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INTERPLANETARY MICROSPACECRAFT DESIGN CHALLENGES: WAR STORIES OF THE FUTURE?

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ABSTRACT

The nature of interplanetary flight, interrelated mission and microspacecraft design drivers, and associated new technology needs can combine to produce serious challenges. While many of these challenges are easy to predict, others are more subtle and surprising. This paper starts with a brief overview of typical issues in both interplanetary spacecraft design and microspacecraft design, and then it focuses on specific challenges that have been recently encountered. Examples are used from the detailed conceptual design of the Multimission Space and Solar Physics Microspacecraft and its mission to better understand and predict hazardous space weather that can cause disruptions in communications, surges in Earth power grids, interference and damage to spacecraft, and hazard to humans in space.

INTERPLANETARY MICROSPACECRAFT

In contrast to Earth-orbiting spacecraft, interplanetary spacecraft have to communicate over much larger distances (with longer round-trip communication times), and their Sun ranges can be much smaller or larger than 1 AU (and often vary considerably) during a mission. Global positioning satellites are not available to help with navigation, and the Earth horizon and magnetic field are not available to help with attitude determination or, in the latter case, with attitude control. Needed launch/injection energies usually are much higher for interplanetary spacecraft than for Earth orbiters, and, frequently, larger onboard ΔV capability is required to correct for launch/injection errors and deterministic maneuvers later in flight.

A primary reason for using microspacecraft is to reduce launch costs (through use of a piggyback launch, smaller launch vehicle, and/or launch of multiple spacecraft with one vehicle), and the cost reductions are magnified with the high injection energies and onboard ΔV requirements of interplanetary missions. At the same time, these benefits come with requirements for low mass and size in the launch configuration, which can magnify challenges associated with interplanetary flight. For example, the smaller mass and size of microspacecraft imply that there is inherently less aperture area available than would be the case for larger spacecraft. Antenna and solar array sizes are constrained to be smaller, while the Earth range and possibly the Sun range are larger.

THE MULTIMISSION SPACE AND SOLAR PHYSICS MICROSPACECRAFT (MSSPM)

Following much smaller related studies in 1994[1] and 1999[2], NASA/JPL completed an MSSPM detailed conceptual design in April 2000[3], and a brief summary of the results was publicly presented in May[4]. The microspacecraft is designed to be capable of multiple missions, but the mission focused on in the most recent study has the objectives of both providing better understanding of the physics of hazardous space weather and of greatly increasing the warning times for these hazards. The mission uses one small launch vehicle and individually tailored Venus gravity assists to disperse nine microspacecraft in a band around the Sun with solar distances ranging from 0.53 to 0.85 AU. Data are collected continuously and analyzed on board. Each microspacecraft returns scientific data to Earth at approximately weekly intervals, and a microspacecraft close to the Sun-Earth line is monitored continuously for hazard alerts.

Since MSSPM is designed for both interplanetary flight and to benefit from a microspacecraft implementation, the combined challenges are present. Earth range can be as high as 1.8 AU, Sun range can vary from 0.5 to 1.0 AU, and nine microspacecraft, their launch integration and

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deployment system, and an injection stage need to fit in one small launch vehicle. If a Taurus with a 1.6-m (63-in.) fairing and STAR 37 FM upper stage are used, the microspacecraft (including their contingency and propellant) are each constrained to roughly 15 kg and dimensions on the order of 1/3–2/3 meter. (All engineering and scientific sensors as well as the other subsystems must fit within these quite limited constraints.) There is a need to have over 90-m/s ΔV capability just weeks after launch to correct for possible launch/injection errors, and propulsion is also needed to support small later trajectory corrections and attitude control. In addition, the flight system must support the navigation for the initial ΔV maneuver, Venus gravity assist targeting, and other maneuvers during the flight.

CHALLENGES AND SOLUTIONS

Significant interrelated challenges associated with these design drivers were present in every major subsystem of the microspacecraft. A few examples of the challenges encountered and the solutions that were identified are discussed below.

