

An Optimized Small Satellite Bus and Structure for the THEMIS mission

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ABSTRACT: The THEMIS project requires the placement of five small spinning satellites, called Probes, into eccentric Earth orbits. The mission requires a lightweight bus with an optimized structure. The final structure design consists primarily of graphite composite panels, titanium and aluminum fittings and brackets, aluminum and non-magnetic steel fasteners, and various thermal isolators. Extensive analysis and development testing was performed on the new composite elements of the structure. This bus design was challenging in every subsystem area. In particular, the Probe structure had to accommodate the complex and sometimes competing set of requirements. The fairing envelope constraint led to a tightly packaged bus with limited external protrusions. The Delta V requirement led to accommodating a complex propulsion system. The temperature extremes drove many aspects of the structure design. The final design meets all of the mission and detailed requirements, achieves a very high packing factor and has extremely low mass. Five identical Probe buses were fabricated. The schedule was demanding and did not allow for an engineering test unit structure. The first proto-flight structure was successfully designed, analyzed, fabricated, assembled and qualification-tested in a historical brief period of time.

INTRODUCTION

The Time History and Macroscale Interactions during Substorms (THEMIS) project, a NASA MIDEX program through the University of California at Berkeley principal investigator team, requires the placement of five small spinning satellites, interchangeably called spacecraft or Probes, into eccentric Earth orbits. To obtain the required orbits, the five Probes will be deployed in nearly simultaneously separations from the spinning Probe Carrier, which is hard mounted to the Delta II 3rd stage. This unique deployment demands a custom-built separation system with a rigorous qualification program. The packaging of five probes within the Delta II fairing limits the envelope of each Probe. Once deployed the Probe requires approximately 900 m/s of delta V to reach its required science orbit and perform its mission. The delta V requirement creates the need for each probe to carry 49 kg of hydrazine via an intricate propulsion system while still maintaining a low-mass probe. The on-orbit mission profile includes long solar eclipses as well as direct sun exposure of all sides of the Probe, which lead to hot and cold temperature extremes. The science instruments drive the probe to have electrostatically-clean external surfaces and a magnetically-clean Probe.

This paper will discuss the overall Probe bus design requirements and the resultant packaging design. The probe bus structural sub-system will be covered in

detail including the size, shape, materials and testing program.

PROBE BUS REQUIREMENTS

The Probe Bus had demanding and sometimes competing requirements. These requirements flowed down and were derived based on Level One science requirements. The main requirements that drove the bus packaging and the structure design are listed below, in no particular order.

- 1) **Propulsion:** The bus needs to accommodate a propulsion system that contains: two fuel tanks, a composite overwrap pressure vessel (COPV) pressurant tank, four thrusters, seven valves, two pressure transducers and 49 kg of hydrazine.
- 2) **Component and instrument accommodation:** All bus components and instruments need to be accommodated in a tight package, be accessible during integration and test (I & T), have clearance for their deployable elements, and have clearance for their fields of view. In addition, certain components or instruments are sensitive to launch environments and the structure has to be designed in such a way to reduce the acoustic environment experienced by these items.
- 3) **Launch Vehicle envelope:** The packaging of five probes within the Delta II fairing limits the envelope of each Probe. The probes require at least 4 inches

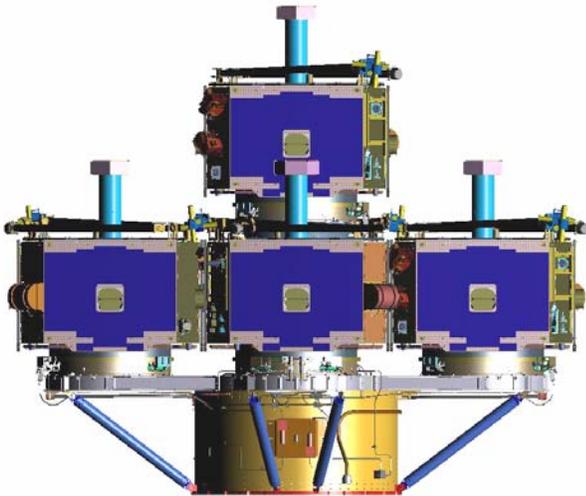


Figure 1: Side View of Probes on Probe Carrier

between each other when mounted on the Probe Carrier to prevent re-contact at deployment. See Figure 1.

4) **Electrostatic Cleanliness (ESC):** All external surfaces of the probe, including the structural elements, are required to be electrically conductive and have a bleed path to spacecraft ground. The only exception to this is that small areas of size .020 inches or smaller are allowed to be non-conductive.

5) **Mass Allocation:** The official mass allocation for all mechanical elements, including solar array substrates, separation system adapter, miscellaneous bracketry, spin balance masses and primary structure is 23.8 kg, or 18% of the not to exceed (NTE) mass of 134 kg for an individual Probe. Of the 23.8 kg about 75% or 18 kg is allocated for primary structure. In addition, the project requested a mechanical sub-system mass much lower than 23.8 kg since other sub-systems were exceeding their mass allocation. So, in effect, the requirement was to “design to minimum mass.”

6) **Spin Balance and Inertia ratio:** The individual Probes have requirements for both center of gravity (CG) offset, 0.18 inches, with respect to the spin axis and principal axis alignment, 1.0 degree, with respect to the spin (Z) axis. The ratio of spin to transverse inertia is required to be greater than 1.04. These requirements determine, in part, the diameter to height ratio of the bus and the available locations for components and instruments to reduce amount of required balance mass.

