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Inner Magnetosphere Imager (IMI) Solar Terrestrial Probe Class Mission Preliminary Design Study Report

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LIST OF ACRONYMS

AO	Announcement of Opportunity
C	Celsius
C&DH	communications and data handling
cm	centimeter
dc	direct current
DSN	Deep Space Network
EOL	end-of-life
FOV	field-of-view
FUV	far ultraviolet
Ga-As	gallium arsenide
Gb	gigabits
GEM	
GN&C	guidance, navigation, and control
I&T	integration and test
IMI	Inner Magnetosphere Imager
I_{sp}	specific impulse
K	Kelvin
kb/s	kilobits per second
kg	kilogram
km	kilometer
lb	pound
LLV	Lockheed Launch Vehicle
m	meter
m^2	square meter
mb/s	megabits per second
N	Newton (kg m/s^2)
Ni-H_2	nickel hydrogen
N-mH	nickel metal hydride
OAM	orbit adjust module
OSC	Orbital Science Corporation
OSR	optical surface reflector
RCS	reaction control system
R_e	Earth radii
RF	radio frequency

RFP	Request for Proposal
rpm	revolutions per minute
S/C	spacecraft
SDT	Science Definition Team
STP	solar terrestrial probe
str	steradian
W	Watts

TECHNICAL MEMORANDUM

INNER MAGNETOSPHERE IMAGER (IMI) SOLAR TERRESTRIAL PROBE CLASS MISSION PRELIMINARY DESIGN STUDY REPORT

INTRODUCTION

Science Rationale

One of the most important discoveries of the space age was that of the Van Allen radiation belts around the Earth. These belts are vast clouds of intense radiation that are caused by the Earth and its rotating magnetic field being impacted by the supersonically expanding atmosphere of the Sun. After 30 years of spacecraft flights through this region, it is known that these radiation clouds contain electrical storms and disturbances that play an important role in the Earth's atmospheric processes.

Through technology advances, pictures of this magnetospheric cloud can be made similar to the satellite photos of ordinary clouds commonly used for weather reports. Thus, NASA is poised to explore and expose this violent and variable region that surrounds the planet with entirely new types of satellite images.

The George C. Marshall Space Flight Center (MSFC) has responsibility for defining potential future space science programs, one of which is the proposed IMI mission. For three decades, magnetospheric field and plasma measurements have been made in situ by diverse instruments flown on spacecraft in many different orbits, widely separated in both space and time, and under various solar and magnetospheric conditions. Scientists have used this information to piece together an intricate, yet incomplete, view of the magnetosphere. A simultaneous global view, using various light wavelengths and energetic neutral atoms, could reveal exciting new data and help explain complex magnetospheric processes, thus providing a clearer picture of this region of space.

To provide these measurements, NASA assembled a Science Definition Team (SDT) to develop the scientific objectives of a magnetospheric imaging mission. Concurrent with the formation of this team, MSFC was given responsibility for defining the mission and subsequently formed an engineering team to begin concept design studies. The result of these efforts is IMI.

In order to better understand this environment, the IMI mission will pose the following questions:

- What does the global magnetosphere look like in quiet and disturbed conditions?
- How do the principal magnetospheric regions globally change in response to internal and external influences?
- How are the principal magnetospheric regions interconnected?
- What are the remote global signatures of the important astrophysical processes occurring in the magnetosphere?

Mission Definition

The IMI was originally conceived to be a part of the Space Physics Division's intermediate class of missions with a cost ceiling of $300M. The engineering studies performed at MSFC indicated that a spinning spacecraft with a despun platform, similar to General Electric's (now Martin Marietta) POLAR and Hughes' HS-376 spacecraft, launched aboard a Delta, could easily accommodate the strawman science instruments defined by the IMI SDT. A summary of the intermediate-class IMI mission spacecraft and instrument complement is listed in table 1.

Table 1. Options for the IMI mission.

	Intermediate Class Mission	Solar Terrestrial Probe Mission
Cost Ceiling ($)	300M	150M
Launch Vehicle	Delta II	Conestoga, Taurus, or Lockheed launch vehicle (LLV)
Orbital Parameters	4,800 by 44,600 km (7 R_e) 90°	4,800 by 44,600 km (7 R_e) 90°
Instruments	Seven (four on spinning spacecraft; three on despun platform)	Three "core" (one is a consolidation of three from the original list) plus up to three "mission enhancing"
Total Spacecraft Mass (kg) (wet; including 30-percent contingency)	1,000 (HS-376) 1,300 (POLAR)	413

In the summer of 1993, the IMI SDT and MSFC engineering team were directed by the Space Physics Division to redefine the IMI mission to fit within a new class of missions, the Solar Terrestrial Probe (STP). STP missions must cost no more than $150M (excluding launch cost) and be accommodated on launch vehicles smaller than a Delta. The IMI SDT met in November 1993 and developed a new strawman instrument complement suitable for a smaller spacecraft but still capable of meeting the core science objectives necessary for magnetospheric imaging. This report summarizes the preliminary results of the IMI STP design study.

SCIENCE INSTRUMENT COMPLEMENT

The IMI SDT developed the initial strawman instrument complement for the IMI mission. It consisted of seven instruments with a total mass of 187 kg, requiring 190 W of power. Four instruments operated in a scanning mode and three instruments operated in a staring mode; therefore, requiring a spinning spacecraft with a despun platform. The STP mission guidelines necessitated the development of a new strawman instrument list by the SDT. This list includes three core instruments and three enhancing instruments as described in table 2. The concepts discussed in this report only accommodate the three core instruments. The addition of enhancing instruments will be considered in future studies. The instruments' technical parameters were also provided by the SDT. Other sources have indicated that a reduction in instrument electronics by 30 percent in volume, mass, and power is possible. This reduction was presented to the SDT and was considered reasonable, but not preferred.

Table 2. IMI STP strawman instrument list.

Instrument Name	Field-of-View (FOV)	Dimensions (W×D×H) m	Mass (kg)	Power (W)	Data (kb/s)	Point Acc. (degree)
CORE						
Hot Plasma Imager (H)	4π str	0.51×0.35×0.51	14.0	4.0	12	5.0
Hot Plasma Imager (L)	4π str	0.30×0.30×0.25	7.0	7.0	6	
Electronics		0.30×0.30×0.30	8.0	12.0		
Plasmasphere Imager (He+304)	135°×180°	0.48×0.16×0.20	7.2	4.5	7	0.5
Electronics		0.23×0.18×0.20	11.8	16.5		
FUV Imager and Electronics	40°×360°	0.70×0.80×0.30	30.0	25.0	15	
Total			78.0	69.0	40	1.0
ENHANCING						
Plasmasphere Imager (O+834)	135°×180°	0.48×0.16×0.20	7.2	4.5	7	0.5
Electronics		0.23×0.18×0.20	11.8	16.5		
Electron Precipitation Imager	3°×3°	0.20×0.20×0.60	24.5	11.0	2	0.3
Electronics		0.25×0.18×0.18	3.0	9.0		
Radio Sounder (four units)		0.22×0.12×0.12	35.2	10.8	6	
Spin Axis Antenna (two units)		0.50×0.20×0.18				
Electronics		0.20×0.18×0.15				

Two of the core instruments are from the original IMI instrument list but the third, the Far Ultraviolet (FUV) Imager, is a combination of three ultraviolet imagers from the original instrument complement: two staring and one scanning. All three core instruments operate in the scanning mode, eliminating the requirement for a despun platform. The total core instrument mass is 78 kg and the power requirement is 69 W. The Electron Precipitation Imager must operate in a staring mode and would require the addition of a despun platform, driving up the cost and complexity of the mission. The other two enhancing instruments operate in a scanning mode, thus making their potential inclusion to the current IMI STP mission somewhat less difficult.