Approach

Low cost was the driver not only for use of a single small launch vehicle and microspacecraft but also for the design approach to the microspacecraft themselves. In particular, many elements from an approach to low cost that was developed in 1993-1995[5] were utilized in the MSSPM design. This approach applied aggressive new technology where appropriate and included making the microspacecraft identical or nearly identical to each other, using onboard analysis/compression and autonomous control, and minimizing needed flight system resources and complexity. This further implied using a small, focused payload; minimizing the size, mass, and power needs of components and assemblies; avoiding use of nuclear fuel or other hazardous materials; and incorporating redundancy only where it would be most cost effective.

In regard to redundancy, the nature of the mission and its nine microspacecraft is such that it can tolerate failure of a microspacecraft with only gradual degradation. With that in mind, along with the constraints and low-cost approach, it was decided to employ block redundancy only where moving mechanical parts were involved (and to minimize the number of those parts).

Shape, Stabilization, and Orientation

The 0.5-AU minimum solar range (with 4-Sun-equivalent thermal environment) was a key MSSPM design driver, and it was decided to incorporate several related thermal design features in the MSSPM design that already had been successfully used by other spacecraft at that or shorter solar ranges.

Two Helios spacecraft flew in the mid-1970s and made fields and particle measurements as close to the Sun as 0.3 AU. Their design employed a spinning, bobbin-like body with a spin axis that was perpendicular to the ecliptic. Since the spacecraft flew in that plane, the Sun and Earth directions were at the side of the spacecraft, at 90-degree cone angles from the spin axis. Fixed, semiconical solar arrays flared outward at the top and bottom of the spacecraft and had solar cells on their outboard surfaces. Radial booms extended outward from a central compartment in the spacecraft body, between the solar arrays, and a fixed, low-gain antenna was located on the spin axis above the body along with another antenna and despun high-gain antenna reflector.

Somewhat similar body shape (as can be seen in Figures 1 and 2), stabilization, and spin axis orientation were employed in the MSSPM design because of their attractive features. Recessed thermal radiators on the top and bottom of the microspacecraft core structure (located between the solar arrays) are shaded from the Sun by the arrays and have broad fields of view of space. Spinning simplifies attitude control and spreads the solar heat load around the full circumference of the spacecraft. Solar cells are feathered with respect to the Sun direction, which reduces their maximum solar heat input. A radial boom attached to the core structure is located in the plane of the microspacecraft center of mass and does not shadow the solar arrays. Earth is continuously in the pattern of a fixed low-gain antenna on the spin axis, and pointing of the downlink communication beam at Earth requires only the appropriately phased despinning of the beam.

Although these MSSPM design characteristics were made similar to Helios for the benefits they provided, changes were also made to reduce MSSPM mass, complexity, and cost. The cone angles of the MSSPM solar arrays were increased, and the axial dimension of each array relative to the core structure was decreased.This makes the microspacecraft more oblate than Helios and, in combination with larger perihelion distance, reduces the maximum solar load on the arrays. In

