Structural Design of the MSX Spacecraft

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The Midcourse Space Experiment (MSX) spacecraft represents the largest structure ever assembled at the Applied Physics Laboratory. The design of the structure involves the application of conventional aluminum framework/honeycomb panel construction with composite materials technology to satisfy both the launch vehicle induced loads and the dimensional stability requirements of the optical sensors and instruments. The 1.59-m^2 by 4.11-m-long structure is divided into three discrete elements consisting of (1) an electronics section, which provides an interface to the Delta II launch vehicle and a mounting platform for instrument and spacecraft electronics; (2) a thermally stable graphite/epoxy composite truss structure for mounting the Spatial Infrared Imaging Telescope III (SPIRIT III) instrument and providing an interface between the instrument and electronics sections; and (3) a temperature-controlled instrument section with embedded heat pipes for mounting delicate sensors and instruments. The instrument section also provides a thermally stable mounting structure for the optical bench and the beacon receiver bench. This article describes the features of the MSX structural design, the structural analysis, and testing efforts associated with qualifying the design. A description of the efforts involved in the development of the graphite/epoxy composite optical bench and beacon receiver bench is also presented.

INTRODUCTION

The Midcourse Space Experiment (MSX) spacecraft, sponsored by the Ballistic Missile Defense Organization, is an observatory that provides a multisensor platform for collecting background, target detection, and tracking phenomenon data. Because of the precise pointing and tracking requirements for the spacecraft, the physical layout of the scientific instruments and sensors and of their electronics packages is critical to mission success. The 5.1-m-long (including optical sensors), 2812-kg spacecraft (Fig. 1) is divided into three discrete sections: the instrument section, the truss section, and the electronics section (Fig. 2). The instrument section contains 11 sensitive optical sensors that are precisely aligned relative to one another so that targets can be viewed and tracked simultaneously by all sensors. The structure of the instrument section contains honeycomb panels with embedded heat pipes to maintain all sensor mounting surfaces at a uniform temperature regardless of the spacecraft's orientation relative to the Sun. Maintaining co-alignment of the optical sensors is critical during target tracking events. The center section of the spacecraft contains the graphite/epoxy (G/E) composite truss structure that supports the Spatial Infrared Imaging Telescope III (SPIRIT III). The truss structure connects the instrument section with the electronics section while thermally isolating the sensors from their heat-generating electronics packages. The electronics section contains the sensor electronics packages in addition to electronics for the attitude control, power, command, and telemetry subsystems necessary to operate the spacecraft. The aft end of the electronics section structure contains the cylindrical payload adapter for mating the spacecraft to the Delta II launch vehicle.

The instrument section also contains two ancillary structures that play an important role in the tracking and attitude control of the spacecraft. The optical bench is a precision inertial measurement platform that is attached to the top of the instrument section structure (Fig. 2). The G/E composite bench supports the ring laser gyroscopes and a star camera for establishing and maintaining spacecraft attitude control. The beacon receiver bench is a precision antenna platform that is attached to the $-y$ axis honeycomb panel for launch (Fig. 2). Once orbit is achieved, the bench deploys into a position perpendicular to the $-y$ axis panel cantilevered off the top edge of the instrument section. The G/E composite bench contains four S-band receiving antennas that can acquire and lock onto a target signal beacon at frequencies of 2219.5 and 2229.5 MHz during spacecraft tracking activities.

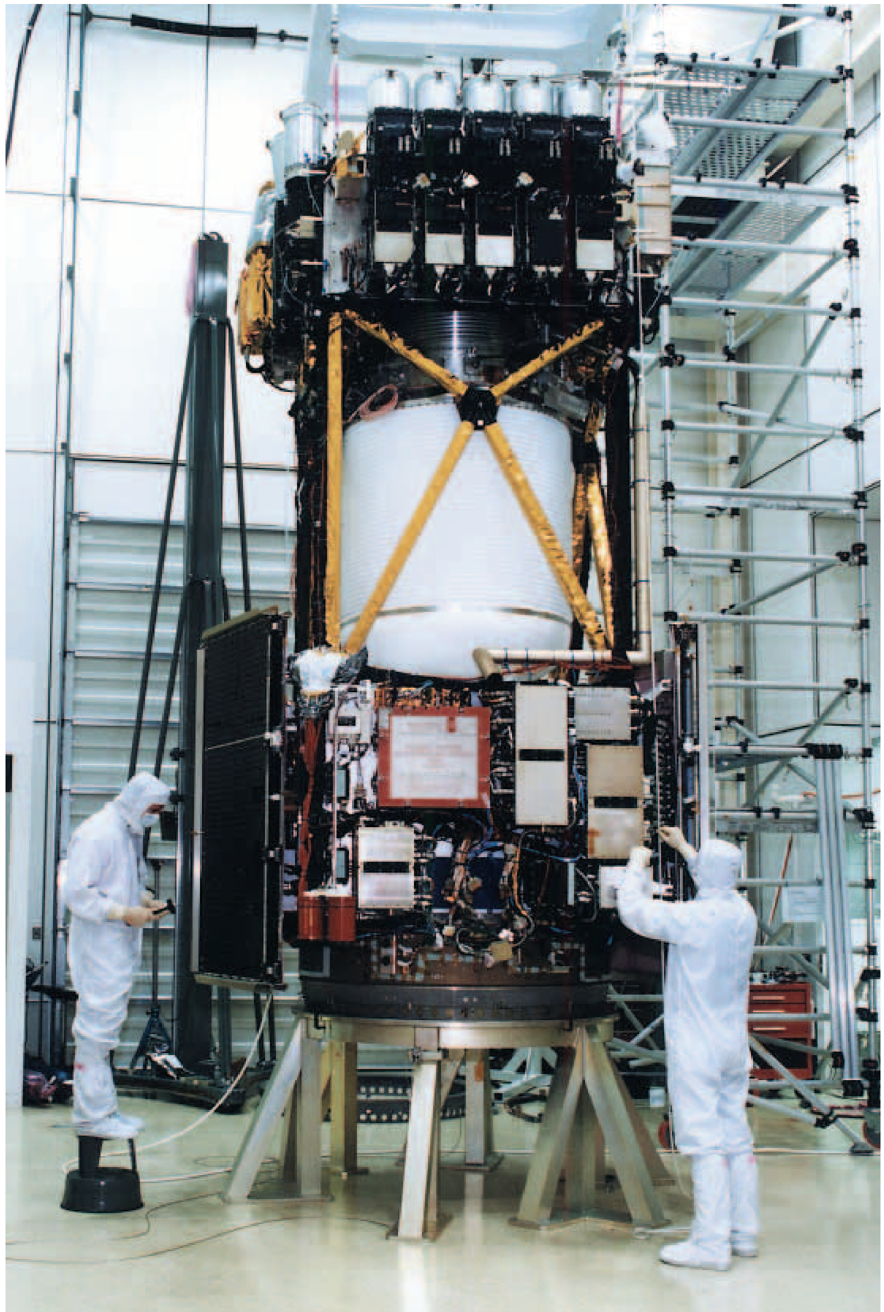

Figure 1. The MSX spacecraft in APL's Kershner Space Building.

ELECTRONICS SECTION

Two important functions in the mission of the MSX spacecraft are performed by the electronics section. First, it is the major structural element in the spacecraft bus providing a direct load path for transferring launch-induced spacecraft loads into the Delta II launch vehicle. Second, it provides the mounting platform for electronics packages associated with the spacecraft attitude control subsystem, spacecraft power and power distribution subsystem, spacecraft commanding subsystem, and spacecraft telemetry subsystem. Figure 2 shows the MSX spacecraft with the electronics section

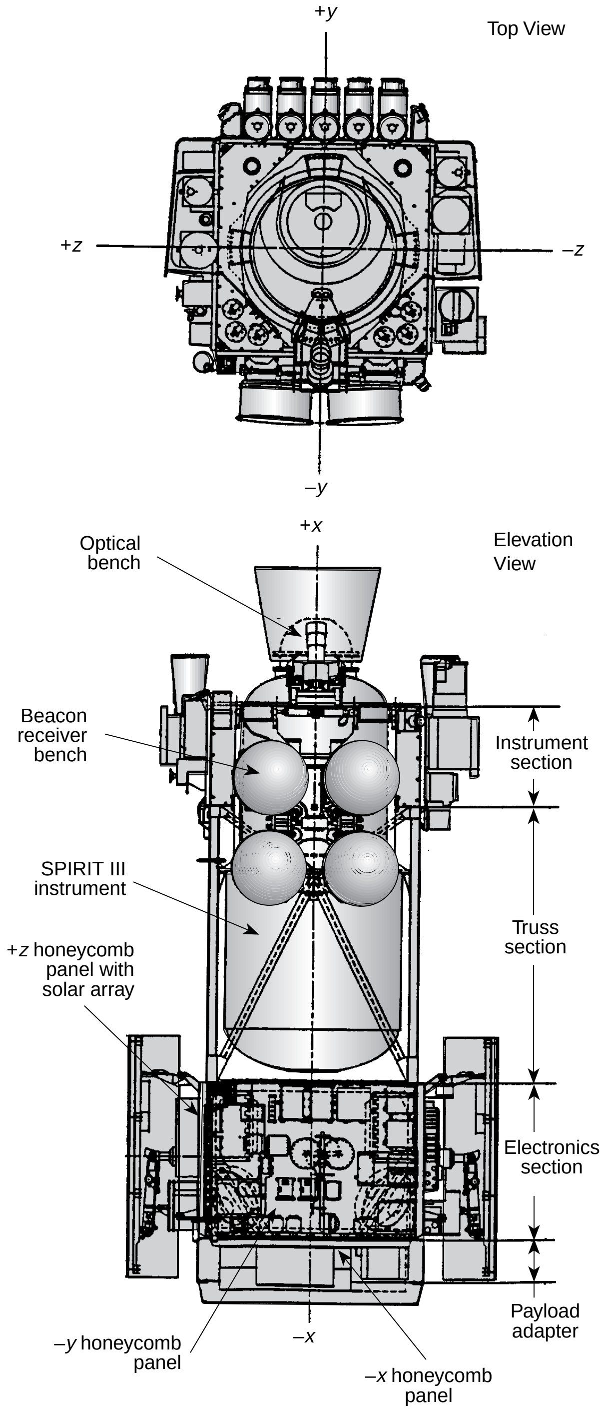

Figure 2. The MSX spacecraft configuration (stowed).

visible immediately beneath the truss structure. The electronics section is a hollow rectangular structure fitted with a cylindrical payload adapter to interface to the payload attach fitting on the second stage of the Delta II launch vehicle. The overall dimensions of the rectangular structure are 1.59 m^2 by 1.12 m long, and the overall dimensions for the payload adapter are 1.60 m in diameter by 0.30 m long. The rectangular structure consists of an internal skeletal frame covered on all sides by aluminum honeycomb panels. The interior and exterior of the honeycomb panels are densely populated with electronics packages used to power and operate the spacecraft subsystems. In addition to providing the square-to-round transition necessary to mate the spacecraft to the launch vehicle, the payload adapter provides an overflow location for mounting electronics packages and the electrical umbilical connections to the spacecraft.