MISSION ANALYSIS

The IMI mission orbit has a perigee altitude of 4,800 km, and an apogee altitude of 44,600 km (7 R_e). The apogee of 7 R_e was a requirement specified by the SDT, and the perigee altitude of 4,800 km was driven by the performance capability of the POLAR spacecraft propulsion system, the intermediate class mission spacecraft mass, and the Delta II launch vehicle performance capability. Other concerns driving the orbit selection included avoiding monatomic oxygen in the upper atmosphere at the 1,000- to 1,500-km altitudes and high plasma densities at altitudes less than 4,800 km. Because of these environmental constraints and to maintain instrument viewing perspective, this orbit is considered nominal for the solar terrestrial probe mission.

The achievable IMI perigee is dependent upon the amount of propellant that can be loaded onto the spacecraft, the spacecraft mass, and the launch vehicle capability. The IMI SDT asked that a trade study be performed to determine the available payload mass as a function of perigee altitude for the current STP option. The results, presented in figure 1, can be summarized by stating that for every 100 km the perigee is reduced, 1 kg of additional mass (science instrument or spacecraft) can be placed into the desired orbit. Any spacecraft subsystem mass changes directly affect the science instrument mass that can be accommodated.

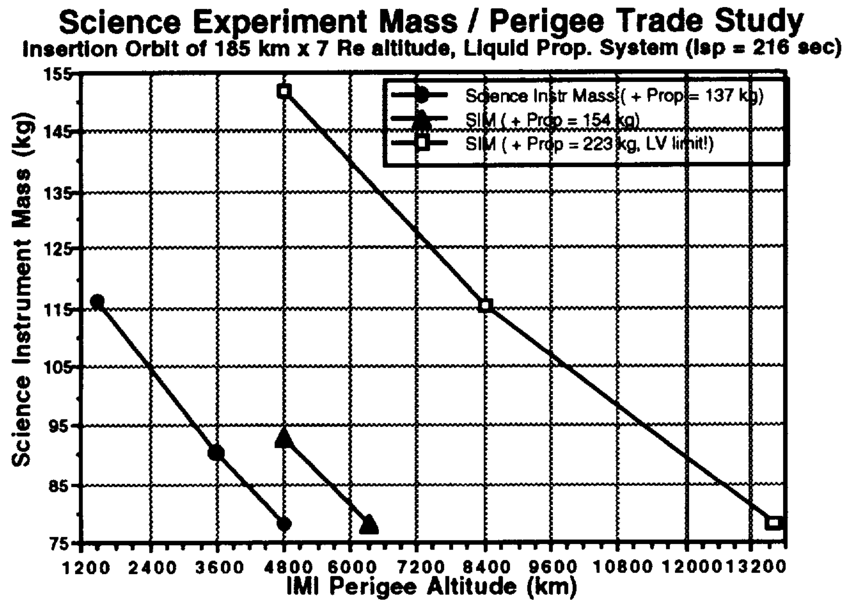


Figure 1. Experiment mass/perigee trade study.

LAUNCH VEHICLES

Three launch vehicles, the Taurus 120 XL/S, Conestoga 3632, and Lockheed LLV3, were considered for the STP option. Performance estimates for the insertion orbit of 185 km by 7 R_e were generated by the respective vehicle manufacturers and the capability of each vehicle is graphically represented in figure 2. The values for the 185-km perigee insertion assume that the spacecraft's propulsion system will be used to achieve the final 4,800 km by 7 R_e orbit.

The configuration description and capabilities of the vehicles that were under consideration are discussed briefly in the following paragraphs.

Taurus 120 XL/S

The Taurus 120 XL/S is being developed by Orbital Sciences Corporation (OSC) and is the smallest launch vehicle that is capable of completing the mission objectives. The first vehicle in the Taurus line was launched successfully in March of 1994. The initial launch of the Taurus XL vehicle is scheduled for mid-1994, while the 120 XL/S is an enhanced model that is not projected to be

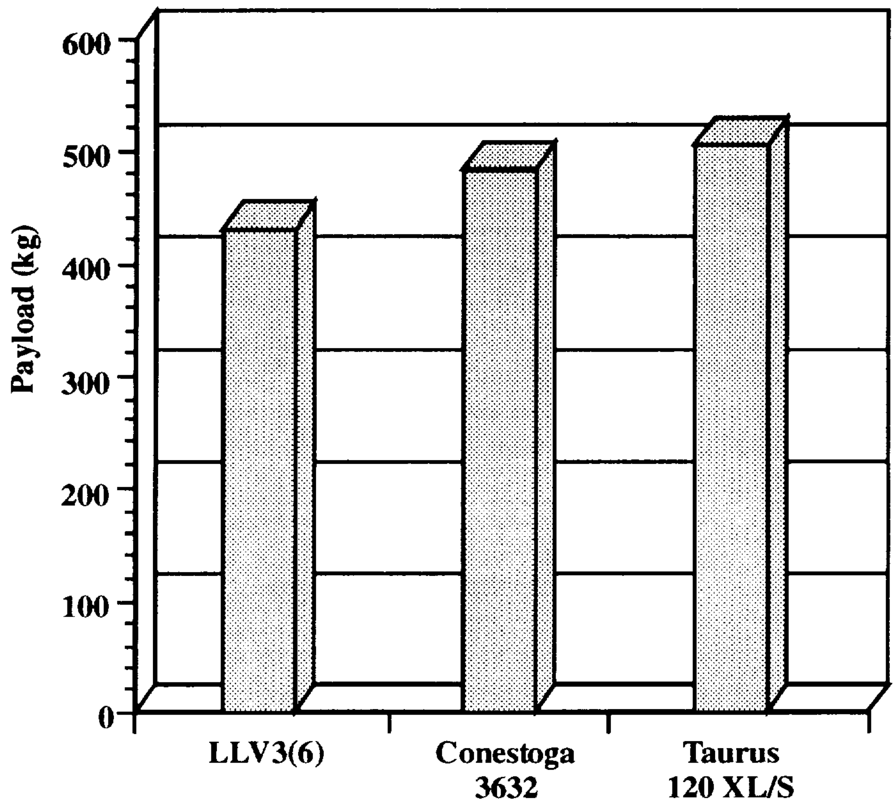


Figure 2. Launch vehicle capabilities.

operational until 1995. The Taurus is composed of Pegasus upper stages lifted by a Thiokol Castor 120 solid motor. In addition, it utilizes two GEM's as strap-on boosters. Based on analysis by OSC, the Taurus will place 500 kg into the 185 km by 7 R_e orbit. The Taurus dynamic payload envelope is 1.4 m, but the IMI spacecraft power and surface area requirements necessitate the consideration of other launch vehicles with larger dynamic payload envelopes.

Conestoga 3632

The Conestoga family of launch vehicles is currently under development by EER Systems, Inc. The scheduled launch of the first all-solid Conestoga vehicle is mid-1994. Commonality of the solid motors is the basis of the Conestoga family. For this mission, the five-stage Conestoga 3632 is necessary. The first three stages are comprised of the core Castor IVB XL and the surrounding two Castor IVA XL and four Castor IVB XL strap-on motors. The fourth and fifth stages are an Orion 50 XL and a Star 48V, respectively.

