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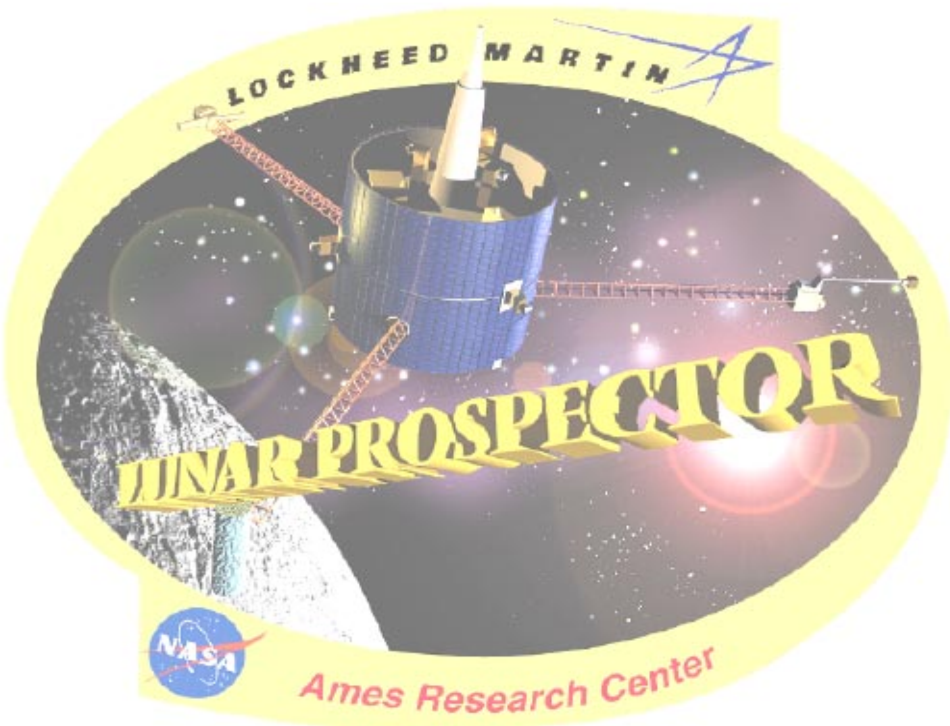
LUNAR PROSPECTOR MISSION HANDBOOK

APRIL 10, 1998



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Acronym List

ADS	Attitude Determination Software
AOS	Acquisition of Signal
APS	Alpha Particle Spectrometer
ARC	NASA Ames Research Center
AS	Attitude Subsystem
ASO	Astrotech Space Operation Company
C&DH	Command and Data Handling
CAS	Collision Avoidance System, TLI Stage
CCAS	Cape Canaveral Air Station
CCSDS	Consultative Committee on Space Data Systems
CGS	Command Generation Software
DSN	Deep Space Network
DFU	Destruct Firing Unit, TLI
DGE	Doppler Gravity Experiment
EBW	Exploding Bridgewire
EMS	Earth / Moon Sensor
EPS	Electrical Power Subsystem
ESBM	Equipment Section Boost Motor, Athena II 3rd stage
EOL	End of Life
EMI	ElectroMagnetic Interference
EMC	ElectroMagnetic Compatibility
ER	Electron Reflectometer
ESD	ElectroStatic Discharge
FDD	Flight Dynamics Division, NASA Goddard
FCDC	Flexible Confined Detonating Cord
FOV	Field of View
FTS	Flight Termination System
GFE	Government Furnished Equipment
GRS	Gamma Ray Spectrometer
GSE	Ground Support Equipment
HTPB	Hydroxy-Terminated Polybutadiene
ICD	Interface Control Document
IMU	Inertial Measurement Unit
JPL	Jet Propulsion Laboratory
LANL	Los Alamos National Laboratory
LMA	Lockheed Martin Astronautics
LMLV	Lockheed Martin Launch Vehicle, also Athena
LMMS	Lockheed Martin Missiles & Space, Co.
LEO	Low Earth Orbit
LOI	Lunar Orbit Insertion
LOS	Linear Ordnance System
LP	Lunar Prospector
LSC	Linear Shaped Charge
LV	Launch Vehicle
MAG	Magnetometer
MCC	Mission Control Center
MGA	Medium Gain Antenna
MGDS	Multi-Mission Ground Data System
MLI	Multi-layer Insulation
MSB	Most Significant Bit
NASA	National Aeronautics & Space Admini
NS	Neutron Spectrometer
NSI	NASA Standard Initiator
OAM	Orbit Adjust Module - Athena II 3rd st
PLF	Payload Fairing
RF	Radio Frequency
SA	Solar Array
SC	Spacecraft
SES	Spectrometer Electronics System
SLC	Space Launch Complex
SEA	Solar Equatorial Angle
SEU	Single Event Upset
SI	Science Instrument
SRA	Solar Roll Angle
SSA	Sun Sensor Assembly
SSD	Sun Sensor Detector
SSE	Sun Sensor Electronics
TDRSS	Tracking Data Relay Satellite System
TLI	Trans-Lunar Injection
UCB	University of California at Berkeley
UTC	Universal Time Coordinated

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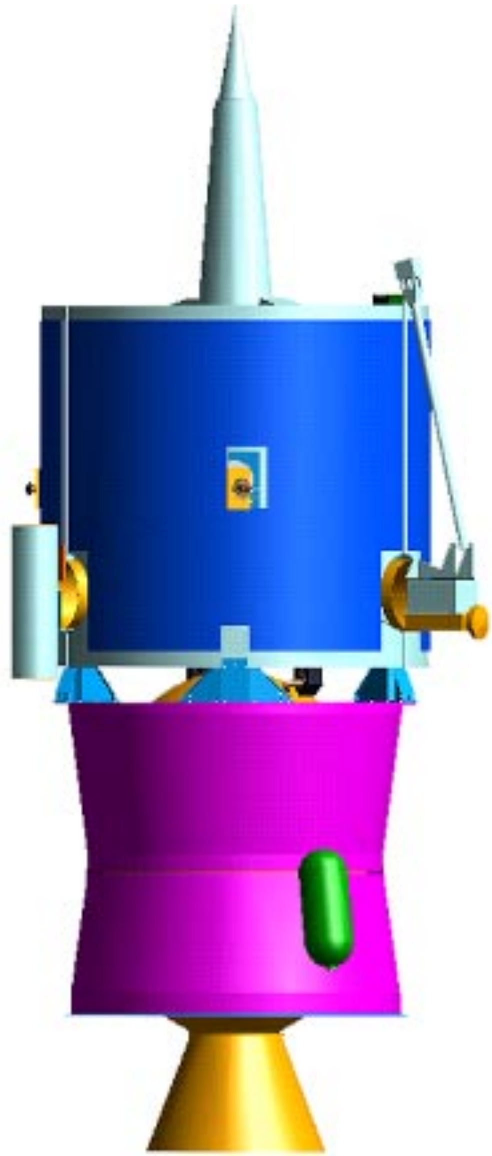
1. INTRODUCTION

1.1 LUNAR PROSPECTOR MISSION OVERVIEW

Lunar Prospector is the first peer reviewed and competitively selected mission in NASA's new Discovery Program of *faster, better, cheaper* solar system exploration. Lunar Prospector was selected on February 28, 1995 and began Phase C/D development in October 1995.

The overall management of the Lunar Prospector mission is the responsibility of Principal Investigator Dr. Alan Binder, Director of the Lunar Research Institute. Mr. G. Scott Hubbard, NASA Ames Research Center, serves as NASA Mission Manager and Mr. Thomas A. Dougherty, Lockheed Martin Missiles & Space, is the project manager.



The Lunar Prospector spacecraft (SC) is a small, simple, reliable, spin stabilized spacecraft designed to map the Moon's surface composition, polar water ice deposits, gravity field, magnetic field, and volatile release activity. The spacecraft and trans-lunar injection (TLI) stage, shown at left, has a combined launch mass of 1523 kg (3357.7 lb). The TLI stage injects the Lunar Prospector spacecraft into a trans-lunar trajectory from a Low Earth Orbit (LEO). In the launch, or stowed, configuration the combined envelope dimensions are approximately 408.9 cm (161.0 in.) in length and 205.2 cm (80.8 in.) in diameter.

Lunar Prospector launched from Space Launch Complex 46 (SLC-46) Cape Canaveral Air Station (CCAS), Florida on January 6, 1998. The launch marked the maiden flight of the Lockheed Martin Athena II (formerly LMLV-2) commercial launch vehicle.

The total mission cost $63 million, a figure which includes the spacecraft, the science instruments, TLI stage, launch vehicle, systems engineering, integration and test, launch services, and a year of mission operations.

During a one-year polar orbiting mission Lunar Prospector will map the Moon's surface composition, gravity and magnetic fields, and volatile release activity. The data will be used to expand the lunar science legacy of the Apollo program and assist in planning future exploration missions. As a Discovery Mission, Lunar Prospector seeks to demonstrate that high quality science missions can be accomplished economically and within a short period time.

The Lunar Prospector (LP) spacecraft was manufactured by Lockheed Martin Missiles & Space (LMMS), Sunnyvale, California. The spacecraft, TLI Stage, and ground support equipment (GSE) was be transported overland to Astrotech Space Operation Company (ASO) in Titusville, FL, for prelaunch processing and encapsulation. As the launch vehicle contractor, Lockheed Martin Astronautics (LMA),Denver CO, was responsible for the mission integration, total system performance, launch and flight operations, as well as production of the launch vehicle.

1.2 SYSTEM REQUIREMENTS

Lunar Prospector began seven years ago as a private initiative to show that small, inexpensive missions could be developed in a short time and could provide a high-quality science data return. To accomplish these goals, mission performance requirements were defined to allow simple spacecraft. Figure 1.2-1 summarizes key program requirements for the Lunar Prospector mission.

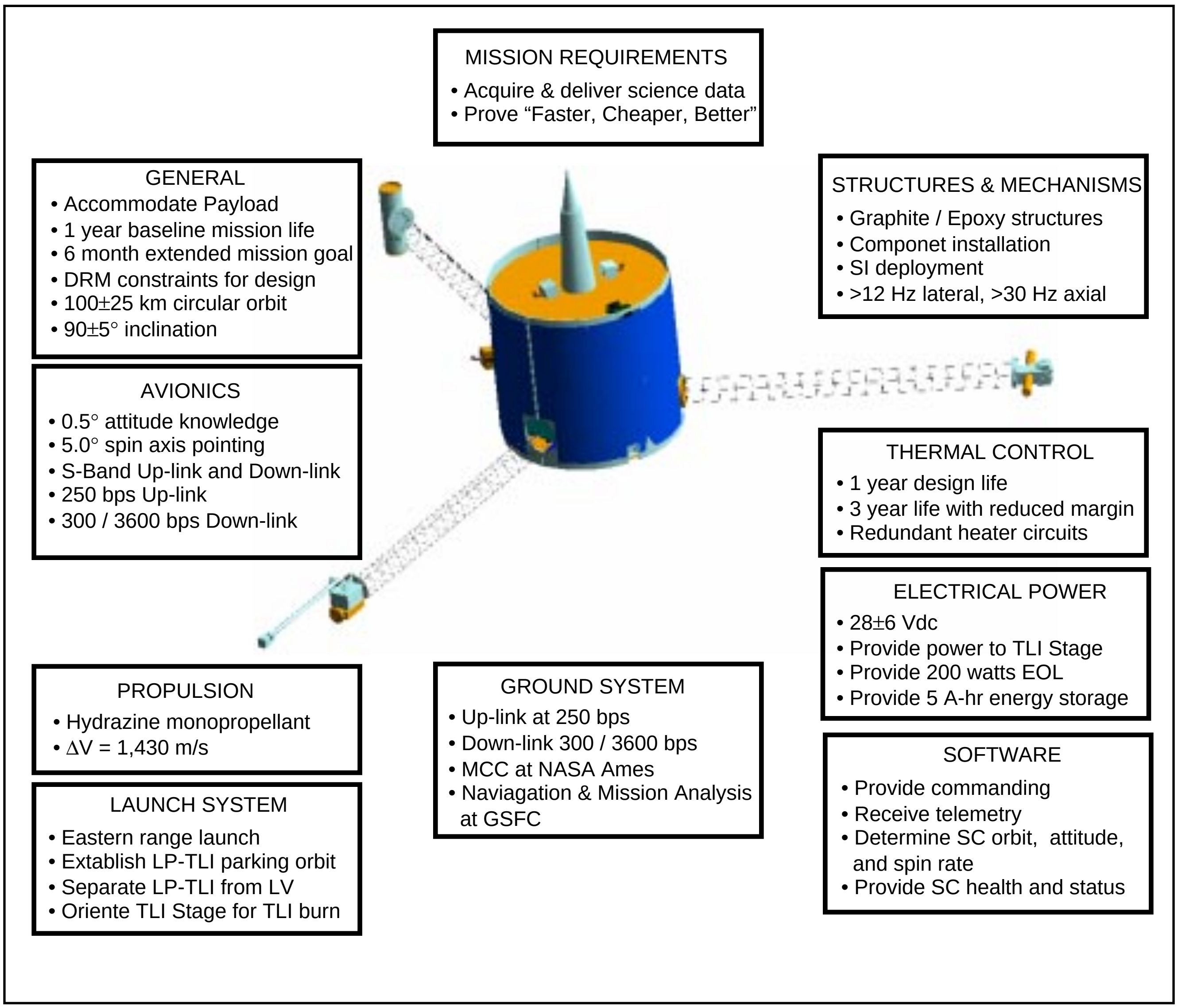


Figure 1.2-1 System Requirement Overview

1.3 APPLICABLE DOCUMENTS

The following referenced documents provide additional information about the Lunar Prospector mission and were used in the preparation of this handbook.

1.3.1 LP DOCUMENTS

LMMS P458382 B	Lunar Prospector Spacecraft and TLI Stage Data Book
LMMS P086796 F	Interface Control Document Spacecraft to Science Instruments
SA 1073-EI-H01445	LP C&DH Electronics Product Functional Description
LMMS P086867 B	LP Command Telemetry Databook
EPS-006	LP Engineering Memo: Charge Controller Operations
EPS-007	LP Engineering Memo: Final Energy Balance Analysis

1.3.2 LP/ATHENA DOCUMENTS

LMLV 1A50104	Lunar Prospector Payload On A LMLV-2 Launch Vehicle ICD
LMLV 1A54940 F	Athena II Lunar Prospector Phase 2 Launch Countdown Manual

2. MISSION DESCRIPTION

2.1 OPERATIONS CONCEPT

Lunar Prospector is a spin stabilized, single string spacecraft design to perform a low cost lunar exploration mission. The spacecraft has no onboard computer or flight controller; all commanding is up-linked from the Mission Control Center (MCC). Spacecraft telemetry are transmitted to the Deep Space Network (DSN) where it is received and sent to the Jet Propulsion Space Flight Operations Facility and relayed to the LP MCC at Ames Research Center (ARC). Telemetry data are processed at ARC and the required commands are selected and generated by the mission controllers and transmitted to the spacecraft via the JPL Multimission Ground Data System (MGDS) command system and the Deep Space network using. The DSN collects and sends tracking data to the Flight Dynamics Division (FDD) at Goddard Space Flight Center. The FDD processes the radiometric data and provides the control team at ARC with orbit predicts, orbit actuals, and maneuver parameters. The mission control team at ARC determines the required commands to implement the maneuvers and transmits them to the spacecraft. The spacecraft receives command signals and transfers the commands to the Command and Data Handling (C&DH) electronics where they are stored in registers. The spacecraft can only execute stored commands after receiving an execute command from the ground. The system is capable of providing a command delay so that command execution occurs after a specified time delay.

2.2 OPERATIONS SEQUENCE

2.2.1 LAUNCH TO TLI SEQUENCE

Lunar Prospector launched from the from SLC-46 of Spaceport Florida on January 6, 1998. The spacecraft and TLI stage was boosted into earth orbit by a Lockheed Martin Athena II launch vehicle. During launch and ascent, the SC remained powered off and did not receive or transmit any signals.

Separation from the booster enables the TLI command timer which outputs a preprogrammed series signals. Half a second after the TLI Stage separates from the Athena II Orbit Adjust Module (OAM), the TLI Stage Spin Motors ignite spinning up the spacecraft and TLI stage to 60 RPM. Twenty seconds after spin up, the TLI stage motor ignites and burns for 63.2 sec sending LP into its trans-lunar injection trajectory. The spacecraft is powered on at the end of the TLI burn and the official MCC mission clock is started. The power on signal turns on the spacecraft receiver and transmitter at 300 bps downlink rate and initiates the C&DH autostart sequence. A fraction of a second before SC-TLI separation, the TLI Collision Avoidance System (CAS) fires for added separation velocity margin and the spacecraft and TLI separate.

Initial spacecraft tracking is provided by TDRSS using two way Doppler tracking on a "best efforts basis" from the end of the TLI burn until 2 minutes before DSN first acquisition. This early tracking data provides the MCC with a first indication of spacecraft health and status and supports the DSN first acquisition. DSN acquisition of signal occurs 23 minutes after SC power-on at the Goldstone station. With AOS at Goldstone, the command uplink-downlink lock is established.

2.2.2 FLIGHT TO THE MOON

After acquisition by the DSN, ground controllers begin to implement the approved flight plan. The plan contains baseline as well as contingency operations in the event of TLI over or under burn conditions. The baseline flight plan calls for a spacecraft reorientation maneuver 40 minutes after spacecraft power-on. The first maneuver reorients the spacecraft to a cruise attitude with the spacecraft antenna normal to the ecliptic plane. The reorientation maneuver is followed by spin rate trim maneuver slowing the vehicle spin rate to 31±2 RPM in preparation for boom deployment.

Approximately one hour after the initial power-on, the boom release sequence for deploying the science instruments is initiated. Ground commands are sent to the spacecraft to release the MAG boom and the three science instrument masts. Boom deployment requires 7 to 15 minutes during which time the SC despins from 31 to 6 RPM. Within an hour of deployment, the SIs are turned on and start collecting calibration data.

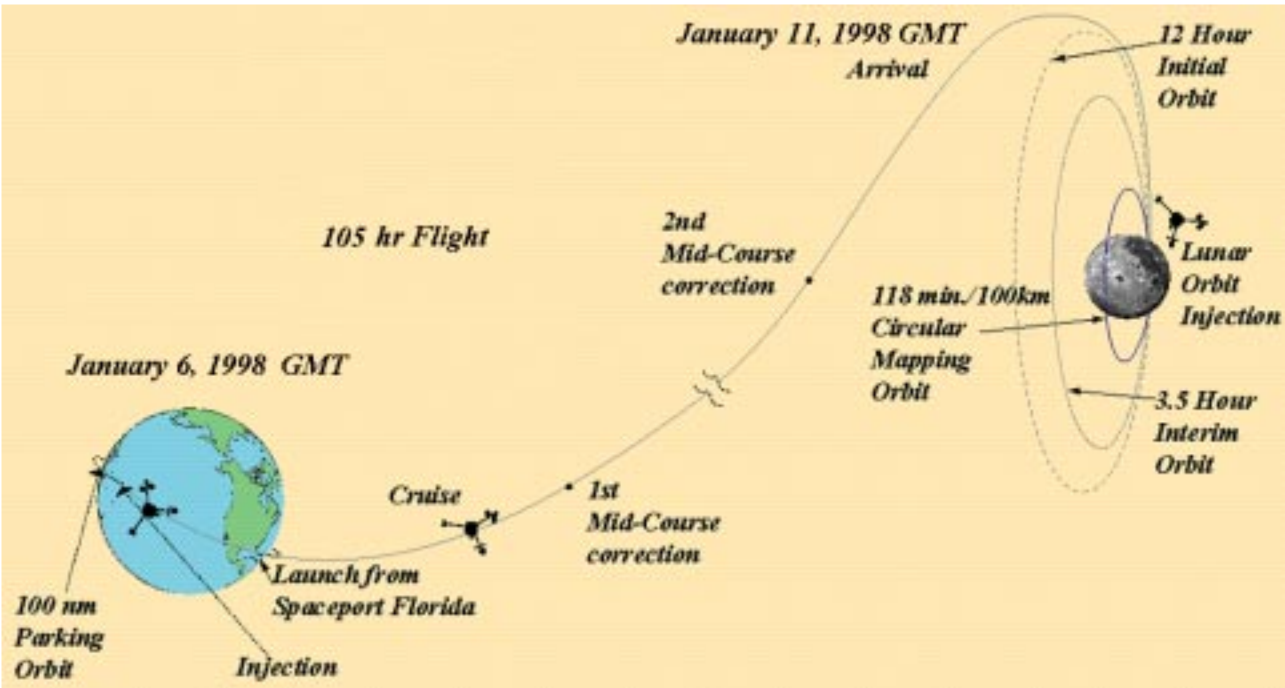


Figure 2.2.2-1 Lunar Prospector Trajectory

Three mid course correction maneuvers are planned during the cruise to the Moon. The first midcourse maneuver takes place four hours into the mission. The second and third midcourse maneuvers are planned for 28 and 76 hours from TLI separation as needed to properly target the trajectory. A propellant budget of 70 m/s has been allocated for these maneuvers. The flight to the Moon will take 105 hours.

2.2.3 LUNAR ORBIT INSERTION

Once the spacecraft reaches the Moon, three lunar orbit insertion (LOI) burns with orbit and attitude correction burns, will be performed. Figure 2.2.2-1 illustrates the LOI burns. The first LOI burn will put the spacecraft into a 12-hour, elliptical capture orbit. A day later, the second LOI maneuver at periselene places the spacecraft in a 3.5 hour elliptical orbit. The final LOI burn, 24-hours later, inserts the spacecraft into a circular, 118-minute, 100 km altitude, polar mapping orbit. At this point, the spacecraft begins its nominal one-year mapping mission. During this phase, periodic orbital maintenance maneuvers will be made to keep the spacecraft in its proper orbit.

2.2.4 SCIENCE MISSION IN LUNAR ORBIT

During the one-year mission, Lunar Prospector's Science Instruments (SIs) will observe the entire lunar surface about 26 times. The SIs will map the Moon's surface composition, magnetic and gravity fields, gas release events, and probe the Moon for any deposits of water ice in the constantly shadowed regions of the lunar poles. These data will expand the Apollo program lunar science legacy and assist in planning future lunar exploration missions.

Lunar Prospector conducts five experiments using five instruments and the spacecraft's radio signal. The five science instruments, packaged on the ends of the spacecraft's three masts, are: the Gamma Ray Spectrometer (GRS), the Alpha Particle Spectrometer/Neutron Spectrometer (APS/NS), and the Magnetometer/Electron Reflectometer (MAG/ER). The three spectrometers determine the composition and evolution of the lunar surface to a depth of more than 10 centimeters. The GRS maps crustal composition; the APS senses volatile release activity; and the NS maps Hydrogen distribution. The Magnetometer/Electron Reflectometer produces a high resolution map of the lunar magnetic field, and the spacecraft's down-link carrier signal is examined for gravity induced Doppler shifts from which a high resolution global gravity map is generated.

