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Lunar Scout

A Project Artemis Proposal

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1.0

INTRODUCTION

1.0 INTRODUCTION

Project Artemis, named after the Greek goddess of the moon, is NASA's current program to put a lander on the lunar surface in the year 1996. The lander will carry payloads such as a lunar telescope, a possible robotic lunar rover or assorted experiments targeted to various lunar locations. The payloads will be used to demonstrate mining equipment to process Hydrogen, Nitrogen and Helium from the moon's soil and to ultimately determine the feasibility of developing a lunar outpost for future manned missions. Thus the lander Lunar Scout which will provide low cost transportation of 200 kg. payloads to survey the lunar surface and provide information as to the feasibility of developing a lunar outpost.

LUNAR SCOUT REQUIREMENTS

- Use off the shelf hardware for low cost
- Provide 2 year spacecraft lifetime
- Systems shutdown during lunar night
- Launch date in 1996
- Soft landing on lunar surface between +/- 60 degrees latitude
- Deliver 200 kg. payload to lunar surface
- Provide 10 W power during two week lunar night to heat spacecraft
- Use Delta II launch vehicle

LUNAR SCOUT

(A PROJECT ARTEMIS PROPOSAL)

DESIGN TEAM:

LEADER - 1/C MATTHEW Mc LAURIN
Structure of the Lunar Lander

DESIGNER - 1/C MIKE LANE
Propulsion, Attitude Control, and Thermal Control

DESIGNER - 1/C JEFF RICKENBACH
Orbital Dynamics

DESIGNER - 1/C JOHN LIN
Power Systems

DESIGNER - 1/C ROBERT DUONG
Communications

2.0

MISSION SUMMARY

2.0 MISSION SUMMARY

The lunar scout will be launched from Cape Canaveral, Florida onboard a Delta II launch vehicle. The Delta II will carry the lander and its payload to a 1367 km orbit. Once it reaches that altitude, a STAR 48A solid rocket motor will kick the spacecraft into a lunar trajectory. After burnout of the lunar insertion motor, it will be jettisoned from the spacecraft. The flight from the earth to the moon will take approximately 106.4 hours. During this time the battery, which was fully charged prior to launch, will provide all power to the spacecraft. Every hour, the spacecraft will use its sun sensors and star trackers to update its position, maintain some stabilization and relay it back to earth using the dipole antennas. At the start of its lunar trajectory, the spacecraft will fire one of its 1.5 N thrusters to spin it at a very small rate. The main reason for this is to prevent one side of the spacecraft from overheating in the sun. When the spacecraft nears the moon, it will orient itself for the main retro burn. At an altitude of 200 km, a 4400 N bipropellant liquid thruster will ignite to slow the spacecraft. During the burn, the radar altimeter will be turned on to guide the spacecraft. The main retro rocket will slow the lander to 10 m/s at an approximate altitude of 40 km above the moon. From there, the space craft will use four 4.5 N hydrazine vertical thrusters and 1.5 N horizontal thrusters to guide the spacecraft to a soft landing. Once on the ground, the lander will shutoff the radar and attitude control systems. After the debris from the impact has settled, the six solar panels will be deployed to begin recharging the batteries and to power up the payload. The feedhorn antenna will then rotate to fix itself on the earth. Once it moves, it will stay in that position for the spacecraft's lifetime. The payload will then be activated to begin the lunar mission.

3.0

LAUNCH VEHICLE

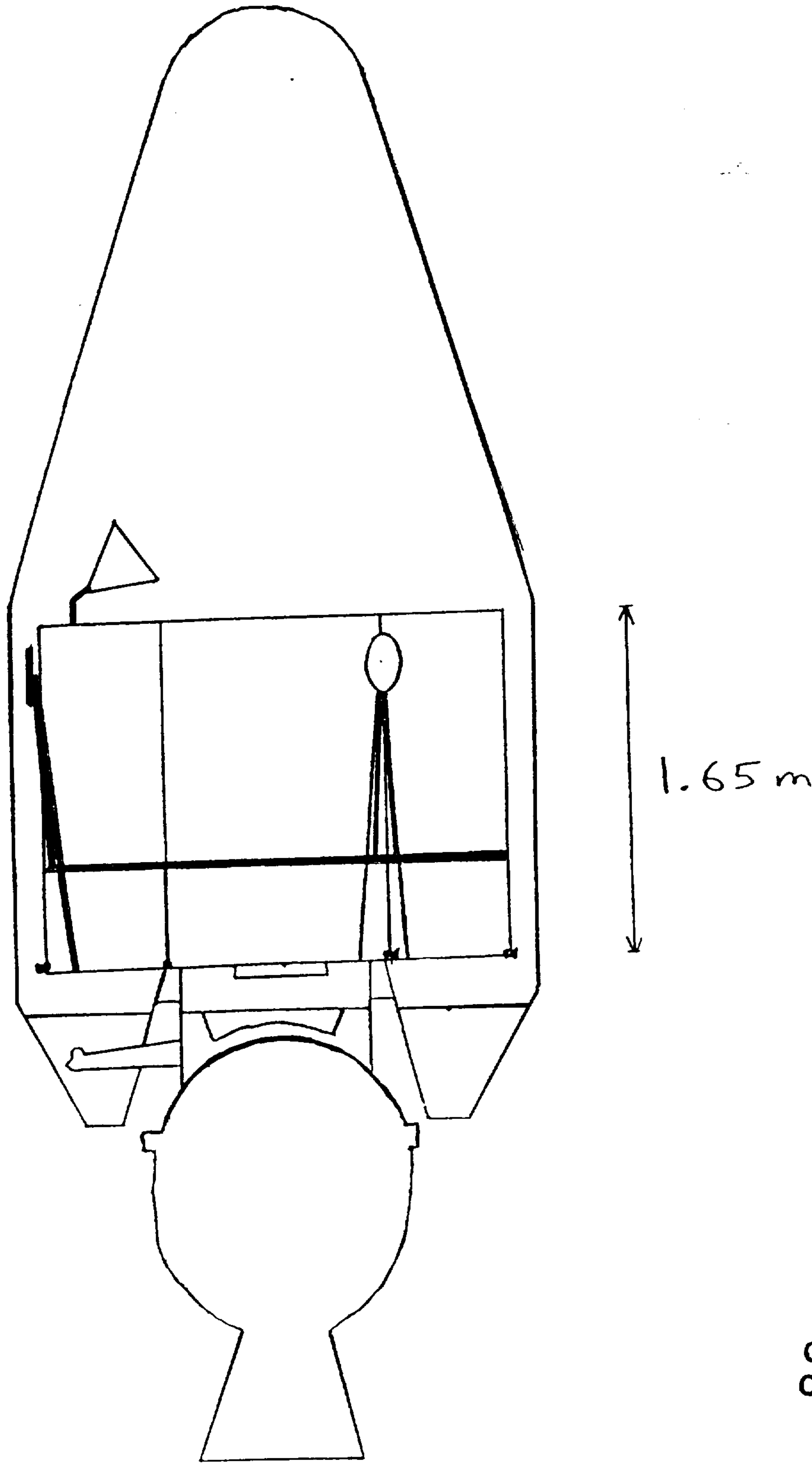
3.0 LAUNCH VEHICLE

The launch vehicle proposed for the Lunar Scout is the Delta 7920. The Delta 7920 will carry the Lunar Scout and the Thiokol Star 48A third stage, with a combined mass of 3775 kg., to a circular orbit with an altitude of 1366.7 km. The proposed payload fairing is the 9.5 ft. Star 48 configuration. The narrower fairing was selected so as to minimize extra space around the spacecraft. A view of the Lunar Scout within the payload fairing is shown in Figure 3.1.

The first stage of the Delta 7920 will burn for 265 seconds. At 278 seconds into flight, the second stage will ignite and carry the spacecraft through a Hohmann transfer to an altitude of 1366.7 km and cut off 620 seconds into the flight. At 302 seconds into the flight, the payload fairing will separate. After second engine cut off there will be a 10 second delay followed by Star 48A ignition to provide injection into a lunar trajectory.