EER Systems estimated the performance of the Conestoga 3632 to insert a payload mass of 481 kg into the 185-km by 7-R_e orbit. This requires the spacecraft to have an onboard propulsion system to achieve the final desired orbit placement. The total payload mass includes not only the separated spacecraft, but also any special attachment structures which may be required. Although the boost capability of this vehicle is less than the Taurus XL/S, the larger payload dynamic envelope of 1.6 m makes it an attractive option.

LLV3(6)

The LLV is a new series of small launch vehicles, with the first flight scheduled in 1994. The LLV3(6) is the smallest member of this family to meet the requirements of the IMI mission. This vehicle, like the Conestoga, is composed of solid motors. The first two stages require a Castor 120, and the third stage is an Orbus 21D. In addition to these motors, there are six first-stage strap-on Castor IVA motors. An Orbit Adjust Module (OAM), located above the Orbus 21D, is attached to the payload. The OAM provides various control functions during flight and can be used for additional maneuvers, such as transfer burns.

With this configuration, Lockheed estimates that 428 kg can be placed into the 185-km by 7-R_e orbit with a payload dynamic envelope of almost 2 m. With additional hydrazine propellant in the OAM, the LLV3(6) can place the spacecraft into the 4,800-km perigee orbit. However, the performance is reduced to 288 kg.

IMI STP Baseline Vehicle

The launch vehicle chosen for the IMI STP mission, based on performance estimates and fairing size, is the Conestoga 3632 presented in figure 3. Included in figure 3 is a closeup view of the 4.9-m payload fairing.

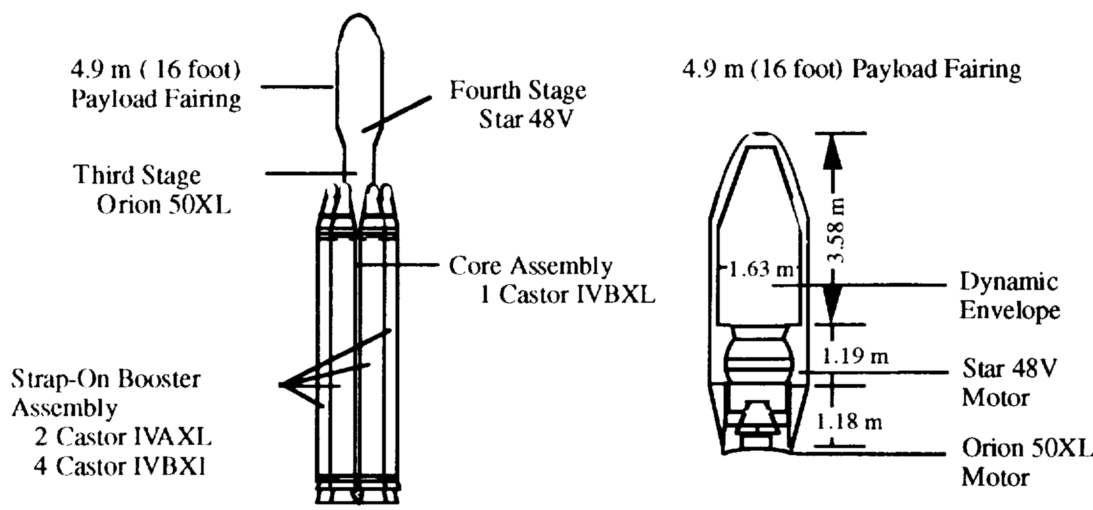


Figure 3. Conestoga 3632 launch vehicle.

CONFIGURATIONS

The baseline configuration for the IMI STP presented in figure 4 is sized to fit a medium launch vehicle, such as the Conestoga 3632 or Lockheed (LLV3) launch vehicle. The instrument complement includes the three core instruments: the Hot Plasma Imager, the Plasmasphere Imager (He+304), and the FUV Imager.

The spacecraft diameter of 1.5 m was chosen as a compromise between launch vehicle payload capacity, power system surface area requirements, and spacecraft stability requirements.

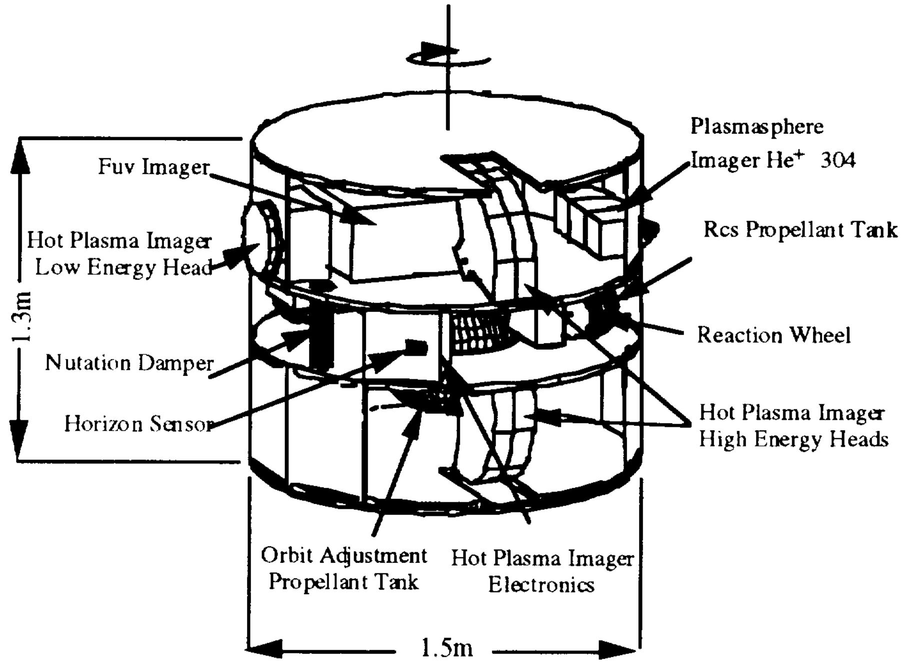


Figure 4. IMI STP preliminary design baseline configuration.

Minimizing the spacecraft size reduces the mass. Maximizing diameter and minimizing spin axis length improves spin stability. The spacecraft length of 1.3 m provides sufficient side wall surface area for solar cells, radiators, antennas, and science instrument view ports. The length is also dictated by the spacecraft subsystems and scientific instrument volumes.

The spacecraft subsystems and science instruments are arranged within the spacecraft to optimize the mass moments of inertia. Placement of the science instruments is restricted by their FOV requirements. The spacecraft subsystems are positioned to account for balancing and compatibility with adjacent components. The addition of any mission-enhancing instruments will necessitate rearrangement of the internal components.

STRUCTURES

The baseline structural design of the IMI spacecraft, shown in figure 5, consists of three aluminum honeycomb plates supported by a side wall and longerons constructed from either aluminum or graphite composite. Modifications to the spacecraft to accommodate the radiator band will result in changes to the plates, which are no longer required to reject heat. These panels may now be fabricated from a graphite composite, although the material selection will be a trade of cost and mass. Construction method, materials selection, and configuration changes will result in a shifting of the structural masses, but no significant mass change is expected. Structural design and analysis will continue to define the configuration of the IMI spacecraft.