7) **Thermal:** The Probe bus thermal design requirement is that the Probe has to survive any attitude with respect to the sun and a three hour eclipse. Very little power is available to the thermal sub-system and the majority of the available power is dedicated to keeping the propulsion system warm. This means the

structure has to survive hot and cold extremes and provide extensive thermal isolation. See Table 1.

Table 1. Structure Thermal Requirements

	Min °C	Max °C
Top Deck	-100	85
Bottom Deck	-60	70
Corner Panels	-105	95
Top Solar Panels	-125	115
Side Solar Panels	-120	115
Bottom Solar Panels	-120	115

PROBE BUS OVERVIEW

The Probe Bus is a compact small satellite bus designed for the components and instruments of the THEMIS mission. The fact that the Probe is a spinning spacecraft in orbit very much determined its final configuration. The bus accommodates the following components:

- Bus Avionics Unit (BAU)
- Battery and Auxiliary Electronics Box (AEB)
- Inertia Reference Unit (IRU) assembly
- Antenna and Transponder
- Digital spinning sun sensor
- Body mounted solar arrays
- Separation system
- Harness
- Propulsion system

The propulsion system is distributed throughout the bus and the propulsion system layout was performed in conjunction with the primary propulsion system vendor. The propellant tanks and a majority of the plumbing are symmetric with respect to the spin axis. The thrusters are located to provide spin up, spin down, side thrusting and axial thrusting capability. The propellant tanks mount to the bottom deck directly above the separation system.

The bus accommodates the following instrument suite provided by UCB:

- Quantity 2 Axial Boom (EFI) in center tube
- Quantity 4 Spin Plane Booms (SPB)
- Quantity 2 Solid State Telescope (SST)
- Electrostatic Analyzer (ESA)
- Instrument Data Processing Unit (IDPU)
- Quantity 2 magnetometer booms (FGM/SCM)

Figures 3-6 are CAD images and photographs of the internal and external elements of the Probe.