FIGURE 3.1

LUNAR SCOUT



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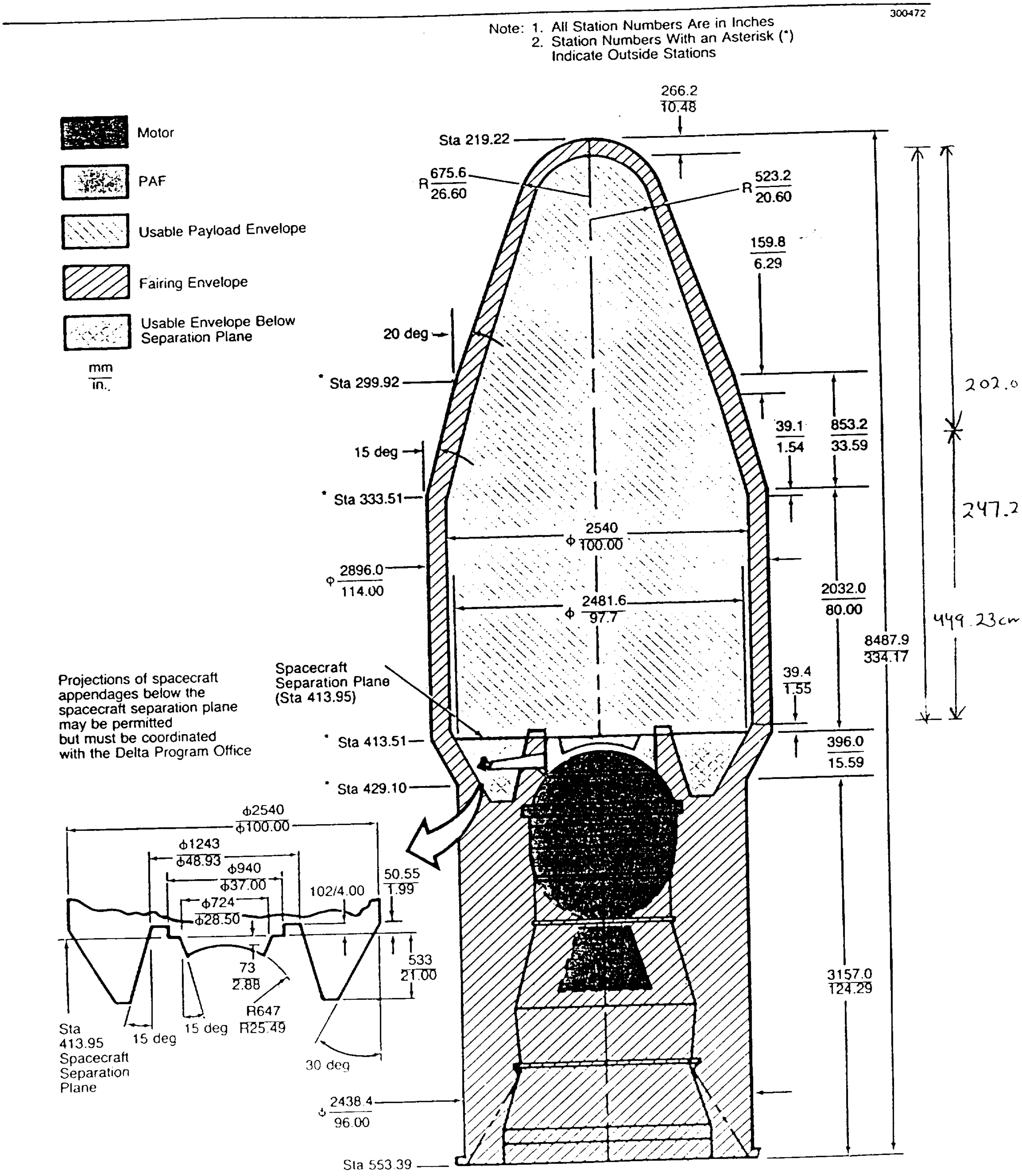
VIEW OF LANDER IN PAYLOAD FAIRING

FIGURE 3.2



Spacecraft Envelope, Star 48 Configuration (3712 PAF)

FIGURE 3.3

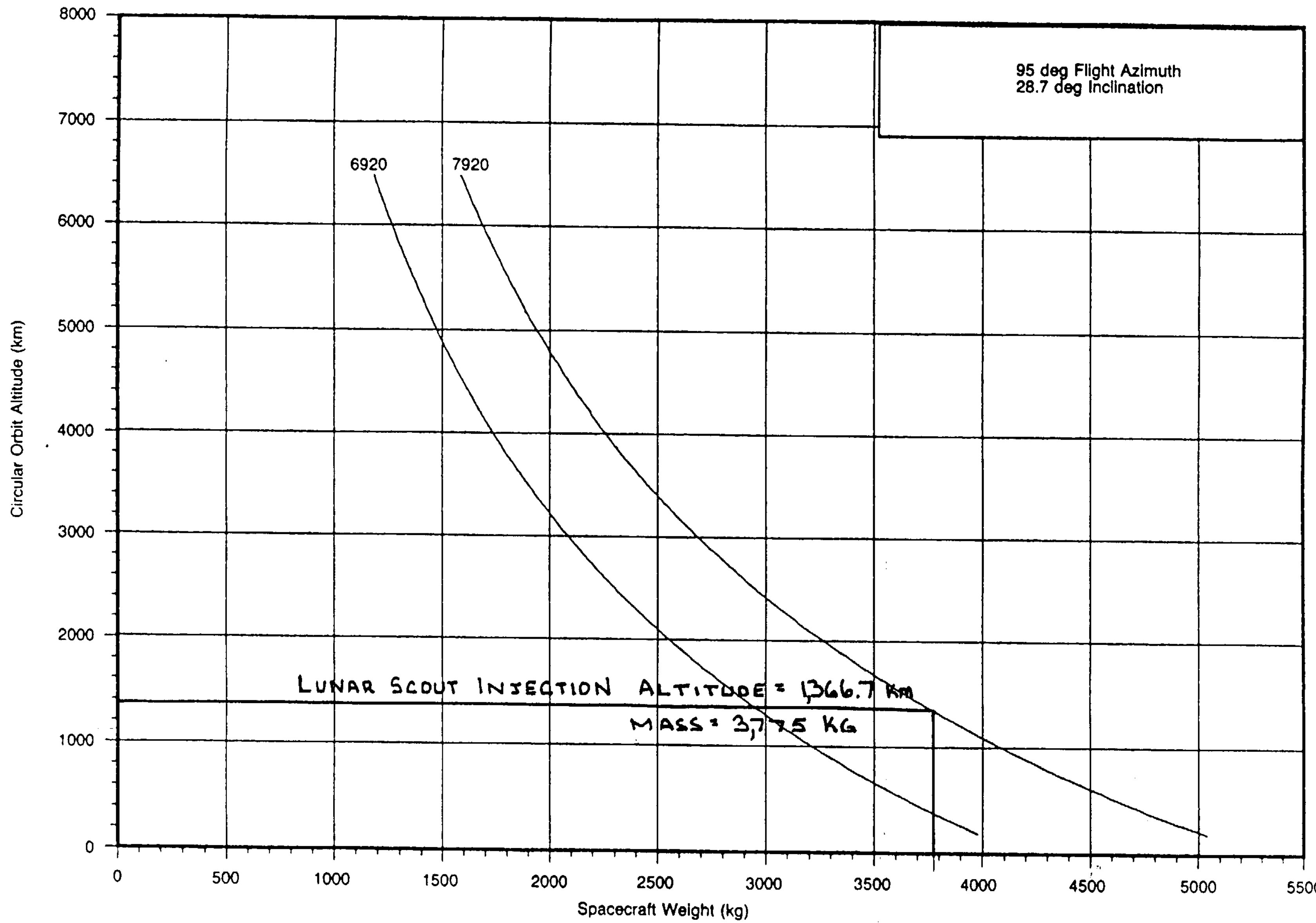
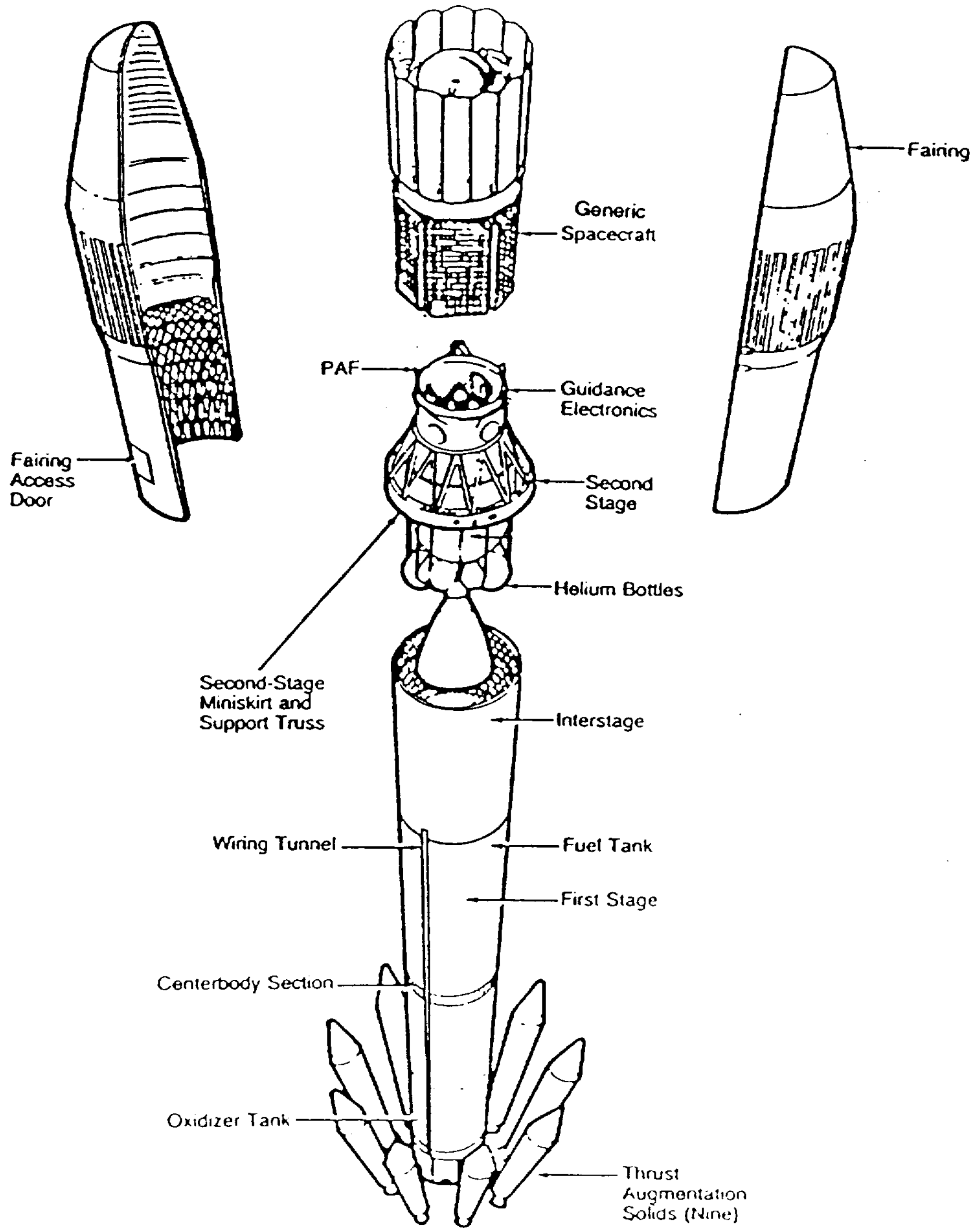