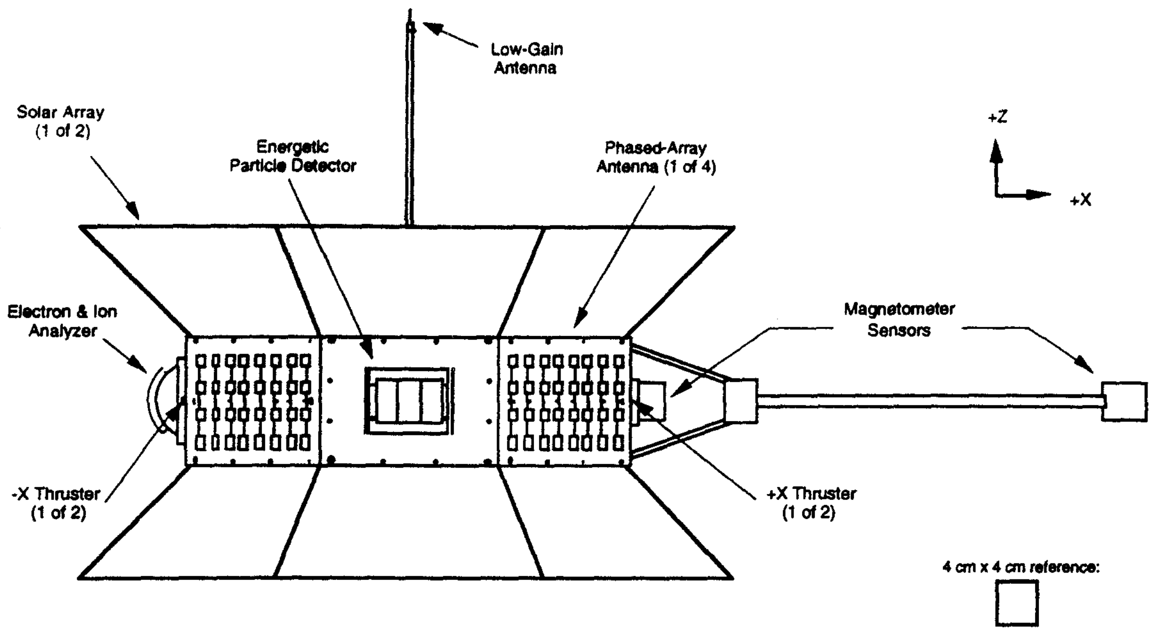

Figure 1: Side View

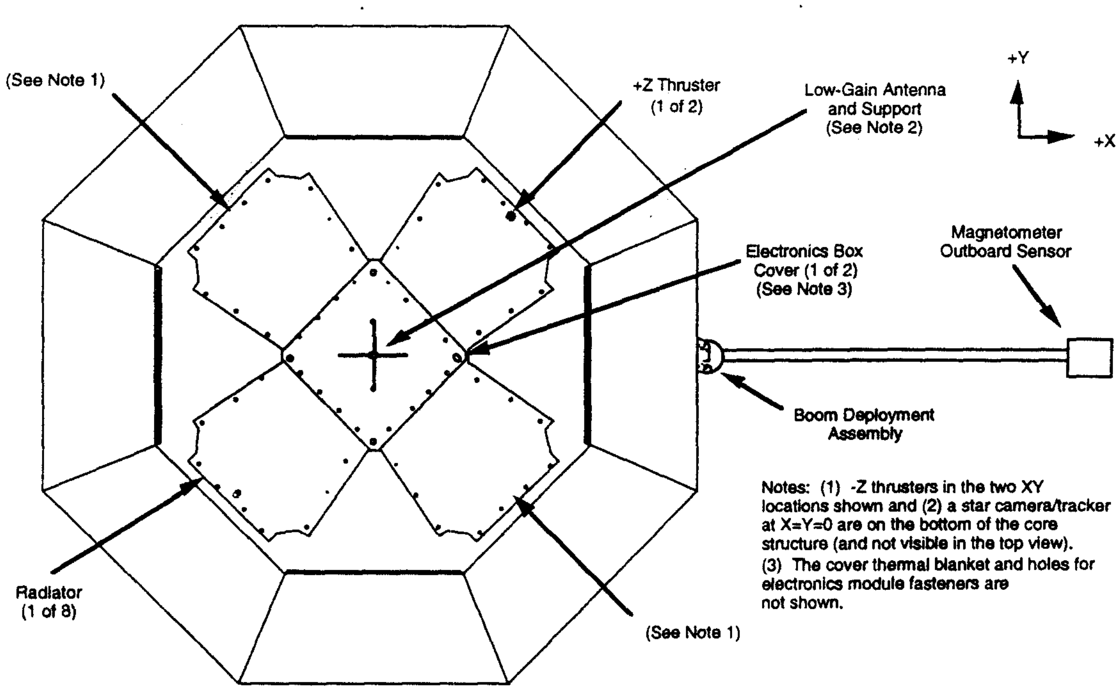

Figure 2: Top View

turn, this makes unnecessary the interspersing of solar cells with second-surface mirrors (that was used on Helios to reduce solar input). It also makes unnecessary the incorporation of fins on the inboard sides of the arrays (that was used on Helios to limit heat radiation into the spacecraft core). Electronic despinning of the downlink communications beam on MSSPM eliminates the need for a despin motor and bearing assembly.