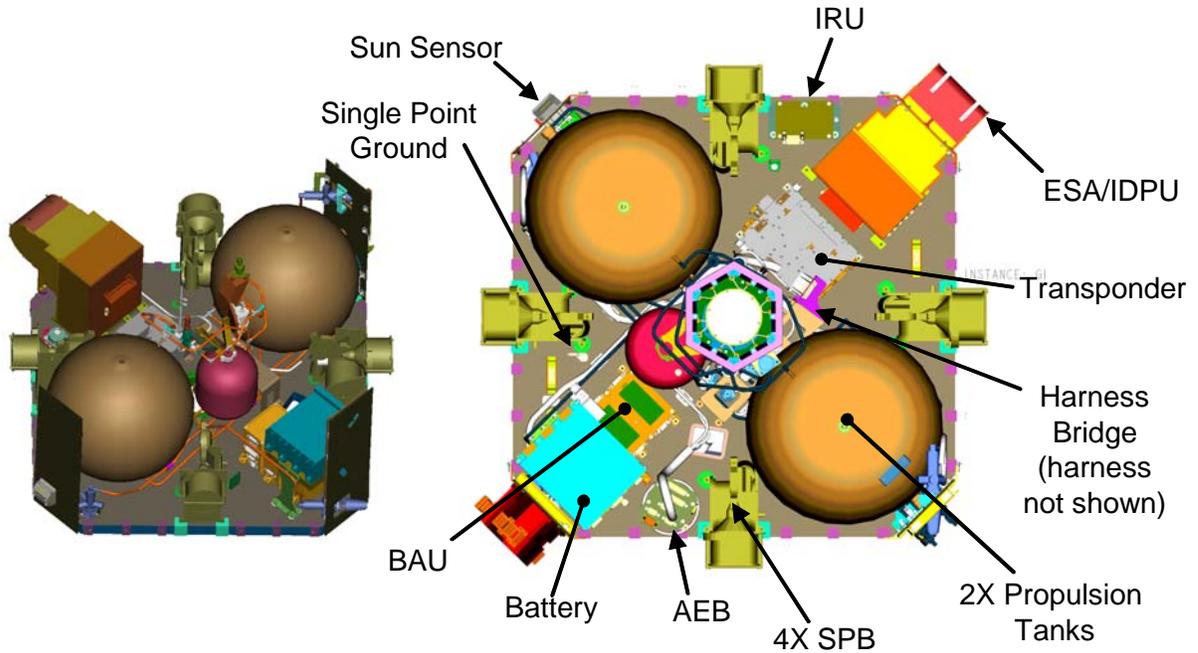


Figure 2. Internal CAD Layout of Probe

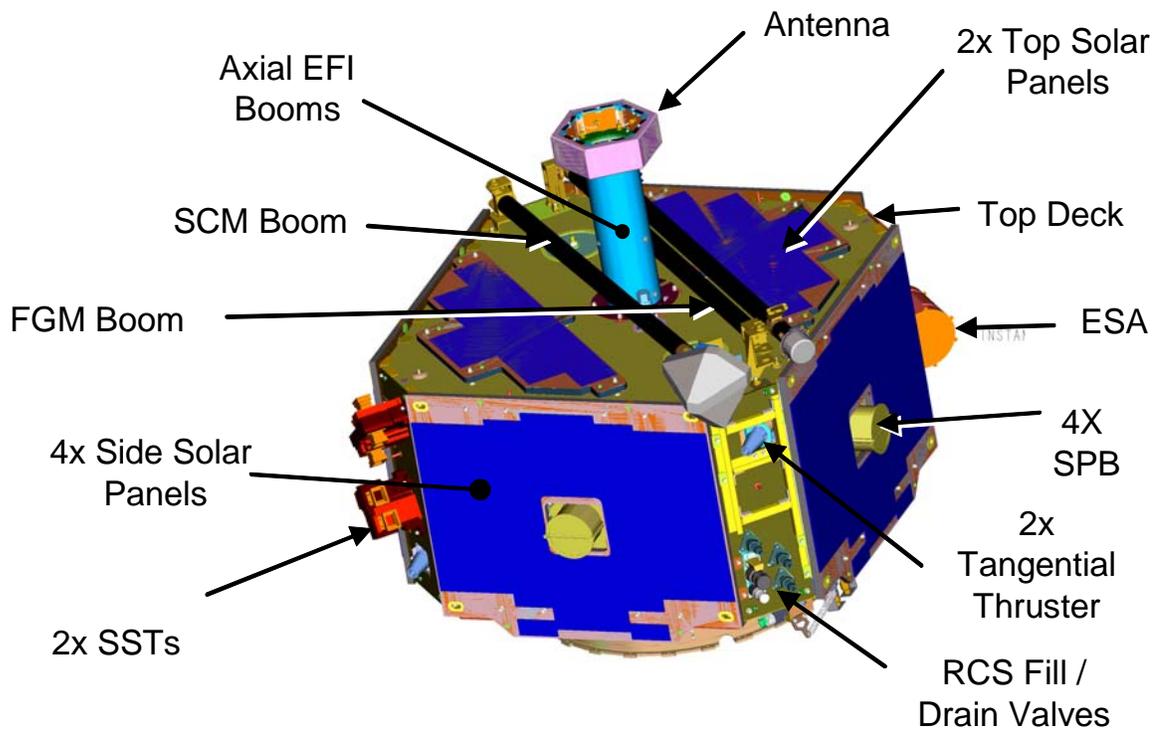


Figure 3: External CAD Image of Probe

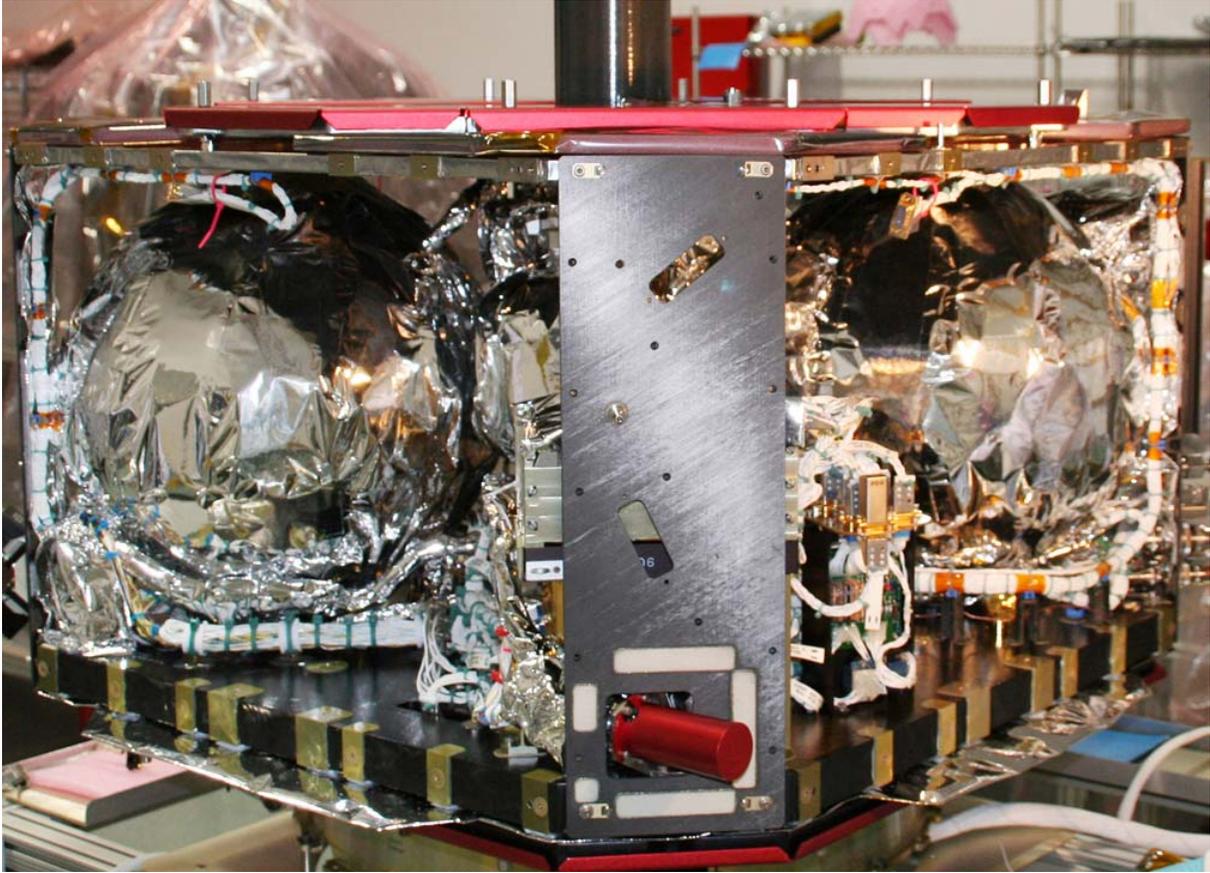


Figure 4. Probe Bus Internal Photo



Figure 5. Probe Bus External Photo

PROBE STRUCTURE DESIGN

A trade study was performed with Aluminum versus graphite composite materials considered for the primary structure. For various reasons, with mass and stiffness being the primary reason, a graphite structure was chosen. The shape of the bus evolved over a period time and was primarily dependent on the nature of the instrument suite and launch vehicle fairing clearance. The final design is a approximately square shaped with eight sides, a top deck, a bottom deck and an axial tube. The axial tube was provided by the instrument team and designed to bus interface specifications. The eight sides form a square with chamfered corners. The corners consist of graphite fiber laminant of various thicknesses. The corner panels provide the primary support for the structure during integration when the side solar panels are removed. The other four side panels are removable body mounted side solar panels. These panels have thin face sheets with low density core. The panels must be removed to gain access for component and instrument integration. The majority of the components and instruments mount to the bottom deck. The top deck supports the magnetometer boom instruments, the top of the propulsion tanks and interfaces with the upper flange of the axial tube. It is a honeycomb panel similar to the solar panel except the face sheets are thicker. The bottom deck is thicker than the top deck or solar arrays, has higher density core and thicker face sheets. See Figures 6 and 7 for an overall layout of the structure.

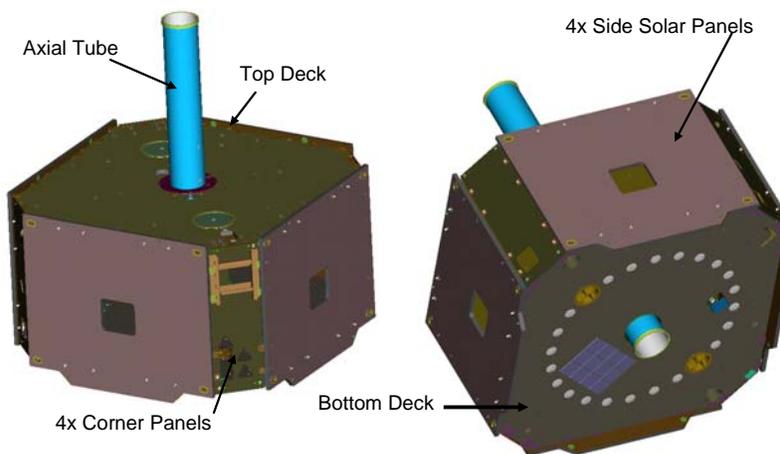


Figure 6. Structure CAD Image with all panels

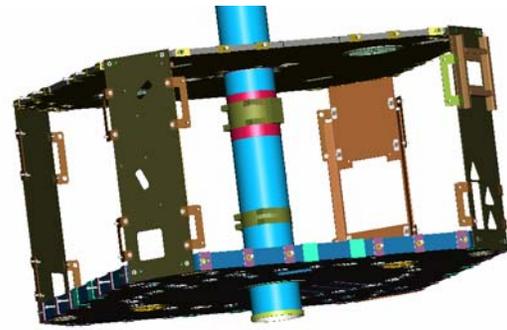


Figure 7. Structure with Solar Panels Removed

