Conceptual Design of a Combined Mercury Orbiter and Solar Physics Mission

R. Gershman
Jet Propulsion Laboratory/California Institute of Technology

Abstract

A Mercury Orbiter mission has been assigned high priority in the Roadmaps of two of NASA's four space science themes: Solar System Exploration (SSE) and Sun Earth Connection (SEC). JPL worked with SSE and SEC Science teams to develop a mission concept capable of satisfying the principal goals of both themes. Mercury has been visited by only one previous mission, Mariner 10 in 1974, and while very successful, that mission raised as many questions as it answered. The key investigations for the new mission are: formation and evolution of Mercury, current internal structure of Mercury and its temporal evolution, structure and dynamics of Mercury's magnetosphere and its interaction with the solar wind and the planet's surface and exosphere, and use of the proximity of Mercury's orbit to the sun for observations of the sun and solar wind. Getting to Mercury and into orbit requires substantial propulsive energy. The baseline case uses SEP with launch on a Delta 2 and a Venus gravity assist trajectory. The baseline orbit is driven mainly by science and thermal considerations. Imaging science requires low passes for high resolution while the thermal control requires time at long range from Mercury to dump heat. This results in a highly elliptical orbit: 200 km by 10,000 km with periapse at the equator. The spacecraft is 3-axis stabilized to provide an appropriate platform for remote sensing and uses a turntable to satisfy the needs of some of the fields and particles instruments for full angular coverage . Thermal control is maintained by absorbing heat into storage media during periapse passes (when the view factor of the planet is high) and radiating in the high altitude portion of each orbit.

1.0 INTRODUCTION

In the period of about a year and a half preceding the compilation of NASA's Space Science Strategic Plan in April, 1997, each of the four science themes represented in the Plan developed roadmaps to guide their activities for the next ten to fifteen years. A mission to Mercury plays a prominent role in the roadmaps of both the Solar System Exploration (SSE) and Sun-Earth Connection (SEC) themes. SEC established a Mercury Science Definition Team chaired by Dan Baker of the University of Colorado while SSE assigned the follow-up to a Campaign Strategy Working Group (CSWG) chaired by Jim Head of Brown University. JPL performed mission concept studies in support of both groups and then, upon realization that the resulting concepts were similar, performed the study reported on here in which the goal was to design a single mission capable of satisfying the most important objectives of both themes at an end-to-end cost (including launch vehicle) not exceeding three hundred million dollars (FY97$).

The resulting concept calls for placing a comprehensive set of instruments into a highly elliptical orbit around Mercury (200 X 10,000 km). To effect this with an affordable launch vehicle (Delta 7925H), substantial technology advances will be needed in a solar electric propulsion (SEP) system relative to that being flown in the DS1 mission and in the design of high temperature solar arrays.

2.0 SCIENCE

Many outstanding science questions regarding Mercury and its environment remain since the investigation by the Mariner 10 spacecraft. It has been recognized for some time that these questions are best addressed via an orbital mission; however, such a mission is demanding on spacecraft subsystems and potentially expensive. The concept described here aims to address many of these outstanding questions and contain the mission cost by the application of advances in instrumentation and spacecraft technologies. Science objectives for the proposed mission fall into the following categories:

- Formation and evolution of Mercury: Determine the planet's global geography, look for evidence of volcanism and tectonics, measure the mineralogical and elemental surface composition, study and characterize the polar ice cap and characterize the atmosphere/exosphere.

- Current internal structure of mercury and its temporal evolution.: Characterize the gravity and magnetic fields, compare with interior models of Venus, Earth, and Mars.
- Structure and dynamics of Mercury's magnetosphere and its interaction with the solar wind and the planet's surface and exosphere: Study the physical processes taking place during magnetospheric substorms, and compare them to those at Earth, study convection and energy transport/storage within the magnetosphere and how it is affected by interplanetary conditions, determine the extent to which Mercury's surface and atmosphere has been modified by the solar and the magnetospheric charged particle populations.
- Use of the proximity of Mercury's orbit to the sun for observations of the sun and solar wind.