All SIs collect data continuously. The data is transmitted back to Earth in near real time (2 second delay) interleaved with recorded data delayed by 53 minutes. The delayed data provide redundancy for data which may lost during occultations. When DSN tracking is available, estimated at 80 percent of mission, the data are collected and sent to ARC for science data processing. The MCC software processes and time tags all data and provides data records to the science team with redundancy eliminated and gaps filled in. Engineering and ephemeris data are also provided to the science team.

2.2.5 EXTENDED MISSION

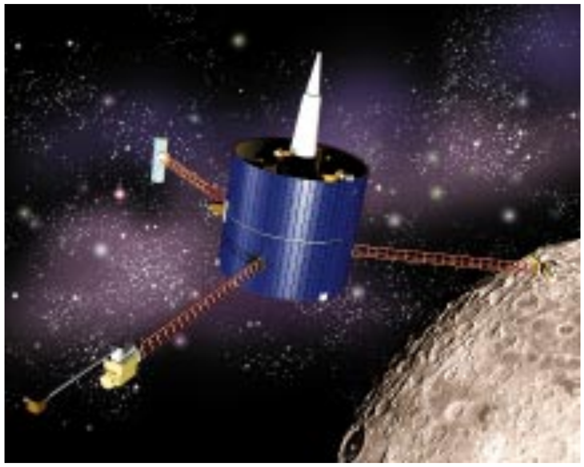
Expecting that fuel is available at the end of the one-year nominal mission, an extended mission is planned with a low altitude 10 kilometer by 100 kilometer orbit. The early part of this mission will have periselene at one pole, later move it to the other pole, and finally move it to the equator. This mission will improve gravity map resolution considerably and will provide excellent close up coverage by the spectrometers. There is a great desire by NASA to achieve this extended mission. The mission will end when the fuel needed for orbital maintenance is depleted and the spacecraft impacts on the lunar surface.

2.3 MISSION OPERATIONS

Spacecraft operations are most intensive following TLI, during trans-lunar cruise, and during LOI maneuvers. After lunar orbit insertion, mission operations are simplified and non time critical. The spacecraft collects and plays back science data continuously. Mission controllers monitor the quality of the data and the status of the ground system. The science data are processed to level 0 and provided to the science teams for level 1 processing at their home institutions before archiving in the project data base. Engineering data are monitored in real time with trend analyses performed on selected parameters.

The spacecraft requires occasional trim maneuvers during the mission. These maneuvers require tracking data processing, orbit determination, maneuver calculation , and command preparation and transmission.

Given that SI anomalies may cause excessive power usage, SIs may be powered down. In the event of an SI failure, the instrument will be turned off. Although spacecraft subsystems are single string, there may be failures modes which result in a graceful degradation of performance. The bottom line is that this spacecraft has been designed for a low cost and simple operations, requiring only a small team of operations personnel to carry out the mission.



3. SYSTEM PERFORMANCE OVERVIEW

3.1 MISSION LEVEL PERFORMANCE

3.1.1 LAUNCH VEHICLE COMPATIBILITY

• Model:	Lockheed Martin Athena II
• Payload fairing:	92 in.
• Payload attach fitting:	SAAB 47” marmon clamp

3.1.2 ORBIT COMPATIBILITY/CONSTRAINTS

• Altitude:	100 km circular
• Inclination:	90° polar orbit
• Period:	118 min.
• Operational altitude range:	Nominal mission: 100 +/- 25 km circular
	Extended mission: as low as 10 km elliptical
• Orbit maintenance capabilities:	Function of desired mission lifetime and orbit

3.1.3 CLEANLINESS LEVELS

• Particulate:	Spacecraft controlled to 300,000 class, instruments to 100,000 class. Special N2 purge capability is provided for instruments
• Molecular :	Hydrocarbon contamination controlled: A/C ≤ 15 ppm; N2 purge ≤ 2 ppm
• Magnetic:	Spacecraft induced magnetic field is less than 0.1 nT at less than 16 Hz for time-varying field; less than 0.5 nT for constant magnetic field

3.1.4 DESIGN LIFETIME

• Reliability :	Designed for up to 3 year mission. Single string subsystems: no redundancy except on solar arrays and thermal control heaters. Parts meet Mil-Std-975 class B.
• Expendables:	Minimum of 1 year mission

3.2 CORE SYSTEM PERFORMANCE

3.2.1 MASS CAPABILITY

• Payload mass capability:	Up to 24 kg (53 lbm.)
• Spacecraft bus mass:	157.8 kg (347.9 lbm.) dry

- Deployables: three 2.5 m (8.2 ft.) masts

3.2.2 ELECTRICAL POWER

- Solar arrays: Fixed, body mounted, high efficiency silicon cells
- Orbit average generated power: Up to 200 Watts at EOL
- Peak power demand: 176 W (including 77 W for battery charging)
- Battery capacity: 5 amp hours
- Bus voltage: 28 ± 6 VDC

3.2.3 PROPULSION SYSTEM PERFORMANCE

Spacecraft:

- System type: Blow-down
- Propellant Hydrazine, 138 kg (304 lbm.)
- Average Thrust 22.2 N (5 lbf.)
- Specific Impulse 220 s (234 s max.)
- Total Impulse: 293,440 N-s (65,968 lbf.-sec).

TLI Stage / Star 37FM:

- System type: solid motor
- Propellant TP-H-3340 , 1,009.98 kg (2,226.63 lbm.)
- Average Thrust 45,794 N (10,295 lbf.)
- Effective Specific Impulse, Isp 289.82 s
- Total Impulse: 2,892,701 N-s (650,305 lbf.-sec.)

3.2.4 ATTITUDE CONTROL SYSTEM PERFORMANCE

- Type: Major axis spin stabilized, thruster controlled
- Pointing accuracy: 5.0 degrees
- Pointing knowledge: 0.5 degree
- Sensors: 1 earth/moon limb and one sun sensor
- Nutation Damping 1 passive nutation damper

3.2.5 COMMAND AND DATA HANDLING PERFORMANCE

- Data storage capacity: None provided; 53 min data replay capacity
- Processor type: None provided

- Command storage: Has capability to execute stored commands in timed sequence
- Ground control: Ground software control capability for formatting of commands, real time control of the spacecraft, and post processing of science and engineering data

3.2.6 COMMUNICATION SYSTEM PERFORMANCE

- Downlink modulation & rates: Phase modulated (PM)
  3600 bps science and engineering data
  300 bps engineering data
- Uplink rates: 250 bps
- Frequency bands: S-band
- Command & telemetry format: CCSDS
- Range capability: Works up to Lunar orbit
- Ground station compatibility: DSN, TDRSS
- Antenna coverage, FOV: Uses omni and medium-gain antennas to minimize spacecraft orientation requirements.

3.2.7 THERMAL CONTROL

- Implementation: Passive coatings, thermal blankets and thermostatically controlled heaters

3.2.8 ELECTRICAL

- EMI & EMC Environment: N/A
- Magnetic Performance: See cleanliness level for magnetic cleanliness
- ESD sensitivities: N/A

3.2.9 RADIATION TOLERANCE

- Total Dose: Parts meet Mil-Std-975 class B
- SEU: N/A

4. CORE SYSTEM AND SUBSYSTEM DESCRIPTION

4.1 SCIENCE INSTRUMENTS & EXPERIMENTS

The Lunar Prospector spacecraft carries five Science Instruments (SIs). An Alpha Particle Spectrometer (APS), a Gamma Ray Spectrometer (GRS), a Neutron Spectrometer (NS), a Magnetometer (MAG) and an Electron Reflectometer (ER). These SIs are mounted on three deployable masts located external to the SC module. In addition to the five SIs, the LP mission performs a Doppler Gravity Experiment (DGE) that provides measurements of the gravity by means of S-Band transponder tracking data.

The Lunar Prospector science payload was chosen from a list of experiments proposed by NASA scientists for lunar mapping missions. The spacecraft carries six of the highest priority experiments, chosen for their scientific value, their ability to be flown on a simple, spin-stabilized spacecraft, and for their low mass, power and data rate requirements. The experiments are:

- Gamma Ray Spectrometer (GRS) - Robert McMurray (Acting Co-I), NASA Ames
- Neutron Spectrometer (NS) - William Feldman, Los Alamos
- Alpha Particle Spectrometer (APS) - Alan Binder, Lunar Research Institute
- Magnetometer (MAG) - Mario Acuña, NASA Goddard; Lon Hood, Univ. of Arizona LPL
- Electron Reflectometer (ER) - Robert Lin, UC Berkeley SSL
- Doppler Gravity Experiment (DGE) - Alex Konopliv, NASA JPL

4.1.1 GAMMA-RAY SPECTROMETER

The gamma-ray spectrometer is a more advanced system than was flown on the Apollo missions and will provide global maps of the elemental composition of the surface layer of the Moon. The main elements mapped are uranium, thorium, potassium, iron, titanium, oxygen, silicon, aluminum, magnesium and calcium. Knowledge of the concentrations of these elements over the entire lunar surface will aid in understanding the composition and evolution of the lunar crust.

Figure 4.1.1-1 Gamma Ray Spectrometer

Though uranium, thorium and potassium are only trace elements, they are found concentrated in a material called KREEP (potassium [K], rare earth elements [REE] and phosphorus [P]). KREEP is not only the main source of these elements, but many other important trace elements such as zirconium, fluorine and chlorine. Mapping the locations and concentrations of KREEP deposits is important to lunar science as it is believed that the material developed late in the formation of the lunar crust and upper mantle, and thus can help define how the crust and mantle formed and evolved.

Gamma Ray Spectrometer (GRS) and Neutron Spectrometer (NS) The GRS and NS will return global data on elemental abundances, which will be used to help understand the evolution of the lunar highland crust and the duration and extent of basaltic volcanism, and to assess lunar resources. The NS will also locate any significant quantities of water ice which may exist in the permanently shadowed areas near the lunar poles.

4.1.2 NEUTRON SPECTROMETER

While the Moon does not have any water of its own, theoretical calculations suggest that water brought to the Moon by comets and water-rich meteoroids may be frozen in the bottom of small craters in the polar regions. The floors of some small craters located within 30 degrees of the north and south poles never see sunlight and have continuous temperatures below -190 degrees centigrade. At such temperatures, water ice would be stable over the lifetime of the solar system. The neutron spectrometer on Lunar Prospector has a water ice detectability limit of better than 0.01%, which means it can locate 200 grams of water (a cup of water) in a cubic meter (about a cubic yard) of regolith (lunar soil). The discovery of lunar polar ice would have a profound effect on the economics and logistics of the exploration and colonization of the Moon and the inner solar system. It would mean that water, necessary for life support and as a source of both oxygen and hydrogen needed to produce rocket propellant, would be available in situ to future lunar explorers.

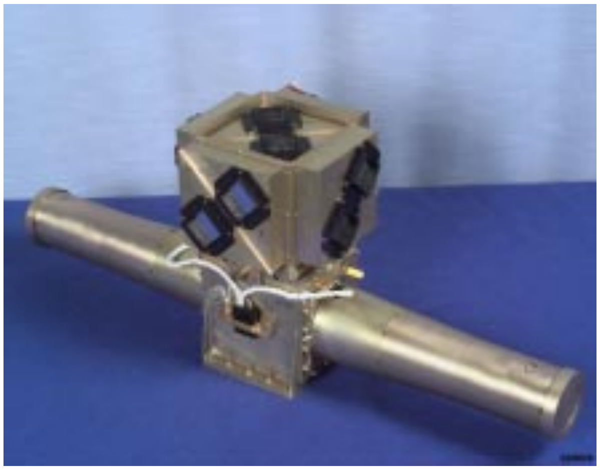


Figure 4.1.2-1 APS/NS Assembly

4.1.3 ALPHA PARTICLE SPECTROMETER

Alpha Particle Spectrometer (APS) The APS instrument will be used to find radon outgassing events on the lunar surface by detecting alpha particles from the radon gas itself and its decay product, polonium. Observations of the frequency and locations of the gas release events will help characterize one possible source of the tenuous lunar atmosphere. Determination of the relationship of outgassing sites with crater age and tectonic features may be possible. This may in turn be used to characterize the current level of lunar tectonic activity.

The alpha particle experiment is an advanced version of an experiment flown on Apollo 15 and 16. It will determine the locations and frequency of gas release events by detecting alpha particles from both radioactive radon gas and its decay product, polonium. This is the only orbital instrument which provides information on the current level of tectonic and volcanic out-gassing activity of the Moon. Thus, it provides unique information about the Moon which, until Apollo, was thought to be tectonically and volcanically dead. As a result of the Apollo surface seismometers and mass spectrometer and the orbiting alpha particle experiment, scientists now know that the Moon is active, though much less so than Earth or Mars.

4.1.4 MAGNETOMETER AND ELECTRON REFLECTOMETER

The Moon does not have a global magnetic field like the Earth, however, weak fields of local extent do exist. Mapping the strengths and distributions of these local fields over the Moon will determine if they were caused by an earlier global magnetic field like the Earth's, if they were caused by meteoroid impacts, or if they have some other origin. The data should also provide information on the size and composition of the lunar core and, coupled with Neutron Spectrometer data, could reveal correlations between magnetic fields and solar wind implanted hydrogen and helium concentrations.

Figure 4.1.4-1 MAG/ER Assembly

Magnetometer (MAG) and Electron Reflectometer (ER) The MAG/ER experiments will return data on the lunar crustal magnetic field and the lunar induced magnetic dipole. These data will help provide an understanding of the origin of lunar paleomagnetism and the degree to which impacts can produce paleomagnetism, and allow constraints on the size and composition of the (possible) lunar core.

On Earth, magnetic mapping is an important tool for locating economically important ore bodies. Similarly, the magnetic experiments will provide information to help understand the economic potential of the Moon.

4.1.5 DOPPLER GRAVITY EXPERIMENT

This investigation will use Doppler tracking of S-Band radio signals to characterize the spacecraft orbit and determine the lunar gravity field. This data will provide information on the lunar interior and, combined with lunar topographic data, will allow modeling of the global crustal asymmetry, crustal structure, and subsurface basin structure. It can also used for planning future lunar missions.

The experiment provides the first complete gravity map of the Moon. Because no earlier mission was in a low polar orbit, NASA does not have an adequate lunar gravity model for planning follow-on unmanned and manned lunar missions. Lunar Prospector will define the lunar gravity field that affects the altitude, and thus the fuel quantity required for orbit maintenance, of all orbiting spacecraft. It will also provide data on density differences in the crust, the internal density of the Moon, and the nature of the core.

4.1.6 SCIENCE INSTRUMENT CHARACTERISTICS

Instrument	**Source**	**Qty.**	**Size** (cm)	**Data Rate**	**Power** (W)	**Mass** (kg)
GRS	LANL	1	55.4 x 17.6 DIA	688 bps	0.36	6.6
APS	LANL	1	19.0 x 19.0 x 13.5	182 bps	0.88	5.6
NS		1	65.8 x 12.7 x 12.8	50 bps	0.06	
SES	SwRI	1	23.1 x 21.8 x 16.5	928 bps*	11.2	6.2
MAG /	Goddard	1	10.2 x 7.2 x 7.2	328 bps	4.5	5.3
ER	UCB SSL	1	30.2 x 20.3 x 17.5	298 bps		

* Total telemetry data rate for LP spectrometers.

4.2 OVERALL SPACECRAFT DESIGN/SYSTEM ARCHITECTURE

The Lunar Prospector spacecraft is of general cylindrical form. The spacecraft bus consists of a 1.37 m (54") in diameter and 1.29 m (51") in axial length drum with an inner advanced graphite-epoxy triangular support structure. Most of the electronic boxes and attitude sensors are mounted on the equipment, or top, panel of the inner triangular support structure. Also on this top panel is the 1.6 m (62") tall antenna assembly consisting of the omni directional antenna (cone on top) and the medium gain antenna (conical-cylinder). The Medium Gain and Low Gain (omni) Antenna systems provide data downlink and command uplink communications capabilities between the SC and a GFE (DSN) network of ground tracking stations. The propellant tanks are located internal to the main SC body. Three deployable masts are located external to the main SC body. The scientific instruments are mounted on the masts. Solar cells are mounted on essentially all the outer surface of the cylindrical body.

The SC envelope, when stowed, is approximately 241.3 cm (95.0 in.) in length and 205.2 cm (80.8 in.) in diameter. When the masts are deployed, the SIs sit 318.6 cm (125.4 in.) from the center of the SC. The exception is the Magnetometer which is on an extension boom and protrudes 452.4 cm (178.1 in.) from the center of the SC. The fully fueled SC mass of 296.3 kg (653.2 lb.), includes 138.0 kg (304.2 lb.) of hydrazine fuel. The Medium Gain and Low Gain (omni) Antenna systems provide data downlink and command uplink communications capabilities between the SC and the DSN ground tracking stations.

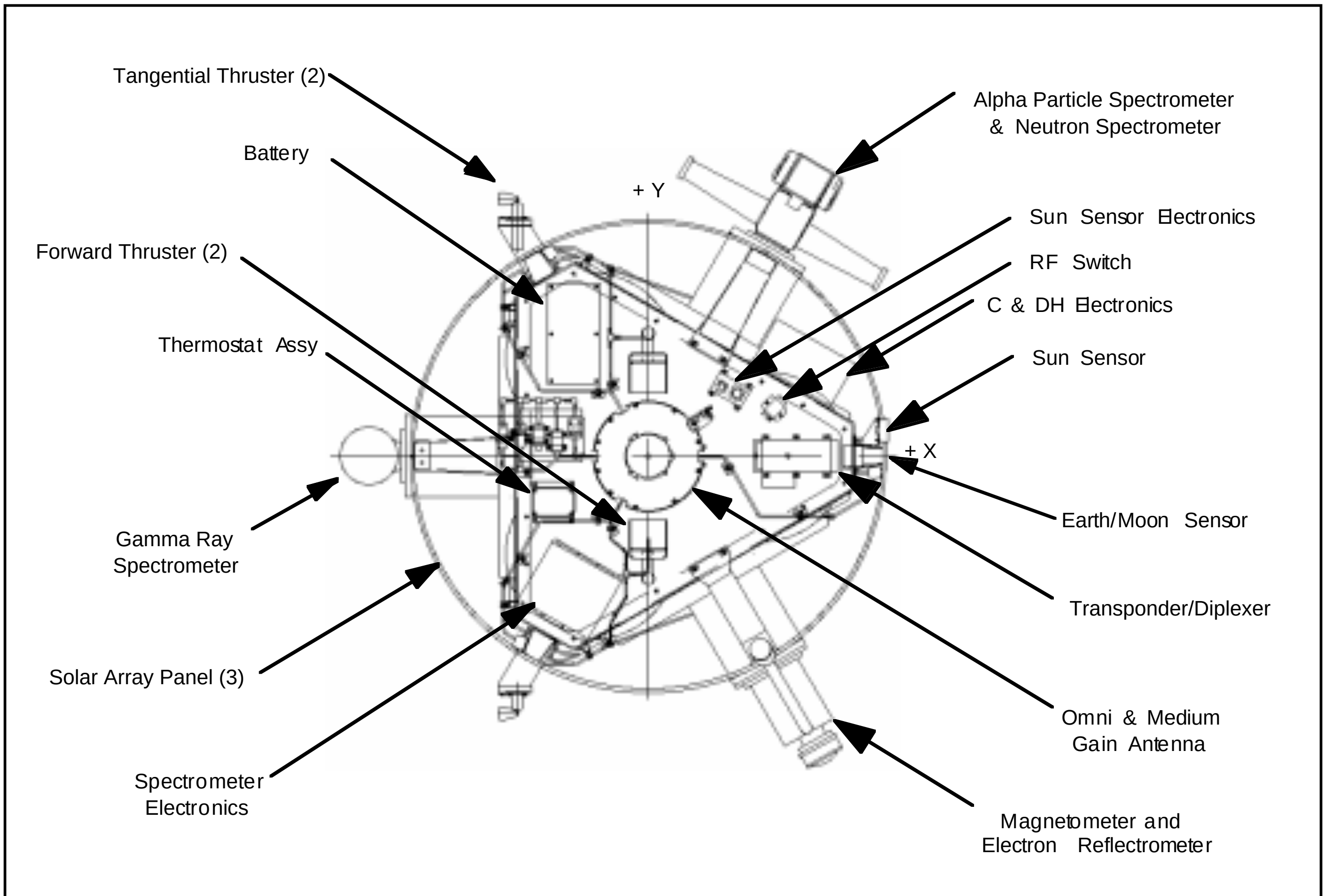


Figure 4.1.6-1 Spacecraft Top View

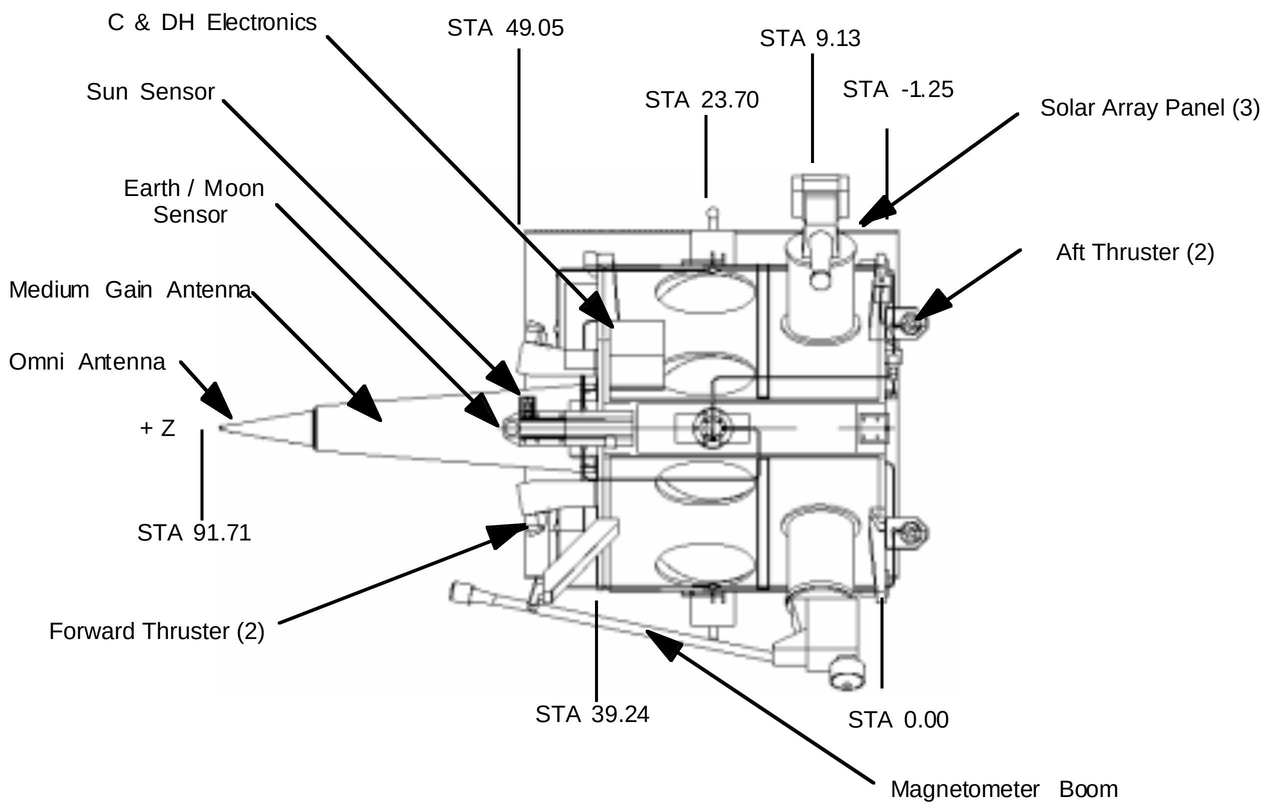


Figure 4.1.6-2 Spacecraft Side View

4.2.1 TLI AND SPACECRAFT BODY COORDINATE SYSTEM

The TLI and Spacecraft Body Coordinate System is shown in Figure 4.2.1-1 and Figure 4.2.1-2. The origin of coordinates is:

a) in the separation plane between the S/C and the adapter of the trans-lunar injection (TLI) stage; and
b) at the geometric center of the equilateral triangle defined by the bolt holes for the separation nuts

The Z axis is normal to the separation plane. The +Z direction is towards the medium gain antenna (forward) end of the S/C. The X axis is in the separation plane between the S/C and the adapter of the TLI stage. The -X direction is parallel to the centerline of instrument boom No. 1 and towards the GRS. The Y axis is defined by the right-hand rule.