A complicating factor for MSSPM, however, is that its solar orbits are inclined a few degrees with respect to the ecliptic. This implies that keeping its spin axis normal to the ecliptic would not prevent some change in the Sun cone angle and, particularly, the Earth cone angle during the MSSPM solar orbits. Although associated reductions in available power were expected to be insignificant, reductions in communications capability and increased difficulty in shading a star camera/tracker (which is recessed in the center of a semiconical solar array) were thought to warrant a modification. Therefore, instead of the Helios orientation, MSSPM normally maintains its spin axis approximately perpendicular to the Sun–microspacecraft–Earth plane. This eliminates significant variations in Sun and Earth cone angles but requires small changes in spin-axis orientation throughout the mission and comes with the further cost of additional propellant.

Payload

The primary MSSPM payload was focused down to those instruments that were essential to the mission. These were instruments that analyzed magnetic fields, solar wind, and energetic particles and had measurement ranges consistent with hazardous space weather. A number of steps were taken to help fit the instruments within the mass, size, and power constraints. These included, where reasonable, using relatively simple instruments, off-loading information processing and control functions to the centralized flight computer, moving electronics to a common electronics module, and customizing instrument sizes and shapes.

A triaxial fluxgate sensor on the end of a simple, deployable boom was incorporated for magnetic field measurements, and all electronics for the magnetometer were located in the common electronics module. The flight system was designed to help minimize potential magnetic interference at the sensor from the microspacecraft, and the selection of spin stabilization was expected to help in interference compensation during data analysis. Even so, with the simple boom moving the sensor out to only about one diameter of the microspacecraft from its center, a second sensor was thought needed to help with the analysis of potential residual interference. Located roughly a third of the way out to the first sensor, the second sensor shares its electronics in the common module.

A modified version of an instrument designed at Caltech for use on the ACE spacecraft was incorporated for energetic particle measurements. The instrument uses a dE/dx vs. total energy technique with a particle telescope and silicon detector stack, and its interface electronics are located in the common module.

Surprisingly, the plasma analyzer presented the greatest instrument challenge. Initially it was thought a simple Faraday cup design could be used. Its large aperture, however, implied there would be a large solar heat load, particularly near perihelion. An instrument using a solid-state detector that is currently under development was thought to be an alternative, but the sensor required cooling to very low temperatures, and that was thought to be impractical in this mission. Finally, personnel at Southwest Research Institute came up with a concept that was incorporated in MSSPM. It uses a dual "top hat" design and is a modification of an instrument SWRI designed for use on the Rosetta spacecraft. Its interface electronics are also located in the common module.

This completed the MSSPM primary payload, but it was thought that a very small amount of flight system resources would still be available and could be used for a selectable instrument. It was even a possibility that different selections could be made for different microspacecraft. Two concerns accompanied these possibilities, however. One was that addition of another instrument, particularly one that was not initially part of the flight system design, could ripple through the design and cause serious increases in mass, load power, complexity, and cost. The second concern was that if more than one selection was made (and the payloads varied from one microspacecraft to the next), it would excessively complicate design, assembly, test, and operations. The identified solution has several parts. Any selected instrument must fit within the existing flight system resources, including the existing envelope where the instrument is allowed to be located. Addition of the instrument is not allowed to degrade investigations of the primary payload. Simple processing and power interfaces are provided by the flight system, and the instrument must be compatible with those interfaces. Tight constraints on mass properties of the instrument are specified. Finally, although not all of the nine microspacecraft have to fly the selected instrument, only one instrument is allowed to be selected. Microspacecraft that do not fly a selected instrument fly a supplemental rechargeable battery instead.

Power

The microspacecraft size constraint inherently limited the solar array area that could be exposed to the Sun. (A deployable/inflatable array could have been used, but that was thought far too complex and, probably, excessively massive.) Feathering the arrays helps cool the solar cells and improve their efficiency, but it also implies they suffer an inherent cosine loss of solar input power, which is a problem early in the mission when solar range is near maximum. The solution chosen was to use high-efficiency triple-junction GaAs cells, minimize flight system power needs, and buffer maximum loads through use of a rechargeable lithium-ion polymer battery and operational scenarios that would keep battery use within an acceptable discharge range. In addition, the operating point of the solar arrays is electronically controlled to optimize their efficiencies when maximum power is needed. (The same subsystem also allows the operating point of the arrays to be intentionally deoptimized when the array outputs would otherwise be much higher than needed by the microspacecraft.)