The honeycomb panels have embedded inserts. Some of the inserts were placed prior to panel consolidation but the majority were installed after consolidation. The primary choice for insert material was titanium but aluminum inserts were used in select areas for mass reduction reasons. Insert design became a critical element due to a co-efficient of thermal expansion (CTE) mismatch between the components and the bottom deck. The graphite honeycomb panels have a very low CTE while the components and instruments have the higher CTE of aluminum. This mismatch puts large stress into the inserts and face sheets when the boxes reach their extreme hot and cold temperatures. Two examples of this are the BAU and separation system ring. The BAU cold survival qualification temperature is -30°C . The bottom deck inserts that hold the BAU had to be specially designed to survive the stress induced by the aluminum housing of the BAU shrinking due to thermal contraction. In addition to the component induced CTE mismatch the face sheet had to survive the CTE due to local mismatch between the insert, face sheet and potting compounds. Similar stresses were induced by the CTE mismatch of the aluminum separation system ring, which was 22.75 inches in diameter with 24 bolts. The cold qualification temperature for the separation system was -35°C .

The side solar panels are held to the corner panels with thin titanium brackets. These brackets bolt to the back of the solar panel to provide maximum available front-side area for solar cells. Each of the eight side panels have a match drilled slip fit pin at each corner. These thirty-two pins provide the capability for the structure to take shear through controlled load paths. The eight side

panels are offset from the bottom deck with thermoplastic Ultem spacers for thermal isolation. The primary load on the structure is from the 49 kg of fuel split between two propellant tanks. The tanks mount to the bottom deck using the same insert as the separation system. This provides the optimal load path to the separation system for the loads induced by the propellant. The top of the propellant tank interfaces with the top deck via a tightly toleranced spherical pin that only puts lateral loads into the top deck. See Figure 8 for the propellant tank mounting.