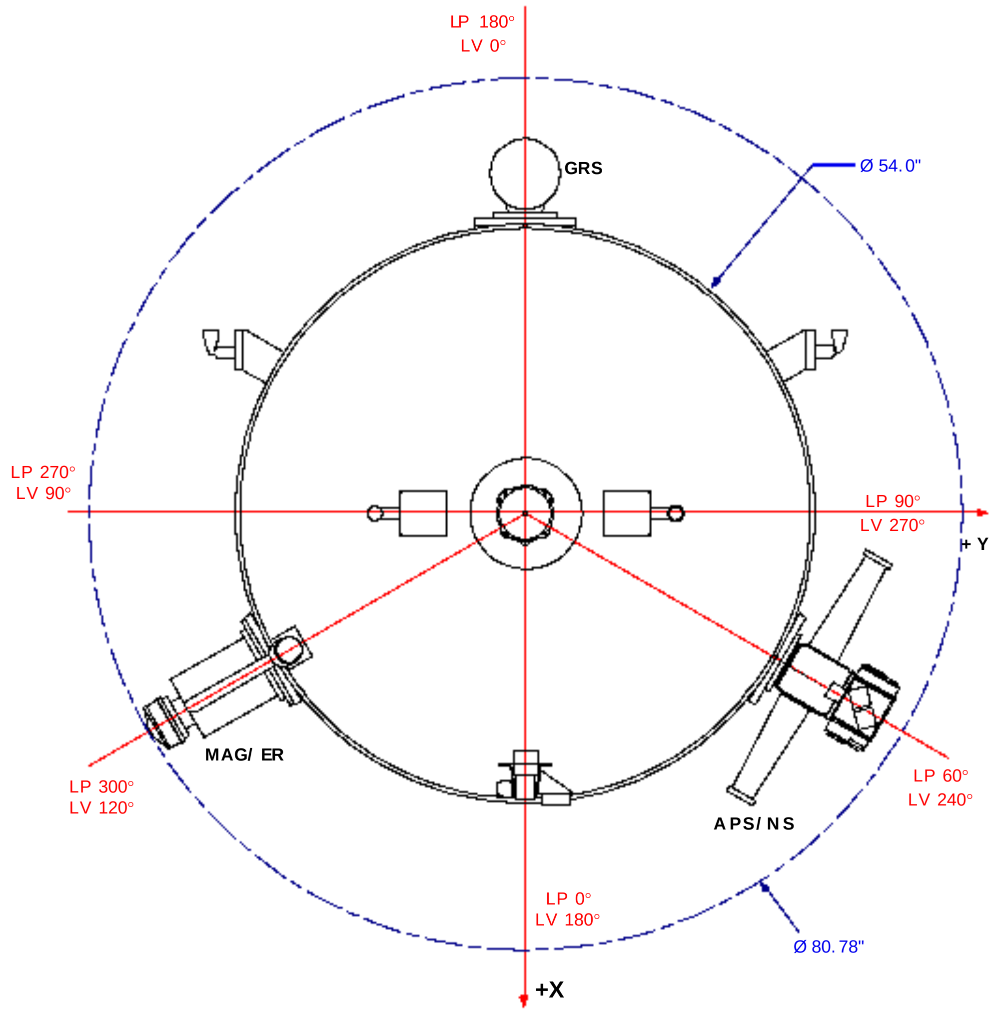


Figure 4.2.1-1 Lunar Prospector Stowed Spacecraft Top View

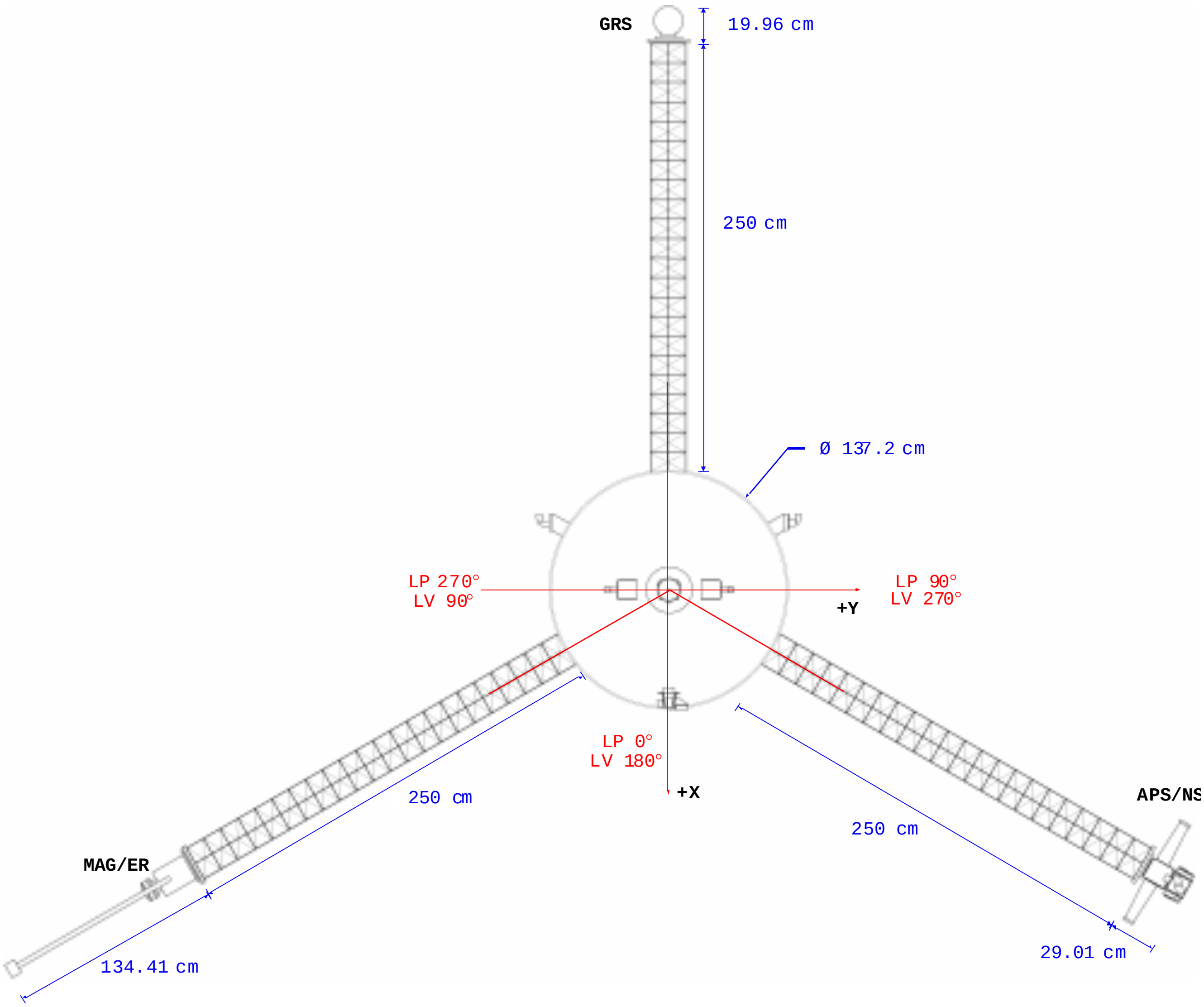


Figure 4.2.1-2 Lunar Prospector Deployed Spacecraft Top View

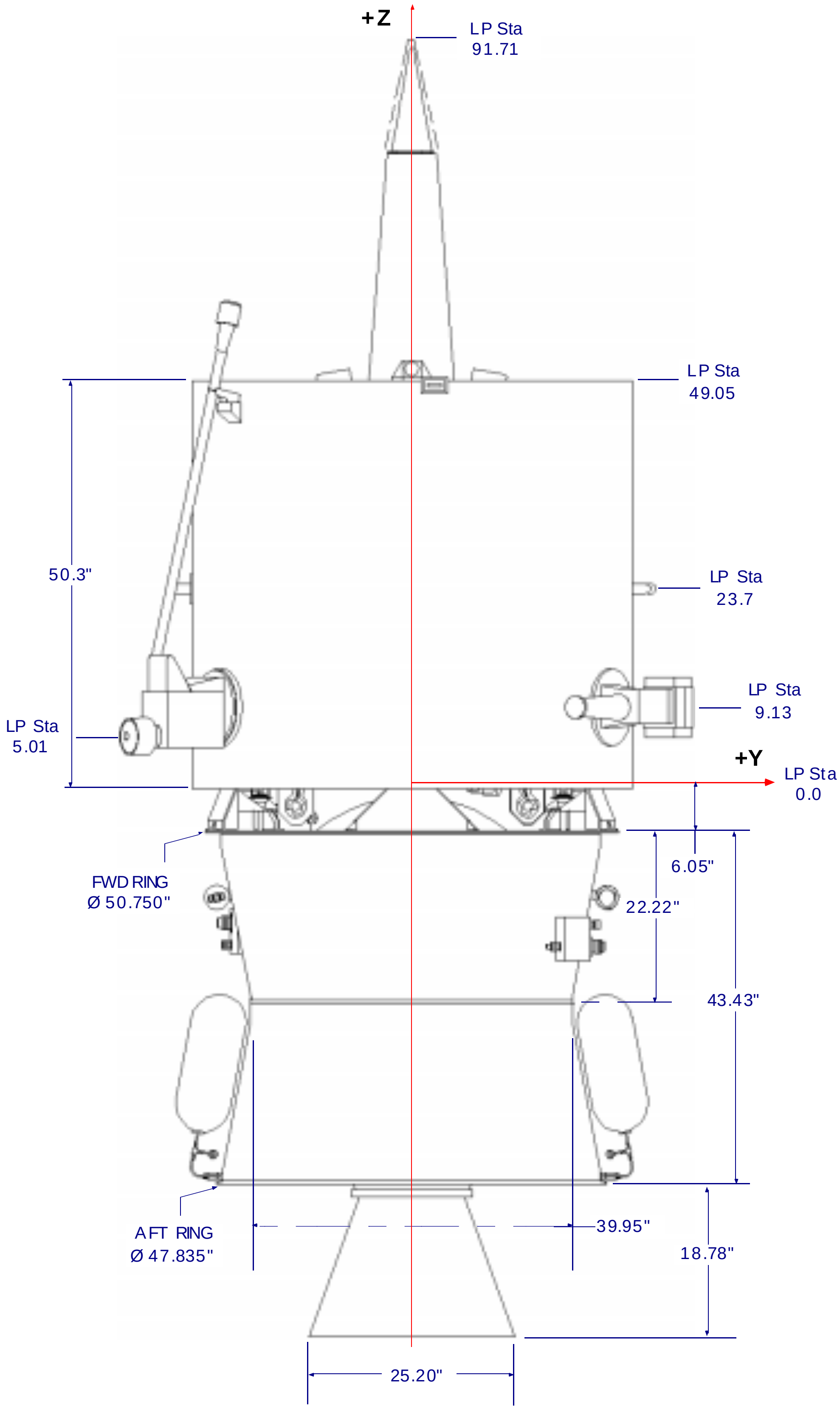


Figure 4.2.1-3 Lunar Prospector Spacecraft & TLI Side View

4.2.2 STRUCTURAL/MECHANICAL SYSTEM

The Lunar Prospector spacecraft has a reasonably simple structure and minimal mechanisms. The triangular structure is a graphite epoxy material. The three solar panels which wrap around the outside of the triangular structure are passive. The Able masts which deploy the Scientific Instruments are coilable lattice structures with s-glass/epoxy members.

4.2.2.1 Component Characteristics

Component	Source	Heritage/ Model	Qty.	Size (cm)	Power (W)	Mass (kg)
Solar Drum Panels	LMMS	new	3	137 D x 129 x 1.3	N/A	8.6
Equipment Shelf	LMMS	Iridium	1	139 D x 1.3	N/A	4.8
Triangular Structure	LMMS	Iridium	1	139 D x 60 x 54	N/A	17.3*
Control Assembly	ABLE	new	1	13 x 78 OD	20**	3
Mast Canister	ABLE	Galileo	3	29 x 23 OD	N/A	0.8 ea.
Instrument Masts	ABLE	Galileo	3	250 x 23 OD	N/A	2.4 ea.

* includes side panels
** transient for mast release

4.2.2.2 Bus Structure

The LP primary load carrying structure consists of a triangular structure, equipment shelf, engine brackets and fittings for attaching to the TLI stage. The structure provides primary structural support for three fuel tanks and secondary structural support for the scientific instruments. A mercury filled nutation damper is attached to the base of the spacecraft. The equipment shelf provides the mounting support for all electronic components, batteries and propulsion system equipment, although equipment can also be mounted on the side panels. The TLI adapter is located on the base side of the structure and provides the interface between the LP and the Star 37 kick motor. The Athena II payload adapter provides the load path from the TLI stage to the launch vehicle. The LP structures are fabricated from high modulus graphite-epoxy material.

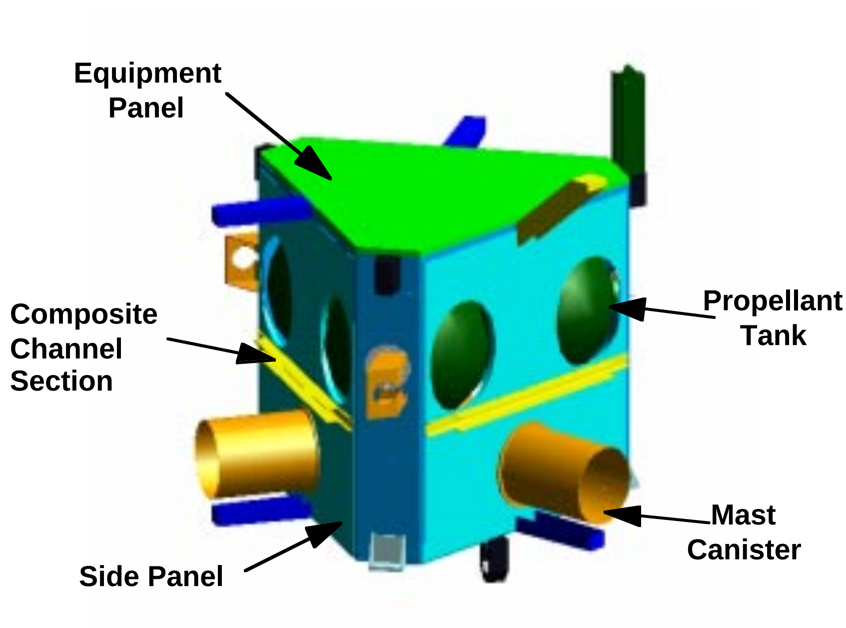


Figure 4.2.2-1 Lunar Prospector Bus Structure

4.2.2.3 Payload Masts

Three masts isolate the science instrument assemblies from the spacecraft. Each self-deploying mast extends 2.5 m (8.2 ft.) from the side of the spacecraft. Tip plates and SIs rotate 4 times through deployment. One instrument, the Magnetometer, is mounted on an extension boom that hinges on the tip of its mast and extends an additional 114.4 cm (45.1 in.). During launch, each mast is tightly coiled and stowed inside a canister 29 cm (11.5 in.) long and 23 cm (9.2 in.) in diameter. After trans-lunar injection and spacecraft separation, the Magnetometer extension boom is released, followed by release of the central restraint mechanism for the 3 masts. Centrifugal force and strain energy in the masts causes them to deploy. The masts are deployed synchronously. Deployment rate is controlled by lanyards connecting each boom tip plate to a common drum and damper mechanism. The masts are deployed in about 12 minutes and reduce the spacecraft spin rate from 31 RPM to 6.5 RPM in preparation for operation around the Moon.

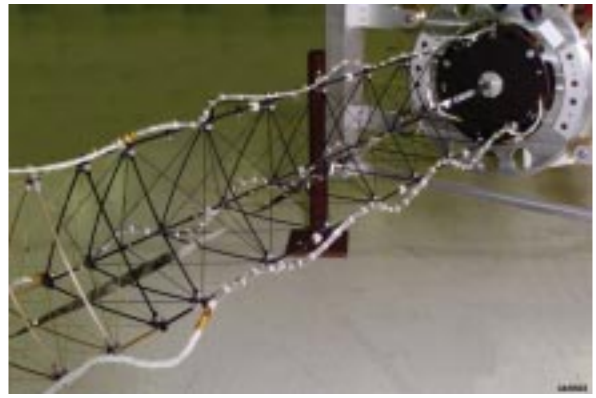


Figure 4.2.2-2 ABLE Mast

4.2.3 ELECTRICAL POWER SUBSYSTEM

Lunar Prospector's Electrical Power Subsystem (EPS) generates, stores, and distributes electrical power to spacecraft electrical equipment and the science instruments. The EPS consists of a 28 volt 5 Amp-hour battery, a 200 Watt solar array, and the associated wire harnesses for power distribution. The EPS maintains a voltage of 28±6 volts direct current for the power-using equipment on board the spacecraft. Battery charge control is accomplished by a circuit board packaged in the C&DH electronics unit. The battery is located on the spacecraft Equipment Shelf. The solar array consists of three curved structural panels surrounding the spacecraft bus. Electrical cables connect all equipment and are routed over the spacecraft bus structure. For detailed discussion of the EPS energy balance, including system budgets, power generation model, system losses, and IV curve characteristics refer to LP Engineering Memo EPS-007: Final Energy Balance Analysis.

During periods when the spacecraft is in shadow, the battery provides all necessary electrical power. When the spacecraft is in sunlight, the solar array delivers enough power to run the spacecraft and recharge the battery in preparation for the next night side pass. A Charge Control Board, located in the C&DH unit, regulates battery charge current by means of pulse-width-modulated transistors in series with the three solar array segments. When the battery reaches full charge, the Charge Controller reduces the charge current to a maintenance or trickle charge rate. Specific information regarding charge controller operations is contained LP Engineering Memo EPS-006: Charge Controller Operations.

The Lunar Prospector Spacecraft employs a Single Point Ground philosophy. All power is returned through the wire harnesses, not through structure. All components are electrically bonded to the structure ground plane using a ground wire or by contact with a metallic mesh embedded in the outer layer of the spacecraft's composite panels.

4.2.3.1 Component Characteristics

Component	Source	Heritage	Qty	Size (cm)	Power (W)	Mass (kg)
Solar Cells	Sharp	Various	2640	2 x 6 and 3 x 6	N/A	5.5
Battery	Eagle-Picher, LMMS	Various	1	15 x 30 x 15	N/A	7.3
Wiring	LMMS	Various	≈800	AR	7.0	6.5

4.2.3.2 Solar Array

The Lunar Prospector Solar Array converts sunlight into electrical current through strings of high efficiency (17%) silicon solar cells manufactured by Sharp. The solar cells are bonded to the outboard surface of the three semi cylindrical substrate panels forming the spacecraft's outer drum. Some 2640 solar cells are connected into 16 strings by micro-welding silvered-molybdenum foil strips from cell-to-cell. Each solar panels is "back-wired" on the rear-side in order to minimize current loop magnetic fields generated by the solar cell strings. The Lunar Prospector solar array delivers about 200 watts of power. Science Instruments and spacecraft electrical equipment require roughly 90 watts of continuous power while the remaining power is used to recharge the battery after eclipse periods.

The Solar Array converts sunlight into electrical current which is carried by Cable Harnesses to the loads and to the Battery for storage. During eclipses electrical current flows from the Battery to the loads by means of the Cable Harnesses.

4.2.3.3 Battery

The Lunar Prospector Battery uses flight-proven, off-the-shelf super nickel-cadmium battery cells. It is rated at an energy storage capacity of 5 Ampere-hours. The Battery has twenty-two prismatically shaped cells clamped between thermally-conducting aluminum plates in a 2x11 configuration. Excess battery heat is radiated to space by means of a radiator plate mounted atop the Battery. During periods of sunlight, the solar array delivers sufficient power to recharge the battery in preparation for the next shadow period.

Each battery cell casing is manufactured from passivated 304L stainless steel and designed to withstand pressures up to 100 psi. with a safety factor of 1.5. Operating pressure typically does not exceed 65 psi. The electrolyte is alkaline KOH at a concentration of 31%. The cells are completely hermetically sealed and, therefore, there are no emissions of any materials, toxic or otherwise.

4.2.4 PROPULSION SYSTEM

The propulsion subsystem provides spacecraft ΔV for translunar mid-course corrections, lunar orbit insertion, and lunar orbit maintenance, as well as attitude control and spin control. The subsystem propellant capacity meets all mission requirements with substantial design margin.

The mono-propellant hydrazine propulsion system provides attitude control and velocity corrections for lunar orbit insertion and orbit maintenance. The propulsion system is integral with the primary spacecraft structure. The simple blowdown system uses three titanium tanks and 304L stainless steel plumbing with flight proven transition joints and thrusters. The tanks are filled with 138 kg (303 lb.) of MIL-P-26536 high purity hydrazine and pressurized with helium to 450 psia. These tanks are located within the composite triangular structure of the spacecraft.