Information Processing and Control

Onboard data analysis was incorporated to greatly simplify communication to Earth of both information on the physics of space weather as well as warnings of hazardous space weather. An accepted consequence of this was the need for more onboard knowledge and processing capability than would otherwise be necessary. Significant ROM and RAM sizes (16 and 64 Mbytes, respectively) and a targeted processing capability of 200 MIPS were therefore incorporated in the flight system. This is a particularly large capability when it is considered that the arrival rate of critical data at the microspacecraft is slow compared to the time available for its processing, and more than 10^{14} instructions can be executed between weekly downlinks of scientific analyses. (An additional factor in acceptance of the processing capability target was the rapid advance of computational hardware combined with the significant time remaining before potential flight of MSSPM in 2005 or 2007.)

Telecommunications and Navigation

The initial (1994) concept[1] behind MSSPM included the capability for a precursor solar early warning mission that placed a single microspacecraft in solar orbit. That concept was developed in the expectation that, prior to the mission, highly advanced technology would have been developed and demonstrated in autonomous analysis, autonomous navigation, autonomous control, and other areas. The combination of modest mission goals and access to highly advanced technology allowed the 1994 design concept to use autonomous optical navigation and not require radio navigation or even a command receiver on the microspacecraft.

In contrast, the nine MSSPM microspacecraft were designed to carry out a much more extensive scientific investigation and provide operational warning coverage. Furthermore, a shorter time was expected for technology development, and, therefore, its use of advanced technology was more conservative. Accordingly, the capability to receive uplink signals from Earth was incorporated in MSSPM. Onboard analysis software can therefore be modified as needed from Earth; navigation uses two-way radio communications; and assistance from Earth is possible if difficulties are encountered in autonomous control of the microspacecraft. These navigation changes and new, optional command capabilities, while eliminating the need for specialized optical navigation hardware and software, did bring new challenges associated with the addition of uplink communications.

Before this change, the microspacecraft main telecommunications hardware consisted of a modulator/exciter, solid-state X-band power amplifier, and phased-array antennas. The downlink beam was despun and pointed at Earth by switching the amplifier output to the array pointed closest to Earth at a given time and then using one-axis scanning by that array to complete the pointing process. With the uplink addition, the first change was to expand the modulator/exciter into a full transponder, and its associated miniaturization needs are now probably the most demanding MSSPM technology development. The addition of a diplexer was also planned, but that was unexpectedly problematic. Although the transponder miniaturization was thought feasible, it was not obvious that a small enough diplexer could be developed within the MSSPM time horizon. In fact, it appeared that at least one of the diplexer dimensions could be the same order of magnitude as the diameter of the microspacecraft core. A further problem was that making the phased-array antennas operate efficiently at both the uplink and downlink frequencies was expected to be difficult. The solution employed was to separate the uplink and downlink signal paths. This eliminated the need for the diplexer and dual frequency use of the phased-array antennas. It came with the cost, though, of adding a simple low-gain antenna on the spin axis and of improving the filtering of uplink signals in the transponder.

Another challenge for MSSPM that proved more difficult than expected was communicating hazardous space weather alerts to Earth. What is necessary here is to continuously monitor transmissions from one microspacecraft, usually the one at that time that is closest to the Sun–Earth line. (Since coronal mass ejections pass through the 0.53–0.85 AU solar range of the band of microspacecraft several hours to two days before they reach Earth, a microspacecraft near the Sun–Earth line would be well positioned to provide early warnings.) The issue was that while the collection of MSSPM microspacecraft would be able to return their analyzed scientific information using one or two passes per week from a Deep Space Network (DSN) 34-m antenna, requiring continuous coverage from the 34-m network would excessively load this critical DSN resource. The obvious alternative was to use Earth stations with smaller antennas, and the DSN 11-m network was selected. Even with some modifications at the stations, though, and with maximum warning ranges of only 0.57 AU, compared with a 1.8 AU maximum range for scientific coverage, the combination of limited MSSPM downlink signal power and limited performance of the 11-m stations prevented assurance of adequate downlink margin for normal telemetry. The limited downlink signal power results from constraints on mass and size of the microspacecraft that limit both available power for the transmitter and antenna area. The solution here was to use a modified version of beacon-mode communications. (Margins can be higher in a beacon system because the effective data rate is very low and more integration time is available on Earth.) Specifically, MSSPM uses a four-tone beacon alert system in which a change in tone provides a hazardous space weather alert and then a sequence of tones over a half-hour period provides data regarding the nature of the perceived hazard. The station on Earth has 116 s to integrate each tone.