The skeletal frame of the electronics section is composed of several intricately machined aluminum structural shapes and fittings that were precisely aligned and assembled using special high-tensile-strength fasteners. The assembled frame includes a unique triangularly shaped bracket used to mount the attitude control subsystem's reaction wheels in each inside corner. The reaction wheel bracket was designed to precisely align the wheel's axis of rotation at 0.44 rad to the y/z spacecraft axes. The 0.44-rad inclination of the four reaction wheels provides redundant attitude and guidance control of the spacecraft in the event of a single reaction wheel failure. The base of the frame includes a specially machined baseplate that initiates a smooth transition from the square cross section of the electronics section to the round cross section of the Delta II launch vehicle. The transition is completed by attaching a cylindrical payload adapter fitting to the underside of the frame baseplate. The skeletal frame was designed and fabricated at APL. Parts of the electronic section frame, the reaction wheel bracket, and the baseplate can be seen in Fig. 3.

The four sides ($\pm y$ and $\pm z$ axes as shown in Fig. 2) and the bottom ($-x$ axis) opening in the frame are covered by 5.2-cm-thick aluminum honeycomb panels with 0.6-mm-thick aluminum face sheets and integral machined edge members. The honeycomb panels are individually bolted to the faces of the frame to provide the lateral stiffness required for the entire assembly. The electronics section was designed so that the honeycomb panels on the $\pm z$ axes sides of the spacecraft can be removed to provide access to the interior of the frame. The hollow center of the frame is used to house several attitude control and data storage devices.

Figure 3 shows the interior of the electronics section frame as viewed with the $+z$ honeycomb panel removed. Visible inside the frame are the following devices:

- Four each — Reaction wheel assemblies mounted on triangularly shaped brackets in all four lower corners.
- Three each — Torque rod assemblies mounted on the inside surface of the $-x$ axis and $+y$ axis honeycomb panels.
- One each — Reaction wheel control electronics package mounted on the inside surface of the $-x$ axis honeycomb panel.

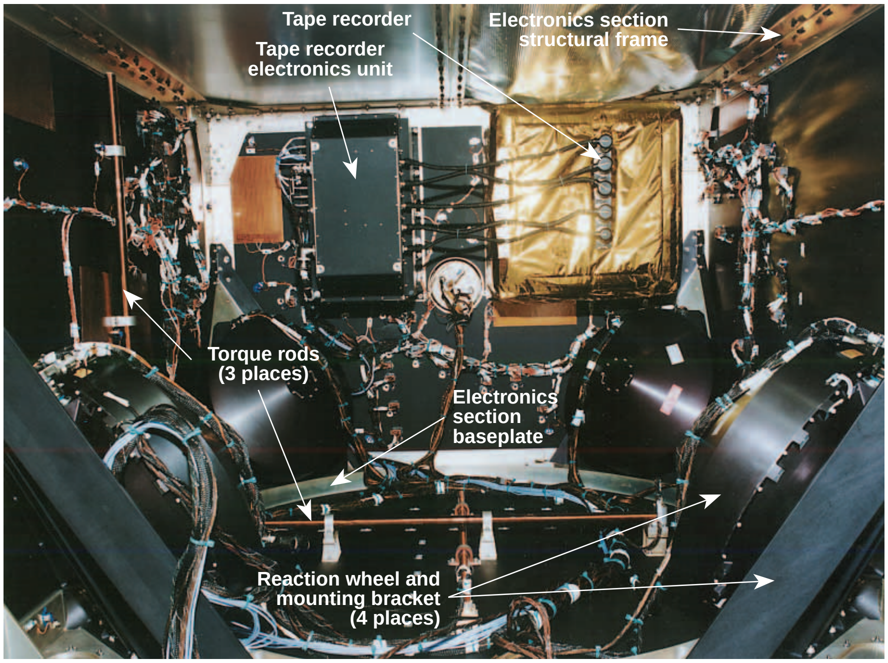

Figure 3. Interior of the electronics section.

- Two each — Tape recorder assemblies (composed of separate electronics and recorder packages) mounted on the inside surfaces of the $\pm z$ axes honeycomb panels.

The exterior surfaces of the $\pm y$, $\pm z$, and $-x$ axes honeycomb panels are used to mount an assortment of electronics packages, communication antennas, solar arrays, batteries, and associated support brackets. The honeycomb panels contain threaded inserts, bonded integral with the panel, used to attach the flight packages. The $+x$ axis side of the electronics section is covered by two half panels, each 1.4 cm thick with 0.6-mm-thick aluminum face sheets and integral spools for attachment to the panels. This structure is primarily required to provide the necessary in-place stiffness for the top of the electronics section structure.

The cylindrical payload adapter was designed jointly by APL and McDonnell Douglas Aerospace and fabricated by McDonnell Douglas Aerospace. The adapter, which completes the transition of the frame from the square-to-round cross section, is 1.60 m in diameter by 0.30 m long with a skin thickness of 4.3 mm. The upper flange of the adapter contains a 48-hole bolt circle that attaches to the baseplate of the frame with special high-tensile-strength fasteners. The lower flange of the adapter is machined to mate with the McDonnell Douglas 6306 payload attach fitting and separation clampband assembly. The hollow center section of the adapter surrounds the electronics packages mounted to the exterior surface of the $-x$ axis honeycomb panel. The primary hardware mounted in this area includes the spacecraft battery and power distribution electronics packages.

SOLAR ARRAY SUBSYSTEM

The MSX solar array subsystem consists of two solar array wings deployed along the z axis of the spacecraft. Each wing consists of four panels in a bifold configuration attached to a drive boom that will acquire and point the solar array toward the Sun. Figure 4 illustrates the deployed solar arrays and their attachment to the electronics section $\pm z$ axes honeycomb panels. The entire surface of each 1.73×0.78 m panel is covered with 3×6 cm silicon solar cells. The complete solar array subsystem uses 4992 solar cells to generate a beginning of life power of approximately 1250 W at 34 V DC.

The solar arrays are located in the center of the $\pm z$ axes honeycomb panels. The panels are spaced off the panel surface (0.10 m) by a solar array drive adapter to provide sufficient room behind the stowed panels for mounting electronics packages. Figure 5 shows the folded panels and the drive boom in the stowed configuration. Two active latches and four passive latches are used to restrain the solar panels and boom to the honeycomb panel during prelaunch and launch. The active latches contain the pyrotechnic pin puller assemblies that allow the folded panels to deploy once orbit has been achieved. The passive latches are used to guide the folded panels as they unfurl to form the deployed solar array configuration. A system of torsion springs, compression springs, and liquid dashpot actuators is used to safely drive the solar boom and panels from the stowed to the deployed position. Active pointing of the solar array is controlled by the solar array control electronics unit, which is located on the $-z$ honeycomb panel of the electronics section. Two pitch Sun sensors, located on each wing, provide inputs to the array control electronics unit, which in turn drives the boom to track the Sun. The drive system can rotate a solar wing through a full 360°.

TRUSS STRUCTURE

The truss structure of the MSX spacecraft, built by Composite Optics, Inc., of San Diego, California, provides the primary support for the SPIRIT III instrument and the instrument section. The critical requirements

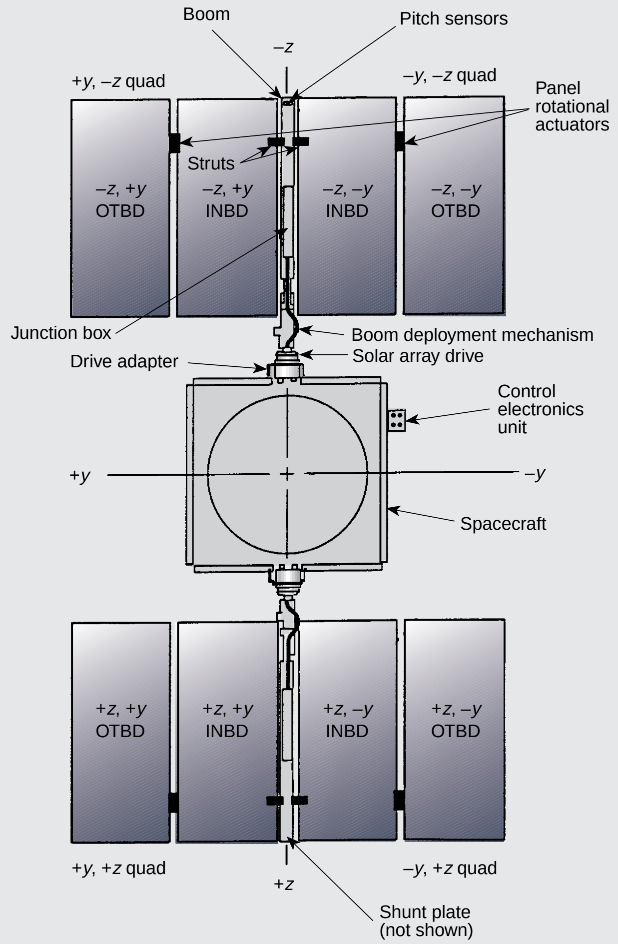

Figure 4. MSX solar arrays: deployed configuration. Arrays are rotated with the panel "cell side" shown. INBD, inboard; OTBD, outboard; Quad, quadrant.

