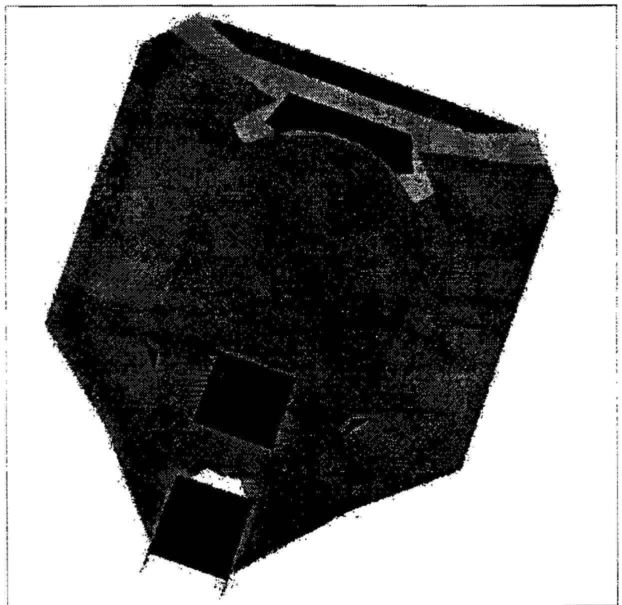


Figure 8: View of the spacecraft from the anti-Sun side as it clears the sunrise on the fourth orbit.

7.5 Model

With the model completed, the surfaces were "stitched" together into one part and the structure was meshed. The fins and dewar shell were free-meshed at a density of 10 cm, and the solar panel was map-meshed at a spacing that would ensure that the nodes of the fins and the nodes of the panel were aligned. This spacing ended up being 10 divisions in the axial direction and 8 in the transverse direction. During the finite-element meshing, materials and shell thickness were also selected. The dewar shell was modeled with a thickness of 6.4 mm, the solar panel at 10 mm, and the fins at 4 mm. The finite element model is included as Figure 6.

With meshing completed, the TMG module was activated, and orbital and thermal boundary conditions were specified. The orbit selected was designed to be a "worst-case" thermal environment. It was a noon-midnight polar orbit at 400 km altitude. The parameters required for TMG are included below:

- Earth albedo value: 0.35
- Solar declination: 0.0
- Eccentricity: 0.0
- Ascending Node Angle: 0.0
- Period: 5500 sec
- Calculation frequency: 11
- Orbital inclination: 90 degrees
- Semimajor axis ratio: 1.0625
- Perigee Angle: 0.0

The spacecraft was set to rotate -360 degrees about its Z-axis each orbit keeping its solar panel always oriented to the Sun. A graphical representation is shown in Figure 9. Thermal boundary conditions were set as follows: the dewar shell initial temperature was set to -23°C (250 K) and the fins and solar panel initial temperature was set to -3°C (270 K). TMG was set to filter out view factors of less than 0.1 between surfaces and radiate them to space. The heat flux and radiation calculations utilized Gebhart's method. The model analysis ran for four orbits, equivalent

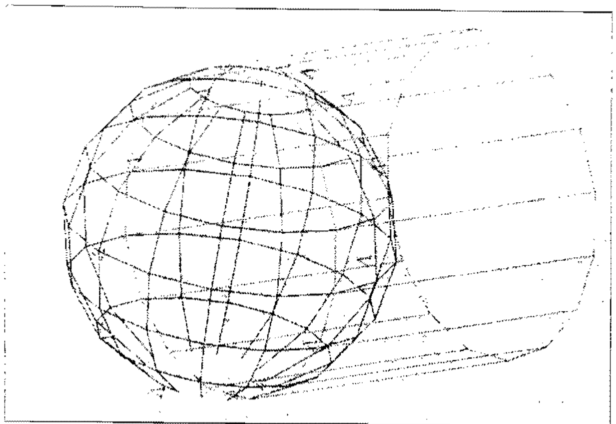


Figure 9: Orbit representation with blue cylinder representing the Earth's shadow.

to 23,000 seconds, with a time constant multiplier of 0.5. The solving method was set to forward-backward because of its unconditional stability, with the results given for every 250-second interval. Maximum iteration was 100 and the temperature was considered stable when variance was below 0.1 K. The model required approximately an hour and a half to solve on a SGI O2.