Figure 2-8. Two-Stage Circular Orbit Capability (Metric Units) – ESMC

12

FIGURE 3.4



DELTA II TWO STAGE SEPARATION

4.0

ORBITAL ANALYSIS

4.0 ORBITAL ANALYSIS

4.1 BACKGROUND

The trajectories involved in the Earth-Moon system are some of the most complex in our solar system. This is for two primary reasons. First, nearly all other problems of motion can be described relatively accurately by a 2-body problem, including both near-Earth satellites and interplanetary probes. However, due to the relative sizes of the Earth and Moon, two centers of attraction are present throughout nearly the entire trajectory. Secondly, the motion of the system itself is very complicated. To begin with, the Moon does not revolve around the Earth, but both bodies revolve around their common center of mass, which is about 3/4 of the way from the center of the Earth to its surface. There are numerous other complexities such as the following: the eccentricity of the moon's orbit experiences small periodic changes at intervals of 31.8 days; the line of nodes of the Moon's orbit rotates westward, making one complete revolution every 18.6 years; the inclination of the moons orbital plane relative to the Earth's equator varies between 18.3 and 28.5 with a period of 18.6 years; and the line of apsides of the lunar orbit rotates in the direction of the Moon's orbital motion, completing one 360 rotation over 8.9 years.

Taking all of these factors into account, the computation of a precision lunar trajectory can only be done by numerical integration of the equations of motion, which requires a high-speed digital computer and many hours of computation time.

Because the complexities in these computations are well beyond the bounds of undergraduate design constraints, simplifying assumptions must be made.

In the simplest case, the mass of the moon is assumed to be zero. For this approximation, the lunar trajectory is described by an unperturbed two-body conic section relative to the Earth. The inclination of the moon's orbital plane relative to Earth's equator and nonequatorial launch sites can also be accounted for in this model. This case can then be used to derive a reasonably accurate set of initial conditions required for the Earth-Moon trajectory. The problems with these simplifications is that the transit time will be overestimated and the error coefficients will be conservative. This can be overcome, however, by implementing a moon-centered conic section. This is accomplished by patching the conic at infinity relative to the moon or at the radius of the Moon's sphere of gravitational influence. This is the highest level of sophistication that can be obtained without resorting to numeric integration.

4.2 SUMMARY

The orbital dynamics calculations for the Lunar Scout are sub-divided into three sections: the simple two-body problem, the third dimension added to the two-body problem, and the patched conic. Because all of the calculations were interdependent, they were done on a spreadsheet to make the manipulation of the numbers faster and easier. This spreadsheet and a detailed explanation of its calculations, including diagrams, is contained in the appendices.

Launch for the Lunar Scout is planned for 1996 from the Eastern Test Range on a Delta II primary launch platform. The Lunar Scout will use a direct trajectory to the Moon without the use of parking orbits around either the Earth or Moon. This type of trajectory is utilized in order to minimize the flight time, minimize command and telemetry communications, and minimize the number of burns required. Insertion to the lunar trajectory will occur at an altitude of approximately 1366.7 kilometers. This insertion altitude is based on the capabilities of the Delta II for the given mass of the spacecraft. An insertion delta V of about 2.87 km/sec will be provided by a Star 48 solid rocket motor.

In the third dimension, launch will be due east at an azimuth of 90 . In 1996, the Moon will be near its maximum declination of 28.5 . The transit angle from the launch point to landing on the lunar surface is estimated at 267.6 . Given these angles, the Lunar Scout will reach the Moon when it is about 1.1 below the plane of the Earth's equator. Using spherical trigonometry, estimations for the site of landing on the Moon will be 10.5 north of the lunar equator and 79.3 west of the lunar nadir to the Earth. The time of transit will be approximately 106.4 hours.

Lastly, a patch conic approximation was applied as the lander approached the Moon. At a radius of 362,284 kilometers, the Lunar Scout entered the Moon's sphere of influence which has a radius of 66,300 kilometers. At this patch point, the spacecraft has a velocity relative to the Moon of 0.81 km/sec. Using this velocity and the angle at which the Scout enters the lunar sphere of influence, the parameters of the orbit of the lander relative to the Moon can be determined.

At an altitude of 200 kilometers above the lunar surface and at a velocity of nearly 2.36 km/sec, the main retrograde burn is initiated. After a burn of 134.9 seconds, the Lunar Scout begins a slow, controlled descent to the surface.

5.0

SPACECRAFT SUBSYSTEMS

5.1 STRUCTURE

In designing the structure of Lunar Scout, the Surveyor, Viking and Soviet Luna landers were researched to provide insight into existing materials and hardware that could be used on the lander. Since low cost and use of off the shelf hardware was one of the requirements, an attempt was made to incorporate some similar materials from these previous projects to reduce the cost of development and testing. It was desired to find a material with a high strength to weight ratio to withstand the high stresses inherent in landing on the moon while providing a lightweight structure. The dimensions of the lander were designed so that the spacecraft would fit within the payload fairing of the Delta II 7920 launch vehicle chosen for use.

The first aspect addressed was the shape of the lander. A hexagonal shape was chosen to make use of the maximum possible area within the circular payload fairing cross section. This shape was also well suited for symmetric attachment of the three lander legs selected for use. Figure 5.1.1 shows the hexagonal shape and the position of the lander leg attachments. Three legs were used rather than four because the stability provided would be adequate and an extra leg would only serve to increase the mass of the spacecraft. Each side of the hexagon will be 1.15 m. in length to provide an internal base area of 4.60 square meters. The height of the basic structure in the fairing will be 1.60 m.