The primary planetology science objective is global spectral mapping of Mercury at a resolution of 200 m or better, preferably 100 m. As yet unmapped areas (more than half of the surface) have the highest priority. When global planetary mapping is mentioned, polar low circular or near-circular orbits come to mind, since they offer the best global coverage at uniform resolution. Unfortunately, Mercury's thermal environment makes low circular orbits extremely challenging and thus costly, requiring some form of active cooling due to the occasional dual load of solar heat from one side plus, on the other side, nearly 2π-steradians of thermal radiation from Mercury's ~700-K surface at the subsolar point. Less demanding and thus less costly mission designs use highly elliptical orbits with equatorial periapses, so the spacecraft has plenty of time far from Mercury to dump the heat absorbed during a dayside periapse pass. Such elliptical orbits make uniform mapping difficult. High latitudes are farther from the spacecraft with concomitantly larger pixel sizes. However, given the current state of Mercury mapping and anticipated budget constraints, an elliptical orbit appears to be the best compromise. The imaging system is designed to provide the specified resolution at the poles and a pixel-averaging strategy implemented in software reduces the highly oversampled region near the equator to more nearly optimal sampling.

For the space physics/magnetosphere objectives, there is strong motivation for a spin-stabilized spacecraft, since some of the instruments need to spin for coverage and noise cancellation reasons, but this is incompatible with imaging requirements. The solution here is to provide a zero-net-momentum spin platform on the otherwise three-axis stabilized spacecraft.

3.0 MISSION DESIGN

The ΔV for the transfer to Mercury is about 10.4 km/s. A SEP transfer augmented by Venus gravity assist was developed for launch on the Delta 7925 in October 2005. The trajectory was designed to use two advanced 30-cm thrusters. The launch energy needed for this transfer is C_3 of 6 km^2/s^2. One advantage of a SEP transfer is that the relative velocity at arrival is close to zero, allowing a final approach from almost any direction, so that orientation of the operations orbit about Mercury can be freely chosen. Upon arrival the SEP spirals in requiring an additional ΔV of 1.7 km/sec.

The baseline orbit at Mercury is 200 x 10,000-km near polar ellipse, with the periherm located above the equator. The period is roughly 8 hr. For this orbit, the periapse ground tracks are spaced about 87 km apart, the altitude over each pole is about 1900 km, and the maximum eclipse 105 minutes) occurs at apoherm when the sun is lined up with the line of apsides. In order to minimize the thermal load on the spacecraft, the orbit is aligned so that the periapsis is at local noon when Mercury is at aphelion. As Mercury's oblateness is negligible, this orbit will remain more or less inertially fixed. (An integrated trajectory is needed to indicate the degree to which this orbit will wander due to solar perturbations.) During the course of the 88-day Mercury year, the subsolar point can be under both the periherm and apoherm, and eclipses will vary from none to the maximum of 105 min. The time for the spacecraft to travel around periherm passage (true anomaly –90° to +90°) is about 54 minutes.

4.0 INSTRUMENTS

The baseline instrument set includes:

- UVVIS five-band global coverage at 100 to 200 m surface resolution.
- Laser ranging with along-track sample spacing of about 1 km.
- Near-IR imaging with six bands.
- UV spectrometer to measure species in the very thin atmosphere.
- Magnetometer to carefully characterize the known intrinsic magnetic field.

- X-ray fluorescence and gamma-ray spectrometers to determine elemental composition of surface materials.
- Plasma wave subsystem (PWS).
- Energetic particles detector (EPD).
- (PEPE).
- Dust detector.

Mass and power estimates for these instruments are shown in Table 1.

Table 1. Instrument Set

Instrument	UVVIS	NIR	XRF	GRS	LIDAR	UV Spect.	MAG	PWS	EPD	PEPE	DUST	Totals
Mass (kg)	0.6	2.0	3.0	1.0	3.0	6.0	2.0	4.0	1.0	5.0	0.3	28
Power (W)	2.0	17.4	2.0	3.0	8.0	18.0	2.0	5.0	3.0	9.0	0.5	70

5.0 SPACECRAFT SYSTEMS

Any Mercury mission is dominated by thermal considerations. The thermal subsystem occupied a greater mass fraction and greater challenges than would be required for any mission except for a close flyby of the Sun. In addition thermal considerations play a much stronger role in the design of the power system and propulsion than would normally be the case. In the case of propulsion, the SEP thrusters must be protected from the high temperatures, and the power system will require technology development items such as special adhesives and welding connections.