This system utilizes six 22 Newton (5 lbf.) thrusters to provide a maximum velocity change of 1430 meters per second (4690 ft./sec.). These thrusters provide all the attitude control necessary. Command and control will be from Earth since the spacecraft does not carry an onboard computer.

The thrusters are mounted in three pairs. Pairs A1/A2 forward and A3/A4 aft fire nearly parallel to the spin axis of the spacecraft. Pair T1/T2 fires nearly tangential to the circumference of the spacecraft and are located on its opposite sides. All thrusters can be pulse fired for as short a time as 15 msec.

The four axial thrusters are used for major translational burns and reorientation maneuvers. Engine pairs A1/A2 and A3/A4 are mutually redundant and any 1 of the 4 axial engines can make all required burns. These thrusters are canted 10° from the spin axis of the spacecraft. Pairs A1/A2 or A3/A4 are fired to make a major translational burn (nominally the 3 LOI burns). To minimize the need for reorientation maneuvers greater than 90°, A1/A2 and A3/A4 are used when the ΔV vector is in the spacecraft's northern and southern hemisphere, respectively. The axial thrusters are generally paired A1/A4 and A2/A3 to balance the torques for reorientation maneuvers. However, a single thruster, reorientation burn can be as small as 0.03°, or 16 time smaller that the attitude control requirement of 0.5°. This maneuver is carried out by alternately pulse firing one thruster from each thruster pair every half spacecraft rotation. This maneuver can be redundantly done by either of the axial pairs or by any 1 of the axial thrusters.

The two tangential thrusters are used for spin up/down maneuvers. T1 and T2 are fired for spin-down and spin-up, respectively. They are also used for small translational burns, during mid course correction maneuvers and lunar orbit maneuver burns, using the pulsed burn mode in conjunction with A1/A2 or A3/A4 burns. T1 and T2 are canted 5° from the tangent to minimize the impingement of the engine exhaust on the spacecraft, but they fire approximately in the same direction.

4.2.4.1 Component Characteristics

Component	Source	Heritage/ Model	Qty.	Size (cm)	Power (W)	Mass (kg)
Thrusters	Olin	MR-106E	6	3.43 D x 18.2	25.29	3.66
Tanks	PSI	80396-1	3	49.07 D sphere	-	15.69
Filter	Vacco	8509432-001	2	2.54 D x 8.51	-	0.26
Fill/Drain Valve	OEA	8547083	1	2.95 D x 6.02	-	0.17
Propellant Isolation Valve	Moog	51-166	2	5.84x3.86 x10.54	negl.	0.66
Pressure Transducer	Paine	8509430-011	2	2.59 D x 7.63	0.5*	0.56
Brackets	LMMS				-	2.00
Lines	LMMS			0.635 OD	-	1.40

* Only one transducer powered during flight

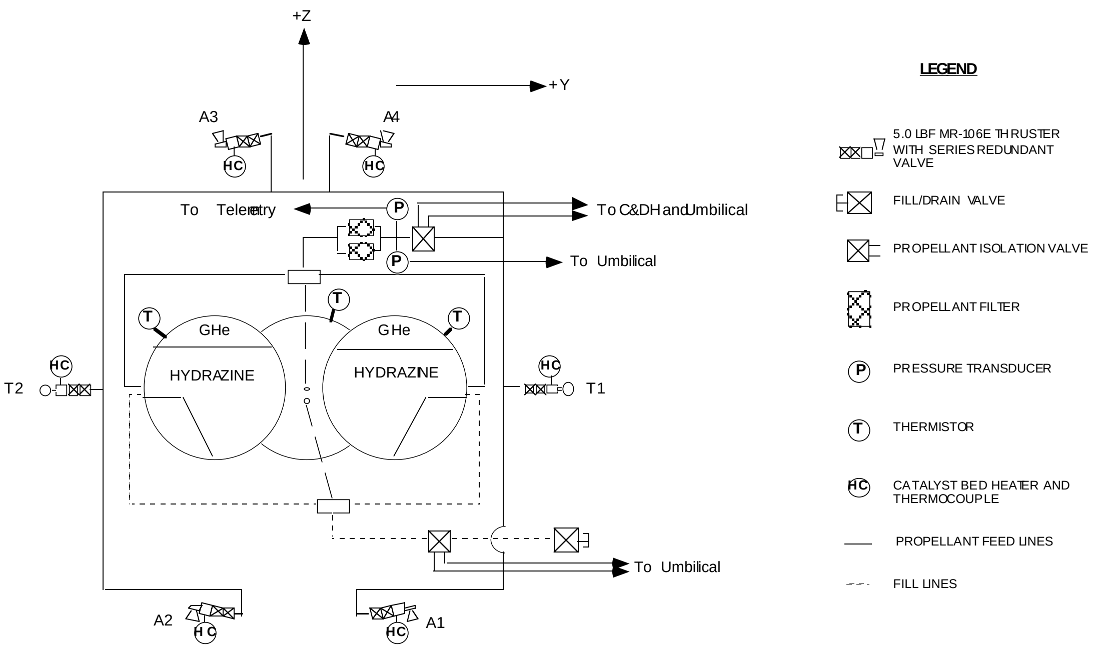


Figure 4.2.4-1 Propulsion System Block Diagram

4.2.4.2 Thrusters

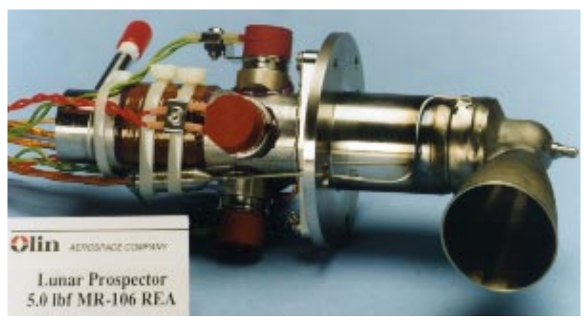


Figure 4.2.4-2 MR-106, 22.2 N Thruster

The thrusters are Rocket Research MR-106, 22.2 N (5 lbf.) monopropellant thrusters, developed and qualified for the Global Positioning Satellites. The MR-106 has a minimum delivered Isp of 220 seconds (234 max.). The thrusters are factory equipped with catalyst and valve heaters, thermostats, thrust chamber thermocouples, and series redundant valves. The catalyst and valve thermostats have set points at 5°C. When commanded to FIRE, the electromagnetically operated, spring loaded valves open and the thruster fires as long as the FIRE command is activated. When the command signal stops, the electromagnet shuts off and the spring automatically closes the valve for and FAIL-SAFE operation. Each thruster valve requires 25 watts to stay open. Engine firing is confirmed by the telemetry data on the thrust chamber temperature.

4.2.4.2.1 Total Impulse

The propulsion subsystem provides a minimum total impulse of 297,105 N-s (66,792 lbf.-sec). The MR-106 has a minimum delivered Isp of 220 seconds (234 sec. max.). The minimum average specific impulse over the specified mission duration is greater than 217 sec.

4.2.4.2.2 Thrust

The thrust at vacuum conditions for the MR-106E thruster is 38.08 N (8.56 lbf.) nominal at an inlet pressure of 3.10 MPa (450 psia.) at 18°C (65°F) (beginning of blowdown) and 11.88 N (2.67 lbf) minimum at 0.69 MPa (100 psia.) (end of blowdown).

4.2.4.3 Propellant Tanks

The propulsion subsystem includes three spherical, titanium hydrazine propellant tanks. The volume of each tank is 61,763.5 cm^3 (3,769 in^3) minimum and uses blowdown pressurization. The tank maximum expected operating pressure is 3.65 MPa (530 psia), with a proof pressure of 7.58 MPa (1100 psig) and a burst pressure of 8.27 MPa (1200 psig). The tanks will be loaded and pressurized by first filling them with helium and then filling with the hydrazine, thereby increasing the helium pressure.

4.2.4.4 Fill and Drain Valve

The propulsion subsystem has a manually-operated service valve for propellant fill and drain, pressurant fill, vent, system inert gas pressurization, and leak check operations.

4.2.4.5 Pressure Transducer

The propulsion subsystem contains a pressure transducer for monitoring tank pressure. The transducer has a range of 0 to 3.35 MPa (500 psia.) and a total error band of ±1.5 percent of full scale. The transducer requires an input voltage of 22 to 34 Vdc and provide an output voltage of 0 to 5 Vdc. The proof pressure capability of the unit is 6.89 MPa (1000 psia.) A second transducer with connections to the launch vehicle umbilical shall be capable of ground readout of tank pressure.

4.2.4.6 Filter

The propulsion subsystem uses two (2) 10 micron propellant filters in parallel for filtering out any contamination.

4.2.4.7 Fuel Lines

The propellant lines are made of 6.4 mm OD (1/4 inch) stainless steel. Wall thickness of all tubing and piping is 0.05 cm (0.020 in). The propellant lines are automatically heated.

4.2.5 ATTITUDE CONTROL SUBSYSTEM

The attitude subsystem provides the data so the ground computer can determine the spacecraft's attitude and spin rate. The subsystem also provides passive nutation damping. Lunar Prospector is spin stabilized, so the attitude subsystem (AS) is very simple. The AS consists of a sun sensor assembly (SSA), infrared (IR) Earth/Moon limb crossing sensor (EMS), and a nutation damper. The SSA and the EMS provide data that can be used by the ground processing system to determine Lunar Prospector's spin rate and inertial orientation. The solar array power output and antenna radiation pattern provide backup attitude data. The EMS provides backup spin rate data. Lunar Prospector's spin rate and attitude control are accomplished via the propulsion subsystem.

4.2.5.1 Component Characteristics

Component	Source	Heritage/ Model	Qty.	Size (cm)	Power (W)	Mass (kg)
Earth/Moon Limb Sensor	Ithaco		1	20.8 x 12.1 x 13.6	0.45	0.84
Sun Sensor	Adcole	Intelsat V	1	3.2 x 6.5 x 2.5	-	0.12
Sun Sensor Electronics & Mount	Adcole	N/A	1	10.3 x 8.4 x 10.7	0.43	0.50
Nutation Damper	LMMS	LMMS	1	Ø 58.4 x 9.5 mm	-	1.59

4.2.5.2 Sun Sensor Assembly

The Lunar Prospector Sun Sensor (P/N P108SA11) is manufactured by Adcole Corporation and is comprised of two separate modules, a sun sensor detector (SSD) and sun sensor electronics (SSE). The sun sensor detects the presence of the sun and the sun's orientation relative to the SSDs location on the spacecraft. As the SSD, in its nominal orientation as mounted on the Attitude Sensor Support Assembly, is swept across the sun, the sensor generates (1) an output (sun reference) pulse when the sun lies in the SSD measurement plane and (2) an output signal representing the angle (solar aspect angle) the sun line makes in the SSD measurement plane with respect to the plane normal to the spacecraft rotational axis. The output reference pulse and the solar aspect angle output signals will be telemetered to the ground and used in conjunction with the output of the Earth/Moon sensor and appropriate ground processing to determine the attitude and spin phase of the spinning spacecraft.

The SSE, which is mounted on the equipment deck near the bottom of the Attitude Sensor Support Assembly, receives +28 Vdc from the C&DH electronics and supplies the sun reference pulse, seven coarse data bits, and three analog voltages (sine, cosine, and bias) to the C&DH electronics. The seven coarse bits encode the ±64° field of view into 1° increments in Gray code format. The sine, cosine, and bias voltages define the fine sun angle which, when combined by ground processing with the coarse data, provides the solar aspect angle to 0.01° resolution. Raw data from the SSD is provided to the SSE through an interface cable.

The SSD is 3.2 cm x 6.5 cm x 2.5 cm (1.25" x 2.55" x 1.00"), weighs 0.12 kg (0.27 lb.), and consumes no power. The SSE is 10.3 cm x 8.4 cm x 10.7 cm (4.06" x 3.31" x 4.20") including the mounting feet, weighs 0.50 kg (1.1 lb.), and consumes 0.43 watt.

4.2.5.3 Earth/Moon Sensor (EMS)

The Lunar Prospector Earth/Moon Sensor (EMS) is manufactured by Ithaco Space Systems. The EMS (P/N P108SA12), which consists of optical elements, infrared filters, a detector/sensing element, and associated electronics, provides electrical signals representing the 30 to 100 micron radiometric profile of objects (earth, moon and sun) as the sensor optics/detector scans across them. The scanning motion is provided by the angular rotation of the Lunar Prospector spacecraft. The analog output signal, representing the radiometric profile, goes to the C&DH electronics where threshold detection defines the leading and trailing edge of the scanned body. The leading and trailing edge threshold transitions are time-tagged and telemetered to the ground for further processing to (a) determine the angle between the LP spin axis and the nadir vector to the scanned body and, in conjunction with data from the sun sensor, (b) determine the dihedral angle between the sun and the scanned body.

The detector cavity of the EMS is evacuated and sealed to preclude deterioration of the detector from exposure to moisture during storage and spacecraft level testing; however, a vent valve incorporated in the design allows the cavity to vent outgassed products after launch. A vent valve release screw holds the vent valve in its closed position to preclude its opening during handling and vibration testing and must be removed before launch.

The EMS, which is mounted on the Attitude Sensor Support Assembly along with the sun sensor detector, receives +15 Vdc and -15 Vdc from the C&DH electronics and supplies its analog output to the C&DH electronics as illustrated in Figure 4.2.5-1.

The EMS is 20.83 cm x 12.09 cm x 11.81 cm (8.20" x 4.76" x 4.65") including the mounting flange, weighs 0.84 kg (1.86 lb.), and draws 14 mA on the +15 Vdc line and 15.4 mA on the -15 Vdc line. The vent valve release screw, which is affixed to the vent valve assembly on the front of the unit below the optical baffle and is removed before launch, extends the depth (12.09 cm) dimension of the EMS envelope to 13.59 cm (5.35 in.)

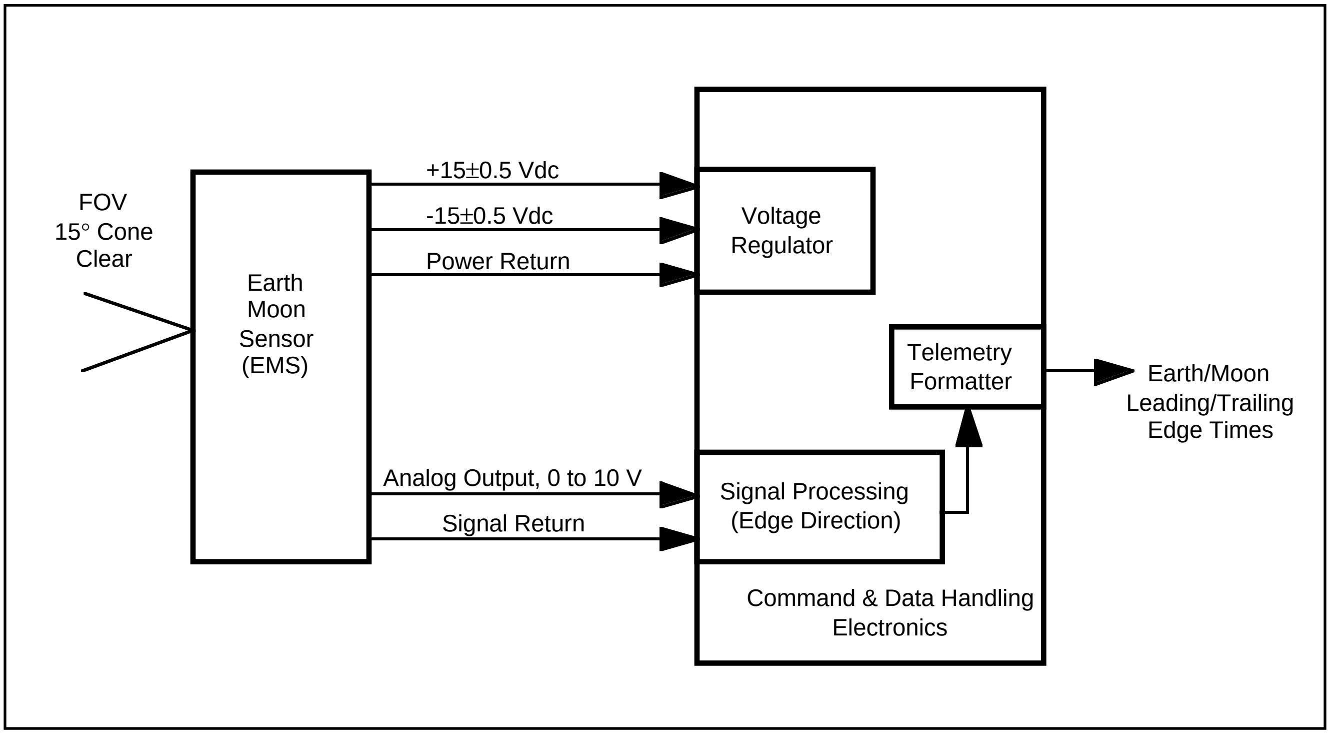


Figure 4.2.5-1 EMS Functional Interface Diagram

4.2.5.4 Secondary Attitude Data

Secondary SEA data is obtained from the solar array (SA) power output since it is a function of SEA. The SA output provides the only way of determining SEAs ≥64° due to SSA FOV limitations. The strength of the radio signals from both of the antennas depends on Lunar Prospector's attitude. Secondary attitude determinations are made with reasonable precision using the MGA signal strength when the Earth is within 20° of Lunar Prospector's equator. Because of the omni's relatively uniform pattern, attitude determinations using it can be made only near the -Z direction. Secondary determinations of Lunar Prospector's spin rate are made using Earth/Moon limb crossing data from the EMS.

4.2.5.5 Attitude And Spin Control

Lunar Prospector's attitude and spin rate are changed using the propulsion system, as discussed above, by direct ground command. A reorientation burn can be as small as 0.03°, or 48 times smaller than the attitude requirement of 0.5°. A spin or down burn can be produce a rotation rate change as small as 0.002 rpm or 3500 times smaller than the 0.7 rpm requirement.

4.2.5.6 Nutation Damping

Spacecraft nutation is removed by a simple, passive mercury filled damper located on the inside face the bottom panel of the triangular bus structure. The nutation damper (PN: P110A104) contains 0.59 kg (1.29 lbs.) of mercury. This damper serves to stabilize the spacecraft and dampen undesired motion. The nutation damper is a 58.42 cm (23 in.) ring attached to the base of the Lunar Prospector spacecraft. This is a passive device with no moving or operating elements other than the mercury. The mercury is contained in a 0.95 cm (3/8 inch) CRES tube with a helium blanket. The CRES tubing is pinch welded shut after filling and dye penetrant inspected.

4.2.6 C&DH SYSTEM

The command and data management subsystem provides stored and ground command capability, science and engineering data formatting and packetizing, and controls all spacecraft subsystems. The Command and Data Handling Electronics unit accepts uplink data streams from the spacecraft S-Band transponder (transmitter/receiver), decodes that data and executes all commands including sending serial commands to the science instruments. The uplink frequency is 2093.0541 MHz. The unit also conditions power from the solar array, provides a charge control function to the vehicle battery and distributes power to other equipment: Spectrometer Electronics, Magnetometer/Electron Reflectometer, Earth-Moon Sensor, Sun Sensor, Transponder and the heaters. The unit acquires signals and data from the science instruments, spacecraft sensors and actuators, formats and stores that data as downlink telemetry frames and then delivers the telemetry streams to the downlink transmitter at either 300 or 3600 bits per second. The downlink frequency is 2273.000 MHz and the transmitter power 5 Watts RF. The unit provides a central timing reference source for system synchronization and provides clocks and strobes to the science instruments. It further monitors attitude sensors and provides indications from which vehicle attitude can be determined on the ground. NOTE: Lunar Prospector does not have an on-board computer; all functions normally carried out by an on-board computer are done by a ground computer. This reduces Lunar Prospector's cost and increases its reliability

4.2.6.1 C&DH Electronics Box

The C&DH box being built by Spectrum Astro, Inc. (SAI) (SAI DWG # CID-200209) is located on one side panel of the triangular graphite epoxy structure. The C&DH is manufactured from aluminum and is 28.2 cm. x 20.3 cm x 15.0 cm (11.1" x 8.0" X 5.9") in size and weighs 6.17 kg (13.6 lbs.) The unit houses electronic circuit cards which perform the following functions:

- Charge controller board that conditions the solar array output, charges the battery and provides internal power to the C&DH.
- A digital input board that conditions bi-level signals for insertion into the telemetry stream.
- An analog input board that conditions and digitizes analog inputs for insertion into the telemetry stream.
- A power output board that contains relays for switching power to various instruments and the heater bus.
- A controller board that decodes uplinked data, verifies its validity and issues commands to the science instruments, the thrusters and the power board. The controller board also formats the telemetry into one of two data rates for transmission to the ground via the transponder. The stored data frames are alternately interleaved with the realtime frames and sent to be transmitted with a 53+ minute (3200sec) delay (half an orbital period).

4.2.6.2 C&DH Functional Description

For a complete functional description C&DH Electronics, Handbook users should refer to the *Product Functional Description for the Command and Data Handling Electronics*, Spectrum Astro document number 1073-EI-H01445. A copy of this document is kept in the MCC and in the Lockheed Martin Lunar Prospector Data Center. The C&DH *Product Functional Description* contains proprietary information and is distribution controlled. The *Product Functional Description* contains detailed technical information including:

- General characteristics and requirements
- A design overview for the Controller Board, Power I/O Board, Charge Control Board, Pulse Code modulation A&D Board, and firmware.
- Interface and circuit board block diagrams
- Thruster Execution flow chart

- VT curves
- Tables for Gain Tables, Telemetry Format, and Command Format.

4.2.7 COMMUNICATIONS SYSTEM

The communications subsystem provides for the reception of ground commands, the transmission of science and engineering data to the ground, and S-Band tracking. The subsystem consists of a medium gain antenna (MGA), an omni antenna, a S-Band transponder, a diplexer, and an RF switch. The system is a single string and transmits down-link telemetry at 22703.0 MHz over a ±1 MHz bandwidth at a nominal power of 5 Watts (peak power 7 Watts). The radiated signal is right-hand circularly polarized and received at 2093.0542 MHz over a ±1 MHz bandwidth.

4.2.7.1 Component Characteristics

Component	Source	Heritage/Model	Qty.	Size (cm)	Power (W)	Mass (kg)
Mid Gain Antenna	LMMS	None	1	3.8 D x 80	-	3.00
Omni Antenna	LMMS	Type 1	1	12 D x 20	-	0.30
Transponder	Loral/Conic	CRSS	1	20 x 18 x 12.5	32.00	8.20
RF Switch		CRSS	1	2.5 x 6.0 x 6.0	neg	0.2

4.2.7.2 Antennas

The Lunar Prospector antenna assembly is a combination of an omnidirectional antenna and a medium gain helix antenna. The assembly is mounted on the forward end of the spacecraft. The omni antenna provides near spherical coverage from 0° to 100° with respect to the spacecraft spin axis. The medium gain antenna provides 10° x 360° conical coverage about a plane parallel to the spacecraft XY plane. Both coverage patterns are symmetrical with respect to the spacecraft spin axis and rated at 5 watts.