Propulsion

Hydrazine was considered as a propellant and would have resulted in less needed propellant and smaller tanks than those eventually used in the design. The expectation, however, was that the mass and size of the thrusters would be excessive and that the mass for propellant management in the tanks would increase. In addition, there was another problem with hydrazine, the relatively high temperature at which it freezes. The concern here results from a consequence of designing the microspacecraft to handle the 4-Sun-equivalent thermal environment that exists at 0.5 AU from the Sun. That same design makes temperatures early in the mission, at 1 AU, low and introduces the potential for the hydrazine to freeze. Another alternative was to use a relatively standard cold gas propulsion subsystem. The problems with this included excessive size and mass of the propellant tanks, as well as valve leakage throughout the mission. Instead of these alternatives, the chosen solution was to use vaporizing liquid ammonia propellant, which has a very low freezing point. The needed propellant volume (and pressure) can be accommodated in the microspacecraft; propellant management within the tanks requires little mass; it is expected that very small microthrusters can be utilized; and a liquid-vapor interface at certain valves can be used to prevent excessive leakage.

The decision to use this type of propulsion, however, brought further challenges. The need to supply the heat of vaporization for the propellant was expected, but the calculated magnitude of that heat was a surprise. It was on the order of two megajoules. Moreover, most of that heat needs to be supplied for the initial ΔV maneuver when the microspacecraft is farthest from the Sun, is coldest, and has the least power available from the solar arrays. The solution here primarily involved stretching out the maneuver times to decrease needed mass flow rates and, therefore, simultaneously reduce the power consumption to an acceptable level. Other challenges in propulsion included the need to prevent cavitation in the tanks during launch and unintended, temperature-related propellant transfer between tanks after launch (both of which were unexpected challenges) and preventing reliquefaction after vaporization (an expected challenge). Briefly stated, the solutions involved adding helium to the tanks to make the total tank pressures higher than the ammonia partial pressures and prevent cavitation, using a valve at each tank to control propellant transfer, and maintaining low enough pressures after vaporization to prevent reliquefaction. The use of helium added another challenge: keeping it in the tanks (particularly when the valves are opened prior to initial spin-up). The solution employed was to add gas-arrestor screens in front of the tank valves.

Guidance and Control

Although the use of spin stabilization simplified MSSPM attitude control, it left significant challenges. For example, to help prevent the launch integration system from being excessively complex and massive, it does not spin up the microspacecraft prior to their release. That job is thus left up to the microspacecraft. In addition, the microspacecraft collectively have multiple orientations when attached to the launch integration structure, and, prior to spin-up, it is desirable to adjust their orientations so that the Sun will be near 90 degrees from the spin axes. Therefore, the microspacecraft were designed to have short-term, three-axis knowledge and control capabilities. Rather than using celestial sensors that were usable only while spinning, the sensors that were used (an axially pointed star camera/tracker and radially pointed Sun camera/scanner) were designed to function also in a staring mode. A three-axis microgyro was added to the design for multiple reasons. It helps in determining the initial attitudes and rates following separation from the launch integration structure. Also, it was expected that periodic lateral thrusting, which is used in the potentially very long ΔV maneuvers, could adversely impact spin axis pointing control, and it was decided to add an active nutation control capability that employed the microgyro reference.