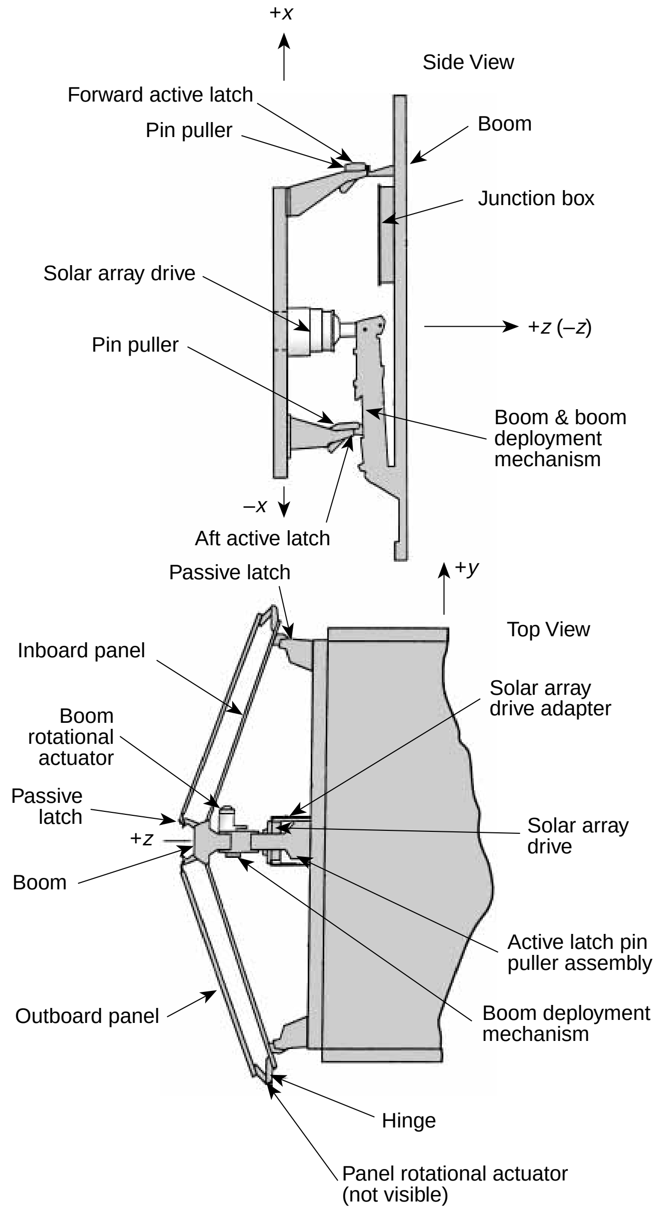

Figure 5. MSX solar arrays: stowed configuration.

that drive the truss structure design are stiffness (spacecraft frequency of 12 Hz lateral and 35 Hz axial), rotational stability of the SPIRIT III mounting plane (0.01°), mass (less than 54.4 kg), and strength (maintain positive margins of safety). A low coefficient of thermal expansion (CTE) (0.54×10^{-6} in./in. per °C) and control of dimensional changes caused by absorption and desorption of moisture were required to achieve the stability requirement. A low thermal conductivity (less than 5.0 W/m·°C) was necessary to minimize the heat transfer between the electronics and instrument sections. Additional requirements considered in the development of this unique part of the MSX structure included stability performance when exposed to temperature gradients as a result of the colder than normal temperature environment (+*y* side, −78°C; −*y* side, −67°C; +*z* side, −71°C; −*z* side, −67°C), survival as a result of temperature extremes (−90°C to +90°C), and a grounding impedance of ≤50 mΩ. This combined set of requirements necessitated the use of composite materials technology in lieu of more conventional materials such as aluminum for the fabrication of the truss section members.

The truss structure spans the full width of the structure and is 1.97 m high. It interfaces at four corners to the top of the electronics section and the bottom of the instrument section (Fig. 2). The assembly uses 20 G/E I-beams connected together with titanium fittings or G/E plates. Three different sizes of I-beams were used to construct the truss; they are classified as lower diagonals, upper diagonals, and verticals. Unique titanium fittings were designed to join the I-beams together and to establish the required interface for the SPIRIT III instrument. The overall configuration for the truss structure (Fig. 6) indicates the typical locations of these

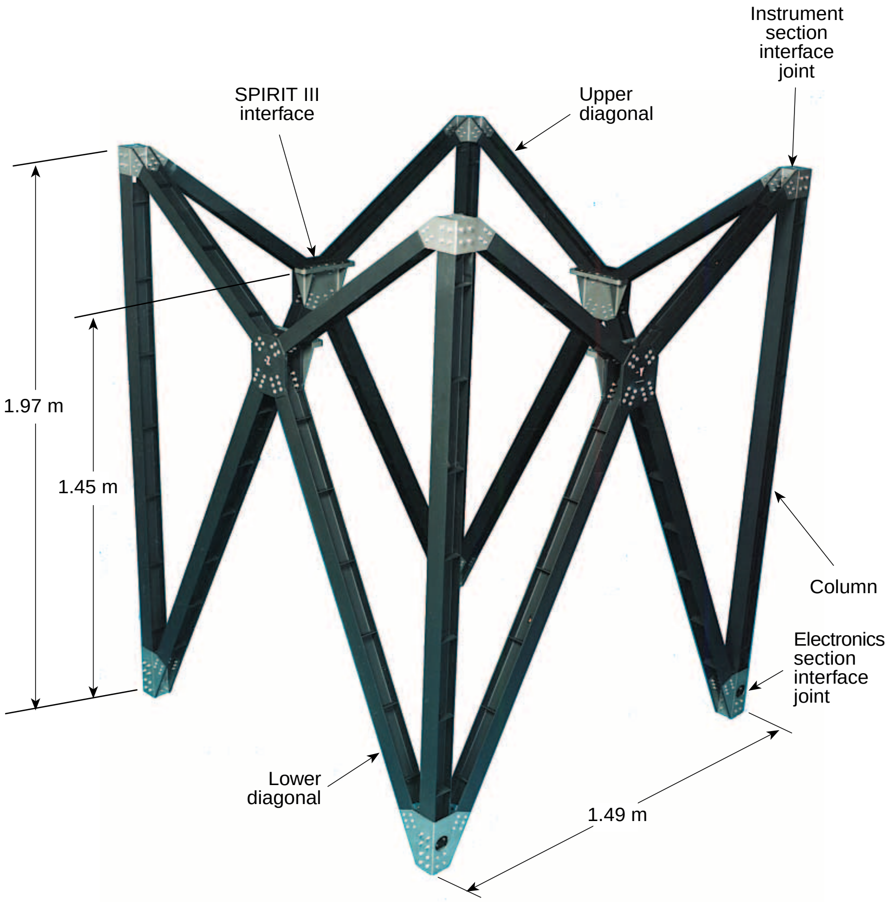

Figure 6. MSX truss structure.

member groups. Figures 7 and 8 depict the different titanium fitting assemblies and illustrate how the detailed parts are interconnected. Special blind fasteners (Composi-loks) were used in these bolted joints to provide a large blind-side footprint that distributes the bearing loads over a large area, thus preventing composite fiber delamination. High-strength titanium fasteners were used at the SPIRIT III instrument, electronics section, and instrument section interfaces. Bushings were used in the electronics section and instrument section mounting holes to provide a close tolerance fit, thus minimizing the possibility of structural movement during the launch environment. Harness clips were bonded to the truss members to provide a means for securing the harness, which spanned the instrument and electronics sections.

The truss elements were made from a T50/ERL1962 G/E composite material configured in a pseudoisotropic laminate stacking sequence of $(0_2°/60°/-60°)_{S3}$. T50 graphite fibers were chosen because of their minimal thermal conductivity and high stiffness characteristics. Amoco's ERL1962 resin was selected because it is a toughened resin system that has excellent mechanical properties and exhibits good resistance to microcracking. Composite materials can be tailored to enhance properties in specific directions of the material, i.e., the material is "directionalized." For the truss, the T50/ERL1962 combination is directionalized to provide enhanced stiffness. Testing was performed to characterize the stiffness, strength, bolt bearing capacity, CTE, and shear strength. The material properties are shown in Table 1. The structure used joint configurations featuring double lap shear. A specimen test was performed to accurately evaluate the load transfer of the G/E I-beam section to the titanium corner fitting. In-plane and thru-thickness thermal conductivity measurements showed that the thermal conductivity was approximately 4 to 5 times greater than the targeted thermal conductivity. As a result of these tests, other potential fiber candidates were reevaluated; however, it was concluded that the T50 fiber selected was the best for this application.

The titanium fittings and G/E plates were machined precisely to ensure proper assembly with the mating I-beams. The I-beams required special tooling to control their thickness within tolerance during the G/E lay-up. These elements were assembled using special fixtures in order to maintain the overall dimensional requirements. Prior to assembly, a eutectic coating was applied to individual G/E elements to control moisture absorption and desorption. Each member was then painted with Chemglaze Z306 flat black polyurethane paint. The eutectic coating, or moisture barrier, consisted of a Composite Optics, Inc., proprietary process that used a precise combination of tin and indium applied to an exact thickness (0.0127 mm). The final weight of the truss assembly was 47.1 kg, well below the budgeted 54.4 kg.