The omni antenna is mounted on top of the MGA and is used for uplink at 250 bps and when engineering data are downlinked at 300 bps. The omni is used to received spacecraft command data during the mission and transmit engineering data to the ground prior to insertion to lunar orbit. At that point the medium gain antenna takes over and transmits scientific and engineering data to the earth. The omni antenna has a 0 dB hemispherical gain and 0 to -10 dB from 91° to 130°.

The medium gain antenna is a bifilar helix that has a minimum gain of 6 dB at the beam peak. The MGA is located on the spin axis, at the top (+Z) of the spacecraft. Its radiation pattern is a 360° x 10° wide, torroidal disk extending out from the spacecraft's spin axis. The MGA is used to downlink the 3600 bps science and engineering data stream.

4.2.7.3 Transponder

The Loral/Conic S-Band transponder is a standard, off the shelf, S-Band transponder that combines the receiver and transmitter into one unit. The transponder receives commands from the ground station along with signals inserted on the uplink for ranging purposes and transmits scientific data and telemetry from the spacecraft to the ground. Range and range rate data are derived from the coherent uplink and downlink S-Band transponder signal. The transponder can be operated in either a non-coherent or coherent mode to support range rate determination. A telemetry point allows the current transponder mode to be monitored. In non-coherent mode, the transmit frequency of 2273 MHz is determined by an internal crystal oscillator. The oscillator stability is ±25 ppm at 1 year and ±30 ppm at 3 years. In coherent mode, the transmitter frequency is derived from the received carrier VCXO signal by the ratio 240/221. The S-Band transmitter output power is ≥ 5 Watts over the life of the mission.

4.2.7.4 RF Switch

Switching from the omni to the helix antenna and back is accomplished by ground commanded. The antenna switch is a break-before-make switch capable of handling at least 5W RF power without any indication of corona or voltage breakdown. The switch can sustain at least 50,000 switching cycles for a period in excess of two years on-orbit operation. The switch latches and remains in the most recent commanded position. Switching the transmitter to the medium gain helix increases the margin, but is not necessary to support the Lunar Prospector mission. The two commands used are “Transmitter to MGA” and “Transmitter to Omni” to connect the transmitter to the medium gain helix antenna and to the omni antenna respectively. A telemetry point allows the state of the RF switch to be monitored.

4.2.8 THERMAL CONTROL SYSTEM

The thermal control subsystem maintains all spacecraft subsystems within their required thermal limits during all phases of the mission. The Lunar Prospector thermal control is achieved by passive means such as surface finishes and multi-layer insulation (MLI) blanketing and redundant thermostatically controlled heaters. All materials used are flight proven and grounded to prevent surface charging. Thermal dissipation of electronic components located on equipment panels is managed by dedicated radiator surfaces and selective blanketing of equipment to limit heat rejection to space and other surfaces. Additionally, a high conductivity graphite epoxy equipment panel is used to effectively conduct heat away from high dissipating units and equipment separation on the panels is determined according to dissipation levels. The sides and bottom of the bus structure are blanketed to minimize heat loss and reduce orbital temperature swings. Lunar Prospector's thermal control is modeled for the translunar phase, longest daytime and nighttime lunar passes, and the longest lunar eclipse. Contamination of surfaces by UV polymerized degassed volatiles from the spacecraft are accounted for.

Thermal control of the propulsion system prevents the propellant from freezing or boiling. The limits are +4° to +49° C (41° F to 120° F). The tanks are thermally isolated from the spacecraft's interior by 8 layer double aluminized mylar and kapton blankets and low conductivity phenolic spacers at the mounting flanges. The tanks have redundant, thermostatically controlled heaters with set-points at 12° C (54° F). The heaters are sized to maintain the tanks at the set-point during the longest lunar eclipse with a 1 heater failure. The propellant lines are similarly treated.

Since the thermal state of the spacecraft is controlled by passive means and automatic thermostatically controlled heaters, there are no direct commands associated with this system other than enabling of the primary and redundant heater buses. In an emergency, overheating of the bus can be controlled by changing the spacecraft's attitude and turning off the science instruments.

4.2.8.1 Component Characteristics

Component	Source	Heritage/Model	Qty.	Size (cm)	Power (W)	Mass (kg)
Propulsion Heaters	TAYCO	Milstar	34	Various	26.1†	1.01
Thermostats	Elmwood.	88	88	1.7 dia.	N/A	0.87
Thermistors	YSI	15	15	0.51 dia.	N/A	0.1
Thermal Blankets	LMMS	various	AR	Various	N/A	4.59

† thermostat controlled heaters; intermittent power-on

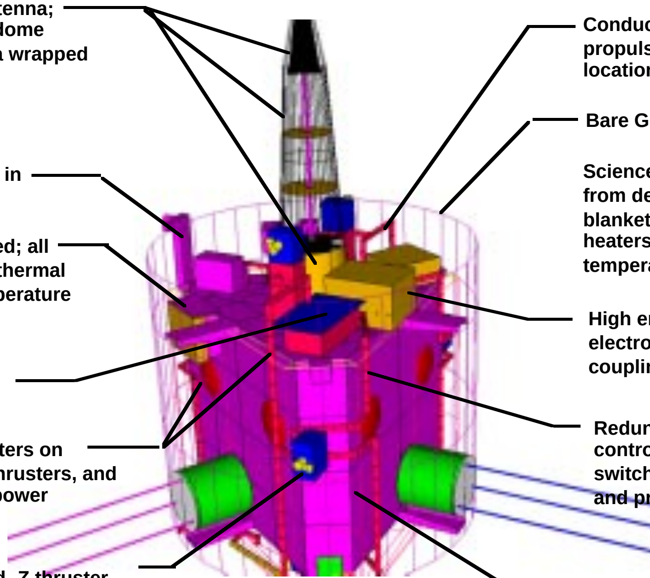


Figure 4.2.8-1 Thermal Control Subsystem

4.2.9 CORE SYSTEMS FLIGHT SOFTWARE

The Lunar Prospector spacecraft does not have an onboard computer. The spacecraft is instead controlled from the ground. This is possible because of Lunar Prospector's engineering simplicity, its simple commanding (the spacecraft is controlled by only 61 on/off commands), the short (2.5s) round trip light-time to the Moon, and essentially all commanding can be done line-of-sight. Spacecraft control via a ground based computer reduces the cost and complexity of the spacecraft, allows upgrades to the flight software without interrupting the mission, and increases overall mission reliability. The ground computer is much more capable than an onboard computer, the former does not have to be space qualified, and in the event of a hardware failure it can be swapped out with another computer without affecting the mission. Basically, instead of the controllers uplinking commands to an onboard computer which "talks" to the spacecraft subsystem via hard wires, the Lunar Prospector controllers "talk", via hard wires, to the ground computer which uplinks the commands to the spacecraft subsystems (albeit through the C&DH). Thus, the "Ground Software" is the "Flight Software" for the Lunar Prospector.

4.2.9.1 Spacecraft Ground Computer Software (Flight Software)

The primary function of the ground software is to perform command transmission, receive spacecraft telemetry, and perform computations for attitude determination, spin rate, hardware status, and spacecraft configuration. Commands are sent via the ground computer by the command controller.

Attitude and spin rate determinations using telemetry data from the sun sensors and the Earth/Moon horizon sensors are computed by the ground software packages. Power output, battery status, and other hardware monitoring for ground controllers to monitor the health is performed by OASIS.

4.2.9.2 Multimission Ground Data System

The ground station software consists mainly of commercial off the shelf (COTS) products with minimal Lockheed Martin developed software used to integrate the COTS. The multimission ground data system

(MGDS) software is used to uplink commands to the spacecraft. OASIS is used for receiving and displaying telemetry from the spacecraft ground computer. OASIS has been integrated with a spacecraft simulator to allow development of CSTOL procedures to support mission operations and validate the attitude and determination software. The simulator is also used as a training tool for mission operations personnel. OASIS has a graphical user interface (TAE+) which allows designers, testers, and mission operators to easily build custom display screens. Since the software is COTS and the "glue" software is already developed, approximately 80% of the required code will be reused from existing code. Most of the work required will be repopulating the ground data base with new command and telemetry definitions.

4.2.9.3 Attitude Determination Software

The Lunar Prospector Attitude Determination Software (ADS) estimates the inertial orientation of the Lunar Prospector spin axis using attitude sensor telemetry, orbit ephemeris data, and Solar/Lunar/Planetary ephemeris data. Attitude sensor telemetry is obtained from spacecraft and stored in attitude bridge files by OASIS. Orbit ephemeris data and the Solar/Lunar/Planetary ephemeris data is provided by the NASA Flight Dynamics Division (FDD). ADS produces attitude estimates using either the combined data from the sun sensor and the Earth/Moon sensor, or using sun sensor data alone. The resulting attitude estimate is expressed in the Mean-ecliptic-of-J2000 inertial reference frame. Detailed instructions for executing ADS are contained in the ADS Users Guide, which is part of the Lunar Prospector Ground Software Users Guide and Programmers Guide (P4583022).

4.2.9.4 Command Generation Software

The Lunar Prospector Command Generation Software (CGS) generates a sequence of spacecraft commands required to execute a thruster maneuver. CGS supports three types of maneuvers: spin rate change, spin axis reorientation, and delta-V. CGS inputs are the current spin rate, spin axis attitude, propellant tank pressure and temperature, and the maneuver target set: target spin rate, target attitude, or target delta-V. CGS determines which thrusters to use and calculates values for each of the thruster parameters. CGS outputs an MGDS-compatible file of command pnemonics for the Flight Controller to uplink to the spacecraft. Detailed instructions for executing CGS are contained in the ADS Users Guide, which is part of the Lunar Prospector Ground Software Users Guide and Programmers Guide (P4583022).

4.3 TRANS-LUNAR INJECTION STAGE

This section describes the Trans-Lunar Injection (TLI) Stage. The TLI Stage was developed by Thiokol Corporation of Elkton, MD under subcontract to Lockheed Martin Missiles & Space. Serving as a fourth stage, the TLI is major element of the Lunar Prospector baseline.

The TLI Stage injects the Lunar Prospector Spacecraft into a trans-lunar trajectory from a low earth parking orbit. The TLI stage consists of a STAR 37 FM solid rocket motor, two spin rockets, a graphite epoxy interstage structural adapter, spacecraft–TLI stage separation system, a cold gas collision avoidance system (CAS), Command Timer, and a flight termination system. The Star 37FM motor provides enough total impulse to increase Lunar Prospector's velocity by 3143 m/s (7030.7 mph). The TLI Stage is approximately 170.8 cm (67.3 in.) long and 128.3 cm (50.8 in.) in diameter. The total fueled mass of the TLI Stage is 1216.5 kg (2682.0 lb.). This mass values includes 1,010.7 kg (2,228.1 lb.) of TP-H-3340 solid rocket propellant in the Star 37FM motor., 1.1 kg (2.4 lb.) of expendable Arcite 377A propellant in the Spin Motors and 3.6 kg (7.7 lb.) of gaseous N2 in the CAS bottles pressurized to 24.13 MPa (3500 psia.). Figure 4.3-1 summarizes the TLI Stage configuration.

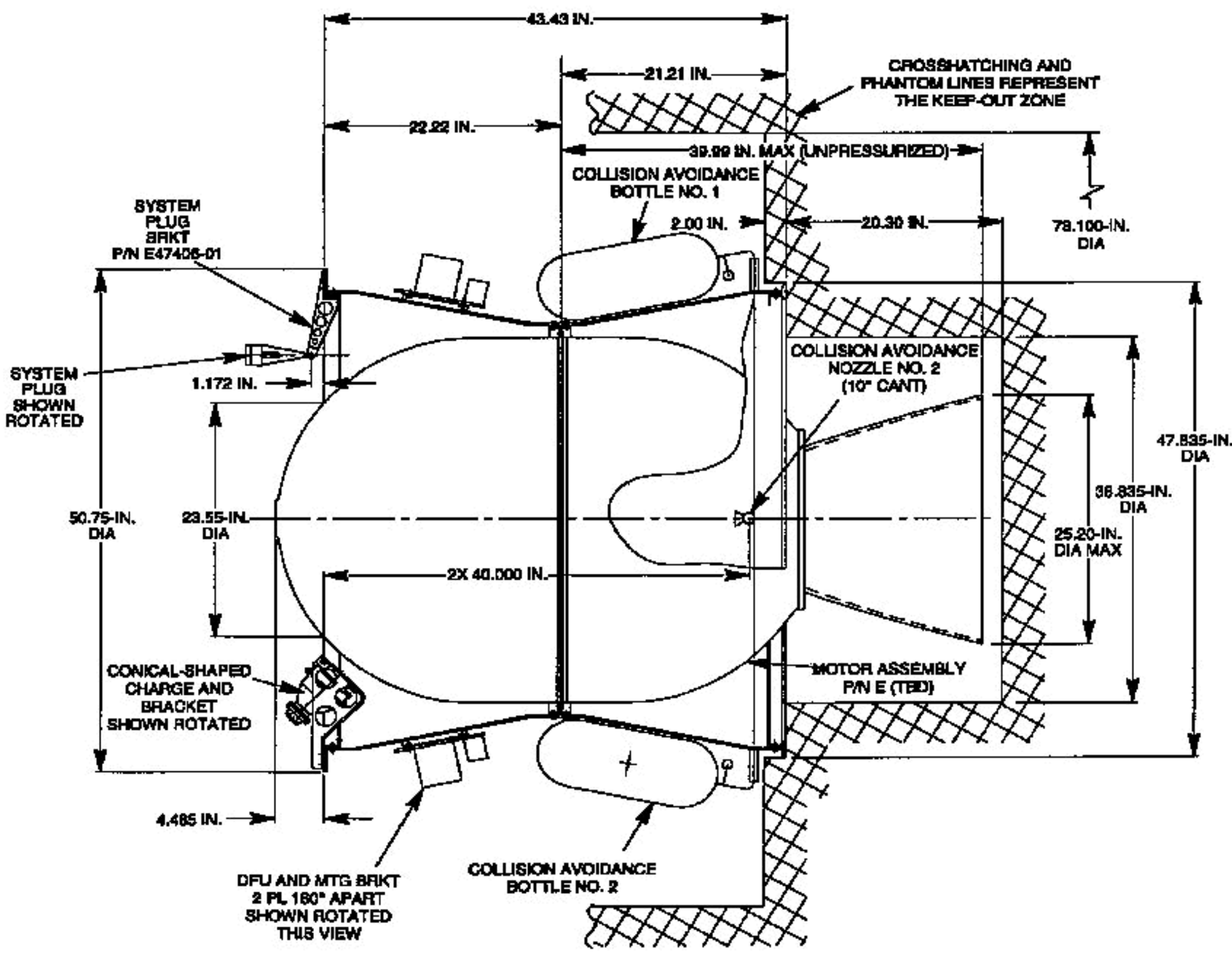


Figure 4.3-1 TLI Stage Design Overview

4.3.1 COMPONENT CHARACTERISTICS

Component	Source	Heritage/ Model	Qty.	Size (cm)	Power (W)	Mass (kg)
Adapter Structure	Thiokol	N/A	1	110.31" x Ø 128.91	-	56.02
Apogee Motor	Thiokol	Star 37FM	1	168.61 x Ø 93.47	-	1092.32
Command Timer	SwRI	N/A	1	13.0 x 15.4 x 7.6	3	4.50
Nutation Damper	LMMS	LMSC	1	Ø 58.4 x 9.5 mm	-	1.59
CAS Tank	SCI	5000245	2	43.7 x Ø 17.3	-	13.66
CAS Pyro Isolation Valve	LMMS	8509436-001	2	15.2 x 8.3 x 8.6	-	0.54
CAS Fill Valve	LMMS	8509432-009	2	6.0 x Ø 2.5	-	0.23
CAS Pressurant, GN2	-	-	N/A	-	-	3.50
CAS Nozzles	LMMS	N/A	2	N/A	-	1.18
Spin Motors	MDAC	KS-210	2	20.78 x Ø 7.77	-	3.52
Conical Shaped Charge	Thiokol	Model 2011	1	15.8 x Ø 13.5	-	1.27
DFU	LMMS	Peace Keeper	2	32.7 x 21.7 x 10.6	-	9.07
LOS	LMMS	1A58201	2	AR	-	0.59

4.3.2 STAR 37 FM

The Star 37FM is a flight qualified solid rocket motor typically used as an apogee kick motor by other launch vehicle systems. Manufactured by Thiokol Corporation of Elkton MD, the STAR 37FM TE-M-783-7, rocket motor uses 1010.65 kg (2228.11 lbm.) of TP-H-3340 solid propellant grain with an eight-star points and radial slots to control the burning surface area. The propellant volume is 498,166 cm^3 (30,400 in^3.) The grain is insulated with silica-filled EPDM rubber and is bonded to a 6Al-4V titanium case. The case minimum ultimate strength is 1172.11 MPa (170,000 psi.) with a minimum yield of 1103.16 MPa (160,000 psi.). The titanium case has an aft-polar boss for mounting the nozzle assembly. The nozzle assembly consists of five major components: throat, exit cone, closure, insulators, and igniter. The throat material is a 3D carbon-carbon composite and it's diameter and erosion rate control motor pressure. The exit cone is carbon-phenolic and is contoured to optimize specific impulse with an area ratio of 50.85. The closure is titanium and it is the primary structure to which all of the other nozzle components are attached. The Pyrogen igniter is mounted on the nozzle assembly and is initiated by two flame sticks which are attached to the ignition train. The motor has an overall length of 168.61 cm (66.38 in.), a diameter (case cylinder) of 93.47 cm (36.8 in.), and an approximate total, loaded weight of 1092.32 kg (2408.16 lbm.). Figure 4.3-2 shows the general arrangement of the Star 37FM motor.

4.3.2.1 Impulse

The predicted vacuum nominal total impulse delivered by the TLIM at 21.1°C (70.0°F) is 2,894,622 N-s (650,737 lbf.-sec.) The propellant specific impulse (Isp) is 291.89 s and its effective Isp is 289.82 s.

4.3.2.2 Thrust

The average vacuum thrust of the Star 37FM is 45,817 N (10,300 lbf.) at 21.1 °C (70.0°F) given an average pressure of 3.62 MPa (525 psia.) The maximum thrust is 53,067 N (11,930 lbf.) and the maximum pressure is 4.03 MPa (585 psia.) The thrust rise rate does not exceed 2,224,110 N/s (500,000 lb./sec.) and no oscillatory pressure component exceeds 172 kPa (25 psi.) from the nominal instantaneous pressure.

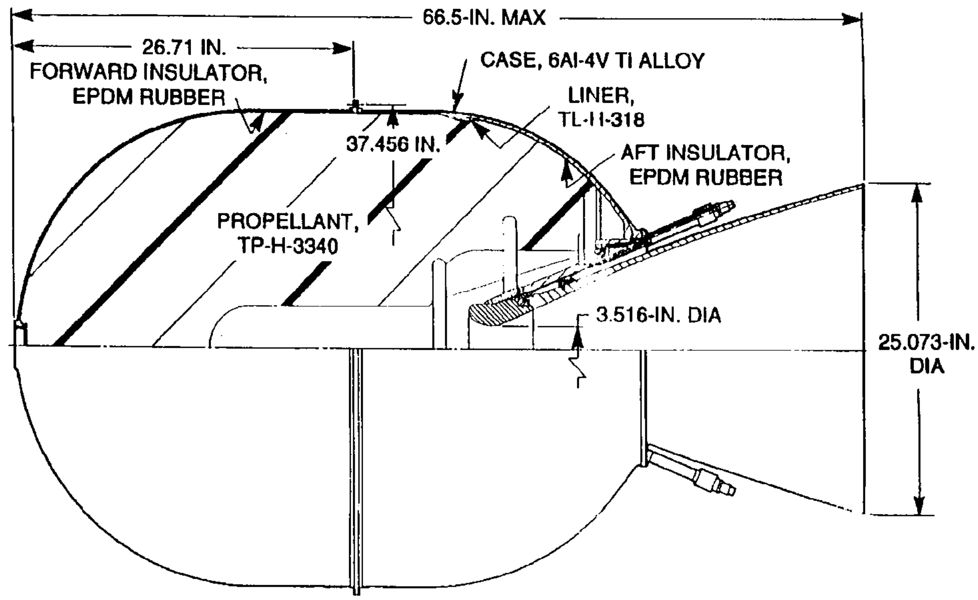


Figure 4.3-2 Star 37FM Motor

Table 4.3-1 TLI Motor Propulsion Characteristics

Dispersion Variable	Nominal Value	3s Uncertainty	Source
Propellant Weight	2228.1 lbm.	0.2 %	Motor Logbook, 7/28/97
Effective Specific Impulse	289.82 lbf.-s/lbm.	0.5 %	Motor Logbook, 7/28/97
Burn Time	62.3 s	3.0 %	Motor Logbook, 7/28/97
Action Time	63.7 s	3.0 %	Motor Logbook, 7/28/97
Igniter Propellant Weight	1.28 lbm.	0.2 %	Motor Logbook, 7/28/97
Expended Inerts	15.9 lbm.	10.3 %	Motor Logbook, 7/28/97
Position Offset (LP XY Plane)	0.002 in.	0.007	Motor Logbook, 7/28/97
Thrust Alignment	0.00029 rad	0.002	Motor Logbook, 7/28/97
PMBT	70.0 °F	3 °F	Motor Logbook, 7/28/97

4.3.3 COMMAND TIMER

The Command Timer, developed under subcontract by Southwest Research Institute, is a hybrid analog-digital device used to control the sequence of flight events immediately following payload–booster separation. The Command Timer detects separation of the TLI from the launch vehicle by means of a break wire and initiates a pre-programmed hardwired command sequence, see Table 4.3-2 The Command Timer weighs 4.5 kg (10 lbm.) and is 13.0 cm x 15.4 cm x 7.6 cm (5.12” x 6.06 “ x 3.00 “) in dimension. The unit mounts on a bracket to the forward ring of the TLI Stage structural adapter. Electrical power is supplied to the Timer from the spacecraft battery through the Spacecraft–TLI separation umbilical.

Table 4.3-2: Command Timer Events Sequence

Time	Command / Event Description
0.50 sec	TLI Spin Rockets Ignition: spin up to 57 rpm
20.00 sec	STAR37FM Ignition: begin trans-lunar injection burn
86.60 sec	LP Spacecraft Power On
91.40 sec	Collision Avoidance System Activated
91.60 sec	LP-TLI Separation: Primary Separation Squibs
91.65 sec	Secondary Separation Squibs (redundant signal)

4.3.4 ADAPTER STRUCTURE

The TLI stage structure, shown in Figure 4.3-4, is a wound filament composite adapter. Manufactured by Thiokol, the adapter serves as both primary structure for the TLI Stage and as the adapter between the Lunar Prospector spacecraft and the Athena II launch vehicle. The adapter is roughly 110.31 cm (43.43 in.) in length and 128.91 cm (50.75 in) in diameter and has a mass of 56.02 kg (123.5 lbm.).