Another issue related to precision of ΔV maneuver magnitude. Ammonia pressure and temperature knowledge combined with thruster calibration data can be used to calculate and control thrust durations, and it may even be possible in some cases to use radio navigation data from Earth to adjust the end times of long maneuvers. Even so, it was thought important to simultaneously improve control of ΔV magnitudes and, if possible, eliminate real-time dependence on help from Earth. The solution identified was to add a one-axis microaccelerometer and integrate its output to help in onboard monitoring and control of the maneuvers.

Packaging and Structure

Mass and size constraints made it important to derive multiple benefits from much of the structure and packaging mass in the microspacecraft, as well as to make efficient use of the volume within the microspacecraft. A box with relatively thick (0.25-cm) aluminum walls at the center of the microspacecraft was utilized not only to house the electronics modules but also to help isothermalize them and shield them from radiation. The box simultaneously serves as a primary structural element for the microspacecraft and provides the interface between the launch integration structure and other structural elements. These include not only dedicated radial, semiradial, and side supports but also horizontal plates that serve both as structure and thermal radiators. In addition, each of the box sidewalls supports both a thin battery cell and the inboard end of a propellant tank. The resulting configuration is shown in Figure 3, which is a top view of the microspacecraft in the launch configuration but with the solar arrays, low-gain antenna, electronics box cover, radiators, and thermal blankets removed.

Temperature Control

As discussed earlier, a shape for the microspacecraft was chosen that would simplify radiative cooling of the spacecraft core and solar arrays while limiting the solar input to the arrays. A serious unexpected challenge, however, was limiting undesired conductive heat transfer from and to the solar arrays and side faces of the microspacecraft core. At minimum solar range, these elements are far above the desired temperatures of the electronics box, tanks, and other elements of the microspacecraft, and the reverse is true at maximum solar range. Using insulating fasteners was considered but appeared to be quite inadequate to achieve the needed thermal resistance. Adding to the difficulty of the conductive heat transfer issue was that the dimensions of the microspacecraft are small and distances available for possible inclusion of other thermal isolators are short. The design solution used was to make key elements of the structure itself out of a very good thermal insulator, G10 fiberglass, and it was used in the side supports, semiradial supports, and most of the radial supports from the electronics box.

Another challenge was that power limitations make it necessary to cycle the transmitting components off at times during the mission. This in turn results in a significant change in power dissipation within the microspacecraft, and, to prevent associated temperature swings, it is necessary to compensate for the power dissipation changes by changes in the heat radiated from the microspacecraft. Initially, adding mechanical louvers to each of the eight radiators was a candidate for controlling the heat radiated, but the mass for that appeared high. Instead, very low-mass electrochromic surfaces were added to the radiators to control their emissivities. (Heaters are used under some conditions to further limit temperature swings.)

Other challenges associated with the temperature control design were met using solar heat shields on half of the microspacecraft bays, carefully selecting surface properties, and using thermal blankets.

SUMMARY

Some examples of the challenges encountered in the MSSPM detailed conceptual design have been discussed in this paper along with the solutions that were identified. While it was anticipated that the greater depth of this most recent study would reveal issues and challenges that had not been fully illuminated previously, the number and complexity of the challenges uncovered was surprising. In retrospect, this resulted from the combination of inherent challenges in the mission, the nature of interplanetary flight, interrelated mission and microspacecraft design drivers, and associated new technology needs. Fortunately, a solution was identified for each challenge, and the conclusion was that the mission is both technically feasible and highly attractive. At the same time, though, this experience suggests that the "war stories" of future interplanetary microspacecraft design may also be characterized by both more challenges and more unexpected challenges than have been routinely encountered in past spacecraft designs.