7.6 Results

Model results were examined using the I-DEAS Post Processing utility and, as expected, the solar panel shield technique was highly effective. The surface of the solar panel showed the hottest temperatures, reaching a maximum of 50°C (323 K) on the fourth orbit. While not the optimal thermal environment for the solar panel, it served admirably to protect the dewar surface from incident solar flux. The G10 fiberglass standoff also kept conductive heat from reaching the dewar. As can be seen in Figures 7 and 8, the dewar temperature remained almost constant at 250 K. The color bar shows the maximum and minimum temperatures and temperature ranges in degrees Celsius. Very little heat entered, and because of the low emissivity of the white paint, very little heat radiated out.

Fin temperature, however, varied greatly, from as high as 310 K at the solar panel interface to as low as 172 K on the anti-Sun side during eclipse. The equipment bays also stayed very cold, because of their small conductive area with the spacecraft and the large area they radiated to space. In the actual spacecraft, these bays would contain instruments that would be producing heat, but with a total thermal output of less than 90 W. It was not judged significant enough to model.

These thermal images show that the spacecraft's temperatures fluctuate throughout a period of four orbits. An attempt was made to capture images that displayed the extremes of temperature. The color bar at the bottom of the page indicated the range of temperatures, in degrees Celsius.

7.7 Implications

The I-DEAS TMG analysis validated our original design concepts, demonstrating that heat can be prevented from reaching the dewar, and the 30-day mission lifetime is feasible under the current design.

Temperatures and thermal cycling in the solar panel are a concern, however. The solar panel swings from temperatures as high as 50°C (323 K) to 1°C (274 K). This thermal cycle of nearly 50 Kelvin would be repeated approximately 520 times in a thirty-day mission, and could damage the solar panels or the interface between them and the spacecraft bus. Solar panel efficiency also degrades at high temperatures and could be a cause for concern.

One of the disadvantages of this design is the inability to keep the solar panels cool. Unlike most solar arrays that radiate their heat from their anti-Sun side into space, the solar array must conduct its heat into the spacecraft fins and bays for subsequent radiation to space. More thermally conductive paths in the spacecraft bus and

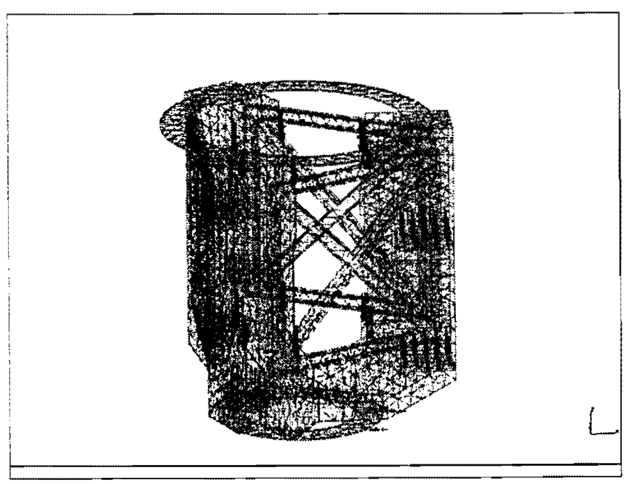


Figure 10: Frame FEM for static analysis.

higher-temperature solar arrays might be merited for this mission.

8.0 Dynamic Analysis

8.1 Constraints

Dynamic constraints for the spacecraft structure require that all static and dynamic loads and accelerations encountered during launch and mission life be accounted for with a sufficient safety margin. Therefore, a structure as light and cost effective as possible was used which would still provide ample strength.

8.2 Preliminary Design

Due to its strength in the longitudinal direction, the primary load bearing member on the spacecraft is the hydrogen dewar. The strength and stiffness of this tank allows it to support the other structural components. Utilizing the dewar as a structural member reduces mass that would otherwise be needed for a separate frame. The six aluminum fins, mounted in sets of two, support the dewar while panels between the fins allow placement of instruments and other components such as batteries and computers. Circular rings at the top and bottom of the fins hold the structure together and add strength in the lateral direction. At the base of the fins, a circular mount allows the spacecraft to be mated to the launch vehicle payload ring. All of these components are composed of 6061-T6 aluminum, heat treated, and bolted together for structural strength. Bolting the bus together maintains a simple and practical structure while reducing costs, and heat treating adds strength to each member. This setup served as the basis for all future analysis, with a few minor iterations to the final configuration.

8.3 Modeling

After modeling this initial setup on I-DEAS, two types of analysis were performed to determine stress concentrations and vibrational responses. It was first necessary to create a finite element model of the structure to perform dynamic testing. Several types of FE models were built for the structure. The first used a planar finite element model which could be adjusted for varying lengths. This type of model helped to show the proper thickness for the fins that could adequately support the structure. Typically, this type of model uses square or triangular elements of varying lengths according to the type of geometry being tested. In order to generate a model such as this for the whole structure, individual planes were modeled separately and then later stitched together. Using this approach to modeling allowed a more precise control of where and what types of elements were used, but proved to be much more time consuming.