INSTRUMENT SECTION

The MSX instrument section must provide a thermally stable platform to accurately support the majority of MSX instruments. This structure was designed to minimize Sun-induced thermal gradients that would vary the co-alignments of the instruments during various spacecraft observations and maneuvers. The instrument section suite includes the Ultraviolet and Visible Imagers and Spectrographic Imagers experiment (nine spectrographs and imagers), two horizon

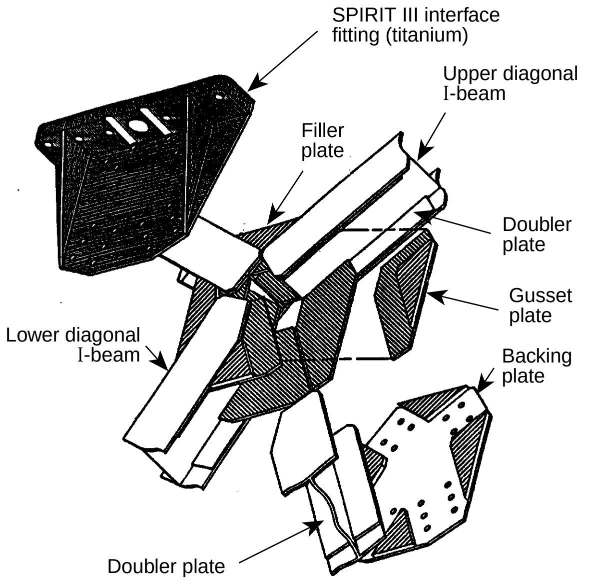

Figure 7. SPIRIT III interface fitting. All components are G/E unless noted.

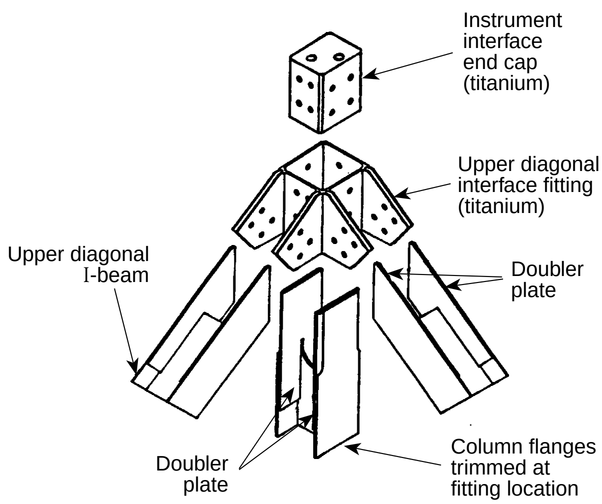

Figure 8. Upper diagonal interface fitting. All components are G/E unless noted.

sensors, three digital solar attitude detectors, a xenon lamp, a total pressure sensor, a mass spectrometer, four temperature-controlled quartz crystal microbalances, a krypton lamp, the Space Based Visible Telescope, six reference object canisters, and several electronics packages. The instrument section assembly is shown installed in the MSX spacecraft at the top of Fig. 1.

The instrument section structural design consists of machined aluminum structural members joined together to form a large "box"-style frame similar to that of the electronics section. It interfaces with the truss structure at four corners and provides the four lift points for the spacecraft. The structural elements of the framework consist of "L"-shaped edge members and "U"-shaped center members joined with intricately machined corner fittings and fasteners. The center members are used on the y and z sides to provide additional framework rigidity and support for the instrument honeycomb panels. The instrument section frame structure measures $1.59 \times 1.59 \times 0.72$ m and is also used for attaching secondary structures that mount several SPIRIT III electronic chasses and the optical bench.

Very precise controls on tolerances and size were required on the internal frame assembly to minimize tolerance buildups as panels, instruments, and other parts of the structure were attached at later dates. The structure was assembled using a large granite table, custom-fabricated squares, and tooling in order to maximize the flatness, parallelism, and perpendicularity of all sides. After all the structural parts of the frame were assembled and adjusted, the assembly was matched drilled and permanent shear and tension Hi-Lok fasteners were installed. The assembly process produced an extremely precise finished structure for mounting the instrument honeycomb panels.

The honeycomb panels, which are attached to the frame structure on the four sides ($\pm y$ and $\pm z$) of the spacecraft, presented several design challenges primarily related to the thermal requirements. Each of these panels is a 3.3-cm-thick rectangular-shaped aluminum honeycomb panel with 0.5-mm aluminum face sheets, and each contains three embedded heat pipes for transferring heat from the instruments to the radiator assemblies. Many of the threaded spool inserts used to mount the instruments are attached directly to the heat pipes, thus providing a conductive heat path to the radiator assemblies. In addition, the $+y$ side panel heat pipes extend beyond both edges of the panel in order to interface with the radiator assemblies mounted on the $+z$ and $-z$

Table 1. Material properties of T50/ERL1962.

Property	Mean
Tensile strength (0°/90°)	$4.34/3.46 \times 10^{8}$ N/m^2
Tensile modulus (0°/90°)	$8.69/9.28 \times 10^{10}$ N/m^2
Tensile Poisson ratio (0°/90°)	0.30/0.34
Compression strength (0°/90°)	$3.29/3.34 \times 10^{8}$ N/m^2
Compression modulus (0°/90°)	$8.07/8.83 \times 10^{10}$ N/m^2
Compression Poisson ratio (0°/90°)	0.32/0.33
Flatwise tensile	1.17×10^{7} N/m^2
Two-rail shear	2.25×10^{8} N/m^2
Short beam shear	7.34×10^{7} N/m^2
CTE (0°/90°)	$0.56/0.45 \times 10^{26}$ in./in. per °C

Table 2. MSX spacecraft limit load factors (Delta II).

Parameter	Liftoff	Main engine cutoff
Thrust (x) axis	−2.4/+0.2 g	−7.1 g
Lateral (y, z) axes	±2.5 g	—

Note: Loads are applicable at the spacecraft's center of gravity. The dash implies compression to launch vehicle interface.

panels. The $-y$ side panel contains few packages; thus heat is transferred through the frame via conduction. Smaller packages that were not well defined when the panels were fabricated are to be attached to the panels with surface-mounted rivenuts installed at a later date. The side-panel internal surfaces were masked and painted with Z306 Chemglaze paint to augment radiating capabilities, thus enhancing isothermal performance. The panels and frame were then coated with thermal grease to maximize heat transfer between the panels and frame. Special "Milson" close-tolerance, high-strength fasteners are used to attach the panels to the frame structure. These fasteners prevent movement of the panels relative to the frame under load. They also provide accurate repositioning of the panels should they need to be removed and reinstalled.

Four "half-moon"-shaped honeycomb panels close out the top and bottom of the instrument section structure and provide additional rigidity and in-plane stiffness to the structure. All panels are made from 2-cm-thick aluminum honeycomb with 0.5-mm-thick face sheets. They are designed to be removed to facilitate the installation of the SPIRIT III instrument and provide adequate clearance around the instrument, when installed. In addition, the top panels (+x side) contain embedded structural members that are used for attaching the six reference object canisters.

SPACECRAFT STRUCTURAL ANALYSIS AND TESTING

The MSX primary structure was designed to provide a safe mounting platform for the spacecraft bus and instruments during the launch scenario and to maintain on-orbit co-alignment requirements of the instruments. To satisfy these requirements, spacecraft structure strength, stiffness, and stability issues were addressed.

The McDonnell Douglas Delta II Payload Planner's Guide recommends spacecraft structure fundamental frequencies above 35 Hz in the thrust axis and 12 Hz in the lateral axes to avoid dynamic coupling between low-frequency vehicle and spacecraft modes. Dominant launch vehicle induced loads result from the liftoff and main engine cutoff flight events. The maximum expected flight loads, or limit loads, are calculated by multiplying the spacecraft weight by a limit load factor (in g). The spacecraft limit load factors specified in the Planner's Guide are listed in Table 2. These two requirements, although independently defined, are related in the sense that failure to achieve the stiffness requirement could have a detrimental impact on the load conditions. Thus, the proper combination of these two requirements is necessary for equipment to survive the launch environment.

To meet instrument co-alignment requirements, the total structural and on-orbit distortion for all of the structure above the SPIRIT III mounting interface was not to exceed 0.042°. This requirement represents a combined mechanical and thermal loads problem that directly affected the design of the truss and instrument section structures.