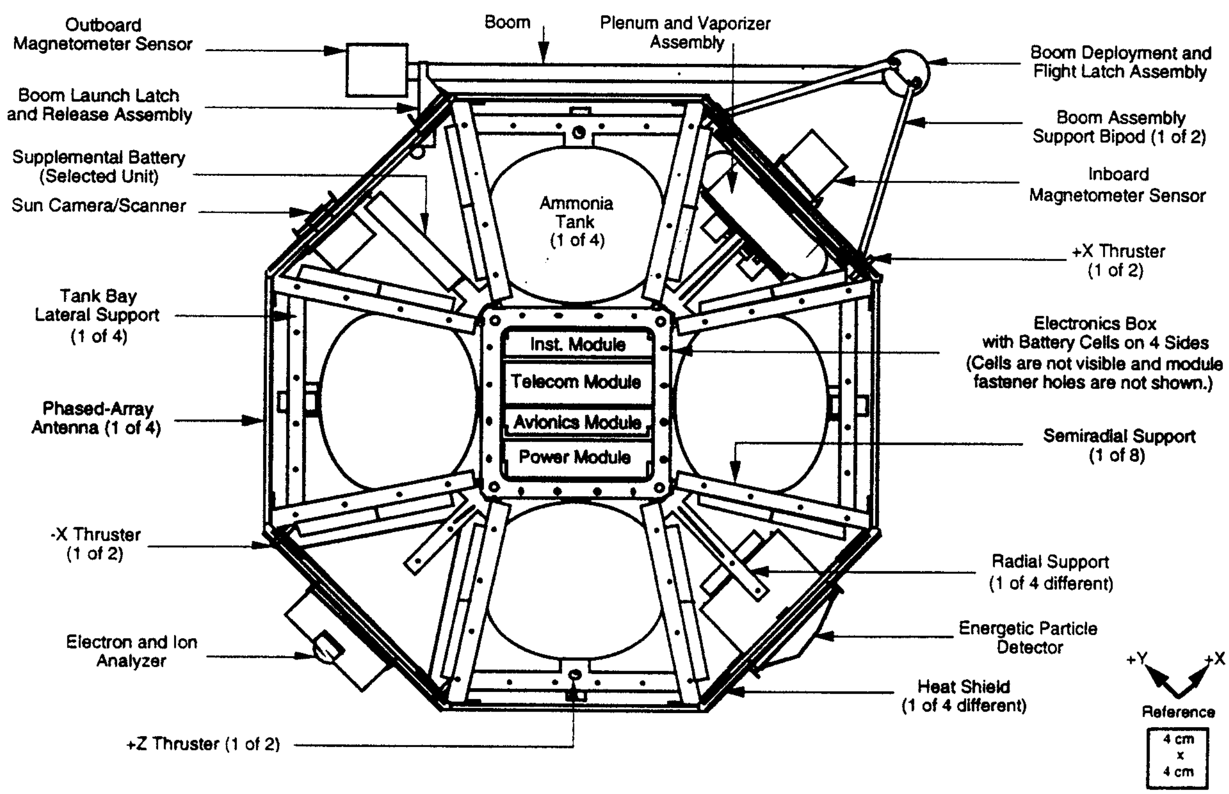

Figure 3: Top View in Launch Configuration

REFERENCES

1. Collins, D. H., Horvath, J. C., et al., "Space Physics Fields and Particles Mission Class Spacecraft," *Second Generation Microspacecraft Final Reports for 1994*, JPL internal document D-12645, Rev. A, Jet Propulsion Laboratory, California Institute of Technology, Pasadena, CA, 1995, pp. 59–89

2. Collins, D., Pomphrey, R., Yen, C., Stallard, M., Cooke, D., et al., "Multimission Space and Solar Physics Microspacecraft," *Air Force Research Laboratory – Jet Propulsion Laboratory Future Collaborations in Microsatellites Study*, AFRL–JPL internal document, Pasadena, CA, Apr. 1999

3. Collins, D. H., et al., *NASA/JPL Multimission Space and Solar Physics Microspacecraft*, JPL internal document JPL-18809-A, Jet propulsion Laboratory, California Institute of Technology, Pasadena, CA, Apr. 2000

4. Collins, D. H., "Multimission Space and Solar Physics Microspacecraft," a paper presented at the Fourth IAA International Conference on Low-Cost Planetary Missions, Laurel, Md., May 2–5, 2000, and submitted (as Revised IAA-L-1110) for publication in *Acta Astronautica*.

5. Collins, D., Kukkonen, C., and Venneri, S., "Miniature, Low-Cost, Highly Autonomous Spacecraft — A Focus for the New Millennium," IAF-95-U.2.06, 46th International Astronautical Congress, Oslo, Norway, Oct. 1995

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