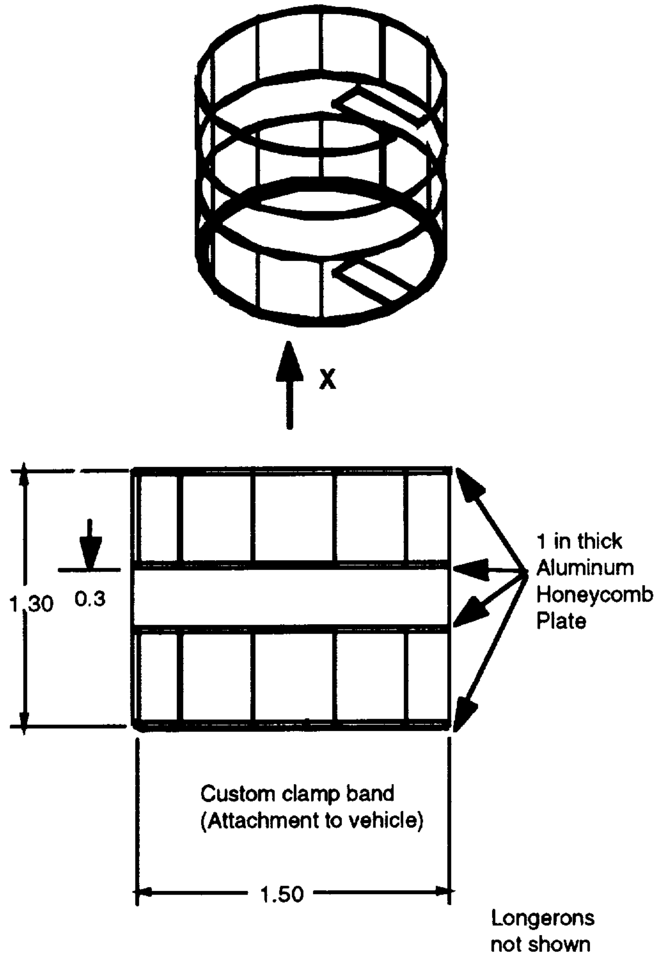


Figure 5. IMI STP structure.

ELECTRICAL POWER SYSTEM

Electrical power load requirements for the IMI STP spacecraft are estimated to be 182.2 W. The total IMI power requirements represent 69 W for the three-instrument payload suite, with the remaining power designated for other subsystems, housekeeping, and contingency. The power subsystem requirements, components, and masses are listed in table 3.

The polar orbiting spacecraft is spin-stabilized with body-mounted solar cells on the cylindrical section and both ends. Total active solar array surface area is estimated to be 7.58 m^2 with a maximum effective illumination area of 2.1 m^2, as shown in figure 6.

Platform orbit orientation of the spin axis is normal to the orbit plane. The results of this orbital profile is a worst-case angle of a ±66.5° between the Sun vector and the orbital plane (beta angle). At these angles, the solar array power output is 363 W. The worst-case beta angle of 0° will give a power output of 252 W. This will result in a power margin of 70 W between the solar array output and the total load demand at end-of-life (EOL) shown in figure 7.

Table 3. Electrical power subsystem summary.

Instrumentation Suite, Core (Full Electronics Power)	
– Hot Plasma Imager (H)	4.0 W
– Hot Plasma Imager (L)	7.0 W
– Electronics	12.0 W
– Plasmasphere Imager(He+304)	4.5 W
– Electronics	16.5 W
– FUV Imager and Electronics	25.0 W
Subtotal:	69.0 W (avg.)
Subsystem Electrical Power Requirements	
– Communications and Data Handling	32.0 W
– Transponder (@ 7-percent duty cycle)	
– Guidance, Navigation, and Control	42.0 W
– Thermal (@ 70-percent duty cycle)	14.0 W
Subtotal:	88.0 W (avg.)
Total Electrical Power Load	
– Instrument Suite	69.0 W
– Subsystems	88.0 W
– Contingency (15 percent)	25.2 W
Total:	182.2 W (avg.)
Surface Area Available (Assumed):	
– Cylindrical Surface: 72 percent	4.40 m^2
– End Surfaces (Two): 90 percent	3.18 m^2
EPS Mass:	
– Solar Arrays	7.74 kg
– Electronics	7.26 kg
– Battery	9.52 kg
– Cabling/Harnesses	20.11 kg
Total:	44.63 kg

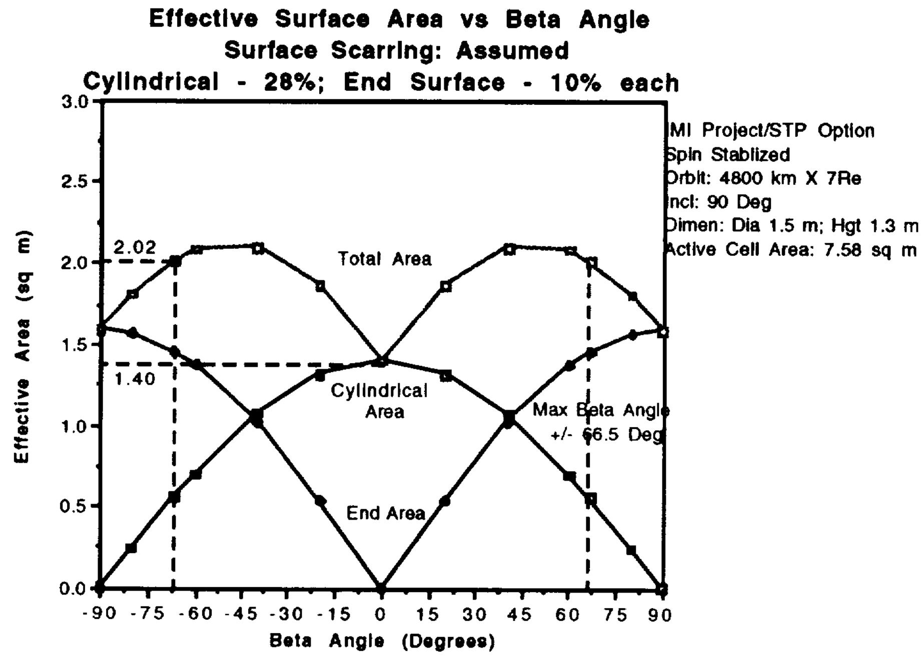


Figure 6. Effective area versus beta angle.

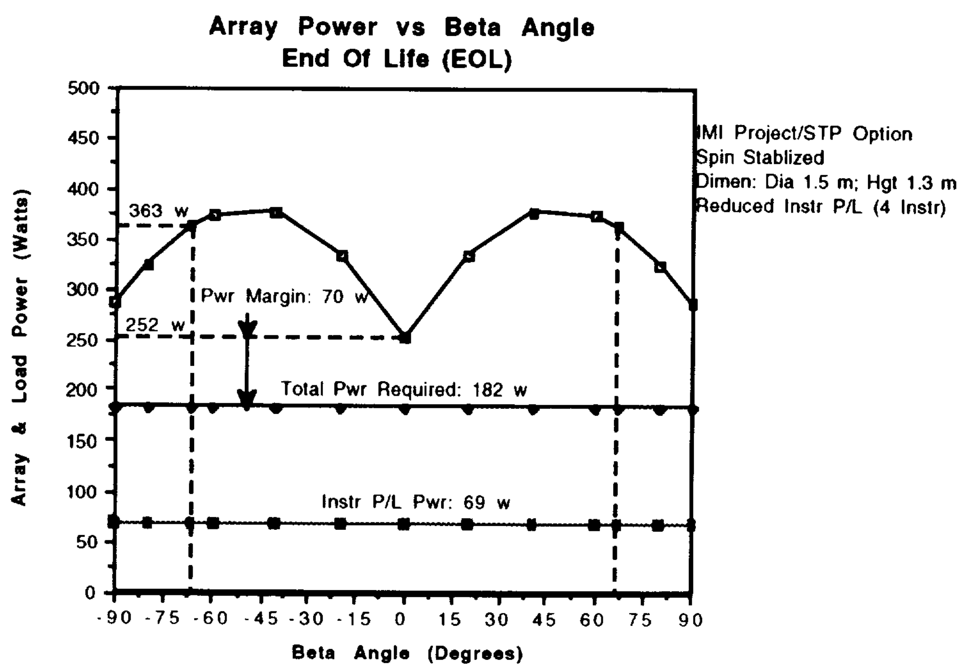


Figure 7. Solar array load power versus beta angle.