To meet the ESC and grounding requirements, every insert on the top deck, and side panels needed to be independently and redundantly grounded to core. Each panel needed a controlled path to single point ground on the bottom deck. Grounding was achieved through the use of conductive adhesive in select areas. In addition, the outer composite surface of each panel needed to have sufficient surface conductivity to meet the ESC requirement. This surface conductivity was achieved through material selection, careful preparation of the material outer surface and use of thermal coatings.

The final bus design met and exceeded the inertia ratio requirement of 1.04. The current on-orbit prediction for the design is 1.30. Two kilograms were allocated for spin balance masses. Current analysis shows that the Probe is naturally quite balanced and will need approximately 0.5 kg of balance mass. Due to volume constraints these masses will be tungsten.

The final, as-weighed, structure design mass was significantly less than the allocation, which at the time of Phase A was considered a potentially challenging allocation. Table 2 below compares the actual mass values to the allocations.

Table 2. Final Mass versus Allocation in kg

	Allocation	Final Mass	% of Probe Dry mass*	% of Probe Wet mass**
Total Mechanical	23.8	19.8	24.2%	14.8%
Structure only	18	14.4	17.6%	10.3%

Notes:

- * The not to exceed(NTE) Probe dry mass is 81.8 kg.
- ** The probe structure design requirement is 134 kg of wet mass. The actual Probe on-orbit mass will be approximately 126 kg.

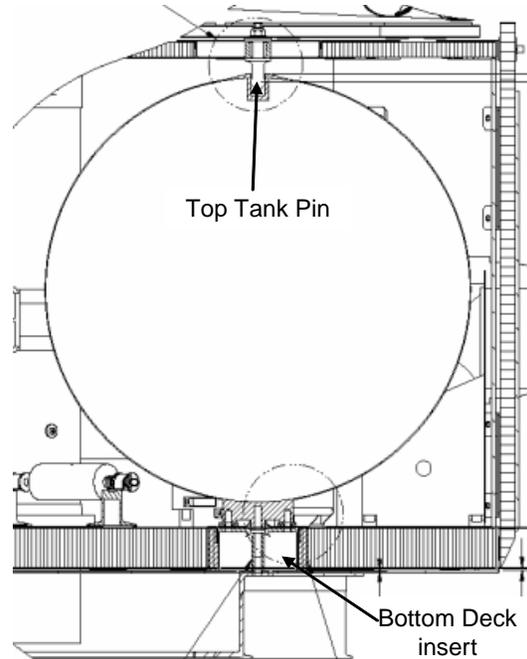


Figure 8. Propellant Tank Mounting

STRUCTURAL ANALYSIS

Analysis efforts focused on demonstrating that the probe would survive the launch loads environment and on-orbit thermal environment. Due to the unique Probe Carrier configuration the launch loads were developed specifically for this mission. This effort included multiple coupled loads analyses and a detailed acoustic analysis. The on-orbit thermal requirements were flowed down from the thermal sub-system and were discussed previously in this paper. The dominant load case for the structure was the quasi-static simultaneous lateral 10.2 G and axially 6.03 G. Design factors of safety were based on NASA specifications for “test” factors and are shown below.

Table 3. Design Factors of Safety

	Yield	Ultimate
Metallic and machined elements	1.25	1.4
Composites	-	1.5
Bonded Joints	-	1.5

A structural math model was developed for the 134 kg Probe and it yielded a normal first mode of 40 Hz. The derived requirement was 35 Hz. See Figure 9.

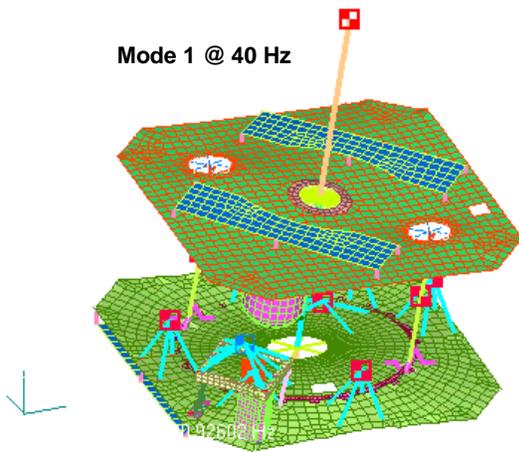


Figure 9. First Normal Mode of Probe (side panels removed for clarity)

For mounting of metallic components onto the graphite structure extensive analysis was performed for CTE mismatch. This analysis included nearly every type of insert, component mounting and local geometry of the bottom deck. CTE mismatch causes a high shear load at screws and inserts. The most likely failure mode was determined to be at the composite face sheet due to either bearing against the insert, combined interlaminar tension and shear or combined compression and tension. Figure 10 shows a typical thermally induced stress strain analysis for the aluminum separation system interface ring.

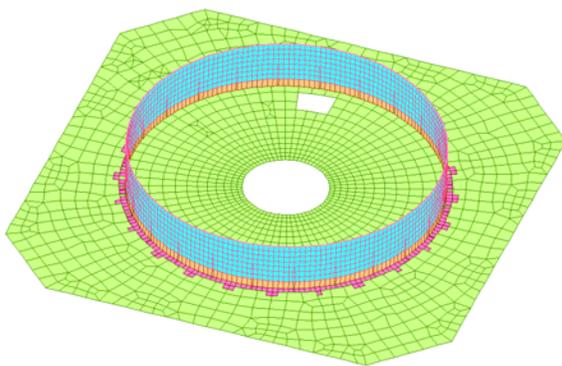


Figure 10. Model for CTE Analysis of Sep System

CTE mismatch analysis lead to an insert spacing criteria for the honeycomb panels. Inserts that are placed too closely put loads into the face sheet that exceed the allowable. The spacing was dependent on insert material plus other factors, and sometimes drove the

material selection. Figure 11 shows an example of a face sheet failure that occurred due to placing inserts too closely together. Figure 12 shows a view looking through the bottom deck area for the IDPU/ESA mounting area. This complex area has overlapping inserts from both sides of the deck and overlapping adhesives.