Each lander leg will consist of two solid members connected at the bottom edge of the spacecraft and two hydraulic shocks

TOP VIEW OF LANDER WITH PANELS UP

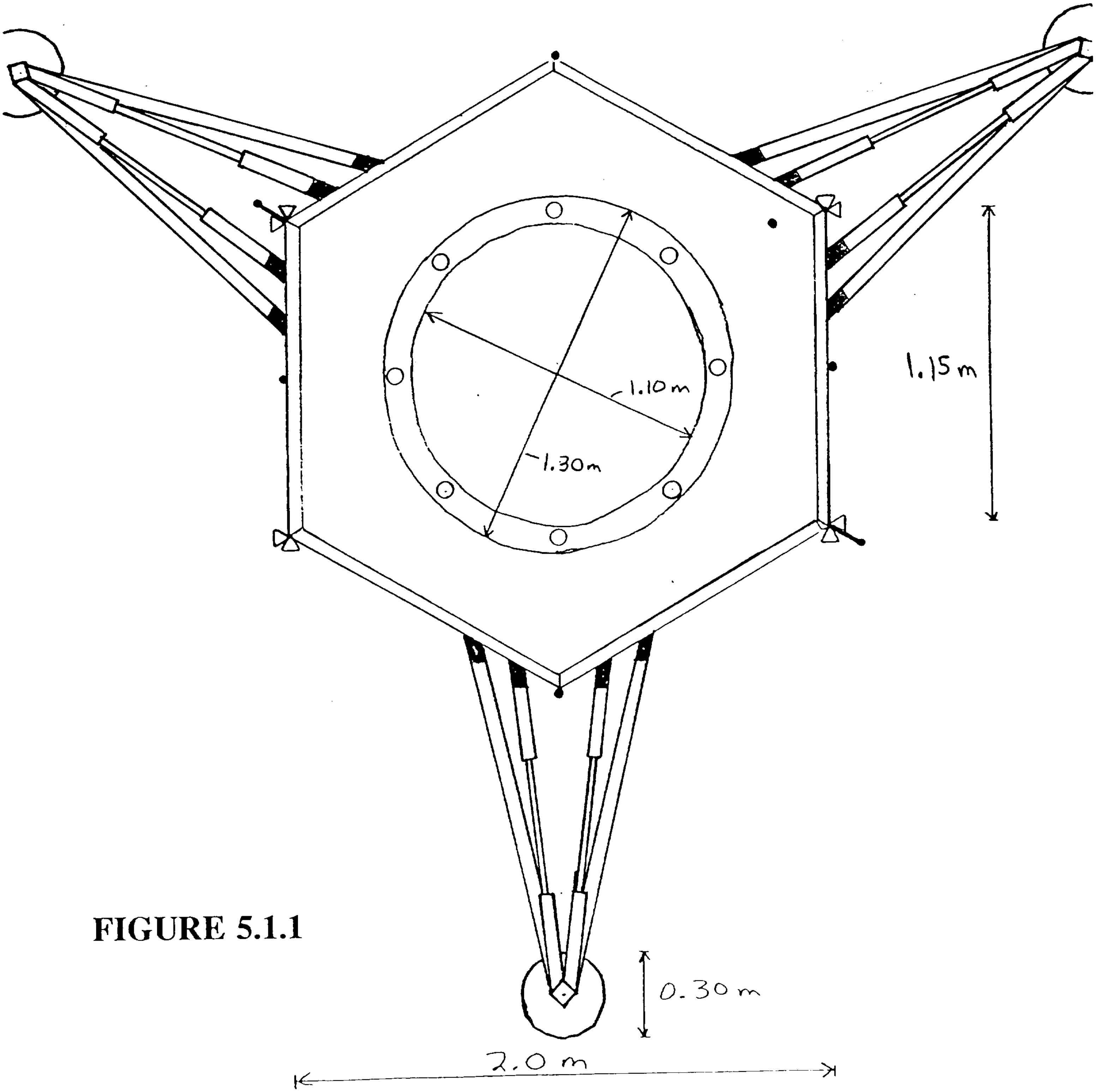
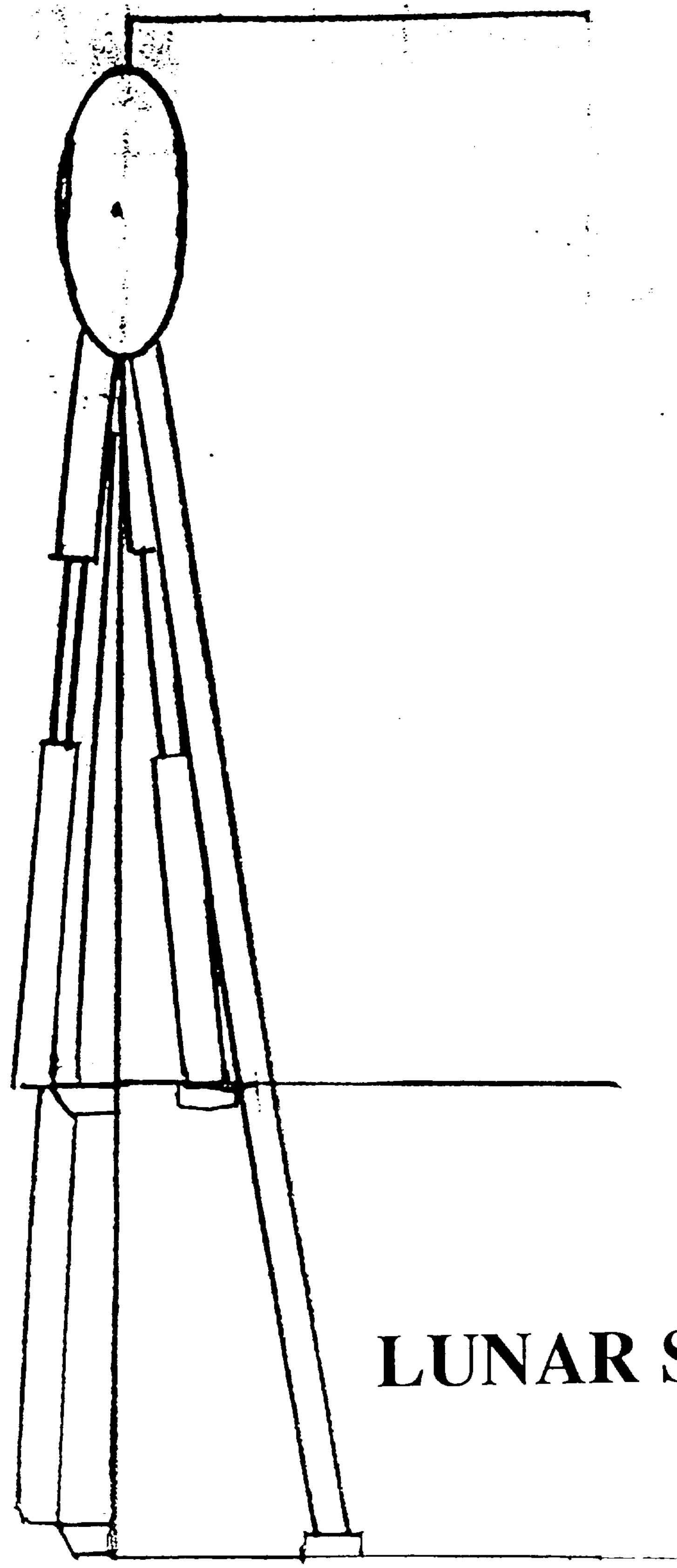


FIGURE 5.1.1

attached 0.50 m. up from the bottom edge of the spacecraft. The hydraulic shocks used will be the same as those used on the Surveyor lander. For stowage within the payload fairing, the legs will be folded up against the side of the lander as shown in Figure 5.1.2. Upon separation of the fairing, the legs will be spring loaded to rotate down and lock into position. This will be accomplished by hinging one edge of each member and attaching bolt-like protrusion which extend 0.635 cm. out on the other three edges around the member. At the attachment point of the spacecraft, spring hinged metal plates with holes for the bolt-like protrusions will be aligned on the three edges to lock the legs into position when the two pieces meet. An enlarged view of the hinged attachment and locking mechanism is shown in Figure 5.1.3. At the end of each lander leg is a 0.30 m. diameter landing pad which will be made of an aluminum honeycomb material to help absorb the stress of impact with the lunar surface on touchdown. An enlarged view of the lander pad is shown in Figure 5.1.4.