Table 2. Spacecraft Mass and Power

	Mass (kg)	Power (W) SEP Cruise @ 0.72AU
Instruments	28	6
Bus		
Attitude Control	36	15
Command & Data	10	11
Power	95	124
Propulsion (hydrazine)	9	.3
SEP	79	5200
Structure	147	0
S/C Adapter	31	
Cabling	13	
Telecom	13	10
Thermal	64	35
Bus Total	498	5395
Spacecraft Total (Dry)	526	5401
Mass/Power Contingency	161	320
Spacecraft with Contingency	687	5721
Propellant & Pressurant	10	
SEP Propellant	450	
Spacecraft Total (Wet)	1147	
7925H Launch Capability	1155	

A high level of redundancy was assumed, since the mission duration is several years, and radiation total dose was estimated for one year in Mercury orbit at 47 krad per year with 100 mils aluminum equivalent shielding margin (RDM). Mass and power requirements for spacecraft systems are shown in Table 2.

5.1 Propulsion

Advanced SEP technology is assumed for this design. The technology is in the thrusters, PPUs, and feed-system components. Thruster performance characteristics differ from NSTAR thruster characteristics in having a greater propellant throughput capability, and higher maximum power input capability with comparable higher thrust capability. Specific impulse at the maximum power input is the same as the NSTAR thruster at its maximum power input. The improved NSTAR thruster characteristics are as follows:

Thruster size	30 cm
Power to the thruster PPU (max)	5.0 kW
Thrust at 5.0 kW	184 mN
Specific impulse at 5.0 kW	3280 s
Service life equivalent	320 kg Xe
Thruster mass	5.0 kg
Thruster gimbal mass	2.0 kg

and the PPU characteristics are as follows:

PPU lifetime is twice thruster lifetime.	
PPU mass	20.0 kg
PPU thermal control mass	10.0 kg

The propellant tank mass for 450 kg of Xe is 21 kg, assuming the Xe is stored at 2000 psia pressure, where its density is about 1900 kg/m^3. Composite propellant tank construction was assumed with significantly advanced materials, design, and fabrication technology that results in a design efficiency factor (PV/M) of 4.1×10^6 cm. An advanced propellant feed system uses magnetostrictive valves and microgas rheostats, and two plenum chambers are included.

A monopropellant hydrazine system provides for momentum wheel unloading torques. The hydrazine system uses eight thrusters of initial 0.9-N thrust in a system of 3:1 blowdown over the mission duration. Two existing titanium tanks with elastomeric diaphragm PMD are used.

5.2 ACS

This subsystem provides:

- three-axis stabilization.
- GRPM spin platform for selected instruments
- dual redundant sets or internally redundant ACS hardware (to meet the mission reliability requirements and goals).
- pointing control of ± 0.5° (3 sigma)
- pointing knowledge of ± 0.1° (3 sigma).

Attitude Determination: Attitude determination sensors consist of coarse digital sun sensors, star cameras, and inertial reference units (IRUs). The coarse digital sun sensors provides 4π-steradian coverage at an accuracy of ±0.5°. The star cameras provide three axes of fine attitude referencing in the range of 2 to 30 arcsec. Both the sun sensors and the star cameras can be inherited from the X2000 program. The IRUs each consist of three micro-machined silicon gyroscopes that are being developed by MDL for the X2000 program and JPL's X-33 Avionics Flight Experiment. The gyroscopes will have a 1° to 10°/hr drift-rate capability. A redundant pair of interface electronics (IFE) placed on multi-chip modules (MCMs) allow the attitude determination hardware communicate with the C&DH System. These electronics are based on New Millennium and X2000 MCM technologies.

Attitude and Articulation Control: Attitude control is primarily provided by a reaction wheel system with hydrazine thrusters for momentum management. Some pitch and yaw attitude control is provided by the SEP thrusters when SEP is on. Each reaction wheel can provide 0.04 Nm of maximum torque and has a angular momentum storage capability of 19.5 Nms. A redundant pair of propulsion valve drive electronics placed on MCMs allow the spacecraft C&DH computer to control the eight hydrazine thrusters and the six gimbaled SEP engines. The propulsion valve drive electronics are inherited from the X2000 program. Two sets of single-axis rotary actuators are required to articulate the solar arrays. The ACS design includes one roll rotary actuator and one yaw rotary actuator per solar array. Each rotary actuator is a stepper motor with internally redundant windings and potentiometers. These actuators are based on the DS-1 solar array actuators. The solar array drives and SEP engine gimbals are driven by the same set of gimbal-drive electronics which are based on the DS-1 gimbal-drive electronics. A spin platform mechanism consisting of a despin drive, a momentum-compensation wheel, and electronics is baselined. The spin-mechanism components are based on commercial technology.