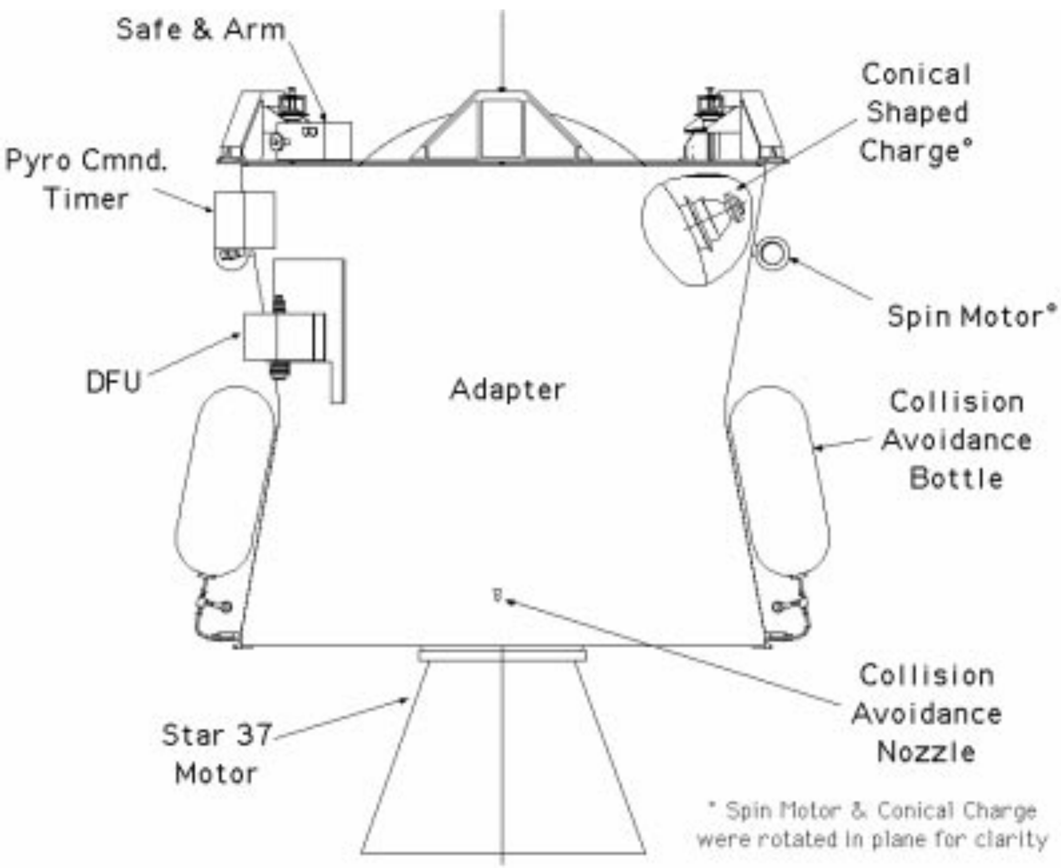


Figure 4.3-3 TLI Adapter Structure and Components

4.3.5 COLLISION AVOIDANCE SYSTEM

The Lunar Prospector Collision Avoidance System (CAS) is designed increase the separation velocity between the spacecraft and TLI stage. This CAS consists of two composite overwrapped pressure bottles, four pyro valves, two fill valves, two nozzles, and associated plumbing.

The bottles are pressurized to 3500 psi. with gaseous nitrogen. The CAS bottles, manufactured by Structural Composites Industries 325 Enterprise Place, Pomona, CA 91768, consists of 6010-T62 Aluminum liner with a graphite epoxy overwrap production model SCI SUPER TANK MODEL NO: AC-5000. The bracketry to hold the tanks, lines and fittings is installed on the TLI stage by Thiokol. Figure 4.3-5 below shows the general configuration of the tanks.

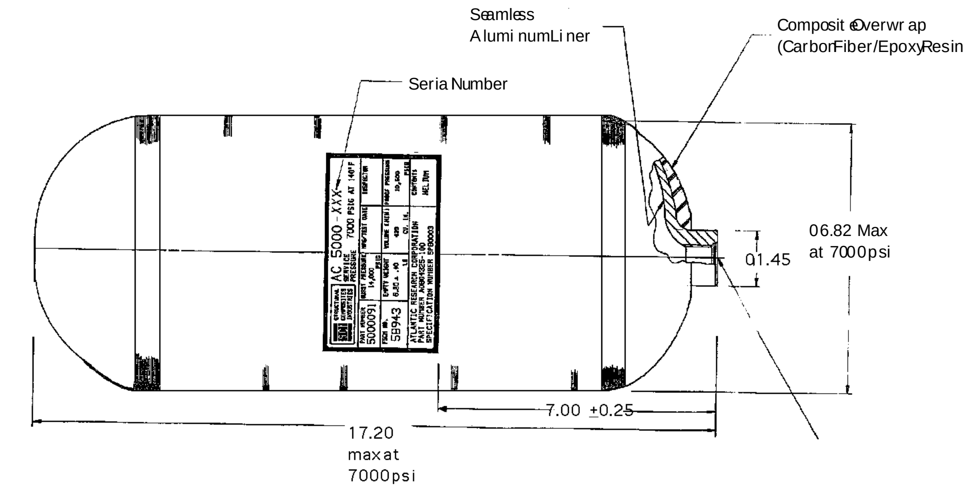


Figure 4.3-4 Composite Overwrapped Tank

4.3.6 SPIN MOTORS

The Spin Motors are located outboard on the TLI stage and spin up the spacecraft–TLI stack after separation from the launch vehicle. The spin rockets contain a total of 1.04 kg (2.3 lbm.) of Arcite 377A propellant. They measure 20.78 cm (8.18 in.) in length and have a maximum diameter of 7.77 cm (3.06 in.). Each motor has a total mass of 1.76 kg (3.88 lbm.).

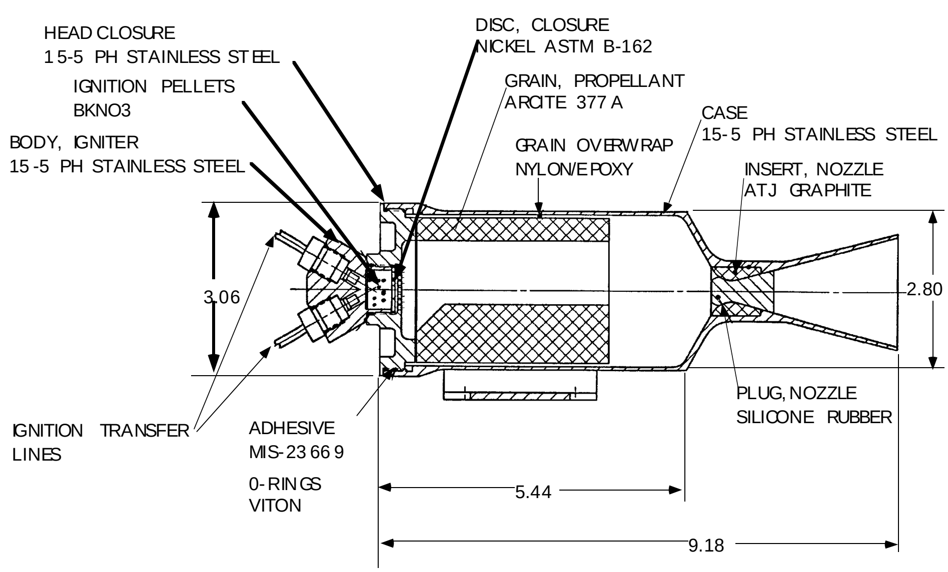


Figure 4.3-5 MDAC KS-210 Spin Rocket

The spin motors are activated from two vendor equivalent NASA Standard Initiators (NSI), each of which initiates two Ignition Transfer Line Assemblies. The transfer line assemblies connected in an OEA Aerospace TLX Manifold, PN 24051-1, which provides for a redundant initiation of the Spin Rocket Motors. The total impulse delivered by each motor in vacuum at 21.1°C (70.0°F) is 1201.5 N-s (270.1 lbf.-sec.). The motor's maximum in vacuum thrust does not exceed 1427.9 N (321.0 lbf.).

Table 4.3-3 TLI Spin Motor Propulsion Characteristics

Dispersion Variable	Nominal Value	3s Uncertainty	Source
Propellant Weight	1.144 lbm.	0.003 lbm.	ARC Datasheets
Specific Impulse	237.3 lbf.-s/lbm.	1.83 lbf.-s/lbm.	ARC Datasheets
Burn Time	1.0810 s	0.055 s	ARC Datasheets
Action Time	1.5073 s	0.036 s	ARC Datasheets
Ignition Delay	0.0124 s	0.0014 s	ARC Datasheets
Expended Inerts	0.01 lbm.	0.005 lbm.	ARC Datasheets
PMBT	70.0 °F	3 °F	ARC Datasheets

4.3.7 DFU

The Destruct Firing Units (DFU), PN 1A59040, SPEC. 1A59386, supplied by the Athena II booster are based on the Peace Keeper DFU units. The DFUs obtain electrical power and command signals from the booster. The DFUs initiate the LOS lines by means of exploding bridge wire detonators connectors. The LOS, in turn, initiate the destruct conical shape charge.

4.3.8 LOS LINES

The LOS lines, LMLV P/N 1A58201-101 & 1A58201-102, connect the two DFU's to the Conical Shaped Charge. The LOS transfer line consists of a length of flexible confined detonating cord (FCDC) terminated at each end with loaded booster cups and end fittings.

The FCDC consists of a linear, aluminum-sheathed explosive core supported along its length by 4 layers of over-wrap. The linear, aluminum-sheathed explosive core is 2.7 grains per foot HNS-IIA explosive mix per WS5003. The over-wrap consists of sequential layers of polyethylene, fiberglass, polyurethane and a CRES wire outer-braid.

Loaded booster cups are attached at each end of the FCDC. The "input" booster cup consists of a drawn stainless steel cup (0.150" diameter) which is loaded with approximately one (1.0) grain of HNS-IA explosive mix per WS5003. The "output" booster cup consists of a drawn stainless steel cup (0.190" diameter) which is loaded with approximately 1.6 grains of HNS-IA explosive mix per WS5003. Each booster cup is attached to a steel fitting, which is in turn mechanically secured to the FCDC. The booster cups, so attached to the FCDC, are then described as "end-tips". A threaded nut is provided at each end-tip to for mating.

4.3.9 CONICAL SHAPE CHARGE

The conical shape charge is the component in the Flight Termination system that destroys the propulsive action of the Star 37FM motor. Mounted on the vertical aspect of the Star 37FM, the charge is initiated by LOS lines and DFUs. The explosive charge drives through the solid rocket motor casing disabling uniaxial thrust through the motor's nozzle. The Thiokol Model 2011 Conical Shape Charge contains 500 grams Composition C4 in the main charge and 17 grams Composition A4 in the booster charge. The shape charge mass is 1.27 kg (2.8 lbm.).

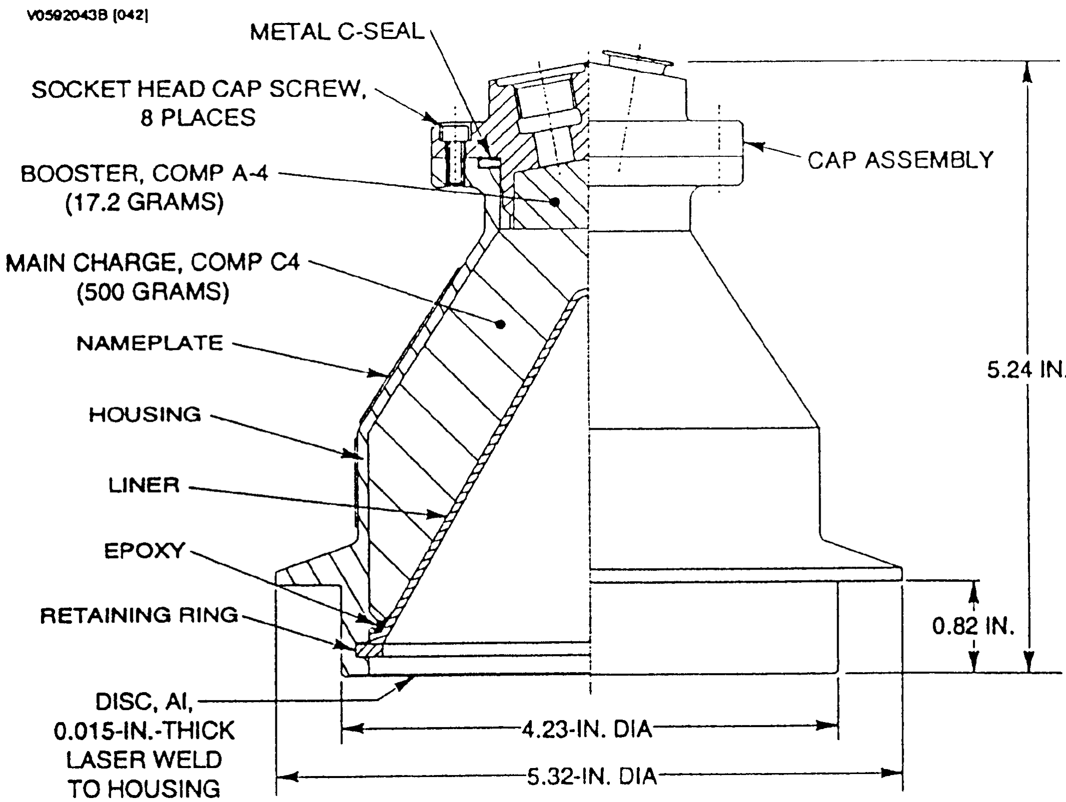


Figure 4.3-6 Model 2011 Conical-Shaped Charge

4.4 LAUNCH VEHICLE

4.4.1 LAUNCH VEHICLE DESCRIPTION

With it's relatively low mass, Lunar Prospector Spacecraft can be launched directly to the Moon on a 3 stage Lockheed Martin Athena II launch vehicle (LV) with a the TLI Stage serving as a kick stage. Athena II, formerly LMLV-2, was selected because of its low cost, launch capability margin, and simplified physical and organizational interface.

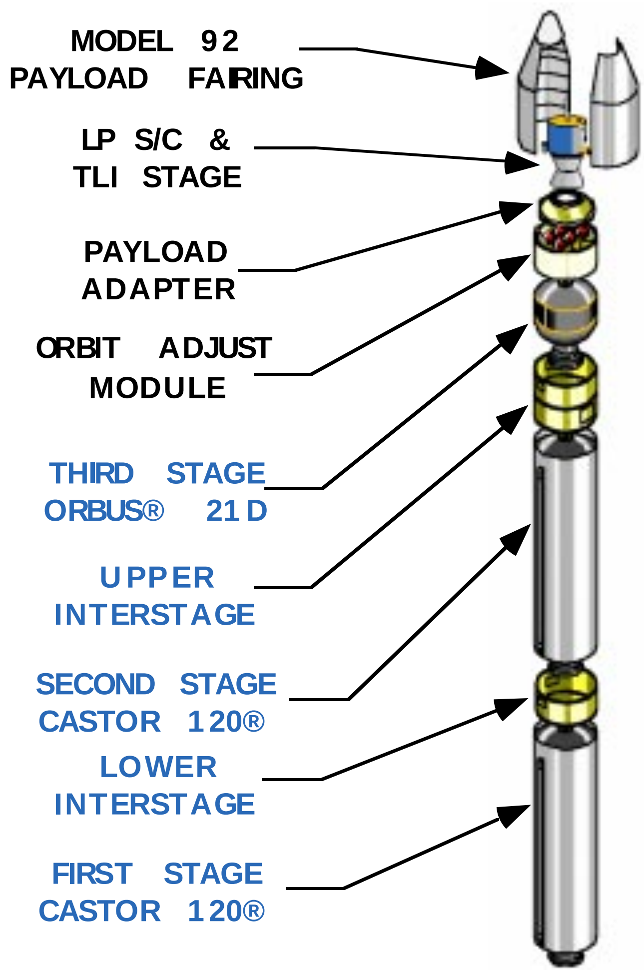


Figure 4.4.1-1 Athena II/Lunar Prospector Configuration

The Athena II is a three stage launch vehicle based on commercially available solid rocket motors. The first and second stages of the Athena II are Thiokol Castor 120™ motors. The third stage is a smaller United Technologies Chemical Systems Division Orbus® 21D motor. A hydrazine monopropellant Orbit Adjust Module (OAM) provides on orbit attitude control. The Athena II for Lunar Prospector uses a 92 inch diameter payload fairing. Figure 4.4.1-1 shows the launch vehicle profile.

The Castor 120™ is a commercial motor derived from Thiokol's Peacekeeper ICBM first stage motor. The Castor 120™ uses HTPB propellant, with thrust vector control provided by a hydraulically-gimbaled nozzle. The lightweight case is made of graphite-epoxy composite. The first and second stage Castor 120™ motors are separated by an aluminum interstage.

The Orbus® 21D third stage is connected to the second stage by an aluminum interstage. The Orbus® 21D has a kevlar case and electromechanical thrust vector control. To avoid confusion with the Athena I configuration, the Orbus® 21D stage is generally referred to as the ESBM, or Equipment Section Boost Motor.

Attitude control for the Athena II is provided by the OAM. The OAM includes a flight computer, inertial measurement unit (IMU), and a hydrazine propulsion system. The OAM provides roll control during first and second stage burns, and three axis attitude control during coasts and after OAM separation from the second stage. Tank pressurization is provided by a helium pressurization system.

4.4.1.1 Athena II First and Second Stage

As shown in Figure 4.4.1-1, the first and second stages of the Athena II are Castor 120™ solid propellant rockets. The Castor 120™ is a commercial motor derived from Thiokol's Peacekeeper Intercontinental Ballistic Missile (ICBM) first stage motor. The Castor 120™ uses 48,988 kg (108,000 lbs.) of Class 1.3 Hydroxy-Terminated Polybutadiene (HTPB) propellant, with thrust vector control provided by a hydraulically gimbaled nozzle. The lightweight case is made of graphite-epoxy composite.

The first and second stage Castor 120™ motors are separated by an aluminum interstage. Castor 120™ motor characteristics are provided in Table 4.4.1-1.

Table 4.4.1-1 Castor 120™ Characteristics

Propellant mass (kg) (weight (lbm.))	48,988 (108,000)		
Initial total mass (kg) (weight (lbm.))	53,070 (117,000)		
Burn time (sec)	75.3 - 88.7		
Burnout mass (kg) (weight (lbm.))	4,082 (9,000)		
Average thrust Newtons (lbm.)	(387,000)		
Maximum thrust Newtons (lbm.)	(486,000)		
Primary propellant constituents	68% Ammonium perchlorate (AP)	20% Aluminum (Al)	12% Hydroxyl Terminated Polybutadiene (HTPB)
Propellant designation	TP-H1246		

4.4.1.2 Orbus 21D Upper Stage

The Orbus® 21 D third stage is connected to the second stage by an aluminum interstage. The Orbus® 21D has a kevlar case and electromechanical thrust vector control. The Orbus® 21D uses 9,766 kg (21,530 lbm.) of Class 1.3 Hydroxy-Terminated Polybutadiene (HTPB) propellant. Orbus® 21D motor characteristics are provided in Table 4.4.1-2.

Table 4.4.1-2 Orbus 21D Characteristics

Propellant mass (kg) (weight (lbm.))	9,766 (21,530)		
Initial total mass (kg) (weight (lbm.))	10,614 (23,399)		
Burn time (sec)	150		
Burnout mass (kg) (weight (lbm.))	848 (1,869)		
Average thrust Newtons (lbm.)	42,040		
Maximum thrust Newtons (lbm.)	47,819		
Primary propellant constituents	71 % Ammonium perchlorate (AP)	18 % Aluminum (Al)	12 % Hydroxyl Terminated Polybutadiene (HTPB)
Propellant designation	TP-H-3340		

4.4.1.3 Orbit Adjust Module

Attitude control for the Athena launch vehicle family is provided by the Orbit Adjust Module (OAM). The OAM includes the flight computer and inertial measurement unit (IMU), as well as hydrazine propellant tanks and thrusters. The OAM provides roll control during first and second stage burns, and three axis attitude control during coasts and after OAM separation from the second stage. In addition, the OAM performs orbit circularization and adjustment functions. The OAM utilizes 236 kg (520 lbm.) of hydrazine pressurized by helium in a blow down system.

4.4.1.4 Payload Fairing (PLF)

During ascent, the LP SC / TLI Stage combination is protected from aerodynamic forces by a 2.34 m (92 in.) payload fairing. as shown in Figure 4.4.1-2. The Athena II payload fairing is an all-aluminum, ring stiffened monocoque structure with field joints for the nose cap and for attachment to the equipment section. The conical section is tapered 15 degrees to reduce aerodynamic drag. Cork, with an external coating of reflecting white paint, is used on the external surface of the nose, conical section and barrel section to insulate the payload from drag-induced heating and to maintain structural temperature to within required limits. The fairing is separated as hinged clam-shells with Zip, a patented clean separation system that severs the fairing circumferentially and longitudinally into separate halves. Thruster springs located at the base of the fairing provide the required energy for separation. Two helper springs located at the upper conical section provide force to overcome separation friction. This arrangement minimizes radial fairing deflections and maximizes payload envelope. The cammed hinges release after 45° of rotation and allow the fairing halves to separate clear of the payload.