The other type of modeling used a solid model generated by the program. Element lengths and node types were set by the user, and then a model was created by the computer. Typically an element length of two inches was used, although this number was decreased to 0.5 inches at intersections and other critical areas. Solid models were used for most of the initial analysis and then more complex and specific models were used when particular points of interest were identified. Figure 10 shows a finite

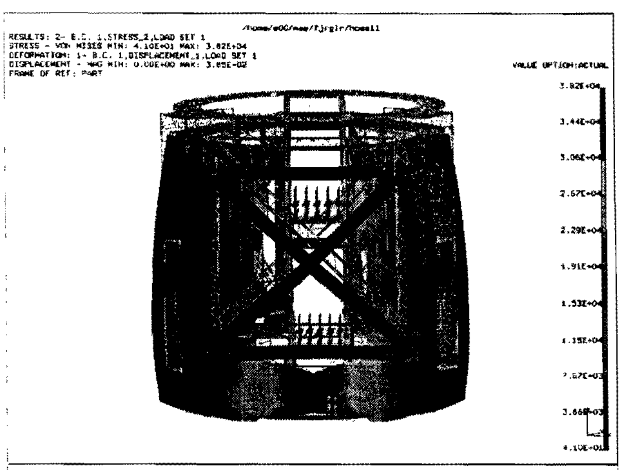


Figure 11: Von Mises stress analysis.

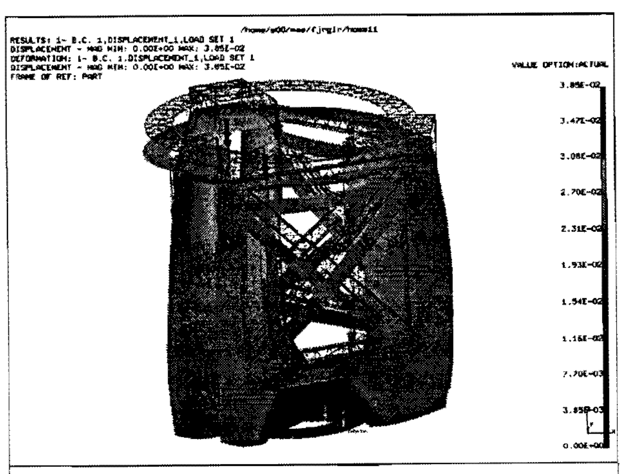


Figure 12: Maximum displacement analysis.

element model for the spacecraft frame. This model is restrained from motion at the bottom (simulating the launch ring), and has forces applied on the various surfaces to represent instruments. A point force is applied at each of the connecting points simulating the weight of the dewar. Elements for the vibrational analysis were larger than that of the static analysis, generally about 3 to 4 inches. The model was restrained at the base and allowed to vibrate freely in all directions. The dewar was modeled as a point mass in the center of the frame, and was connected by rigid rods to each connect point. After finite element models had been created for the different geometries, analyses for stress and vibrations were performed.

8.4 Static Analysis

Using the models previously created, the first type of analysis involved identifying the location of stress concentrations and how the structure deflected under different loads. At first, different parts of the structure were analyzed separately to examine their individual characteristics. These parts were loaded with 1 gravity (g) and 20 g static loadings.

A model of the fin and bay substructure revealed an optimum thickness of 0.25 inches with minimal deflections in the longitudinal direction. Because of their length and tendency to bow, some type of lateral support structure was needed to reduce side deflections. Later, using a model of the full structure, large deflections in the fins demonstrated this point. After including trusses to connect the fins, these deflections were greatly reduced. Figures 11 and 12 show the structure subjected to a 20 g static load. The first shows Von Mises stress concentrations and the other demonstrates deflections. With a few exceptions, the structure adequately supported the dewar and other subsystems.

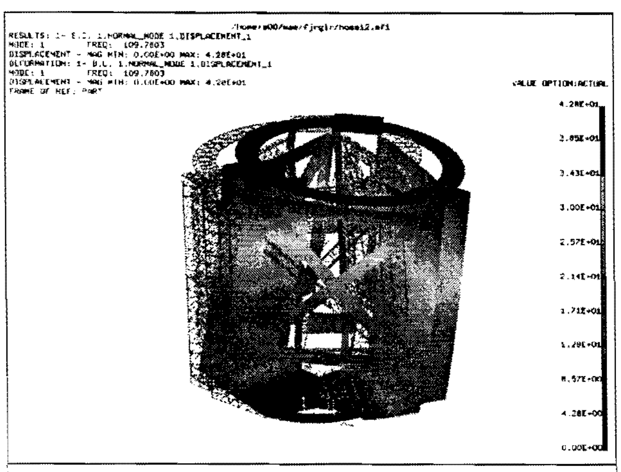


Figure 13: First mode natural frequency diagram.