To address the structural requirements, a finite element model (FEM) representing the characteristics and properties of the primary spacecraft structural elements was developed. Portions of the 1839-element model shown in Fig. 9 are based on models developed by external organizations supplying hardware, e.g., Engineering Consulting Resources Associates supplied the SPIRIT III model, Rockwell supplied a solar array model, McDonnell Douglas supplied the spacecraft adapter model, and Composite Optics supplied the truss and beacon receiver bench models. Individual FEMs were also developed for the instrument and electronics sections, the truss, and the adapter to better evaluate local detailed stresses as a result of the applied load conditions. The NASTRAN software developed by MacNeal Schwendler Corporation was used to analyze the model when it was subjected to the quasi-static loads in Table 2 and to determine the fundamental dynamic characteristics (normal modes) of the spacecraft. For quasi-static loads analysis, factors of safety (FS) were applied to the limit loads to account for test load margins and to ensure that an adequate design margin exists. The FS used on the MSX program are as follows:

Material yield FS = 1.4
Material ultimate FS = 1.8
Material buckling FS = 2.0

Margin of safety (MS) calculations were made to indicate the degree of strength capabilities beyond the desired factors, i.e., MS ≥ 0 is acceptable, and is defined as

$$\mathrm{MS} = \frac{\text{material allowable strength}}{\mathrm{FS} \times \text{applied stress}} - 1.0 ,$$

where applied stress is the material stress resulting from the applied limit loads. A summary of the minimum

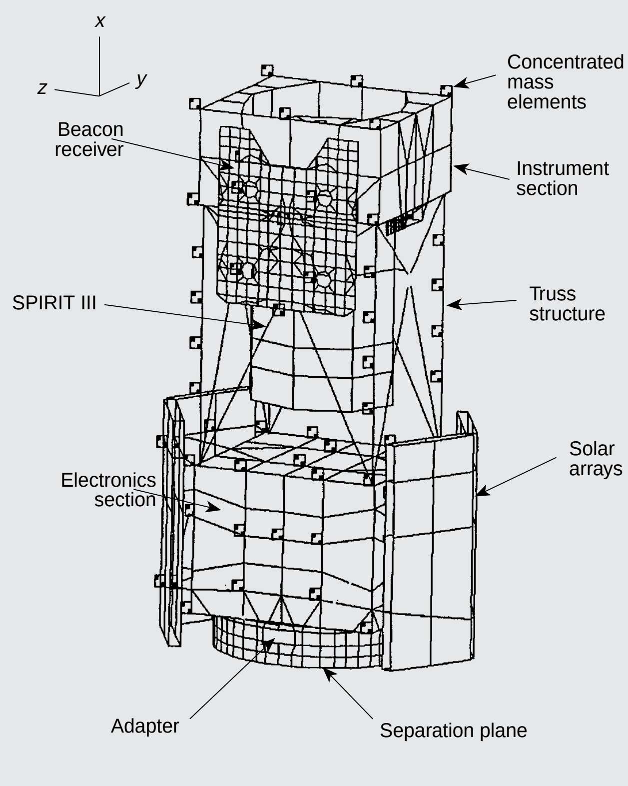

Figure 9. MSX launch configuration finite element model.

margins of safety for the quasi-static analysis of the MSX primary structure is presented in Table 3. The fundamental dynamic characteristics are defined in Table 4.

To address the on-orbit alignment issues, worst-case thermal temperature distributions and 1 *g* effects were examined. The results from the finite element analysis indicated that maximum distortions were less than 0.02°.

The spacecraft FEM was analytically coupled to a Delta II launch vehicle model to predict spacecraft responses as a result of launch events. Although the primary spacecraft modes (8.4 and 29.9 Hz) do not meet the minimum Delta II recommendations (12 and 35 Hz), results from the coupled loads analysis performed by McDonnell Douglas verified that the limit load factors presented in Table 2 did not change.

Once fabricated, the primary structural elements were tested to demonstrate acceptable structural performance. Because the size of the MSX spacecraft precluded structural testing at the system level, static load tests were performed on each of the spacecraft's major structural sections. These structures were proof tested to verify that they were capable of surviving the worst-case predicted flight loads plus a 25% margin. Test scenarios were based on generating worst-case load conditions in critical structural members. Strain gauges and deflection indicators were used in each test sequence to monitor the stresses in key structural elements to ensure that the design stress levels were not exceeded. The data collected were used to verify and calibrate the analytical model of the spacecraft used in previous loads analysis.

Two separate tests were performed for each of the instrument and electronics sections. The first, depicted in Fig. 10, was performed to verify the adequacy of the corner fittings and attachments as a result of flight or lifting load situations. Tensile loads of 4535 kg and 1135 kg were applied at each of the four corners for the electronics and instrument sections, respectively. The overall structural integrity of each section was verified by application of a lateral load in combination with a compression load as shown in Fig. 11. Lateral loads of 9070 kg and 1815 kg and compression loads of 2040 kg and 726 kg for the electronics and instrument sections, respectively, were applied in four orthogonal directions to thoroughly check all structural elements and all possible loading scenarios. Results of the testing indicated that both structural sections exhibited more than adequate strength to withstand the maximum anticipated flight loads. Although stresses were significantly lower and deflections somewhat higher than anticipated, this

Table 3. Minimum margins of safety: MSX spacecraft primary structure.

Structural component	Margin of safety
Instrument section structure	0.33
Instrument section fasteners	0.17
Electronics section structure	0.14
Electronics section fasteners	0.18
Truss structure elements	0.31
Truss fasteners	0.23
Adapter structure	0.11

Table 4. MSX fundamental modes.

Description	Frequency (Hz)
Spacecraft bending modes: *y* and *z* axes	8.4
Solar array bending modes: *z* axis	19.6
SPIRIT III (*y* and *z* axes) and solar array (*z* axis) bending modes	19.8
Spacecraft torsion mode	19.9
Spacecraft axial mode	29.9

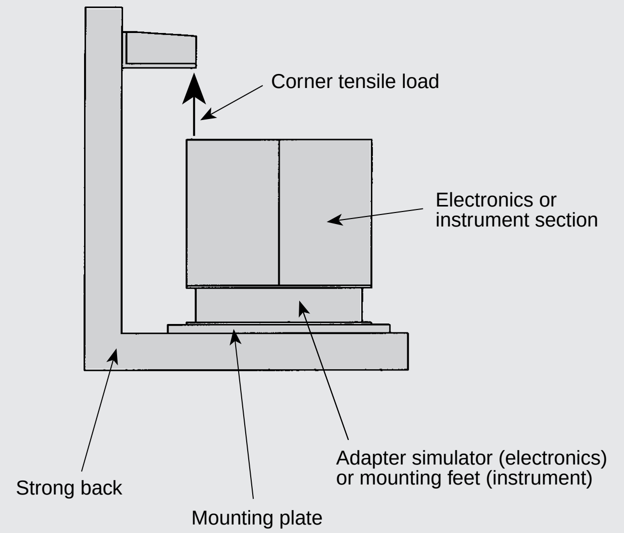

Figure 10. Static load test set-up: corner fittings.

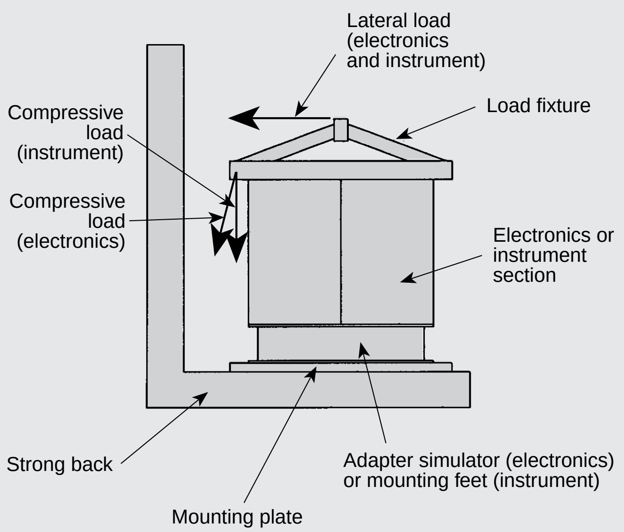

Figure 11. Static load test set-up: overall structural integrity.

was attributed to joint slippage that occurred between the structural frame and the honeycomb panels.

The truss section test, depicted in Fig. 12, was designed to test the lower diagonal member of the truss, the most critical member as defined by previous analyses, to a compressive load of 4322 kg. Although proper loading was achieved, deflection results indicated a lateral deflection 2.5 times the predicted value. This discrepancy was attributed to inaccuracies in the model used for predicting test results, movement of the truss relative to the base fixture/floor, and rigid body rotation of the truss/base fixture assembly. Relevant findings were incorporated into the analysis model, which was subsequently used to verify that there was no impact on the spacecraft loading conditions.

OPTICAL BENCH

During science missions, alignment shifts between the optical bench components (two ring laser gyros, one star camera, and two autocollimators) should be maintained within 2-arcsec relative rotation and 0.0254-mm relative displacement in order to accurately determine the SPIRIT III boresight direction. Any unknown or unmeasured rotations in the bench structure will affect the component measurements and would be transformed into an apparent error in the SPIRIT III boresight direction. These requirements dictated the need for a very precise and stable platform to support these components through the various on-orbit conditions. Use of conventional materials such as aluminum for the bench structure would not provide the desired stable features, thus dictating the use of a G/E composite material. Because of the stringent requirements, the on-orbit operating temperature of the structure will be maintained at 20°C and thermal gradients will be maintained within 1°C in the plane of the bench and between the top and bottom face sheets of the bench. Furthermore, the dimensional and rotational stability also dictated that the CTE for the laminate be minimized (a goal of 0.0) and that the bench mounting system would not allow a rigid body rotation greater than 0.006° (22 arcsec) from its nominal position. The latter requirement coupled with the 2-arcsec requirement dictated the use of a special, unique kinematic mounting system.

The optical bench structure is mounted on top of the instrument section structure near the aperture for the SPIRIT III instrument. The system consists of a mounting platform with two skewed equipment mounting planes and three kinematic mounts to provide support. The system weighs 4.45 kg and is shown in Fig. 13. The distinctive geometry for the 0.34-m-wide by 0.97-m-long by 0.038-m-thick structure was dictated by sensor mounting requirements and the space restrictions in the MSX configuration. The bench structural arrangement is a sandwich construction consisting of a discrete rib core with top and bottom face sheets. Both the ribs and face sheets were made from a unidirectional tape with graphite fibers embedded in Amoco's ERL1962 resin. The P75S fibers (5.17×10^{11} N/m^2 modulus) were selected because of the demanding thermal stability and stiffness requirements for the system; the ERL1962 resin was selected for the same reasons as specified for the truss structure. An added benefit of using this G/E system is the reasonable cost of the materials and the substantial materials database that already exists as a result of previous use. EA 9394 adhesive was selected for bonding the G/E parts together because it is less ductile and more stable for optical bench applications. Components were mounted to the bench structure via inserts that were directly attached to the ribs.