Figure 11. Top Deck Face Sheet CTE Failure

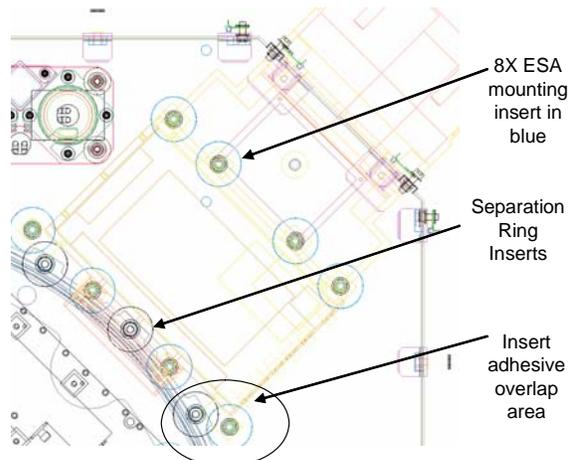


Figure 12. Overlap of Inserts and Adhesive in Bottom Deck

DEVLEOPMENT TESTING

Since the majority of the insert designs were unique for this structure extensive development coupon testing was performed prior to finalizing the design. The approach consisted of designing a coupon that represented the flight element of interest, thermal cycling the coupon and then performing destructive mechanical testing. This process proved to be quite valuable and key to gaining confidence in design elements with little or no previous flight heritage. In

some cases the design was more capable than the analysis indicated and in at least one case the reverse was true. Figure 13 shows a coupon that was used to demonstrate various elements of the top deck insert design. Figure 14 shows a coupon during destructive testing.



Figure 13. Top Deck Test Coupon

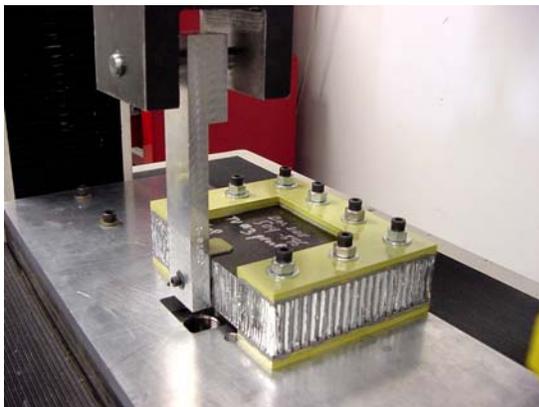


Figure 14. Coupon Destructive Testing

Development testing was also required to make final adhesive selections. There was little or no manufacturer's data available for the desired adhesives. The Goddard Space Flight Center (GSFC) materials branch performed testing that helped determine the glass transition temperature (T_g) and modulus of elasticity versus temperature of the potting compound. Testing was performed on multiple coupons that were cured at different temperatures as some adhesive was cured at elevated temperature and while others were cured at room temperature.

STRUCTURE ASSEMBLY AND TEST

The structure assembly and test sequence began when the individual panels were completed. The overall sequence is shown in Figure 15. Two tests comprise the strength qualification of the structure, thermal cycle and sine burst. Due to the very different thermal

requirements for each individual panel the panels were tested by type. The top deck, corner panels and solar arrays were straight forward thermal cycle tests. Since the majority of the aluminum components were mounted to the bottom deck, its test was significantly more complicated. Each major component had to have a stiffness mock-up mounted to the deck. Since the components had different temperature extremes and the deck design did not have sufficient margin to test to the most extreme component, a bottom deck thermal test with multiple heater zones was performed. This test involved heaters and temperature controllers that kept the various stiffness mock-ups at the appropriate temperature. Figure 16 is a photograph of the test set-up.

Once the panels completed thermal cycle testing the structure assembly began. The assembly process involved carefully designed tooling that was used to properly align the structure and match drill the various pin holes. The individual solar array panels had a requirement to be interchangeable among all five probe structures. To meet this requirement a single template was used to drill all of the pin holes in the solar arrays, top and bottom decks. Figure 17 is a photograph of the structure during assembly.

Once assembly was complete the structure was configured for sine burst testing. The sine burst test configuration involved component and instrument mass and CG mock-ups or ETUs. For the propulsion system two flight like tanks were used and filled with water. All mock-ups were secured using flight fasteners per the flight drawing and torque values. Instrumentation, via three axis accelerometers, was installed throughout the areas of interest in the test article. Once the structure was fully configured for testing alignment measurements were taken using a portable coordinate measuring machine (CMM) to characterize the pre-test alignment. The completed test article was shipped to GSFC for testing. Figure 18 shows the test article configuration with one solar panel removed.

The sine burst test consisted of three individual tests of one per axis to 1.25 times the limit loads. The GSFC GEVS-SE guideline of testing all composite elements of series hardware, as opposed to qualification by similarity was observed. This meant that each individual structure was tested to the same test plan. All structures passed the tests without issue and the data was used to correlate the finite element model. Figure 19 shows the test article on the GSFC vibration table for Z axis testing.