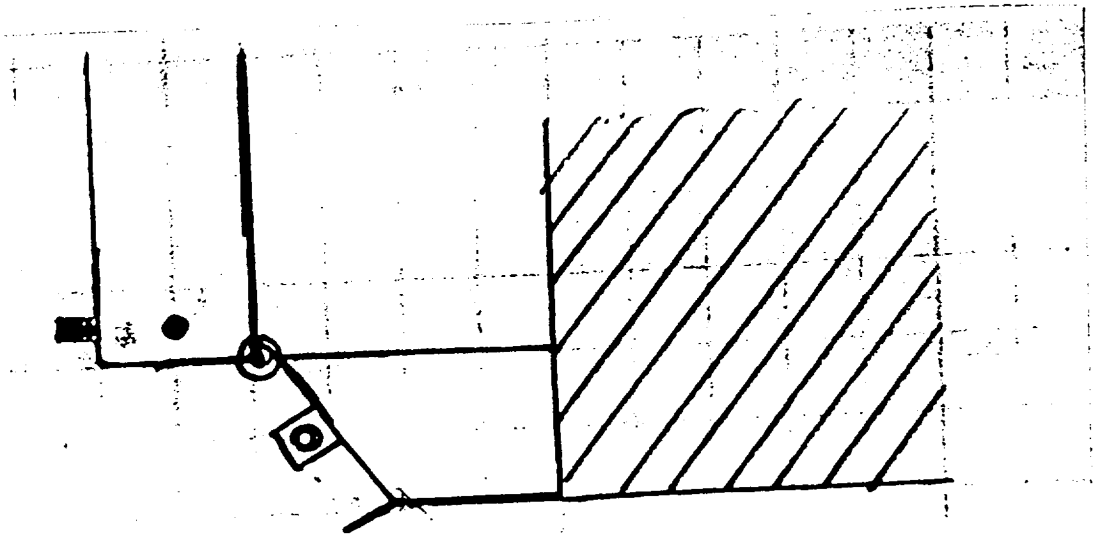
The next major decision made was how to incorporate the solar panels into the design of the spacecraft. The first approach was to use a single large solar panel mounted on a rotating ring with pivot point attachments. This would have given three axis control of the panel so that optimum power output from the solar array would have always been provided. This design, however, presented a problem of placement within the payload fairing due to the sloped walls as you approach the cone of the Delta II fairing. It also would require mechanical

FIGURE 5.1.2



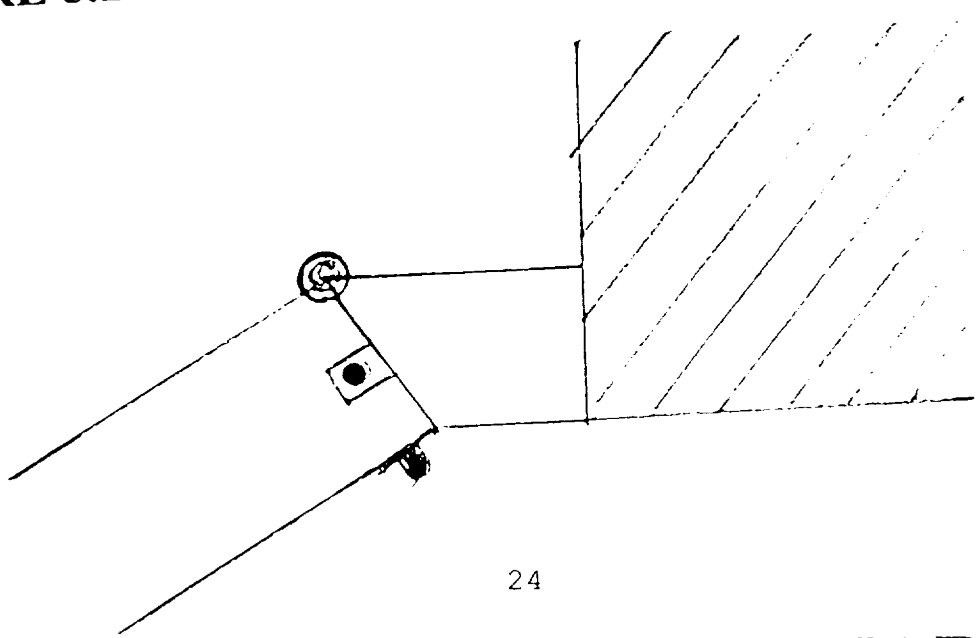
LUNAR SCOUT

ENLARGED VIEW OF LANDER LEG[23]
IN FOLDED UP POSITION



VIEW OF LANDER LEG IN
FOLDED UP POSITION

FIGURE 5.1.3



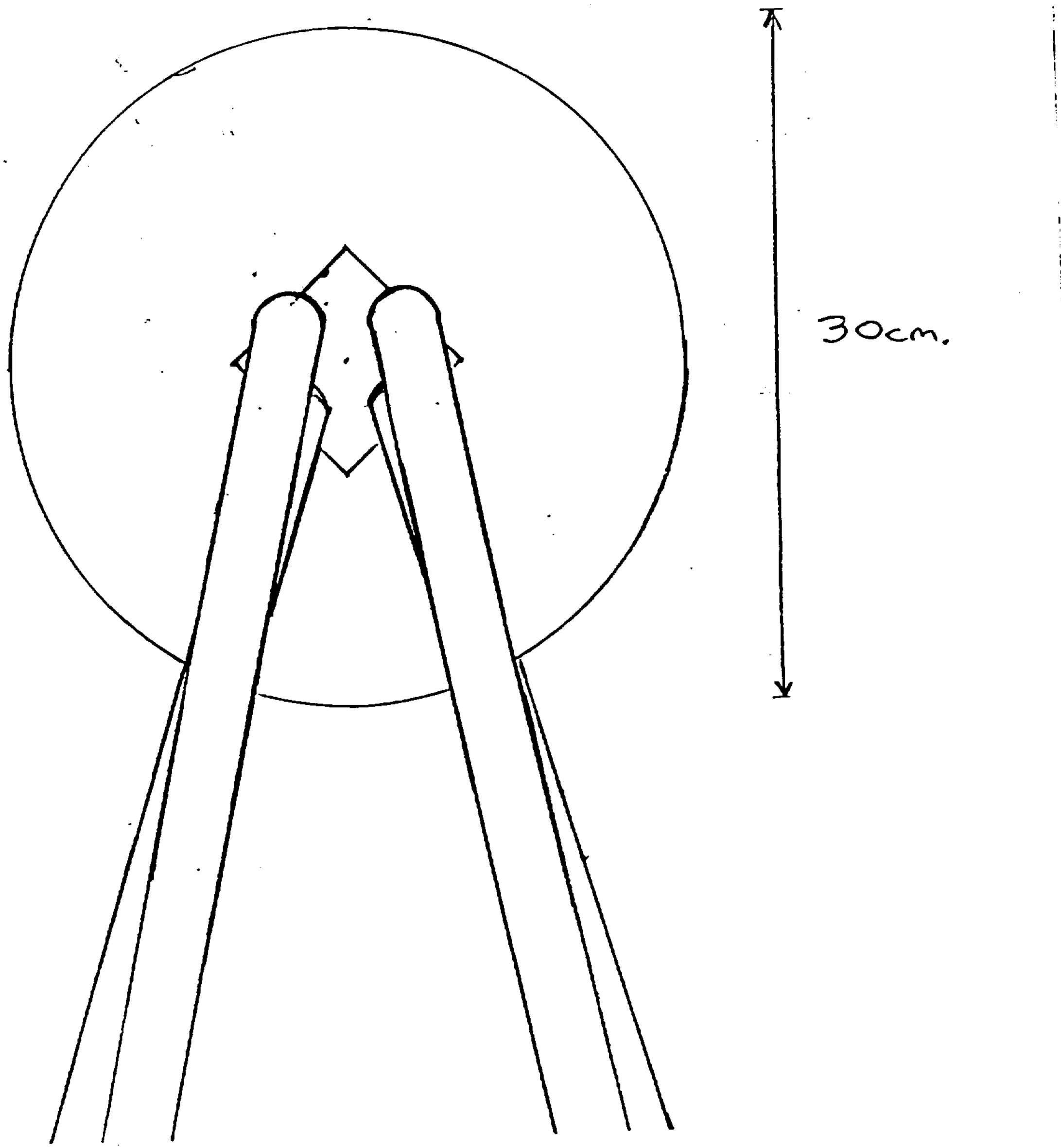
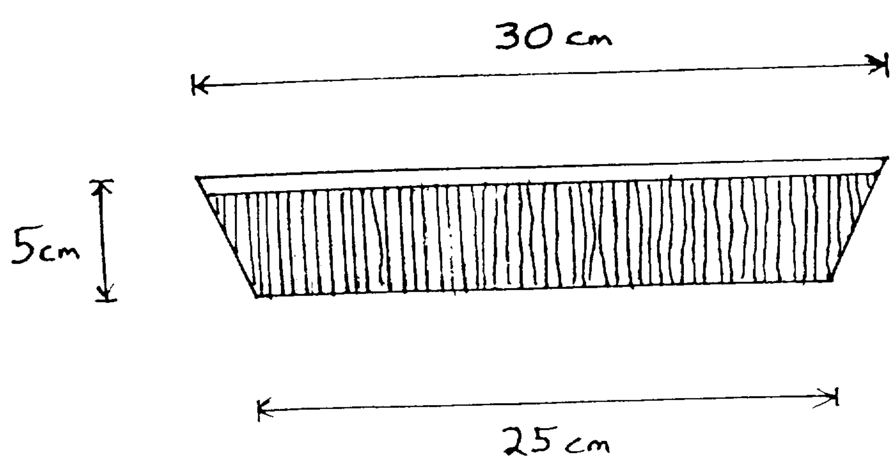