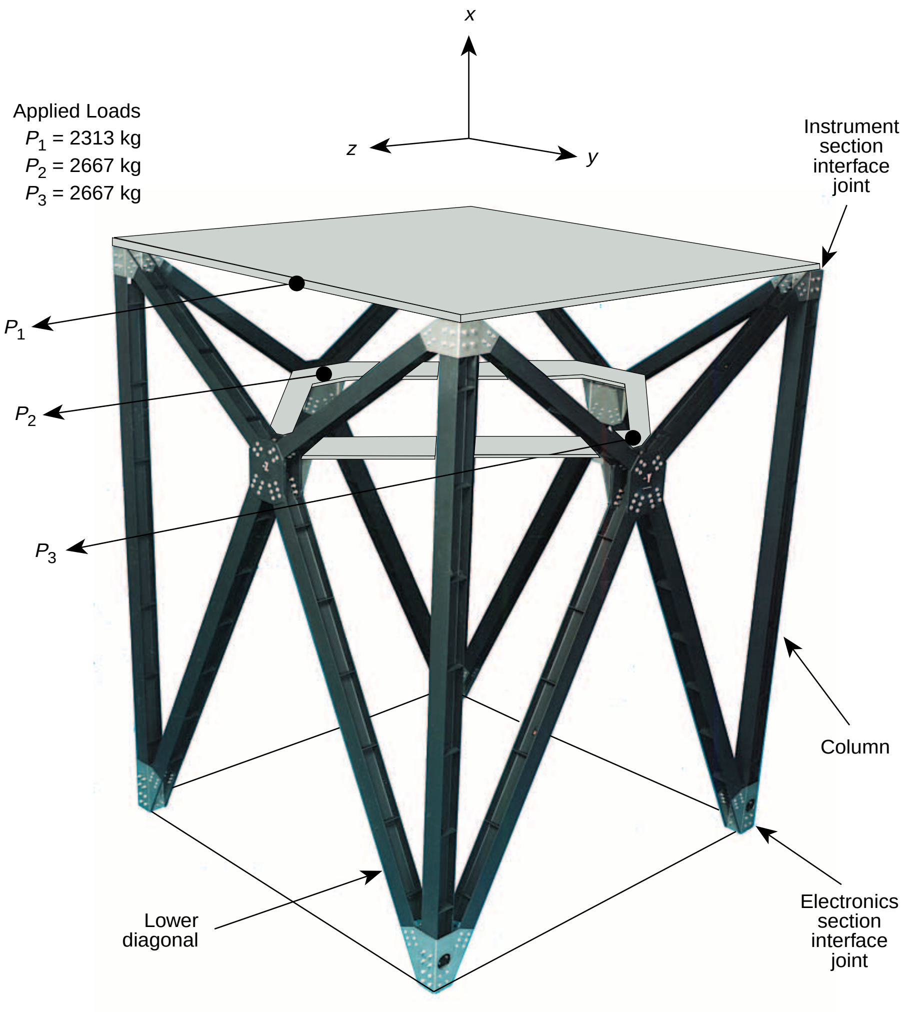

Figure 12. Truss structure static load test set-up.

The structure constituents (ribs, face sheets, and gusset plates) were constructed from multiple layers of the unidirectional tape oriented in a (0°, 30°, 60°, 90°, 120°, 150°)$_{SN}$ lay-up to yield the pseudoisotropic condition. The core ribs were 12-ply and the face sheets were 24-ply laminates. Material properties for this configuration are given in Table 5. A tin-indium moisture barrier similar to that used on the truss was applied to preclude the effects of moisture on the stability of the optical bench. This barrier also helps to minimize moisture contamination of the sensor optics located on the forward part of the structure.

The configuration for the kinematic mount system uses a ball mount concept that provides nonredundant translational support in one, two, and three directions for the three mounts. The attachment of the mounts to the optical bench is located at the central plane of the bench thickness. This method of attachment limits the moment transfer between the spacecraft and optical bench, thus minimizing the distortion of the mounted structure. The mounts were made from titanium to provide good thermal compatibility (low CTE) with the composite bench; in addition, titanium is a relatively poor conductor, thus providing thermal isolation between the bench and the instrument section.

The optical bench structure must meet all requirements after exposure to temperature extremes of −40°C and +60°C. Quasistatic load strength requirements of −12.5 *g*/+6.5 *g* in the spacecraft thrust direction (*x*) and ±6.5 *g* in the spacecraft lateral direction (*y* and *z*) were imposed to ensure that the hardware will survive the launch environment. In addition, load constraints were placed on component attachments (±35 *g* normal and ±20 *g* parallel to the component's mounting plane) and lifting attachments (±3 *g* in the lift direction [*x*]) to guarantee the integrity of these attachments. A stiffness requirement of 100 Hz was desired to preclude dynamic interactions with other structural components. Given all these requirements, the system's mass was not to exceed 6.8 kg.

To predict the capability of the optical bench system, an FEM was generated to examine quasi-static loading conditions, to determine fundamental dynamic characteristics, and to evaluate bench distortions. After examining numerous concepts and iterating to the final version, the fundamental dynamic characteristics were determined to be 111 Hz (bench bending in the thrust direction), 138 Hz (twisting of the star camera mounting plane), and 203 Hz (bench bending in the lateral direction). Results of the quasi-static loading analysis indicated that, in general, except for certain local areas, the structural design exhibits ample structural margins under the given conditions. For components such as the optical bench, which are driven by stiffness requirements, moderate stress levels as a result of mechanical loads are typical. A summary of the minimum margins of safety for each of the bench structural components is presented in Table 6.

To determine compliance with the alignment and stability requirements, a thermal distortion analysis for the operational conditions was performed using the FEM. This included not only the thermal gradient conditions but also the effects of the breakaway torque

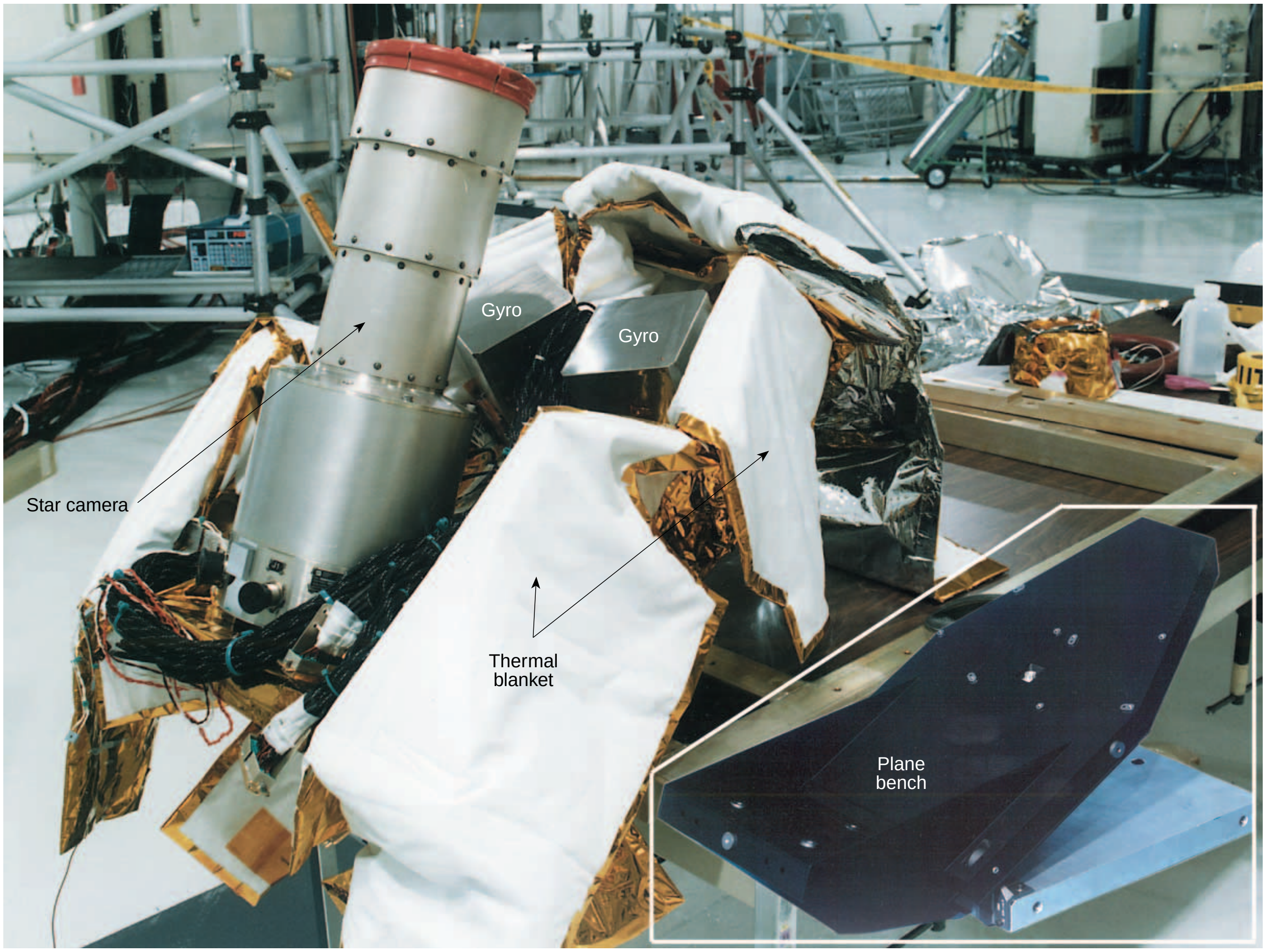

Figure 13. MSX optical bench (structure shown in lower right).

of the kinematic mounts. The results indicate a maximum relative displacement of 0.00635 mm and a maximum relative distortion of 1.35 arcsec for the operating conditions. This maximum occurs between the star camera and autocollimators. For each inch-pound of breakaway torque, an additional 0.1 arcsec of rotational deformation would occur. To minimize the impact of this, bearings with breakaway torques not to exceed 0.17 kg·m were selected for this application. The total anticipated distortion for this configuration is thus 1.65 arcsec. Breakaway torque has a negligible impact on displacement.