After the test each unit was inspected, including 100% torque check, disassembled and inspected again. The

complete disassembly of a primary spacecraft structure after testing is likely unique to this program. The reason for this requirement is that every structural panel except the corner panels needed to go to another location, after testing, for further processing. The bottom deck went to the propulsion vendor to have the propulsion system installed. The top deck went to

GSFC to have a custom vapor deposit gold coating applied. Finally, the solar panels went to the cell vendor to have the solar cells and ESC treatment applied.

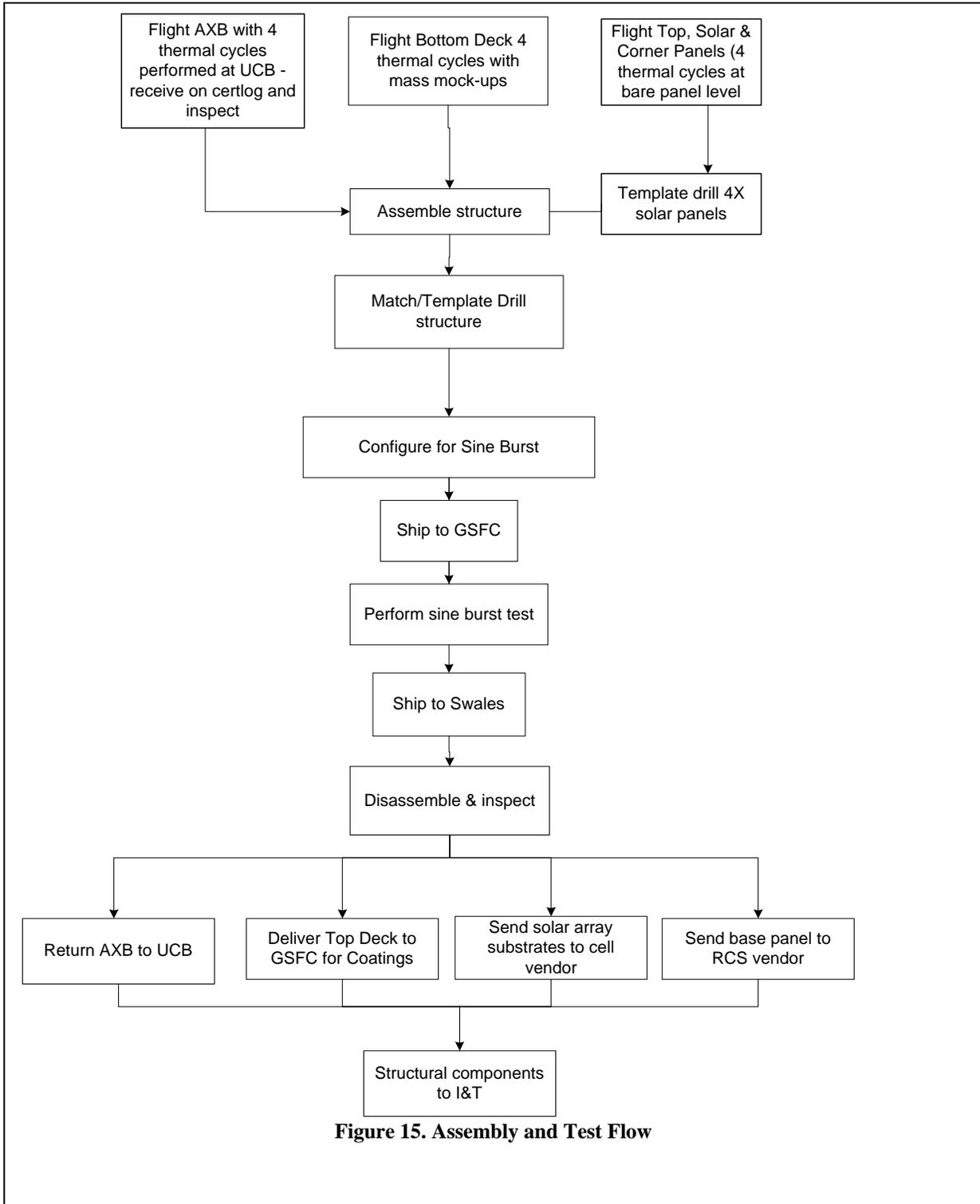


Figure 15. Assembly and Test Flow



Figure 16. Bottom Deck Thermal Cycle Test

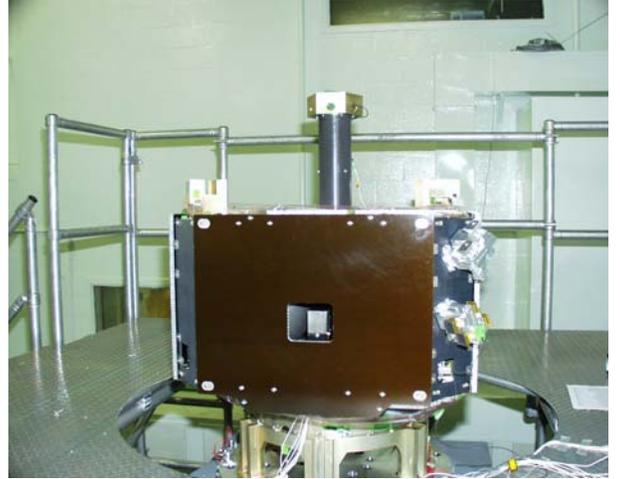


Figure 19. Sine Burst Testing



Figure 17. In-process Structure Assembly

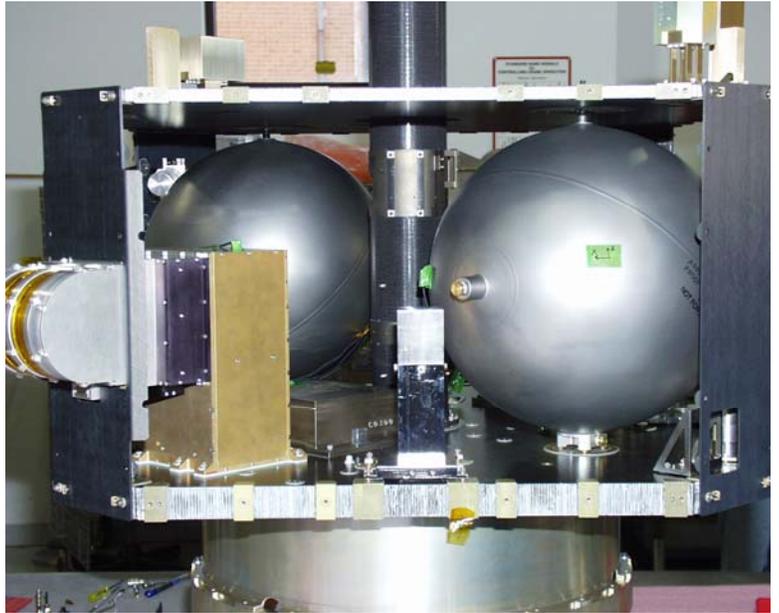


Figure 18. Sine Burst Test Configuration

STRUCTURE AND BUS I&T

Once the structural elements returned from the various vendors spacecraft integration and test began. The first step of I&T was to reassemble the primary structure. There was some concern that the precision pin holes would not re-align with the addition of various elements by the vendors. This proved not to be the case. The fact that we had no difficulties re-assembling this precision structure is a testament to the quality of the tooling and manufacturing workmanship that was applied to all five structures. At the time of writing this paper all five buses have been integrated and shipped to the customer for instrument integration. The first bus to be delivered has been integrated with instruments and been through environmental testing which was successfully completed. Figure 19 shows the bottom deck and corner with an installed propulsion system, as it was received from the propulsion vendor .

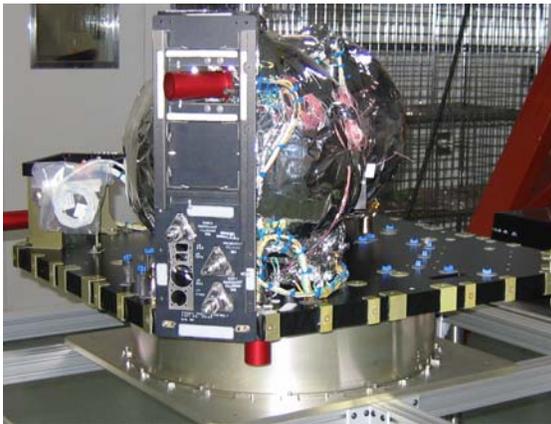


Figure 19. Bottom Deck with Propulsion System
PROGRAM NOTES