FIGURE 5.1.4

ENLARGED VIEW OF LANDER LEG
IN FOLDED DOWN POSITION

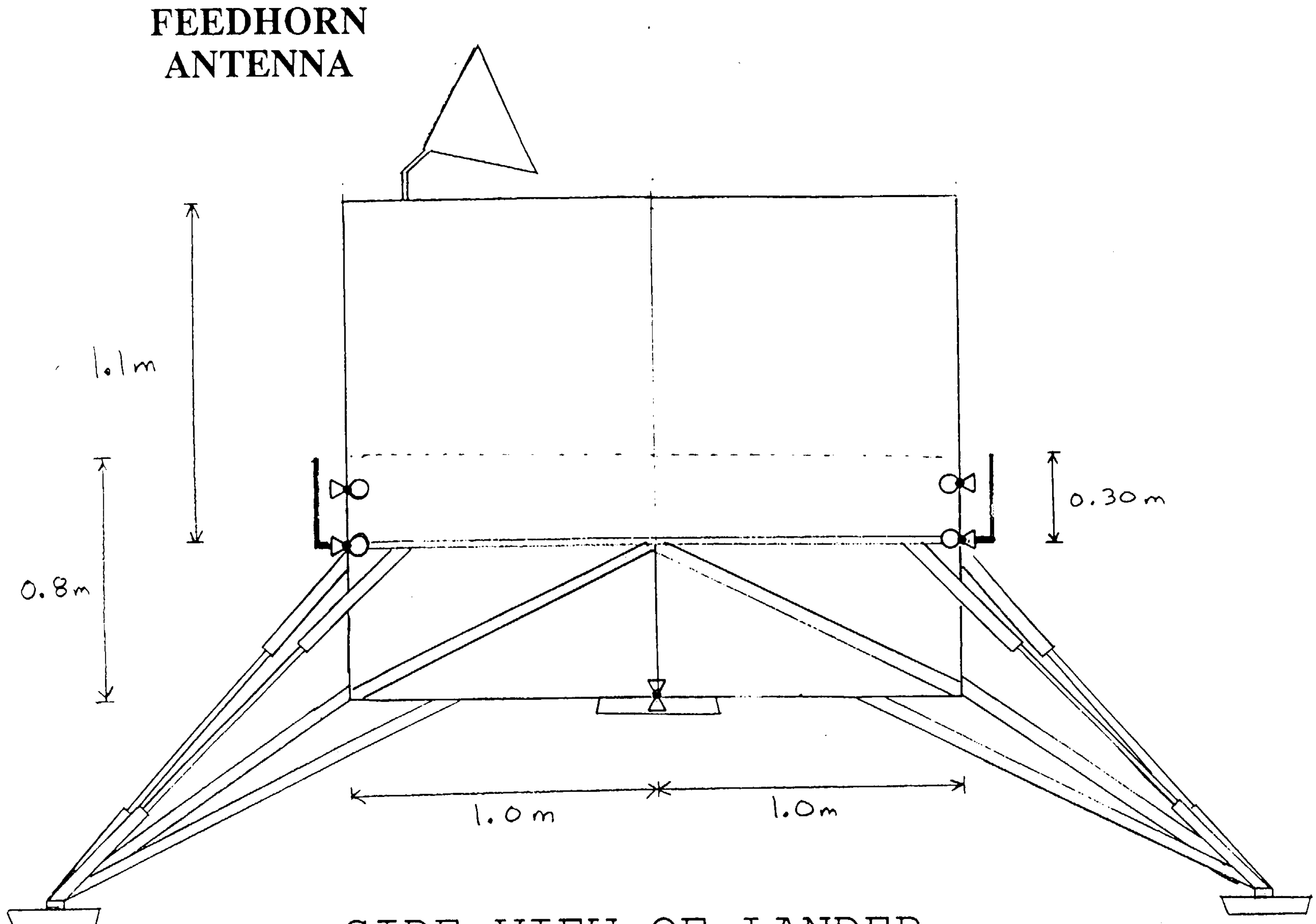


NLARGED VIEW OF

components to move the panel, thus heating would need to be provided by the spacecraft during the lunar night to avoid freeze up, and it would also need a command system to coordinate the movements of the panel. All of these components would add to the mass and complexity of the lander. After researching the problem further, it was decided to use six 1.1 square meter panels that will deploy once upon landing and be stationary for the remaining lifetime of the lander. The panels will be in the upright position as shown in Figure 5.1.5 during the transit to the lunar surface. Then following a set time delay upon touchdown to allow for dust and debris to settle, the panels will be spring activated to release and fold down into position. When deployed, each panel will rest on a small edge of the lander leg at a 45 degree angle to the lunar surface. Figure 5.1.6 shows a top view of the lander with the panels deployed.

Since the payloads of Lunar Scout will be varied and need access to the lunar environment, they will be connected via a 1.30 m. diameter attachment ring located on the top portion of the lander. This ring is shown in Figure 5.1.1. The lower portion of the lander structure will house the liquid retrograde propulsion sub-system, and all other systems such as the communications hardware, attitude control sensors and thrusters, computer, batteries, radar altimeter and thermal control components. Figure 5.1.7 shows a bottom and side view of the liquid propellant tanks, helium tanks and thrust nozzle while Figure 5.1.8 shows a bottom view of the other sub-systems.