One location where adverse stress concentrations proved to be a problem was the point where the dewar mounted onto the frame. Stresses of up to 38 kpsi were detected in this area under a 20 g static load. The yield stress for T6 aluminum is 40 kpsi so the stress at this point needed to be reduced. To accomplish this, a set of brackets dispersing the load of the dewar was included. Not only did this reduce the stress concentration at this point but it also made the dewar more stable in vibration. The remaining parts of the structure have deflections and stresses within allowable limits. All deflections and stresses in these images are exaggerated.

8.5 Dynamic Analysis

After examining stress concentrations, the structure was analyzed for responses to vibration. I-DEAS has several types of solvers for normal mode dynamics; the Guyan method, Simultaneous Vector Iteration (SVI), and the Lanczos method. Due to its quick run times (approximately 20 minutes for a large model), good accuracy, and the fact that it is the most widely used method in industry, the Lanczos method was selected. After becoming familiar with the procedure for constraining and solving a model, several frame configurations were tested. This type of analysis solved for natural frequencies and deflections. Typically, a launch vehicle manufacturer will require any satellite to have a first mode natural frequency greater than 30 Hz. Therefore, the design of this structure reflected that requirement.

For a first cut, the frame surrounding the dewar was analyzed. The FEM model was analyzed as a geometric entity and fixed at the base. It was subjected to normal mode vibrations and yielded a first mode natural frequency of 121 Hz. Deflections were limited to 0.10 inches in a side to side motion at the top of the structure. The second mode peaked at 185 Hz and yielded a twisting motion about the base.

To simulate the dewar without modeling its whole geometry, a point mass was modeled at the center of gravity of the dewar. This was attached by rigid connections to the frame at the dewar mount points. After modeling the FEM with similar constraints as before, analysis revealed a first mode natural frequency for the frame and dewar of 109 Hz. The results of this analysis are shown on Figure 13. The results of this analysis demonstrate the spacecraft is well within design requirements specified by the launch provider. Due to the large margin, no further analysis was performed with instruments mounted on the frame. Most instruments can be assumed to have high natural frequencies.

The results from the two above analyses indicate that the current satellite configuration is more than adequate for any type of static or dynamic loading encountered during launch. Stress configurations and vibrational deflections and modes are within all set limits. Therefore, the satellite structure is sufficient for all required needs.

9.0 Conclusion

Designing and manufacturing a custom spacecraft bus for the HOSS technology demonstrator could significantly reduce mission cost and the time required for deployment. The simplicity of the mission requirements allows a simple spacecraft with only a minimum of attitude control, power, and communications onboard. Using the liquid hydrogen dewar as a primary load bearing member reduces the amount of additional structure needed to support instruments. Thus, mass and cost could be minimized. Materials and manufacturing processes were selected from those well established by past programs, and a simple design ensures rapid building. As long as the primary mission requirements were met, supporting the dewar in orbit for 30 days at an appropriate temperature, other minor considerations were disregarded. The offer to launch on the Intrepid launch vehicle provided the opportunity for additional cost savings.

Once a design satisfactory to mission requirements was completed, it was modeled and analyzed for dynamic and thermal responses. I-DEAS Master Series 5 software was used to create FE models and test for deflections, natural frequencies, and temperatures. Initial static loading revealed excessive stress concentrations in the frame which were corrected with the use of mounting brackets. The current structural configuration ensures integrity past a worst case scenario of 20 g's, much higher than loading encountered during launch. Vibrational testing solved for a first mode natural frequency of 109 Hz, leaving a margin of 70 to 75 Hz above the limit specified by launch providers. The structure has sufficient stiffness to limit deflections encountered during launch to acceptable limits. The I-DEAS thermal solver calculated temperatures for the dewar as the spacecraft orbits the earth. As long as certain precautions in component placement and structural connections are taken, the dewar remains within thermal limits. Therefore, the spacecraft meets all structural, dynamic, and thermal requirements.

Given mission requirements, the current spacecraft configuration ensures all objectives could be met at minimal cost and in a reasonable time frame. Once the decision to launch is made, more detailed attitude control, communications, and data handling systems can be designed and integrated with the existing structure. The spacecraft can then be manufactured and prepared for launch.

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