This was a challenging program from both the technical and program points of view. A completely new low risk bus had to be developed for a NASA mid explorer class mission. The structure design was dependent on the items it was to carry and for various reason these items were late to be defined. Schedule savings were achieved, with some risk, by concurrently designing the structure panels and the tooling that would assembly the panels. In addition, insert designs were completed and released to manufacturing prior to the completion of the panel assemblies. We were able to perform these various steps concurrently because we had all of the design, analysis and manufacturing capability in one location. With that said, the structure design, manufacturing, assembly, test and delivery schedule was thought to be the program critical path but turned out to be completed well ahead of the critical path. Table 4 below lists some of the major milestone for the structure.

Table 4. Structure Schedule

Item	Date	Month#
System Requirements Review	July 2003	2
Preliminary Design Review	Oct. 2003	5
Critical Design Review	June 2004	12
Panel fab complete 1 st unit	Jan. 2005	19
Assembly & test complete – delivery 1 st unit	March 2005	21
Assembly & test complete – delivery 5 th (final) unit	Aug. 2005	26

CONCLUSION

The THEMIS bus structure and bus packaging design met all of the requirements and goals of the THEMIS mission. In the case of the mass allocation, the graphite composite bus exceeded the requirement. The structure mass is only approximately 10% of the design wet mass of the Probe. For a small, one of a kind, satellite, this is a historical small percentage. The graphite structure employed existing technology and materials in unique geometries for its design. The launch and on-orbit environments were somewhat typical of other missions but the combination of tight requirements and the uniqueness of the Probe Carrier caused the structure design to be tailored for these environments. The structural analyses were comprehensive, critical to the success of the design, and may be unique to this program. Five nearly identical bus structures, were fabricated, assembled, tested, disassembled, shipped to vendors, and reassembled in a relatively brief period of time.

ACKNOWLEDGEMENTS

This material was based on worked performed for the THEMIS project under NASA contract NAS5-02099. The following individuals contributed significantly to this effort: Dr. Chia Chung Lee, David Heckle, Dr. Ben Rodini, John Pindell, Chris Lashley, Tim Keepers, Roger Johnson and Gale Barnes.

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