5.3 CDS

The spacecraft controller consists of two identical units operating in a string A and string B fashion. The CDS requires 10 MIPs of computing power to support the ACS pointing accuracy, as well as data handling and compression. The CDS collects 17.1 Gb of science and SOH data during a 24-hr time period and performs various lossy and lossless data-compression algorithms on the science data. Data compression ratios range from 1:1 to 4:1. Compression of the science data reduces the science and engineering data volume to 13 Gb. The data will be downlinked during a 6.5-hr period every third orbit. Each CDS string has 14 MCMs weighing 10.4 kg per string. The estimated TID environment at the electronic packages is ~60 krad with an RDM of 2. Radiation-hardened electronics parts are recommended for this mission. Electronic components should be radiation-tolerant Mil-Std-883B screened devices. The electronic components should have SEL and SEU immunity no less than 75 MeV/mg-cm^2. The MCMs provide for the following:

- Analog Inputs: Temperature, voltage and current measurements are captured and stored in memory as SOH (state-of-health) engineering data
- Power PC 604 MIPS processing includes spacecraft commands and science data compression. Power strobe controls power down MCMs when they are not in use. MCMs that are powered off are less susceptible to the effects to the TID environment.
- Low-Power Serial I/O Bus: The low-power serial bus (LPSB) architecture uses Mil-Std-1553B protocol and RS-485 transceivers to communicate with science instruments and spacecraft subsystems. Science data and controls transfer over the redundant bus at 760 kbps.
- RFS Uplink/Downlink: The interface link to the Telecom Subsystem processes uplink data at 2 kbps and downlink science and engineering data 200 kbps. The hardware command decoder (HCD) are included on this MCM.
- Serial I/O: Discrete I/O controls and monitors key functions in the Power, Propulsion and Telecom Subsystems.
- FPGA, 60- to 230-kb Gates: The FPGA captures and compresses raw science data before transferring to non-volatile memory for storage. (6.4 Mbps science data)
- Flash NV Memory (512 MB): Power strobes the flash NV memory are powered off when not in use to increase the reliability of the devices regarding the TID environment.
- 512 KB per CDS string PROM: Mission flight software code is stored twice in each of the CDS strings.

5.4 Power

Power System Architecture: All spacecraft power—including SEP—is generated by a solar array. The solar array also generates power for recharging a secondary battery that supplies electrical energy for non-SEP loads during launch, maneuvers, orbital eclipses, and array over-temperature safing. The solar array generates an unregulated high-voltage bus. A high-voltage power conditioning unit (HVPCU) distributes the high-voltage bus to the SEP system, and also converts it to a regulated 28-V bus for non-SEP power. Power management and distribution (PMAD) electronics manages the battery, converts the 28-V bus to various voltages, and distributes them to non-SEP loads. SEP power is conditioned by dedicated power-processing units within the propulsion system that are not part of the spacecraft power system.

Power Generation: The main power system challenge for this mission is the development of a high-temperature solar cell and array. The 1-yr science phase is conducted at an average solar distance 0.387 AU; however, minimum Sun distance can go as low as 0.307 AU. Assuming that the Sun is the only significant thermal source, the computed temperature of a planar, wing-mounted GaAs array pointed directly at the sun at a solar range of 0.307 AU is 335 °C. Thus off-pointing of the array is required and an operational solar-array temperature limit of 160 °C is assumed. The array needs high-temperature adhesives, high-temperature solder, low-outgassing materials, and welded connections. Some arrays of this type have been built; for example, Helios operated to 0.27 AU. However, long-term operational survival at elevated temperature remains a key developmental issue. The power density assumed for the high-temperature GaAs solar array is 65 W/kg.

Energy Storage: Secondary batteries are used to support certain spacecraft modes (launch, eclipses, and high-temperature safing) when the solar array cannot provide the required power. Battery performance at temperatures above 100 °C is unknown. Operational temperature limits of state-of-the-art secondary batteries are 40 °C to 50 °C. Several battery technologies (e.g., Li-polymer and Na-S systems) in development may provide increased high-temperature performance, but they do not appear to be ready in time for the technology cut-off dates, and operational performance has not been demonstrated. For these reasons the Li-ion secondary battery technology was baselined. This battery technology is currently under development and is expected to be available in the year 2002. This system is predicted to have a capability of 80 to 90 W/kg and 140 W/L; however, without the development of alternate electrolyte systems this technology will face an operational temperature limit of 40 °C to 50 °C.