After selecting a suitable design for the lander, materials



SIDE VIEW OF LANDER
WITH PANELS UP

FIGURE 5.1.5

LUNAR SCOUT

TOP VIEW OF LANDER WITH
SOLAR PANELS FOLDED OUT

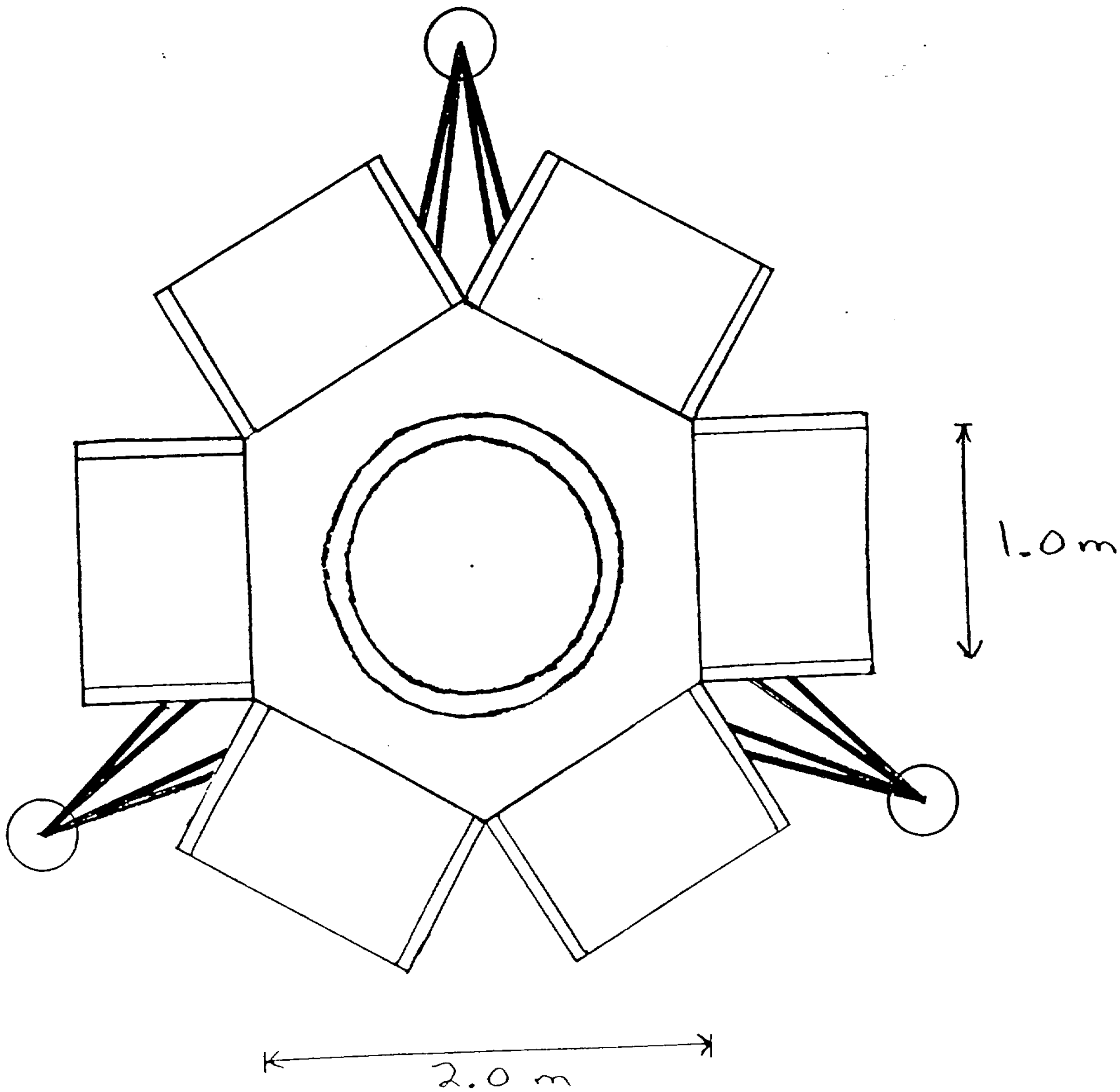
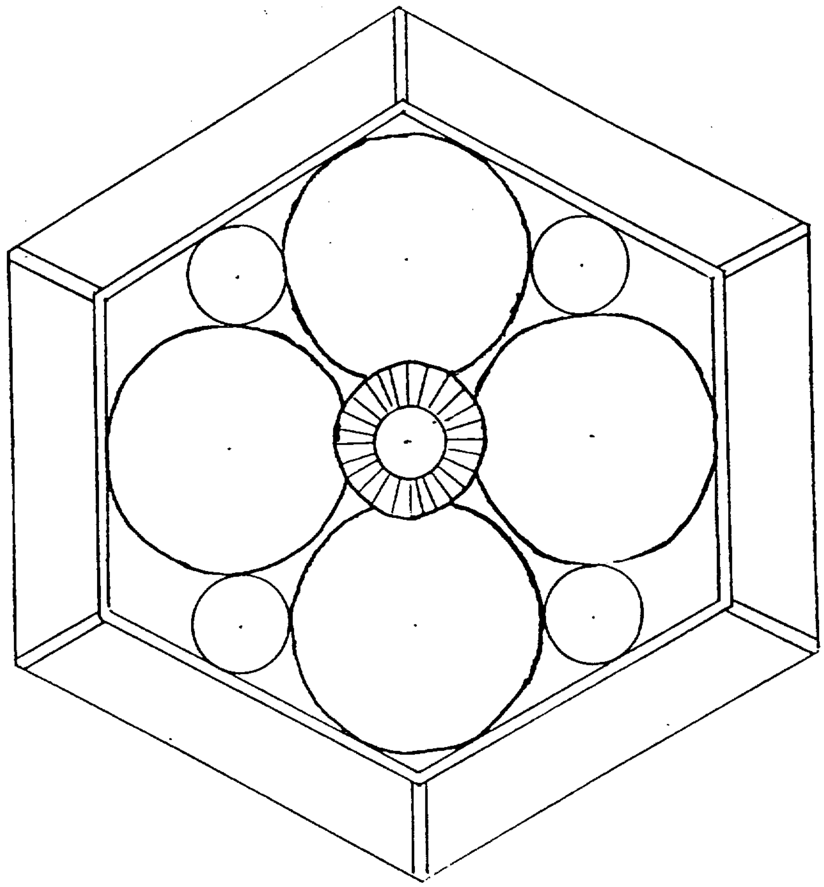