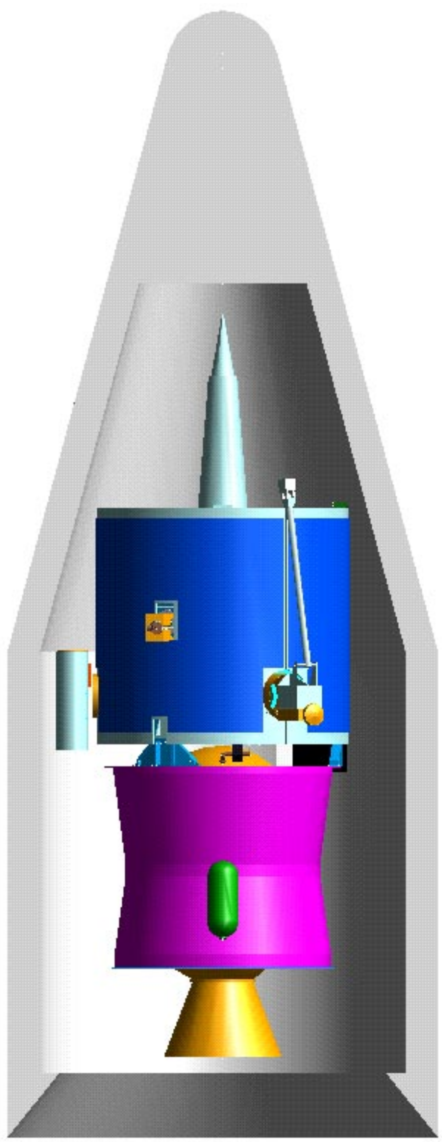


Figure 4.4.1-2 Encapsulated Lunar Prospector Spacecraft & TLI Stage

4.4.1.5 Flight Termination System (FTS)

The Eastern Range Safety Office will establish flight safety limits for the trajectory of the LP launch vehicle. These limits are established to ensure that errant launch vehicles (or debris resulting from a launch failure) do not pose a danger to human life or property. These flight safety limits are determined before launch by calculating the range of possible flight azimuths using predicted values for winds, explosively produced fragment velocities, human reaction time, data delay time, and other pertinent data. During a launch, if the vehicle trajectory indicates that these limits would be exceeded, the Range Safety Officer would take appropriate action, including destruction of the vehicle.

As specified by Range Safety requirements, the LP launch vehicle is equipped with a Flight Termination System (FTS). This system is capable of destroying the vehicle based on commands sent from the Range Safety Officer.

The purpose of the FTS is to ensure that the Athena II flight path is contained in the range boundaries. If there is a flight anomaly (e.g., an obviously erratic flight, a loss of tracking data, a violation of destruct limit lines, or a premature stage separation) the Athena II is destroyed. Flight termination results from either command or automatic initiation of the vehicle's destruct charges. These charges destroy the pressure integrity of the motor cases producing a zero net-thrust or tumbling condition.

All FTS electrical and mechanical components meet or exceed EWR 127-1 Range Safety Requirements. The FTS is a redundant system consisting of two channels (A and B), with either channel capable of independently:

1. Providing command destruct to all stages upon receipt of a Range transmitted DESTRUCT command, and
2. Initiating auto-destruct on a departing stage in the event of a premature or inadvertent stage separation.

The FTS consists of the following components:

1. Two Destruct Batteries, one per channel,
2. Two Destruct Interlocks (DI), one per channel,
3. Eight Destruct Firing Units (DFUs), two per channel, two per stage,
4. Two Command Receiver Decoders (CDRs), one per channel,
5. Two destruct antennas mounted on the equipment section of the vehicle,
6. RF transmission cable and Hybrid Coupler,
7. Four sets of destruct ordnance, including EBW initiators, initiation fittings, energy transfer system, and destruct charges,
8. All associated connectors and interconnecting electrical cables, and
9. All associated attaching hardware.

The Exploding Bridge-Wire (EBW) forms the transition between the electrical and ordnance sections of the destruct system. The initiator is a high energy, electro-explosive unit containing a bridgewire. Destruct initiation starts when a 1200-amp signal of 0.35 micro-seconds rise time is received from the DFU. The heat and shock of the EBW are sufficient to start a high-order detonation of the explosive contained in the detonator. The initiator in turn initiates the Linear Ordnance System and the destruct charges.

The Linear Ordnance System (LOS) provides the capability for initiating high-order detonation of the Linear Shaped Charge (LSC) in the destruct assemblies that are installed on the first , second, and third stage motors as well as on the TLI. The LOS initiation harnesses consist of single EBW initiator adapters, that provide the interface between the EBW initiator and the ETLs that terminate at the LSC.

An ETL consists of a length of flexible confined detonating cord (FCDC), with a 2.7 grain per foot explosive core, terminated at each end with a booster end fitting having an input/output explosive charge of HNS-IA.

5. MASS PROPERTIES CHARACTERISTICS

5.1 PERFORMANCE WEIGHT SUMMARY

5.2 INFLIGHT MASS PROPERTIES

Payload mass at launch:

	X = 0.00 cm	Ixx = 638.61 kg-m2	Ixy = 3.89 kg-m2
Mass 1521.23 kg	Y = 0.00 cm	Iyy = 630.36 kg-m2	Ixz = 0.00 kg-m2
	Z = -39.77 cm	Izz = 217.43 kg-m2	Iyz = 0.00 kg-m2

5.2.1 TLI STAGE

TLI mass at launch:

	X= 0.00 cm	Ixx = 194.31 kg-m2	Ixy = 2.30 kg-m2
Mass 1224.93 kg	Y= 0.00 cm	Iy y= 186.79 kg-m2	Ixz = 0.0 kg-m2
	Z= -65.00 cm	Izz = 152.41 kg-m2	Iyz = 0.0 kg-m2

This weight values includes 1010.6 kg of expendable TP-H-3340 propellant in the TLI Motor, 0.58 kg of igniter propellant, 1.14 kg of expendable Arcite 377A propellant in the Spin Motors, and 3.5 kg of gaseous N2 in the CAS at 3500 psia.

5.2.2 SPACECRAFT

Spacecraft actual mass, including 138 kg (304 lbm.) of MIL-P-26536 high purity grade hydrazine fuel, prior to separation from TLI stage:

	X= -0.01 cm	Ixx = 59.56 kg-m2	Ixy = 1.53 kg-m2
Mass 296.3 kg	Y= 0.04 cm	Iyy = 58.83 kg-m2	Ixz = 0.14 kg-m2
	Z= 60.36 cm	Izz = 65.02 kg-m2	Iyz = -0.27 kg-m2

PRINCIPAL MOIs (kg-m2) =	60.755	57.618	65.042

To provide margin for possible weight growth, a maximum spacecraft weight of 310 kg was assumed for the TLI propellant load calculations. The difference in the possible payload weights (310 - 296.3 = 13.7 kg) is to be carried as ballast mounted on the TLI stage. Since the exact ballast required will be known at the time of payload mating with the TLI stage at Astrotech, access to both the forward and aft ring of the TLI stage is available and should be used for attaching the ballast. The above TLI mass includes this ballast as well as 4.1 kg of ballast at the month-of-the year location.

5.2.3 UNCERTAINTY

The following 3σ uncertainties were used for the above mass properties.

Mass	0.1%

Z cg	5 cm
Y cg & X cg	0.5 cm

Ixx, Iyy, Izz	1%
Ixy, Ixz, Iyz	±0.1 kg-m2

6. TRAJECTORY DATA

6.1 LAUNCH DATE AND WINDOW

6.1.1 NOMINAL

The nominal launch specification for the January 1998 launch window is given in the table below. An on time launch is requested, but a four minute "traffic" window is allowable.

	6 Jan 1998 (GMT)		7 Jan 1998 (GMT)	
Launch Time (GMT)	01:31:12.1		02:28:42.7	
Launch Time (EST)	20:31:12.1	(5 Jan98)	21:31:12.1	(6 Jan98)

6.1.2 LAUNCH SLIP

Alternate launch dates and alternate launch windows were provided by official letter. If the 5 January date was missed, 6 January, followed by 4 and 5 February were alternate launch dates.

6.1.3 TLI STAGE STATE VECTORS AT LP SPACECRAFT SEPARATION

The Athena II launch vehicle was to place the Lunar Prospector – TLI Stage stack into a low Earth parking orbit. The OAM would then spin-up the stack to 3 rpm, orientation the stack for TLI, and release the payload at the precise point in space. The TLI-Lunar Prospector spacecraft post-separation trajectory (epoch date/time, Cartesian state vector and Keplerian orbital elements) is given below.

Date	January 6, 1998
Injection epoch (hh:mm:ss.sss) UTC	02:27:38.846

Earth-centered inertial Earth-true-equator of date

Cartesian position and velocity coordinates

X-position	(km)	-466.88701625
Y-position	(km)	-6197.04599807
Z-position	(km)	-2110.12233839
VX-velocity	(km/s)	9.94523802
VY-velocity	(km/s)	-2.13195652
VZ-velocity	(km/s)	4.00084136

Keplerian orbital elements

semi-major axis	(km)	198523.63341
eccentricity	--	0.9669406804
inclination	(deg)	29.18467
longitude of asc. node	(deg)	303.13082
argument of perigee	(deg)	318.54484
true anomaly	(deg)	0.20515

Note: The state vector above is based upon the following assumptions:

(1) January 6, 1998 launch date with 105 hr time of flight
(2) 97.3 deg nominal launch azimuth
(3) 185.2 km altitude parking orbit
(4) Long coast (i.e. ascending node injection)
(5) LOI north approach with periselene on 1/10/98 1100 GMT
(6) LOI 89.8 deg inclination
(7) 1821.8 km periselene radius-of-closest approach
(8) LP propellant usage 1 v4 model (after 90 m/s midcourse total)
(a) Pre-LOI #1 s/c mass of 284.4 kg
(b) Pre-LOI #1 tank pressure of 365.3 psia

6.1.4 LAUNCH SITE LOCATION

Lunar Prospector launched from Space Launch Complex 46 (SLC-46) located at Cape Canaveral Air Station (CCAS), Florida. The launch site latitude: 28.3°N, longitude: 279.5° E.

6.2 TRAJECTORY EVENT CONSTRAINTS

6.2.1 CONSTRAINTS ON EVENT TIMING

The following constraints were judged to be logical and critical to the success of the mission. Some of the factors that will determine the launch date and window are:

Spacecraft Power Constraints:

- LP battery is sufficient to cover the power requirements during the 47 minute lunar night-time pass. To provide the maximum available battery power for emergency procedures during the post-launch phase, LP should avoid or minimize being in the Earth's or the Moon's shadow.
- The periods where shadows should be avoided are during or immediately after LP's turn-on at S/C-TLI separation, during any part of the 12 hr capture orbit phase, and during any part of the 3.5 hr capture orbit phase of the mission.
- The nominal Solar Equatorial Angle (SEA) at launch should be less than 51° in order to insure that LP has sufficient solar array power to support the critical initialization operations without using the battery.

Spacecraft Thermal Control:

- Thermal design limits dictate that periods total darkness be limited in duration to the maximum lunar night-side pass of 47 minutes.

Attitude Determination:

- The Solar Equatorial Angel (SEA) of the spacecraft can be determined using sun sensor data only for SEA's ± 64°.
- The Solar Roll Angle (SRA) determination using the Earth/Moon limb crossing sensor data is only accurate when the Earth or Moon-Spacecraft-Sun Angle (ES/CS or MS/CS) is 90±60° and when Earth or Moon are >8° in diameter. This method for SRA determination will be ineffective for most of the critical trans-lunar cruise phase of the mission.
- An alternate SEA determination method, using sun sensor data only will be employed during trans-lunar cruise.

6.3 REFERENCE MISSION EVENT TIMELINE

The Lunar Prospector booster mission begins with launch of an Athena II from the Eastern Range (ER), on a launch azimuth of approximately 93 degrees (measured from north). The reference trajectory profile uses a three stage burn to place the Lunar Prospector spacecraft–TLI payload into the specified parking orbit. The profile begins with first and second stage sequential burns, followed by a short coast to allow the free molecular heating rate to decrease to the required levels before jettisoning the payload fairing. After payload fairing separation, the third stage is ignited. A coast phase follows orbital insertion and ESBM burnout. During this coast, the OAM will spin-up, orient, and position for the payload for the TLI burn. The time sequence of major events for the nominal trajectory described above are listed in Table 6.3-3.

Table 6.3-3 Athena II-Lunar Prospector Nominal Event Sequence*

Athena II/TLI Event Sequence for Launch time (UTC hh:mm:ss)

Event	Time (sec)
Stage 1 Ignition	0.0
Max. Dynamic Pressure	75.0
Stage 1 Separation, Stage 2 Ignition	89.0
Stage 2 Burnout	180.8
PLF Separation	232.0
Stage 2 Separation, ESBM Ignition	238.0
ESBM Burnout, OAM Ignition	392.0
OAM Shutdown	844.0
OAM, TLI Stage Spin-up	845.0
LP Separation from OAM (SEP)	3315.0
TLI Spin Rocket Ignition	SEP + 0.5
TLI Star 37FM Motor Ignition	SEP + 20.0
Star 37FM Burnout (3s long)	SEP + 85.6
LP Power On	SEP + 86.6
Collision Avoidance Initiation	SEP + 91.40
LP/TLI Separation (Primary)	SEP + 91.60
LP/TLI Separation (Secondary)	SEP + 91.65
C&DH Startup Sequence Begins	SEP + 92.60

* SOURCE: Athena II Lunar Prospector Mission Phase 2 Launch Countdown Manual, 1A54940, rev. F 27 December 1997.

7. INTERFACES

7.1 LAUNCH VEHICLE INTERFACE

7.1.1 PAYLOAD SEPARATION MECHANISM

The Athena II launch vehicle (LV) provides a 47” Marmon clamp type Payload/Launch Vehicle separation system including the associated pyrotechnic devices, firing signal wiring assemblies and separation springs. This system has redundant firing mechanisms. The separation system mounts on the forward ring of the LV payload adapter and clamps to the aft end of the adapter mounted on the payload.

7.1.2 PAYLOAD ACCESS

Two payload access doors are provided in the payload fairing (PLF). The doors provide access to the spacecraft’s propellant fill/drain valve, TLI stage DFUs, TLI Safe & Arm device, and Command Timer safe and arm plugs. The doors are located on the cylindrical section of the payload fairing at 60° and 240°, respectively, from the Athena II +Y axis between the support ribs at LV stations 2041 and 2065. Door locations are shown in Figure 7.1.2-1.

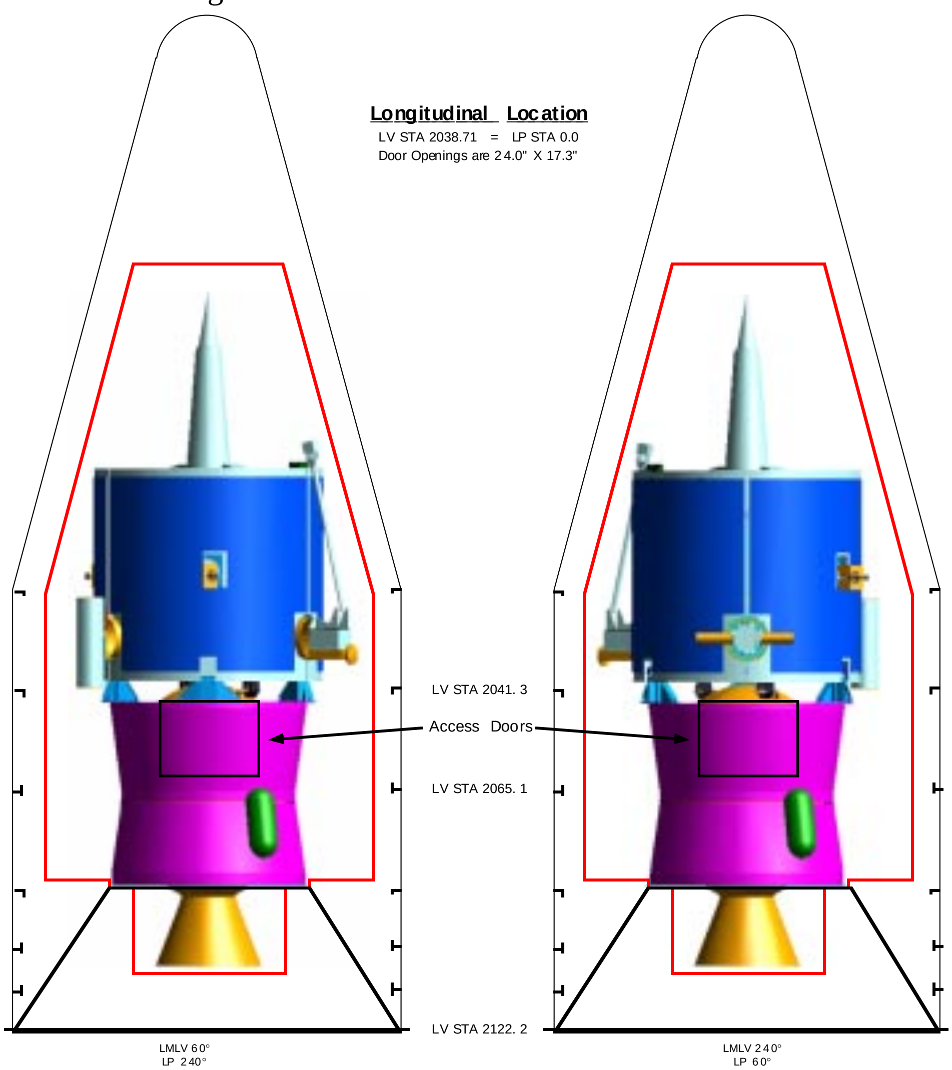


Figure 7.1.2-1 Axial Location of Payload Access Doors

7.2 SCIENCE INSTRUMENTS INTERFACES

This section describes the electrical, mechanical, and thermal interfaces between LP SIs and the spacecraft. In general, Lunar Prospector simplified the instrument interfaces by selecting omni directional, boom mounted instruments. The instruments did not have driving requirements for field of view, alignment, or duty cycle. The spacecraft provided the required power, command and data handling capabilities, boom mounting, spin rate, and attitude stability with sufficient margins. The appropriate constraints for loads and contamination control during ground handling, transportation, and installation easily satisfied. Special accommodation issues are discussed below.

7.2.1 PHYSICAL REQUIREMENTS

7.2.1.1 Mounting Points and Constraints

The SI sensors are mounted on boom tip plates. The spectrometer electronics was mounted on the spacecraft Equipment Panel.

7.2.1.1.1 Available Payload Envelope

Payloads mounted on the deployable masts are a maximum of 29.21 cm (11.50”) in length, 55.40 cm (21.81”) in height, and 65.84 cm (25.92”) in width.

7.2.1.1.2 Mass and Center of Mass Properties

The mass and center of mass of each SI was measured by the SI provided and accurate to within 0.1%. SI mass centers were measured to within ±0.25 cm (±0.1 in.) prior to installation to the spacecraft. Ballast was added to two SI tip plates in order to balance those SIs with the heaviest SI. Tip plate end mass did not exceed 7 kg (15 lbm.).

7.2.1.1.3 SIs C.G. Locations

Each IS C.G. was made coincident with the center line of the boom. The location of the SI CG did not exceed 12.7 cm (5 in.) outboard from the corresponding mast tip plate plane.

7.2.1.1.4 SI to SC Mounting Interface

All SIs were mounted using bolts and 1.27 cm (0.5”) standoffs are provided and installed by the SC contractor. Standoffs are used between the SIs and the masts’ tip plates. All mounting surfaces and devices must use non-magnetic materials near instruments. The SI mounting bracket designs allows for installation and removal of the SIs while the booms are in the stowed configuration.

7.2.1.2 Ground Handling Interface

All SIs were designed for installation or removal during ground operations while the SC was in the vertical position (+Z pointing up) and with the booms in the deployed or in the stowed position. SC design provide adequate access to allow SI installation or removal without degradation or damage to the flight hardware.

7.2.2 ELECTRICAL POWER AND SIGNALS/DATA INTERFACE

7.2.2.1 Power Buses

The SI electronic boxes were provided with a switched connection to the spacecraft’s 28 volt power bus. SI primary power returns were isolated from the SC structure by a minimum electrical DC resistance of 1 MΩ and are returned through the SI interface connector to the SC negative return bus.

7.2.2.2 Power Consumption

The maximum allowable electrical power consumption for all SIs is 17.0 watts.

7.2.2.3 Voltage

7.2.2.3.1 Operating Voltage

Electrical power is provided to each SI at a voltage of 28.0 Vdc ± 6.0 Vdc at the Instrument input connectors. The SIs are not required to operate within performance specification above or below such limits. Each SI is capable of surviving, without permanent degradation, the application of a steady-state voltage of a maximum of 40.0 Vdc and a minimum of 18.0 Vdc.

7.2.2.3.2 SI High-Voltage

SIs which utilized high-voltage power supplies, contained current limiting. If an SI's high-voltage supply could not be operated at atmospheric pressure, it was capable of being disabled by manual means which otherwise allowed ambient testing.

7.2.2.4 Over-Current Protection

All over-current protection devices which are internal to the SI are accessible at the SC integration level without requiring the disassembly of the SI.

7.2.2.5 Power and Signal Routing

7.2.2.5.1 Primary Power Routing

The delivery of primary electrical power to the SI is through twisted conductor cables (pair, quad, etc.) with power and return leads enclosed in the same cable with an electrical shield. Each SI primary power lead has a distinct, isolated return.

7.2.2.5.2 Primary Power Shielding

Each SI power connector provides two pins terminated to chassis as close as possible to the inside of the SI connector shell.

7.2.2.5.3 Power and Signal Isolation

The 28 volt primary power leads and returns are isolated from signal return by a DC resistance of greater than 1 Mega-Ohm when measured at the Instrument input.

7.2.2.5.4 Single Point Ground

The reference point for primary power voltage control is a Single Point Ground which is located in the spacecraft.

7.2.2.5.5 Secondary Power Routing

Current return leads for secondary power circuits are isolated from the primary power return by a DC resistance of greater than 1 Mega-Ohm.

7.2.2.5.6 Signal Routing

SI return leads are provided to the SC for each signal or group of signals generated by a SI component.

7.2.2.6 Grounding

7.2.2.6.1 SI Grounding

Each SI is mounted on thermal isolation stand-offs from its associated mast tip plate. These stand-offs isolate the SI electrically from the rest of the spacecraft. The spacecraft cabling on each mast will provide to each SI location a wire which is connected inside the spacecraft to the spacecraft structure in the neighborhood of Single Point Ground. The SI housing, the mast tip plate on which it is mounted, or the thermal blankets which are associated with each SI may be connected to this structure wire for grounding purposes. No power will be returned through this wire.

Each SI provides an electrically conductive structural enclosure with at least one fastener location on its housing which is suitable for the attachment of grounding terminals. The terminals will be in the form of crimp lugs on straps from thermal blankets, from the mast cable structure wire, or from the mast cable bundle braid.

7.2.2.6.2 Thermal Blanket Grounding

Each SI thermal insulation blanket is designed such that its metallized and conductive layers are electrically interconnected with a DC resistance between layers not to exceed 10 Ohms. Each thermal blanket is equipped with a minimum of two grounding wires such that the DC resistance from the blanket to its grounding point does not exceed 10 Ohms. The thermal blanket for each SI is connected locally to the SC chassis ground.