Power calculations are based upon an 18.5-percent efficient gallium arsenide (Ga-As) cell. A trade study was performed on several types of Ga-As cells:

- 2- by 4-cm, 3.5-mils thick, 18.5-percent efficient (baseline)
- 4- by 4-cm, 5.5-mils thick, 18.6-percent efficient
- 5.5- by 6.5-cm, 5.5-mils thick, 18.1-percent efficient.

The 2- by 4-cm cell was chosen as the baseline solar cell for the IMI STP mission. Two batteries were considered: a new small satellite nickel hydrogen ($Ni\text{-}H_2$) cell design and a nickel metal hydride (Ni-MH) battery. The $Ni\text{-}H_2$ cell design was chosen for the baseline.

THERMAL CONTROL SYSTEM

The possible addition of the radio sounder, with the attendant requirement to not perform a 180° spacecraft maneuver every 6 months, presents some solar incident radiation problems for the thermal control system. Without flipping the spacecraft, the surfaces used for thermal radiators will be exposed to solar heating for extended periods, thereby degrading the performance of the radiators. Furthermore, there is no position on the spacecraft that radiator panels could be located that would not at some time during the IMI mission be exposed to the Sun. The 180° flip provides an ideal heat sink to deep space for the spacecraft systems thermal loads and the FUV detector which needs to be maintained at about –100 °C.

Two options, shown in figures 8 and 9, were considered for thermal control of the spacecraft in the absence of an orbital "flip" maneuver: (1) locating the radiator surfaces on the ends of the spacecraft and (2) locating the radiator on the cylindrical body of the spacecraft. The thermal control system design was forced to consider impacts on the electrical power system design because both require part of the scarce surface area of the spacecraft body. Option 1 would require that the radiator and solar arrays share the ends of the spacecraft. A requirement of the electrical power system only allows the thermal radiators 30 percent of the spacecraft ends which is about 0.5 m^2 for each. Option 2 requires that the solar arrays and the thermal radiators share the cylindrical portion of the spacecraft, leaving the ends free for solar arrays. The radiating surfaces would need to have optical properties similar to those of the shuttle orbiter radiators, which have a low absorptivity ($\alpha = 0.09$) and a high emissivity ($\varepsilon = 0.81$). This optical surface reflector (OSR) would limit the solar radiation absorbed by the radiator while still allowing the surface to radiate effectively.

Steady-state thermal analyses were performed to evaluate the performance of the two concepts. The end-mounted radiators were modeled in their worst-case condition, where one end of the spacecraft is facing the Sun and the other is opposite the Sun. The analysis of this concept showed that only about 168 W of heat could be rejected at 273 K. Using 0.6 m^2 or 34 percent of the end surface area, 193 W could be rejected, which is about 10 W more than the 182 W required. Results of the analysis of the radiator mounted on the cylindrical portion of the spacecraft, shown in figure 10, indicate that a band approximately 0.2-m wide about the circumference of the body would reject the 193 W in the worst-case condition when the spacecraft cylinder is normal to the solar vector. Therefore, the baseline design is to locate the thermal radiator on the cylindrical portion of the spacecraft.

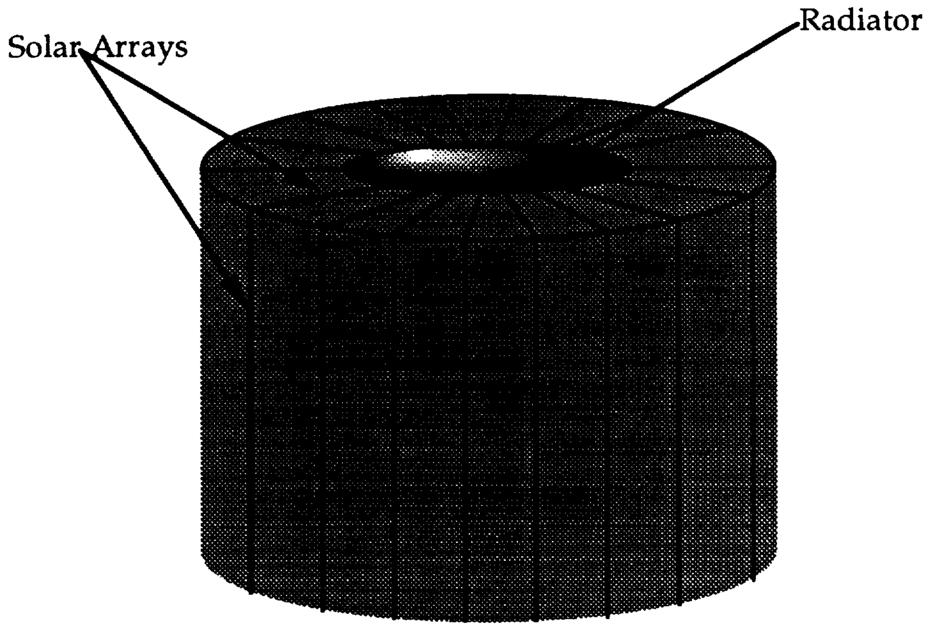


Figure 8. IMI with radiators on spacecraft ends.

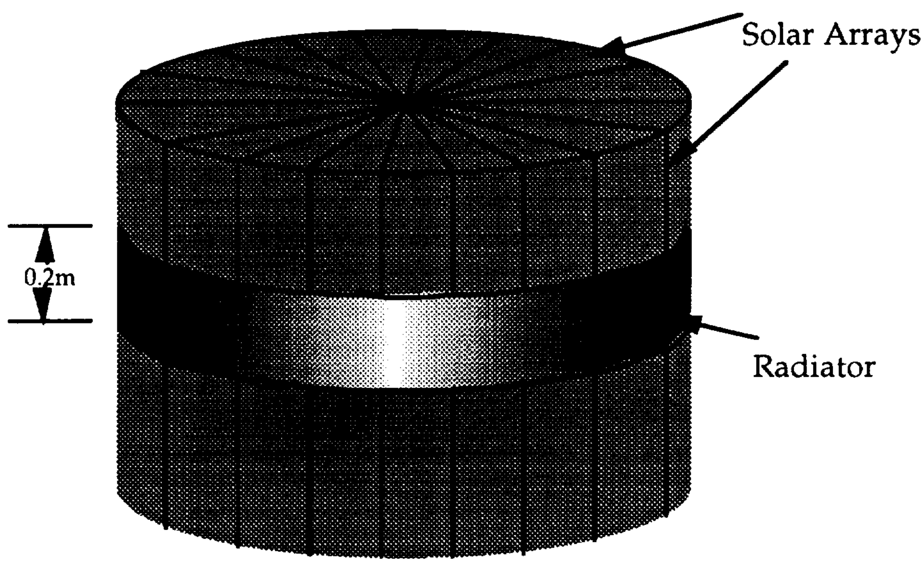


Figure 9. IMI with radiator on cylindrical section of spacecraft.