Power Electronics: The orbiter power electronics are based on rad-hard X2000 technology. MCM packaging is assumed for the power electronics designs presented in this study. Development of first-generation power elements is expected in the 2002 time frame, but may not be available in time for a 2002 launch. Laser pyro-drive cards are assumed for both study cases. The power-electronics power density assumed for both cases is 125 W/kg. Radiation and thermal effects will present a design and thermal control issue for this mission.
Radiation shielding was estimated to be approximately 100% of the electronics' mass. Thermal control and regulation are required to keep the electronics under 105 °C.

5.5 Thermal

Design: The thermal control system must use a variable thermal isolation/radiation, so that the spacecraft is isolated from the environment during close approach with the ability to store thermal energy within the spacecraft, and the ability to reject thermal energy during the apogee of the orbit. Further there may be a necessity to transfer thermal energy within the spacecraft. The thermal design concept uses MLI to isolate the spacecraft elements from the solar and planet environment. There is, in addition, a thermal radiator with a movable thermal shield to radiate thermal energy when the spacecraft is away from the planet, and isolate the spacecraft from the planet-radiated thermal energy. There is a thermal storage device to store thermal energy during the time that the thermal shields are closed. This could be a phase-change device. Thermal energy is transferred within the spacecraft, the most efficient device would be a heat pipe or a pumped loop (Mars Pathfinder used a pumped loop).

Technology: To support the science mission, the Mercury distance at periherm must be at 200 km. New technology needed to support this altitude follows:

- Thermal energy transfer devices--heat pipes/pumped loops.
- Thermal storage--including phase-change elements.
- Thermal isolation/radiation--movable blockers, etc.
- High-performance MLI, thermal surfaces, etc.
- Thermal conduction isolation.
- High-temperature electronic elements.

5.6 Configuration

The structure subsystem mass was estimated parametrically based on the masses of the other subsystems which the structure supports, plus specific mechanisms and components. The baseline assumption is a non-metallic composite structure. An initial configuration concept which minimizes the number of deployed and actuated elements by using a movable sunshade/parasol to control the thermal problems is shown in Figure 1.

5.7 Telecommunications

Telecommunication system hardware consists of:

- One X-/Ka-band HGA (53.74-dBi gain, 3-dB beamwidth of 0.28°)
- Two TDST (X-band uplink/X- and Ka-band downlink)
- Two XSSPA of 15-W RF, to be used for the normal mode of downlink.
- Two LGAs (~ 6.6-dBi gain, 3-dB beamwidth of 65°) for emergencies and low-rate downlink.