7.2.2.7 Electro-Magnetic Compatibility (EMC)

Lockheed's standard EMC design practices result in system level margins of >6 dB. Our use of a 28 v switched array battery regulated electrical bus has demonstrated EMI suppression on many programs. Our standard design techniques include the use of twisted pair wiring, selection of harness size to accommodate bus impedance criteria, and single point ground returns to minimize structure currents. Our selection of filter networks to reduce conducted emissions to <130 dBμA (tapering to 20 dBμA at 50 MHz) minimizes bus noise. Conducted susceptibility levels of 1.5 vRMS ensures the integrated spacecraft meets all design margins. Control of high speed digital rise times reduces component cross coupling and increases margin. Inrush and turn on transients are limited to less than 2 x steady state current (<5 msec) and <30 v excursions on the system bus. Radiated emissions and susceptibility levels are tailored to 20 volts (susceptibility) and <20 dBμV/meter (emissions) at S-Band. Magnetic fields are controlled and boom length designed to control magnetic fields to <0.1 nT at the interface. Control of electrostatic discharge is accomplished through the use of conductive coatings such as carbon Kapton and Indium Tin Oxide and spot shielding at sensitive interfaces. Lockheed's use of a graphite epoxy composite spacecraft enclosure with imbedded ASTROSTRIKE™ microgrid expanded metal foil compensate for the lower conductivity levels of the composites. This results in a homogeneous ground plane that minimizes noise generated from structure currents and mitigates electrostatic charge buildup. Test data shows the composite ASTROSTRIKE™ shielding effectiveness to meet program requirements.

7.2.2.8 Avionics Interfaces

The following paragraphs define the characteristics of the various interface signal types between the SC and the SIs.

7.2.2.8.1 28 V dc Latch

The SI electronics receives operating and heater power and selected pulse commands from the C&DH

electronics in response to ground commands. The SI returns current drawn from a particular 28 VDC latch/driver to the return. Figure 7.2.2-2 is the C&DH electronics 28 VDC distribution schematic and Figure 7.2.2-3 indicates the characteristics of the 28V pulse.

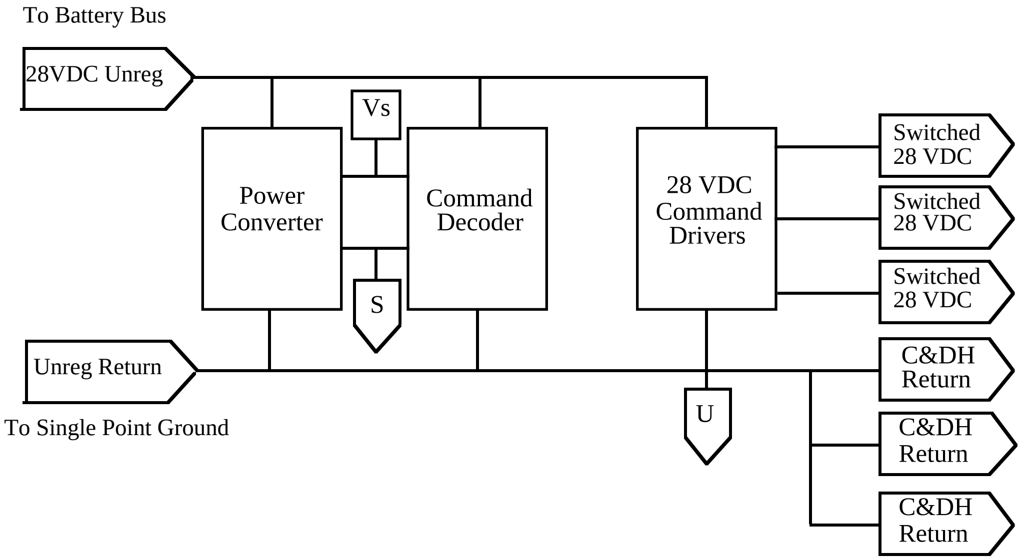


Figure 7.2.2-2 C&DH 28 V dc Command Distribution Schematic

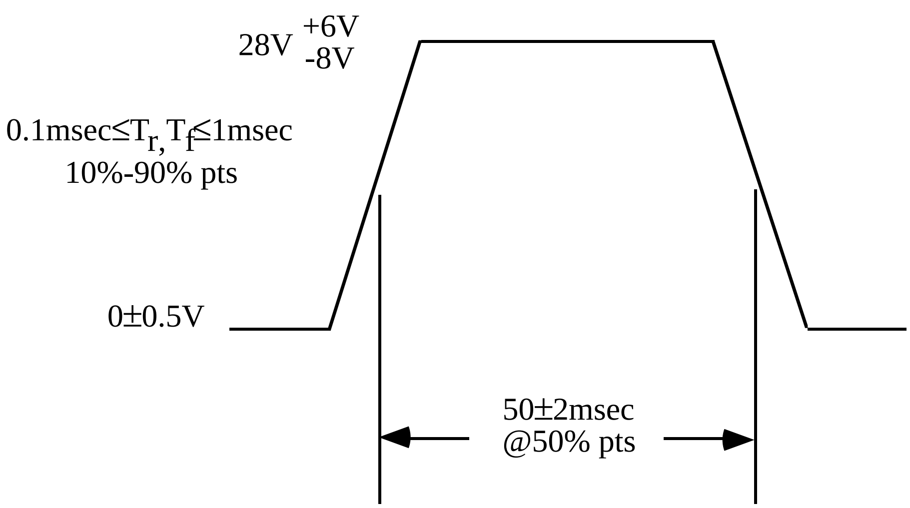


Figure 7.2.2-3 28V Pulse Characteristics

7.2.2.8.2 5 Vdc Digital

The C&DH Electronics supplies synchronization strobes, timing clocks, and digital command data to the SI electronics. The SI electronics provides serial telemetry data to the C&DH Electronics. All digital interfaces use a common signal ground provided by the C&DH Electronics. Figure 7.2.2-4 is a 5 Vdc interface signal schematic.

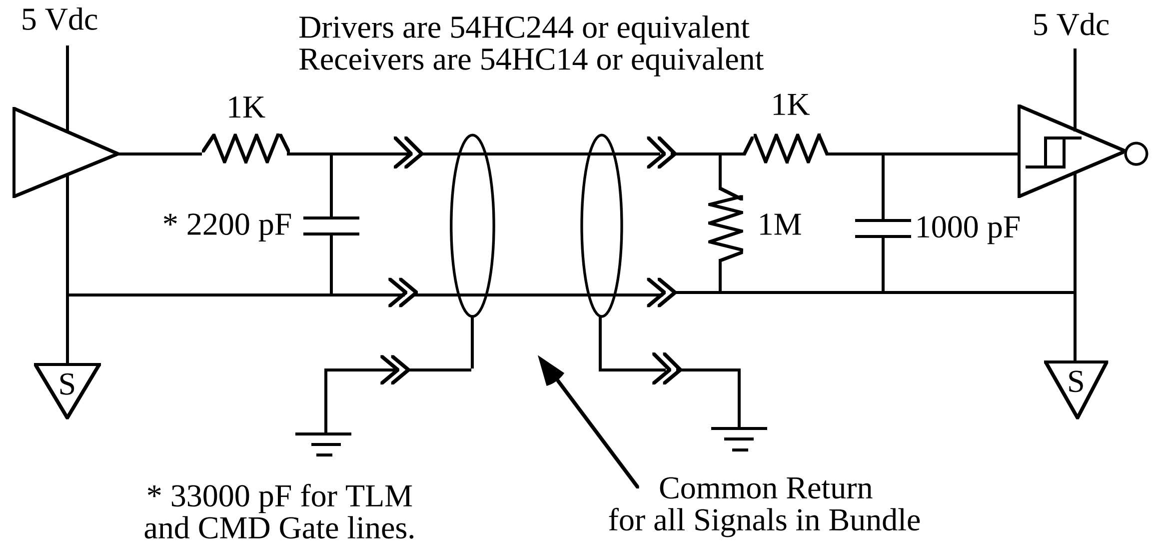


Figure 7.2.2-4   5 Vdc Interface Signal Schematic

7.2.2.8.2.1 System Clock

The C&DH Electronics provides a continuous clock at the telemetry downlink acquisition frequency (1.8 kHz nominal - Figure 7.2.2-4, Figure 7.2.2-5).

7.2.2.8.2.2 Telemetry Main Frame Strobe

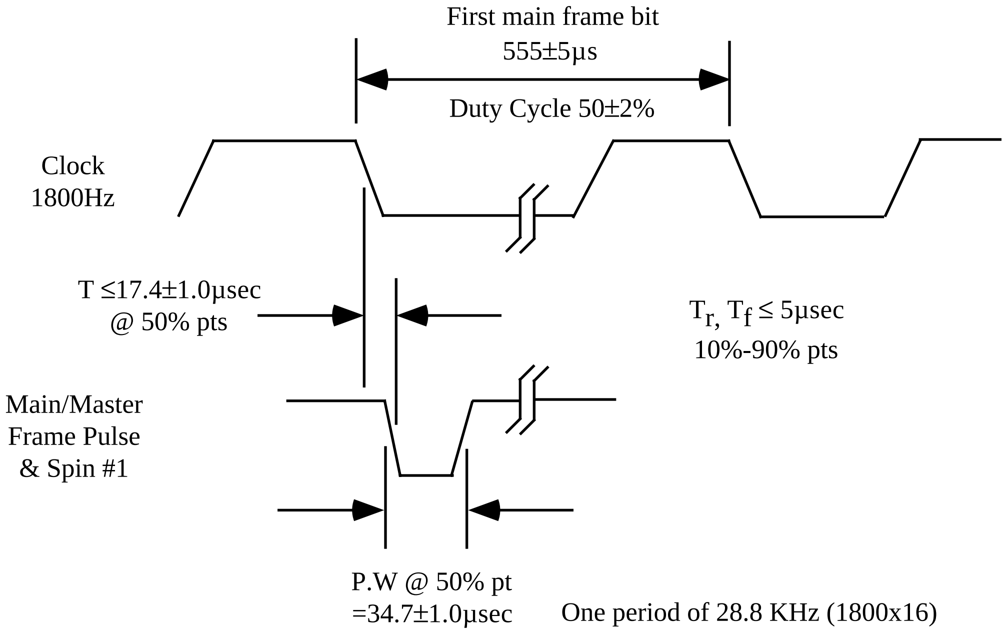
The C&DH Electronics provides a strobe at the start of each real-time telemetry main frame. This mainframe strobe occurs every 2 seconds (7200 clocks - telemetry is at 3600 bps.) See Figure 7.2.2-5.

7.2.2.8.2.3 Telemetry Master Frame Strobe

The C&DH Unit provides a strobe at the start of each real-time telemetry Master frame every 16 mainframes. This strobe is synchronized with a main frame strobe and occurs every 32 seconds. See Figure 7.2.2-5

7.2.2.8.2.4 Spin #1 Synchronization Strobe

The C&DH Unit provides a strobe indicating sun crossing detection from sun crossing detector. The SIs provide the capability to operate normally in the event that the strobe is absent for any period of time. See Figure 7.2.2-5.



Note: For Spin #1, only pulse width applies.
It does not have any correlation to the 1800Hz.

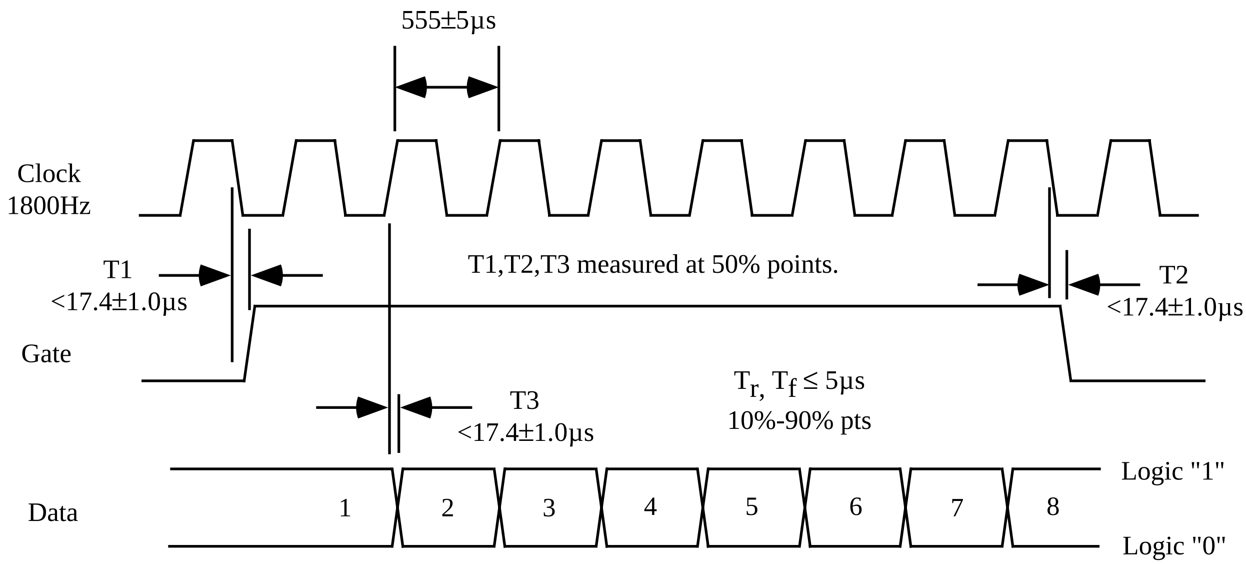
Figure 7.2.2-5 Frame Strobe and Spin Timing

7.2.2.8.2.5 *Serial Commands*

Serial commands are provided from the C&DH electronics in response to uplink commands and delivered as 24 bit words via the serial command port to the addressed SI as defined by the serial port protocol defined next.

7.2.2.8.2.6 *Serial Port Protocol*

The C&DH unit provides Command Gate, Command Data and Telemetry Gate signal lines and accepts a Telemetry Data line from each SI. Data transfers are accomplished during periods where the associated Gate signal is in the enabled state as shown in Figure 7.2.2-6 using the System Clock signal defined in 7.2.2.8.2.1.



Shows 8 bit byte transfer; actual length is application dependent.
Bit Numbering within transfer byte is as shown (continues for longer words).
Data is presented MSB first (where applicable) by convention.
Incompatibility between Gate Duration and Data size is an error condition.
Next data word transfer will normally commence at next Gate enable
All times are measured at the connector.

Figure 7.2.2-6 Serial Port Protocol

7.2.2.8.3 Analog Telemetry

The SIs provide passive analog and 0-5.12 Vdc active signals referenced to instrumentation return. Figure 7.2.2-7 is a sketch of the typical analog telemetry interface. The C&DH unit samples, optionally conditions passive analog signals and reports, via SC telemetry, the value of the resulting analog voltages to an accuracy of 40 mV with a quantization of 20 mV over the range of 0-5.12 V dc. The SI provides analog signal calibration data. It is possible to reconstruct on the ground the time when the analog sample was taken to the nearest msec relative to the Main Frame Strobe. The resistance range for the Passive Analog is 300 Ω- 280 KΩ.

7.2.2.8.4 Science Telemetry

The SI provides its science data as a single block of data for each mainframe, 3200 bits, as shown in Figure 7.2.2-6, to the C&DH unit, according to the serial port protocol defined in 7.2.2.8.2.6. The SIs are capable of delivering their data no later than 284.4±0.2 milliseconds following a main frame strobe. The data provided during one main frame is downlinked in the following mainframe.

7.2.2.9 Software Applications.

The current LP configuration does not require any science instrument flight software. All SI ground software is provided by the SI contractor.

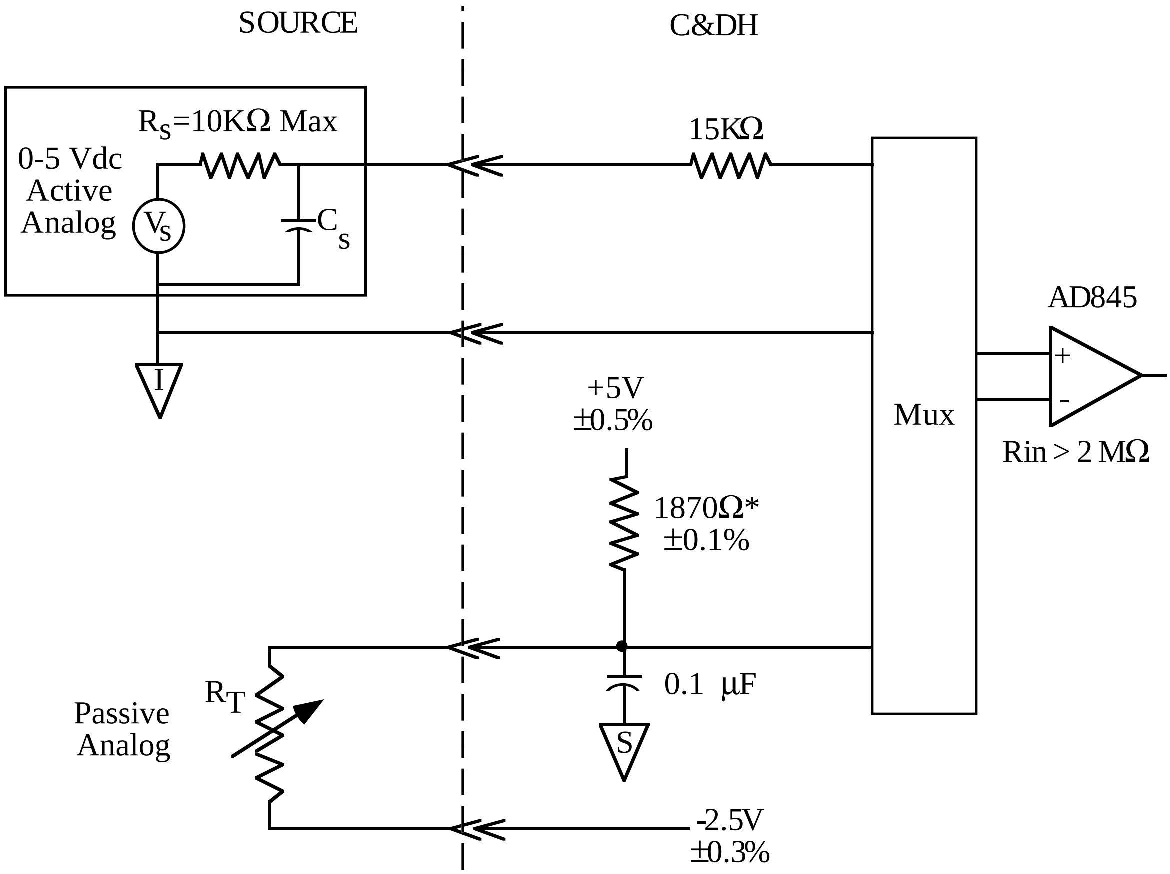


Figure 7.2.2-7 Typical Analog Telemetry Interface

7.2.3 PAYLOAD ENVIRONMENTAL REQUIREMENTS

The environmental requirements stated below are specific to the Lunar Prospector Mission.

7.2.3.1 Thermal Interfaces

SI component housings can be maintained at the operate temperature limits between -31 °C and 45 °C and survival temperature limits between -55 °C and 50 °C. Operate limits apply when components are powered; survival limits apply when components are not powered. SI contractors are responsible for verifying that all SI internal electronic piece part temperature requirements are satisfied for the housing temperatures stated above. The SC contractor defines the SI thermal control required to maintain temperatures specified above.

7.2.3.1.1 Heat Transfer

Each boom mounted SI is designed such that 100% of its internally generated heat is transferred by radiation from the housing. Bus mounted equipment is designed such that at least 80% of its internally generated heat is transferred by radiation from the housing.

7.2.3.1.2 Pre-Flight

The pre-flight environments include transportation, storage, pre-launch and ground testing.

7.2.3.1.3 Transportation and Storage

During storage and transportation the environment surrounding the vehicle is controlled to the following:

a. Cleanliness: Fed Std 209, Class 300,000 or better.
b. Temperature: 21°C ± 8°C with short excursions (less than 12 hours) to 21°C ± 17°C.
c. Relative Humidity: 20% to 60% with no condensation permitted.

7.2.3.1.4 Pre-launch and Ground Testing

During prelaunch operations and ground testing, the environment surrounding the vehicle is controlled to the following:

a. Cleanliness: Fed Std 209, Class 100,000 or better
b. Temperature: 21°C ± 8°C with short excursions (less than 12 hours) to 21°C ± 17°C.
c. Relative Humidity: 35% to 60% with no condensation permitted.

7.2.3.2 Magnetic Environment

The maximum magnetic field caused by the SIs and the SC in any orientation within the SI envelope does not exceed the levels specified below:

a. 0.05 nT at less than 16 Hz for time-varying field
b. 0.5 nT constant magnetic field

7.2.4 IONIZING PARTICLE RADIATION

The radioactive level of any of the SIs and/or of the SC is such that the photon counts at the SI are less than 0.03 photons/cm^2/s.

7.2.4.1 Humidity

During all ground, transportation, storage, and pre-launch operations the relative humidity is 35 to 60 percent without condensation.

7.2.4.2 Pressure Environment

The pressure environment for the SIs during launch varies in accordance with the internal pressure characteristics of the launch vehicle payload fairing.

7.2.5 CONTAMINATION CONTROL

7.2.5.1 Surface Cleanliness

The surfaces of the SC and SIs are visibly clean (VC) to the levels shown below. SIs remain covered with protective packaging to the extent possible.

Item	VC Level	Incident Light Level	Observation Dist.	Notes
S/C	Sensitive	> 50 foot-candles	2 to 4 ft.	1,2,3,5
SIs	Highly Sensitive	> 100 foot-candles	6 to 18 in.	1,3,4

Notes:

1. One foot candle (lumens per square foot) is equivalent to 10.76 lumens per square meter
2. Cleaning is required if the surface in question does not meet VC under the specified incident light and observation distance criteria
3. Exposed and accessible surfaces only
4. Initial cleaning is mandatory
5. Areas of suspected contamination may be examined at distances closer than specified for final verification

7.2.5.2 Cleaning Solvents

Only isopropyl alcohol is used for cleaning. After cleaning, the SIs and the SC can only be handled with gloves. SIs will be cleaned only by the SI contractors.

7.2.5.3 Material Outgassing

Outgassing from the SIs and the SC is minimized.

7.2.5.4 Helium Exposure

The SIs can be bagged and purged with dry N_2 whenever He is used in or close to the spacecraft and/or GRS.

7.2.5.5 Purging

A Nitrogen gas purge system was provided continuously while the SIs are at the SC assembly facility. The purge is terminated before encapsulation of the spacecraft inside the booster payload fairing. The purge fitting is part of the SI assembly. The quality of the purge gas is as follows:

GAS	PURITY
Top grade bottled N_2 with the following allowable impurities	99.9998±0.0001%
Water (PPM)	< 3
Hydrocarbons (PPM)	< 1

7.2.5.6 Metals to Avoid

Because the Gamma Ray Spectrometer experiment was designed to detect specific elements in the lunar crust, the spacecraft design sought to eliminate or minimize the use of the following metals:

1. Iron (Fe)
2. Titanium (Ti)
3. Aluminum (Al)
4. Magnesium (Mg)
5. Potassium (K)

7.2.6 SAFETY

The LP Safety Lead ensures that all operations on the payload and spacecraft meet safety standards.