ATTITUDE CONTROL SYSTEM

The IMI attitude control system should provide a stable spinning platform that meets the science instrument pointing requirements of 0.5° for knowledge, accuracy, and stability over a 1-min period. The spacecraft system should also provide guidance, navigation, and control during orbit transfer from separation of the launch vehicle upper stage to the IMI orbit perigee. Requirements during orbit transfer include a full inertial reference system with sensors and algorithms for orbit and

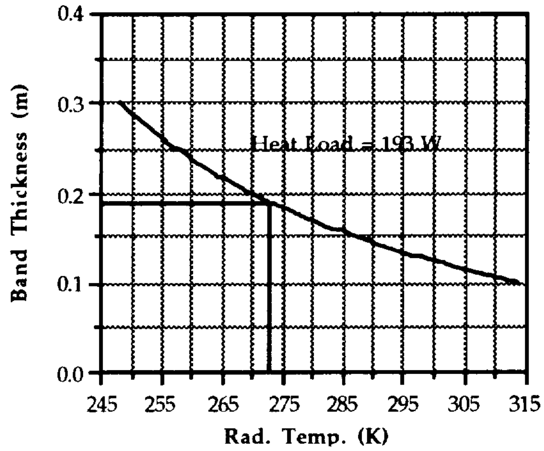


Figure 10. Radiator band size versus radiator temperature.

attitude determinations and a complement of reaction control system (RCS) thrusters to maintain vehicle attitude during orbit transfer. After the spacecraft attains orbit perigee, the RCS thrusters will align the spacecraft spin axis along the orbit normal, remove attitude errors, and then spin the spacecraft to the required 10 rpm. Attitude sensors include rate gyros, fine and coarse Sun sensors, and horizon sensors. A spin-axis damper located at the spacecraft perimeter will provide passive nutation damping. The RCS could augment this nutation damping and provide spin axis control, if needed.

To avoid orbit perturbations due to RCS forces, six pairs of thrusters apply pure couples on the spacecraft and employ simpler control algorithms than those needed for single thrusters. A representation of the spacecraft attitude control system is shown in figure 11. The four pairs of pitch-thrusters will be replaced by two pairs.

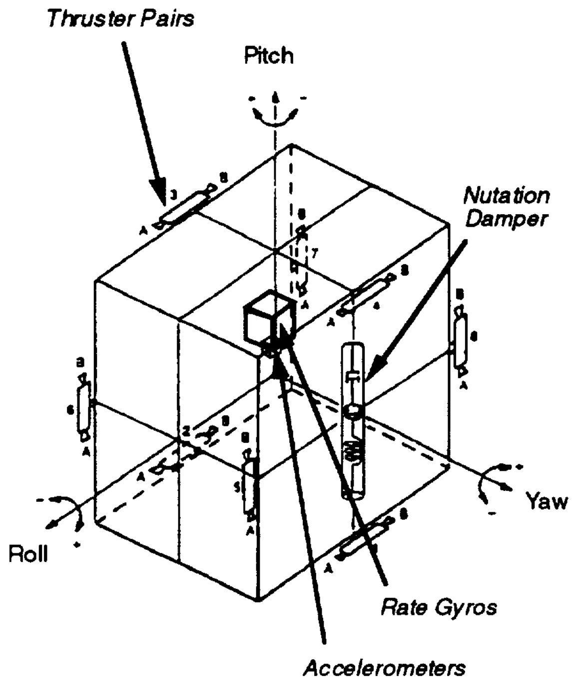
The directional stability of the spacecraft and damper system was assessed for the 1.3- by 1.5-m configuration. The equations of motion in the body-fixed reference frame $\{x_r, y_r, z_r\}$ of figure 12 are:

$$\dot{\boldsymbol{p}} = -\boldsymbol{\omega}\times\boldsymbol{p}+\boldsymbol{f} ,$$

$$\dot{\boldsymbol{h}} = -\boldsymbol{\omega}\times\boldsymbol{h}-\boldsymbol{v}\times\boldsymbol{p}+\boldsymbol{g} , \tag{1}$$

$$\dot{p}_d = m_d\boldsymbol{\omega}^T\boldsymbol{j}\times(\boldsymbol{v}-\boldsymbol{r}_d\times\boldsymbol{\omega})-c_d\dot{\xi}-k_d\xi ,$$

where $\boldsymbol{p}$ is the system linear momentum, $\boldsymbol{h}$ is the system angular momentum, $\boldsymbol{v}$ the body velocity, ω the angular velocity, and $\boldsymbol{r}_d$ is the location of the damper with respect to the body center-of-mass, where $\boldsymbol{r}_d = b\boldsymbol{i} + \xi\boldsymbol{j}$. The sum of external disturbance forces and thruster control forces is $\boldsymbol{f}$, and the sum of external disturbance torques and RCS torques is $\boldsymbol{g}$. The attitude equations are linearized for small perturbations, using small attitude angles α_1, α_2, α_3 . Perturbations about the spin axis are not stable and will not be influenced by the axial damper, so the states are chosen to be $\{\alpha_1, \alpha_3, \xi\}$. The linear system is stable if the system eigenvalues are in the left half plane. Necessary and sufficient conditions for stability can be determined by the following Routh-Hurwitz criteria.



Attitude Sensors:
2 coarse Sun sensors
2 fine Sun sensors
2 high resolution
horizon sensors

Figure 11. Attitude control system components.