All G/E parts for the bench structure were machined from pseudoisotropic flat stock. Special fixtures were fabricated that allowed accurate assembly of the discrete core details in relation to the equipment attachment fittings and the kinematic mount interface details. All G/E components were cleaned, plated with the moisture barrier, and painted before assembly with the kinematic mount metal parts. Once the bench structure was assembled, the optical bench was subjected to the thermal extremes, modal survey, static loads, and thermal distortion tests. Although not a specific requirement, a ±1 *g* distortion assessment test was also conducted to evaluate the change in bench dis-

Table 5. P75S/ERL1962 material property summary.

Property	Mean
Tensile strength (0°)	2.951×10^8 N/m^2
Tensile modulus (0°/90°)	$1.05/1.03 \times 10^{11}$ N/m^2
Tensile Poisson ratio (0°/90°)	0.32/0.32
Compression strength (0°/90°)	$1.68/1.78 \times 10^8$ N/m^2
Compression modulus (0°/90°)	$0.95/1.04 \times 10^{11}$ N/m^2
Compression Poisson ratio (0°/90°)	0.32/0.32
Flatwise tensile	1.17×10^7 N/m^2
In-plane shear strength	1.81×10^8 N/m^2
CTE (0°/90°)	-0.22×10^{-6} in./in. per °C

Table 6. Structural margin of safety summary.

Component	Minimum margin of safety
G/E facesheets	1.5
Ribs	0.7
Adhesive bond line	0.02
Kinematic mounts	0.4
Hardware attachments	0.01

tortion between the ground and orbit environments. A brief summary of each of these tests is presented next.

The thermal extremes and static loads tests were conducted primarily to verify the integrity of the bench structure. First the bench was subjected to 15 thermal cycles between −40°C and +60°C to verify bond line integrity. Subsequently, two static load test cases representing the worst-case tensile and compressive load cases were performed. The bench was mounted to a thick aluminum plate, and mass simulators were used to allow the loads to be applied at the component centers of gravity. A test factor of 1.25 was applied to all loads to demonstrate margin over the worst-case expected loads. Post-test inspections revealed no damage to the bench structure or any bond lines.

Prior to static loads testing, a modal survey test was conducted to verify the dynamic characteristics of the configuration. The configuration consisted of the bench assembly, without components, mounted to a thick aluminum plate. A post-static loads test was also performed to verify that no changes occurred as a result of the structural testing. The results of 352 Hz and 360 Hz (pre- and post-static test, respectively) compared fairly well with the predicted value of 402 Hz. Examination of the analytical model indicated the difference was due to artificial rigidity introduced in interelement connections. When the model was adjusted for more realistic flexibilities, a natural frequency of approximately 370 Hz was obtained.

Bench stability was determined by examining the thermal distortion between the star camera and the autocollimator mounting planes when a thermal gradient was applied through the thickness of the bench structure. A laser, beamsplitter, and mirrors were used to measure the relative difference between the two locations. Results of 14 test cases indicate an average distortion of 2.48 arcsec/°C. Although this measurement exceeds the goal of 2 arcsec/°C, it was determined that the slightly larger error would not have a significant impact on scientific measurements.

BEACON RECEIVER BENCH

The beacon receiver subsystem must acquire cooperative targets with an initial pointing uncertainty of ±5° × ±5° at a range of up to 8000 km and track them with a ±0.1° × ±0.1° residual rms error during the 5-year orbital mission of the MSX spacecraft. To accomplish this objective, the deployed beacon receiver structure must contribute no more than 0.5° to the initial pointing error of and provide stability within 0.03° for the system's four antenna arrays. These requirements must be satisfied over a fairly wide temperature range (−65°C to −15°C) and when the structure is subjected to large thermal gradients (up to 30°C). In addition, stiffness requirements dictate that the fundamental frequency of the stowed bench structure must be greater than 50 Hz and less than 65 Hz in the spacecraft $\pm y$ direction to preclude any dynamic coupling with the spacecraft primary structure. Long-term creep in the bench as a result of the 362.8-kg prelaunch center release preload must be minimized. When viewed as a complete package, the demanding and sometimes conflicting requirements of accurate initial pointing after deployment, long-term pointing stability, passive thermal management, and very low operational temperature with room temperature calibration all combine to produce a very challenging mechanical design problem.

The beacon receiver bench assembly consists of a G/E composite bench, four aluminum dish antenna arrays, a magnetometer, two hinge and damper assemblies, electronics, and a latch and release assembly. The approximate size of the assembly is 1.397 × 1.143 × 0.305 m and it weighs approximately 36.3 kg. Five titanium brackets attach the receiver structure to the $-y$ experiment section honeycomb panel and provide support, hinging, and latching functions. This system, shown in Fig. 14, will be stowed during launch and deployed within 4 hours of separation of MSX from the launch vehicle.

The G/E bench structure is designed to provide a mechanically and thermally stable platform to support the receiver antenna array and associated electronics after deployment from the stowed or launch configuration. G/E composite materials were chosen in lieu of more conventional materials because of the combination of stringent requirements for alignment, stability, distortion, and stiffness. The bench structure, built by Composite Optics, Inc., was chosen to be similar in construction to the MSX optical bench, using G/E face skins separated by a G/E discrete rib core. The same G/E material system (0.1-mm-thick P75S/ERL1962 unidirectional tape) was also used to fabricate the face sheets and ribs because of the similarity in requirements, although a different pseudoisotropic lay-up $[(0°/36°/72°/108°/144°)_S]$ was used to provide a minimal in-plane CTE. Material properties for the pseudoisotropic lay-up are given in Table 5. Various additional G/E angle clips and doublers are used in areas of high stress. Threaded titanium fittings, bonded onto the ribs and face sheets, were used for bolted attachments. As

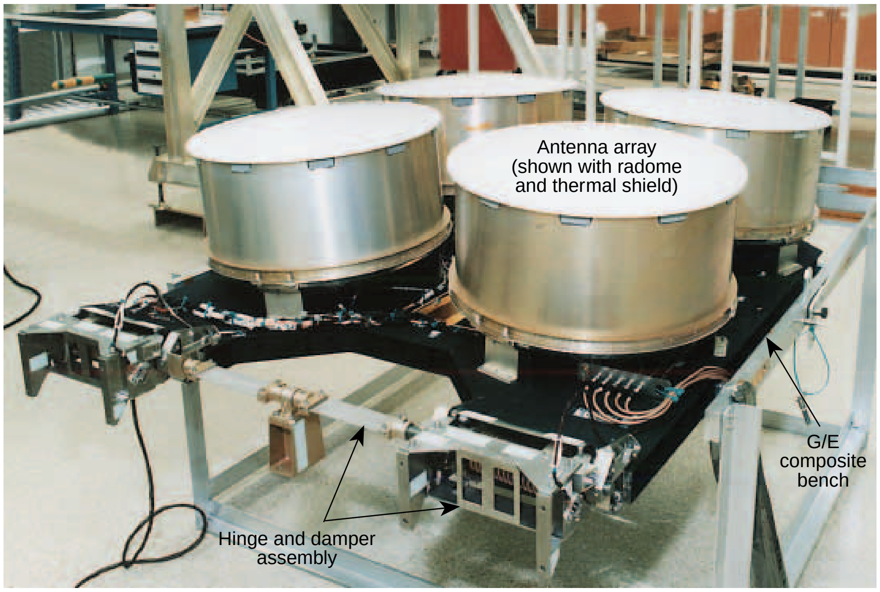

Figure 14. MSX beacon receiver subsystem.

with the truss and optical bench structures, a tin-indium metallic coating was applied to internal and external surfaces after assembly to ensure long-term dimensional stability against the effects of moisture.

The concept for the hinge design evolved from smaller hinges used successfully on previous APL missions. The nonmagnetic beacon deployment hinge assembly uses beryllium copper torsion springs, titanium supports and shafts, bronze bushings, and stainless steel dampers and functions much like an old-fashioned saloon door. The hinge is designed to deploy the structure such that the antenna array's mechanical boresight is initially aligned within ±0.5° of the spacecraft's *x* axis and is maintained within 0.01°. Stored energy in the preloaded torsion springs deploys the bench when released and positions the bench when fully deployed. A torsion damper assembly consisting of 3.175-mm-thick stainless steel strips was used in the system to prevent excessive overtravel during deployment, thus avoiding interference between the beacon receiver and other spacecraft hardware. During deployment, the damper assembly absorbs the kinetic energy of the system when it travels past its deployed position, and the torsion springs wind back up to provide a resisting torque that forces the bench to seek the final, deployed position. Clearances internal to the hinge compensate for different CTEs between the G/E composite bench and the aluminum instrument section. The center release and pin puller assemblies function to accurately preload the bench assembly during launch as well as to release it for deployment after launch. The 362.8-kg preload is provided by a beryllium copper bolt that is instrumented with a strain gauge to ensure that the desired tension is set accurately. The joint design allows liberal tolerance accumulations to be accounted for between the bench and instrument section structure during fabrication and assembly through the use of a rod end bearing, which provides rotation and pivot. Upon command from the spacecraft, the pyrotechnic activated pin puller retracts the pin and the bench is released. After the beacon receiver has been determined to be in the deployed position, the pyrotechnic activated orbit latch is engaged to provide position stability to the bench against spacecraft torques during on-orbit maneuvers.

An FEM of the bench assembly was constructed to analyze quasi-static launch loads, latch preload effects, system dynamic characteristics, random vibration loading, and thermal stress and distortion. Results of this analysis indicate that the first natural frequency of the structure is 55 Hz. All stresses in the structure caused by the center release preload were shown to meet the goal of being less than 10% of the G/E material allowables, thus minimizing long-term creep effects. Minimum margins of safety for other structural load cases were determined to be acceptable and are shown in Table 7 for various parts of the structure. Table 8 gives the boresight pointing results for the thermal distortion cases analyzed.