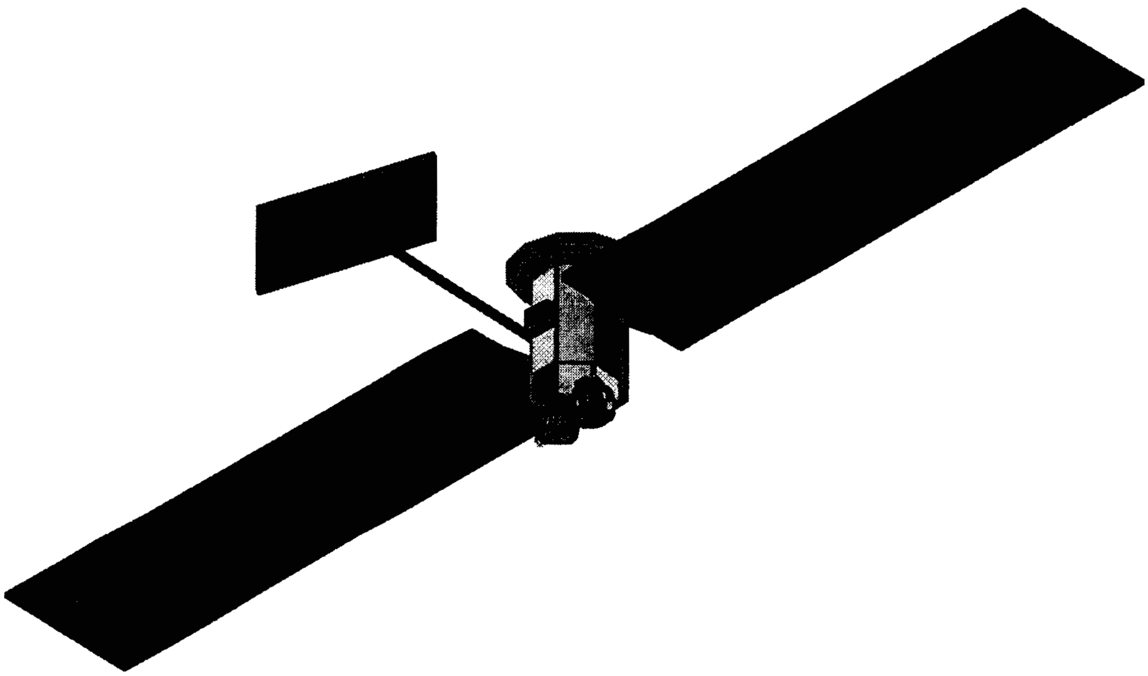

Figure 1: Spacecraft Configuration

Launch/Cruise Mode Communications: Launch mode communications uses smaller DSN antennas such as the DSN 11-m. Once the spacecraft leaves the parking orbit, the DSN's 34-m BWG antennas are used to communicate with the spacecraft using an X-band frequency. The spacecraft uses the 1.5-m high-gain antenna (HGA) for communications. The telecom system uses rate 1/6 constraint length 15 convolutional code concatenated with the JPL standard 223/255 Reed-Solomon code for error control. This code needs about 0.8 dB of bit-energy-to-noise density ratio to support a bit-error rate of 10^{-6}. The link supports a bit rate of about 30 kbps. The radiated power is about 15 W (RF). The ground uses the Block 5 receiver and the link will deliver 3 dB data margin and 6 dB carrier margin. The carrier tracking loop bandwidth of the receiver is assumed to be 10 Hz and a threshold of 12 dB.

Encounter Mode Communications: Science data will be taken two consecutive orbits and is the next orbit the data will be dumped to the DSN station. The telecommunications link uses X-band (8450 MHz). The data rate of 200 kbps is needed during the encounter mode to downlink all the data. To support this rate, the DSN 70-m antenna (DSS 14) will be used.
The telecom system parameters are the same as in the Launch / Cruise phase, The radiated power is still 15 W (the system uses the same power amplifier) RF. The data margin is 3 dB and the carrier margin is 6 dB.

Emergency/Low Rate Mode Communications: This mode is provided for low-rate communications if the high rate fails. The low-rate mode uses the low-gain antenna placed right on the top of the HGA or in the vicinity of the HGA such that when the ACS points for the HGA, the LGA is automatically optimally pointed. Since the spacecraft is a three-axis stabilized spacecraft, there are two LGA antennas at diametrically opposite locations. The gain of the antenna is about 8 dB and the three dB end-to-end beamwidth is about 65°; hence, it needs pointing too. The bit-error rate is assumed to be 10^{-4}. With the same coding as before and 15 W radiated power, a data margin of about 3 dB and the carrier margin of about 4 dB is possible.

6.0 COST

The overall project development cost estimate (phases A, B, C/D) is summarized in Table 3. These costs are based on the the 33-month development life cycle planned for JPL projects launching in the middle of the next decade. Almost every subsystem cost is impacted by the use of SEP technology. Mission and Project Engineering costs are also impacted by the use of SEP technology and these costs have been augmented. Table 4 lists the Mission Operations (Phase E) costs and the Total Mission costs.

Table 3: Phase A, B, C/D Cost Summary in FY '98$M

Project Management	5.5
Outreach	2.4
Launch Approval	1.0
Project and Mission Engineering	5.6
Payload	54.6
Instrument I&T Support	3.9
Spacecraft	84.0
ATLO	4.8
Science	3.6
Mission Operations	8.9
A, B, C/D Sub total without LV	167
Launch Vehicle	60.0
Reserves at 20%	33.4
A, B, C/D total with LV and reserves	260

Table 4: Phase E and Overall Mission Cost Summary in FY '98 $M

Project Management	3.2
Science	5.0
Mission Operations	17.7
Phase E Subtotal	26
Reserves at 10 %	3
Phase E Total	29
Phases A, B, C/D Total	260
Mission Total (Phases A through E)	289

7.0 CONCLUSIONS

A Mercury Orbiter mission satisfying the combined goals of both the SSE and SEC communities could be feasible for launch in the 2005 timeframe with appropriate technology investment in the areas of solar electric propulsion, high temperature solar arrays, and thermal control systems.

8.0 ACKNOWLEDGMENTS

The author gratefully acknowledges the assistance of all the JPL personnel who contributed their time and expertise to this study, and especially Ross Jones and Dick Wallace who coordinated the SEC side of the study, Bob Nelson and Daniel Winterhalter who helped translate the science objectives into a detailed measurement plan, and JPL's Team X who did the bulk of the analysis.

The work described in this paper was carried out at the Jet Propulsion Laboratory, California Institute of Technology, under contract with the National Aeronautics and Space Administration.