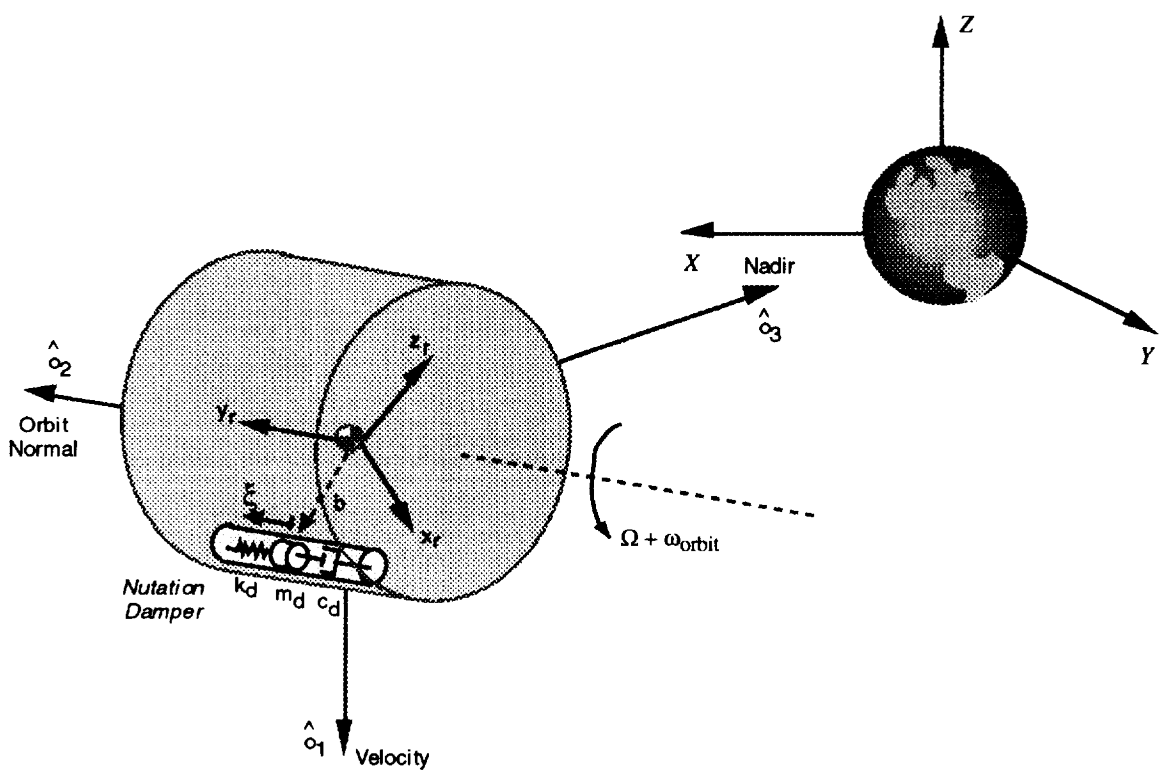


Figure 12. Coordinate frames.

$$k_1 = (I_2 - I_3) / I_1 > 0$$
$$k_{3d} = k_3 - v^2 \frac{m_d b^2 / I_3}{k_d / m_d} > 0 \tag{2}$$

where

$k_3 = (I_2 - I_1)/I_3$.

The principal moments of inertia used for this analysis are I_1 = 101.2 kg m^2, I_2 = 114.5 kg m^2, and I_3 = 94.6 kg m^2. The corresponding spacecraft mass used is 320.5 kg which is an early mass estimate including a 30-percent contingency. The damper mass is 1 kg, and the spin rate v is 10 rpm. The damper properties can be chosen by selecting the damper eigenvalues to match those of the system without the damper, or by examining a root locus of the system eigenvalues for a range of damper parameters. The nutation eigenvalues in the root locus range from damping factors of $\zeta = 0.4$ for k_d of 0.08, to $\zeta = 0.0013$ for k_d of 1.1, with corresponding settling times of approximately 1 min to 4 h. Both techniques yield a directionally stable system using the criteria from equations (2). The damper characteristics will be selected to meet the mission requirements determined by the science instruments.

An estimate of disturbance torques for IMI is shown in figures 13 to 16. The orbit is 4,800-km altitude by 7 R_e, on March 21, 2001, using a 2σ Jacchia density model. Magnitudes of the solar radiation torque, gravity gradient torque, and aerodynamic torque are plotted in figures 13, 14, and 15, respectively. Figure 16 shows the sum of these torques about the spacecraft x, y, and z axes. RCS propellant usage to manage these torques is estimated at 1 kg over the 2-yr lifetime. An additional 1 kg of propellant is needed for initial reorientation and spin-up after orbit acquisition, and 5 kg of propellant is estimated for RCS control during the orbit insertion.

The attitude control system equipment list includes one nutation damper, two coarse and two fine Sun sensors, two high-resolution horizon sensors, three rate gyros, two single-axis accelerometers, control electronics, and cabling. The total system mass estimate is 22 kg, with a total power estimate of 42 W.

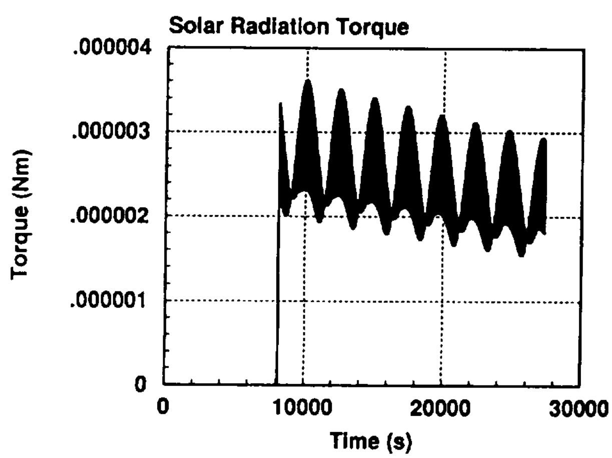


Figure 13. Solar radiation torque.

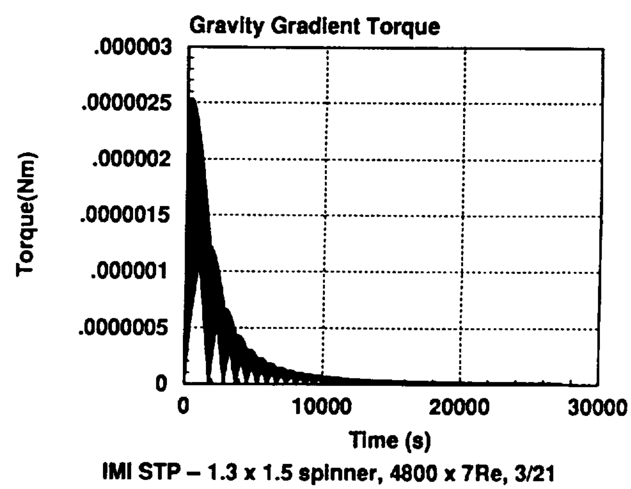


Figure 14. Gravity gradient torque.

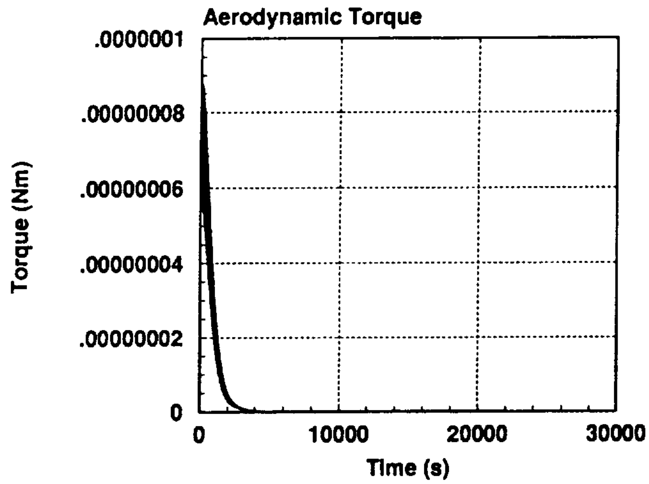


Figure 15. Aerodynamic torque.

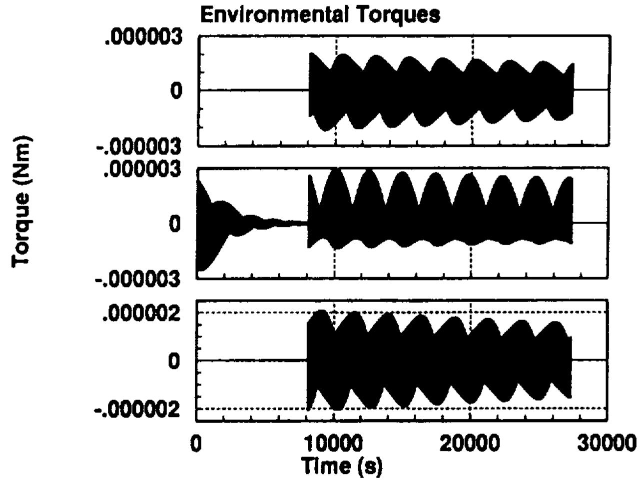


Figure 16. Environmental torques.