In addition, a thermal distortion analysis was performed on an FEM of a single aluminum dish to analytically determine the effect on radio-frequency performance of the CTE mismatch between the aluminum antenna dishes and the G/E bench structure. Results of this analysis[1] show dish distortion to be within budget.

To evaluate system positioning and accuracies during hinge deployment, a mass simulator of the receiver structure was fabricated. The simulator was fabricated from Invar to approximate the CTE of the G/E bench structure. It also served as a mockup for cable and coax service loops and thermal blanket design and tie downs. An aluminum support frame was fabricated to provide five support points and a *g*-negation system for hinge deployment and stability testing. Optical flats were mounted on the frame and a series of manual and pyrotechnic release deployments of the assembly were performed to determine the repeatability of the deployed position. In addition, the orbit latch was engaged for three tests to determine an average angular measurement of the deviation from the deployed to the locked position. Results from this testing indicate that the hinge would consistently deploy the receiver assembly to within 0.115° of the proper position, well within the 0.5° requirement. After the orbit latch was engaged, the hinge was stable in the presence of disturbance torques to 0.003°, well within the 0.01° requirement for hinge play. The simulator and frame assembly were then placed in the large APL thermal vacuum chamber and a cold deployment was successfully accomplished. The orbit lock was successfully engaged during the subsequent cold soak to complete hinge, latch, and orbit lock verification testing.

Prior to assembly, the G/E structure was successfully subjected to a static loads test to 1.25 times the quasistatic design loads, a temperature extremes test, and a moisture loss test, thus verifying performance. The bench-mounted electronics as well as the orbit latch were subjected to sine and random vibration at the component levels. The beacon receiver assembly, including the composite material structure, dishes, thermal shield, multilayer insulation, hinge, support brackets, and center release, was subjected to three-axis sine and random vibration testing to verify system-level capability. The first natural frequency of the receiver assembly was determined to be 49 Hz, slightly below the initial 50-Hz requirement but nevertheless satisfactory from a stiffness viewpoint. Before and after vibration testing, the center release mechanism was actuated and the unit was walked out through its full range of motion to verify proper operation of the deployment mechanism. All testing was completed satisfactorily with no problems encountered.

GROUND SUPPORT EQUIPMENT

The MSX spacecraft, at 2812 kg, 5.1 m in height, and 3.3 m in diameter, is 2 to 3 times the size and weight of any other spacecraft that has been integrated and tested in APL's Kershner Space Building. Because of its size and weight, several unique pieces of mechanical ground support equipment were required for integration and transportation of the spacecraft. The two most significant pieces of hardware, the "elephant stand" and shipping container, are described here.

The high density of packages on the spacecraft necessitated the use of the space inside the adapter to accommodate placement of the battery, the second largest package on the spacecraft (the SPIRIT III instrument is the largest). The 90.7-kg battery is mounted on the $-x$ honeycomb panel surface (essentially the underside of the spacecraft) and requires the use of a hydraulic table for mechanical integration. This requirement dictates a special fixture to sufficiently elevate the spacecraft above the ground and provide access underneath for mechanical operations (using the hydraulic table), wiring, and thermal blanketing. This fixture (shown at the bottom of Fig. 1), named the "elephant stand," supports the spacecraft 0.9 m above the floor, giving adequate room beneath the spacecraft for all required tasks. One of the six gusseted legs that support the spacecraft mounting surface is removable to provide access for use of the hydraulic table underneath the spacecraft for installation of the battery. An extra-stiff spacecraft mounting surface is required because of the need to remove one of the leg supports. The elephant stand was proof tested to 8163 kg in both configurations to demonstrate adequate design margin. At times during integration and test, it was necessary to move the spacecraft and elephant stand together, a configuration weighing in excess of 3447 kg. The elephant stand was designed with special pockets under the base plate to allow air bearings to be used to accomplish this task. With a mere 35.6-N (8-lb) force, the spacecraft and elephant stand could be moved together, for example, between test chambers at Goddard Space Flight Center (GSFC) during spacecraft environmental testing.

A special shipping container and transporter system was designed to move the assembled spacecraft. Size, building limitations, contamination issues, and modes of transportation were important issues considered in the hardware development. As with typical containers, shock and vibration isolation design features were incorporated to protect the spacecraft during transportation. The spacecraft was designed to be lifted vertically and installed on the container base by an overhead crane. Four container sections, required because of limited hook height in the Kershner Space Building clean room, are lowered

Table 7. Beacon receiver margin of safety summary.

Component	Minimum margin of safety
G/E facesheets	0.17
Ribs	0.21
Titanium fittings	1.41
Adhesive bond line	0.01
Inserts	0.56

Table 8. Beacon receiver dimensional stability results.

Requirement	Actual	Allowable
Array boresight alignment with spacecraft x axis	0.0000471 rad	0.00524 rad
Dish-to-dish boresight relative angles	0.000259 rad	0.00175 rad
Dish-to-dish relative translations in the dish mounting plane	0.02245 mm	0.2032 mm
Dish-to-dish relative translations normal to the dish mounting plane	0.02972 mm	0.0762 mm
Dish rotation about a dish boresight axis	0.0000029 rad	0.00262 rad

over the spacecraft to completely enclose it. Each section joint has an O-ring seal to provide an airtight configuration during transportation. The rotation fixture attaches to the enclosure and is designed to lift the loaded container, then rotate it 90° so that it mates with the transporter (Fig. 15). Rotation to the horizontal position is necessary so that the hardware can be transported in the C5 aircraft (to Vandenberg Air Force Base) or air-ride lowboy truck (to GSFC). Pressure inside the container is regulated within 1723.7 N/m^2 of ambient pressure by banks of breather valves located in the base of the container. The sensitive optical instruments are protected from contamination during changes in external pressure while on the C5 aircraft by special HEPA-filters mounted inside the breather valves to clean the incoming air. As with all ground support equipment, the shipping container, transporter, and rotation fixture were load tested before they were used to verify that their capabilities were adequate. The container/transporter assembly was also subjected to an instrumented 0.15-m drop test to verify the shock isolation system.

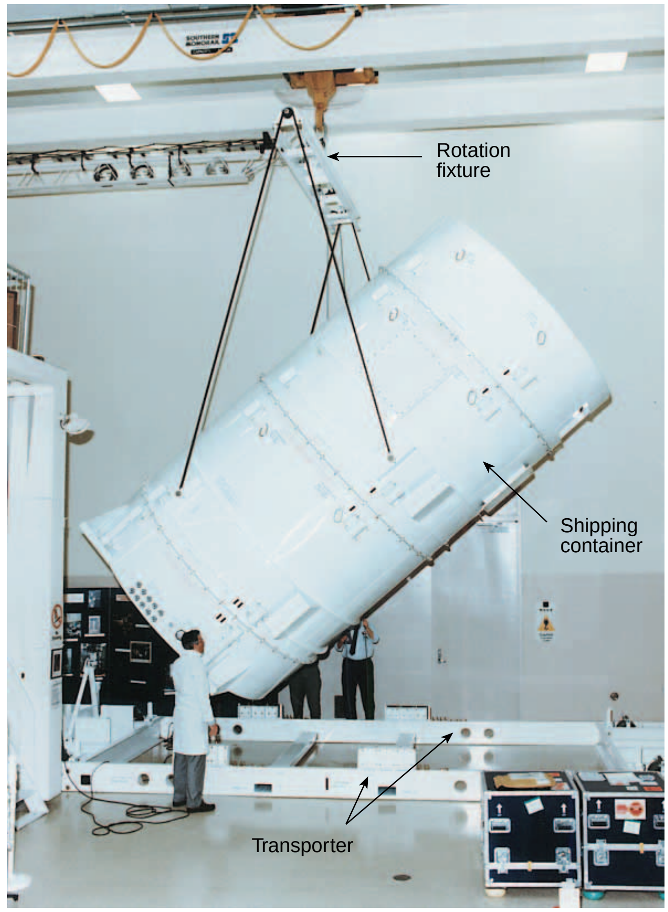

Figure 15. MSX shipping container and lift/rotation fixture.

SUMMARY

The configuration developed for the MSX spacecraft represents the largest structure ever assembled by the Applied Physics Laboratory. The design of each of the three major structural components—the electronics section, the truss structure, and the instrument section—was based on somewhat unique sets of requirements involving packaging, structural, thermal, and stability constraints. These constraints resulted in the application of technologies in the areas of composite materials and heat transfer. Two thermally stable, precision alignment platforms—the optical bench and the beacon receiver bench—were developed to provide support for critical attitude control and radio-frequency system components. All structural components were successfully tested and are currently integrated with the instruments and spacecraft electronics awaiting launch.

REFERENCE

[1]Jablon, A. R., and Persons, D. F., "Spacecraft Reflector Antenna Thermal Distortion Analysis," in *RF Expo West Proc.* (18–20 Mar 1992).

ACKNOWLEDGMENTS: The MSX structural design and analysis efforts have been a major effort involving many APL staff and external participants. We wish to acknowledge the support provided by staff members in the Technical Services Department; Kelly Dodson, Helmuth Dorth and the team at Composite Optics, Inc., in San Diego, California; and Ted Sholar and other contributors from Swales and Associates in Beltsville, Maryland. The MSX mission is sponsored by the Ballistic Missile Defense Organization. This work was supported under contract N00039-94-C-0001.

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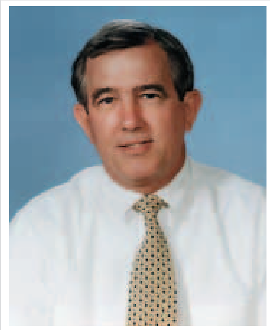

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