PROPULSION

Two options were considered for the spacecraft propulsion system. The first baselined an off-loaded Star 17 solid propellant motor for the orbital reboost with a blowdown monopropellant hydrazine RCS. The mass summary for this option indicates that the total payload mass exceeds the launch vehicle capability: structural accommodation would require a thrust structure, a mission specific payload attachment fixture, and a minimum of two separation systems. An all-liquid blowdown monopropellant hydrazine propulsion subsystem was, therefore, selected following a trade study which indicated that the overall payload weight was reduced due to elimination of the motor structural accommodations required.

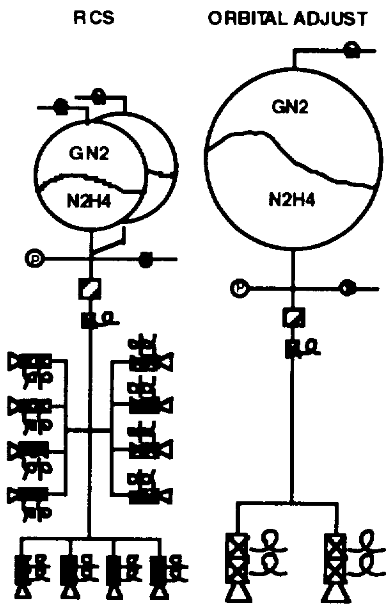
A trade of the options for number of tanks required, placement of the tanks, and the systems operation were all taken into account in the design of the propulsion system. The design consists of two systems: RCS and orbit adjust. The orbit adjust is accommodated by using a single 55.73-cm (21.94-in) diameter tank located along the vehicle centerline with two nominal 66.75-N (15-lbf) thrusters on the spacecraft aft end. A single string isolation system is assumed with all hardware being "off-the-shelf."

The RCS is a similar design based on the same philosophy. Two 23.29-cm (9.17-in) diameter tanks are required in order to keep the spinning spacecraft balanced as the propellant is depleted. The tanks are located in the plane of the vehicle center-of-mass. Bladders are also required as the RCS provides attitude control during orbit transfer. The tanks are purposely oversized in order to maintain the high thrust during the mission. Total propulsion system weight is estimated to be on the order of 100 kg. A summary of the all liquid propulsion system is shown in figure 17.

COMMUNICATIONS AND DATA HANDLING

The performance of the communications and data handling (C&DH) system depends primarily on the data rate and the transmitter power output. The data rate is fixed at 40 kilobits per second (kb/s) for the three core instruments, which yields about 2 Gb of data for each 15-h orbit. The data would be stored on a solid-state recorder and downlinked once per orbit. The recorder, with a minimum of 2.5 Gb capability, was selected based upon mass and power restrictions.

IMI SOLAR TERRESTRIAL PROBE
ALL LIQUID SPINNER SPACECRAFT



NUMBER	EQUIPMENT	WEIGHT (LB)	WEIGHT (KG)
5	FILL & DRAIN VALVES	2.50	1.14
1	O/A PROPELLANT TANK (22.14 ")	12.50	5.68
2	RCS PROPELLANT TANK (9.41 ")	6.50	2.95
2	PRESSURE TRANSDUCER	2.20	1.00
2	FILTER	2.00	0.91
2	ISOLATION VALVE	4.20	1.91
12	THRUSTERS (1 LBf)	9.60	4.36
2	THRUSTERS (15 LBf)	5.00	2.27
	MISCELLANEOUS (15%)	6.68	3.03
	TOTAL DRY WEIGHT	51.18	23.26
	PROPELLANT - USABLE	165.20	75.09
	RESIDUAL	3.37	1.53
	PRESSURANT	2.11	0.96
	TOTAL SUBSYSTEM WEIGHT	221.86	100.85

Figure 17. Schematic for propulsion system.

The downlink time and rate are dependent on antenna gain and transmitter power. With a spinning spacecraft, an omni-directional antenna with little or no gain is dictated. Transmitter output is limited by the dc power availability on the spacecraft. For the IMI STP, 10 W of radio frequency (RF) power was chosen as an acceptable compromise. A downlink rate of 1.5 mb/s was chosen as a value that will give acceptable transmission times at positive link margins. The 1.5-mb/s rate and 10 W of RF power will give positive link margins out to about four Earth radii with either the Deep Space Network (DSN) 26- or 34-m antennas, and a downlink time of about 24 min for one orbit of data. A minimum of 24 min of contact time with one of the DSN stations will be available on most of the IMI orbits.

Commands to the spacecraft will be at a much lower data rate and should be possible at any point in the orbit. Refinements of the communications and data handling system may be possible by varying data rate, transmitter power, or antenna type. These factors will be reexamined as the design of the system matures.

MASS PROPERTIES

The current IMI STP baseline system mass summary, shown in table 4, is a top-level summary of each spacecraft subsystem, the three core science instruments (table 2), and the propellant required for orbit boost and RCS. The current total launch mass is 413 kg with a 30-percent contingency on the spacecraft subsystems and science instrument masses. The launch margin of 68 kg is calculated using the estimated performance capability of the Conestoga 3632 to place 481 kg in the insertion orbit of 185 km by 7 R_e.

Table 4. Mass summary for the current IMI STP baseline.

	Baseline Core Instruments Full Size Electronics
Structures	60.4
Thermal Control System	4.8
Attitude Control System	22.0
Electrical Power System	24.5
Cabling/Harness Assembly	20.1
Communications and Data Handling	25.0
Propulsion System (dry)	23.3
Spacecraft Contingency (30 percent)	54.0
Spacecraft Dry Mass	234.1
Total Propellant	77.6
Science Instruments	78.0
SI Contingency (30 percent)	23.4
Total Launch Mass	413.1
Launch Margin (Conestoga)	67.7

APPROVAL

INNER MAGNETOSPHERE IMAGER (IMI) SOLAR TERRESTRIAL PROBE CLASS MISSION PRELIMINARY DESIGN STUDY REPORT

By M. Herrmann and L. Johnson

The information in this report has been reviewed for technical content. Review of any information concerning Department of Defense or nuclear energy activities or programs has been made by the MSFC Security Classification Officer. This report, in its entirety, has been determined to be unclassified.

J.M. McMILLION
Director, Program Development Directorate

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For three decades, magnetospheric field and plasma measurements have been made by diverse instruments flown on spacecraft in many different orbits, widely separated in space and time, and under various solar and magnetospheric conditions. Scientists have used this information to piece together an intricate, yet incomplete view of the magnetosphere. A simultaneous global view, using various light wavelengths and energetic neutral atoms, could reveal exciting new data and help explain complex magnetospheric processes, thus providing us with a clear picture of this region of space.

The George C. Marshall Space Flight Center (MSFC) is responsible for defining the IMI mission which will study this region of space. NASA's Space Physics Division of the Office of Space Science placed the IMI third in its queue of Solar Terrestrial Probe missions for launch in the 1990's. A core instrument complement of three images (with the potential addition of one or more mission enhancing instruments) will fly in an elliptical, polar Earth orbit with an apogee of 44,600 km and a perigee of 4,800 km. This paper will address the mission objectives, spacecraft design considerations, interim results of the MSFC concept definition study, and future plans.

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