FIGURE 5.1.6

BOTTOM VIEW OF PROPULSION TANKS
AND THRUST NOZZLE



SIDE VIEW OF PROPULSION TANKS
AND THRUST NOZZLE

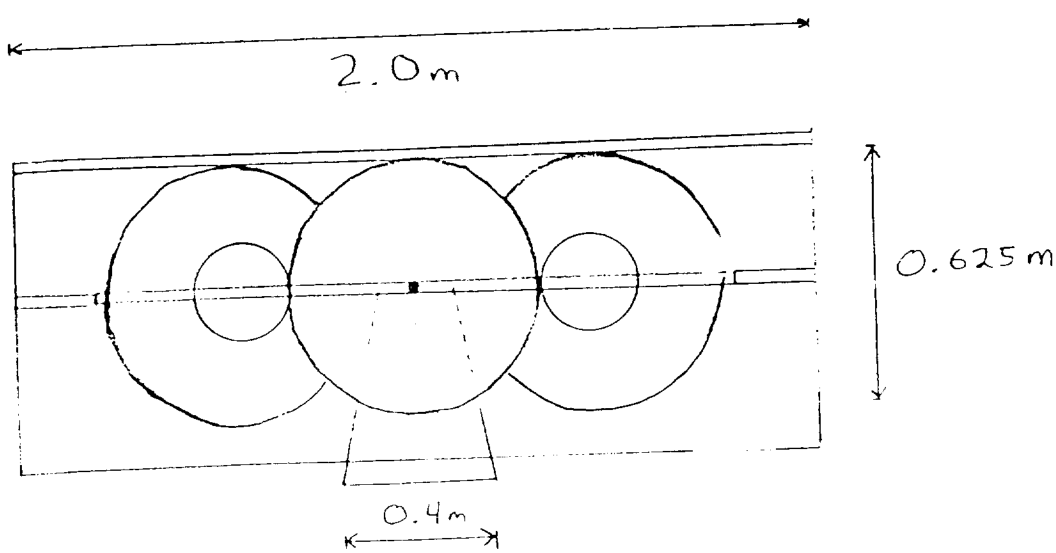


FIGURE 5.1.7

BOTTOM VIEW OF SENSORS AND EQUIPMENT

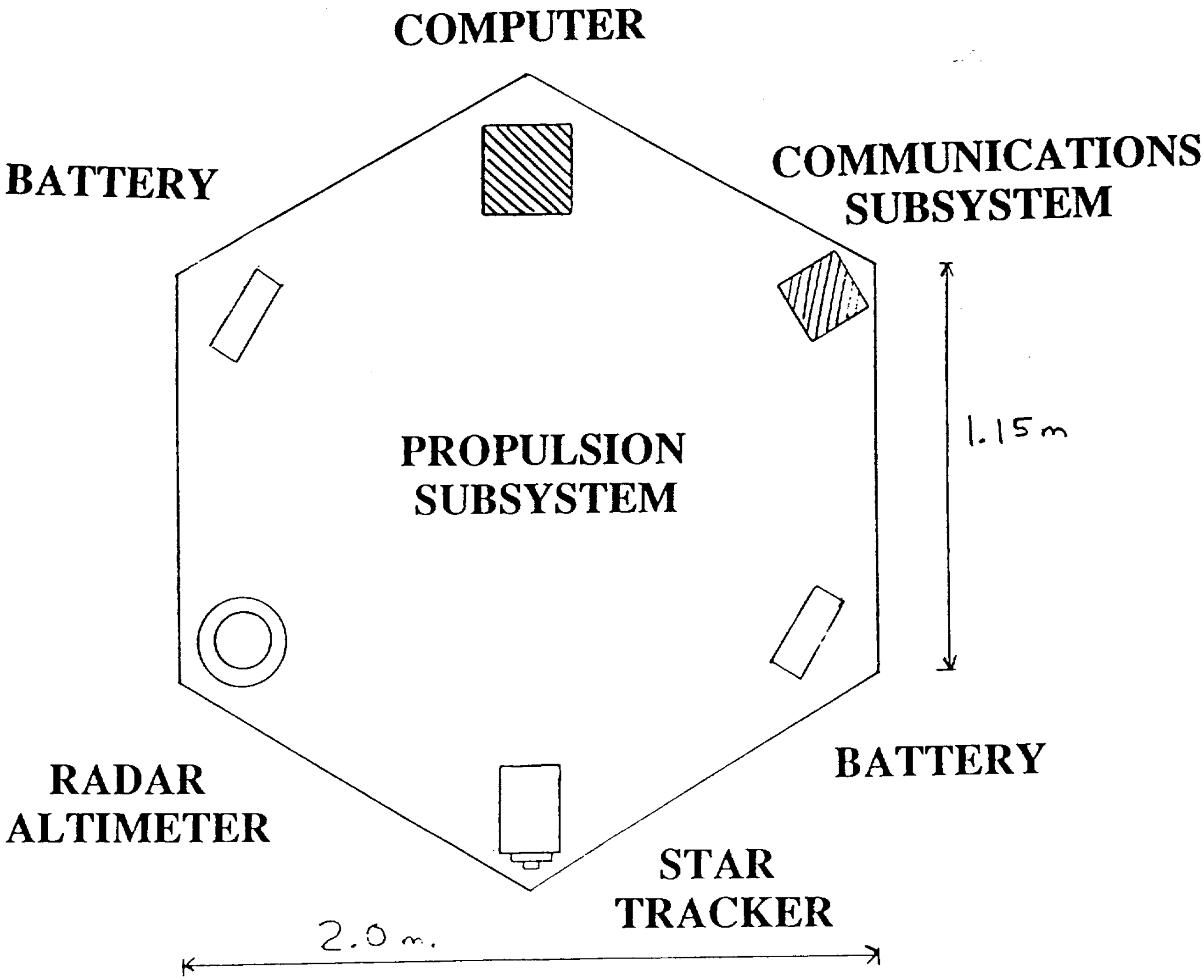


FIGURE 5.1.8

were looked at that could be used to construct the structure. The materials needed to be lightweight, yet strong enough to support the loads that would be encountered during a landing on the moon. To meet the high strength to weight ratio needed, thin walled 6061-T6 aluminum tubing will be used for the majority of the structural members. The outer vertical members and solid leg members will consist of tubing with a diameter of 3.175 cm. and thickness of 0.3175 cm. Horizontal and inner members around the propellant tanks will incorporate tubing with a 2.54 cm. diameter and thickness of 0.3175 cm. The critical cylinder buckling stress for the thin tubing is 94.28 MPa and the maximum stress in any member for a worst case situation with one landing pad hitting down first would be 3.04 MPa which is well below this value. The payload attachment ring will be made out of a sheet of the same aluminum material with a thickness of 0.625 cm. Between the payload and the propulsion sub-system there will be a panel made of ACG-⅛-.003 aluminum honeycomb material to absorb the thrust provided by the liquid retrograde rocket. The lander pads will be composed of 5052-F80-.0025 aluminum flex-core material to absorb the stress of impact with the lunar surface. The compressive strength of the flex-core material is 12.34 MPa and the maximum stress that one landing pad will encounter is 3.04 MPa if the spacecraft landed with all its weight directed to one landing pad so the material is well within strength standards. The natural frequency requirements for the Delta II launch vehicle are that the axial frequency be greater than 35 Hz and the lateral frequency be greater than 15 Hz. For the materials on Lunar Scout, the natural axial frequency will be 500 Hz and the natural lateral frequency will be 1951 Hz. which are both well over the requirements.

The total mass of the structural members of Lunar Scout will be 45 kg. This total mass breaks down as follows:

Leg tubing and vertical support members	-	10.16 kg.
Horizontal and diagonal support members	-	12.30 kg.
Leg shock absorber members	-	10.00 kg.
Three aluminum flex-core landing pads	-	1.00 kg.
Aluminum honeycomb panel under payload	-	3.65 kg.
Payload attachment ring	-	6.29 kg.
Bolts and fittings	-	1.60 kg.
TOTAL STRUCTURE MASS		45.00 kg.

5.2 THERMAL CONTROL SUBSYSTEM

5.2.1 MISSION REQUIREMENTS:

FOR A 110 hr TRANSIT AND A TWO YEAR PERIOD ON THE LUNAR SURFACE:

- ■MAINTAIN SPACECRAFT ELECTRONICS BETWEEN 0°C AND 40°C.
- ■MAINTAIN BATTERIES BETWEEN 5°C AND 20°C.
- ■MAINTAIN FUEL TANKS BETWEEN 7°C AND 35°C.
- ■MAINTAIN SOLAR CELLS BETWEEN 0°C AND 40°C.

5.2.2 TRANSIT ENVIRONMENT:

- ■ WHILE IN TRANSIT TO THE MOON THE LUNAR SCOUT WILL BE EXPOSED TO A SOLAR FLUX OF 1358 W/m^2.

5.2.3 LUNAR ENVIRONMENT:

- ■DURING THE LUNAR DAY THE TEMPERATURE ON THE SURFACE OF THE MOON IS NEAR 117°C.
- ■DURING EARTH ECLIPSE THE TEMPERATURE DROPS TO NEAR -93°C.
- ■DURING THE LUNAR NIGHT THE SURFACE TEMPERATURE OF THE MOON REACHES -183°C.
- ■THE PERIOD OF THIS CHANGE IS 709 hrs.

5.2.4 ACTIVE HEATING:

- ■IN ORDER TO MEET MISSION REQUIREMENTS, THE LANDER MUST ACTIVELY MONITOR THE TEMPERATURE OF ITS COMPONENTS AND DISSIPATE POWER IN THEM.

- ■COVERING A 0.1079 m^2 BATTERY SURFACE AREA WITH MULTI LAYER INSULATION BLANKETING OF EFFECTIVE EMISSIVITY = 0.035 AND DISSIPATING BETWEEN 1.279 W AND 1.578 W WILL KEEP THE BATTERIES BETWEEN 5°C AN 20°C.

- ■COVERING A 0.160 m^2 ELECTRONICS RADIATING SURFACE AREA AND DISSIPATING BETWEEN 1.76W AND 3.04W WILL KEEP THE ELECTRONICS BETWEEN 0°C AND 40°C.

5.2.5 Thermo design of solar panels

In appendix D, there are the calculations to find the operating temperature of each solar panel. Several assumptions were made to arrive at the answer. First, it was assumed that the sun was shining directly on the panels for worst case. This allows for the highest possible temperature to be calculated. Second, the moon albedo coefficient was assumed to be 0.2. This seems logical since the earth's is 0.3. Third, since the solar panels are near the surface of the moon, we assumed that only solar flux hits the solar cells on top while only the moon's albedo flux hits the bottom of the panel. Fourth, the emittance of the solar cells was estimated to be nearly the same as the absorption. Dr. Severns, a solar cell expert stated that generally the ratio of the absorption to the emittance is one.

The temperature calculated at first was a little high for what we wanted. To fix this, 15 2 cm x 1 m thin fins were added on the back of panel to increase the area. This allows more heat to be emitted. Also, the fins increase the stiffness and strength of the panel to help it survive the impact on the moon and the falling on the leg when the panel is deployed.

5.3 ATTITUDE CONTROL SYSTEM

5.3.1 SYSTEM REQUIREMENTS:

- PROVIDE THREE AXIS CONTROL AND STABILIZATION FOR TRANSIT STAGE FROM 1366.7 km EARTH ORBIT TO LUNAR SURFACE.
- USE LOW-COST "OFF THE SHELF HARDWARE."
- PROVIDE 5° POINTING ACCURACY FOR LUNAR DESCENT.

5.3.2 ATTITUDE DETERMINATION SENSORS

SENSOR	MASS	POWER REQUIRED
STAR TRACKER BALL AEROSPACE	8.77 KG	10 W
RADAR ALTIMETER MARTIN MARIETTA	11.11 KG	135 W
WIDE ANGLE SUN SENSOR LOCKHEED	0.155 KG	0 W
	20.06 KG	145.0 W

5.3.3 ATTITUDE CONTROL THRUSTERS

- 16, 1.12 N MR-50L ROCKET RESEARCH HYDRAZINE THRUSTERS ARRANGED IN 4, 4 ENGINE CLUSTERS FOR REDUNDACY. LOCATED AT CENTER OF MASS BEFORE AND AFTER RETROGRADE BURN.

- 4, 4.5 N MR-120 ROCKET RESEARCH HYDRAZINE THRUSTERS, POINTED DOWN FROM LOWEST EXTREMITY OF LANDER.

5.3.4 MASS MOMENT OF INERTIA

1. BEFORE LIQUID RETRO BURN
   - CENTER OF MASS LOCATED 0.5m ABOVE BOTTOM OF SPACECRAFT BODY.

   $I_{xx} = I_{yy} = 145.55 \text{ kg}\cdot\text{m}^2 \qquad I_{zz} = 357.81 \text{ kg}\cdot\text{m}^2$

2. AFTER LIQUID RETRO BURN
   - CENTER OF MASS LOCATED 0.71m ABOVE BOTTOM OF SPACECRAFT BODY.

   $I_{xx} = I_{yy} = 72.98 \text{ kg}\cdot\text{m}^2 \qquad I_{zz} = 196.50 \text{ kg}\cdot\text{m}^2$

5.4 COMMUNICATION

Undoubtedly one of the most important component of any spacecraft is the communication subsystem. Every other system of the vehicle could be functioning perfectly, but if communication is not possible then the spacecraft is considered useless. The purpose of this section of the report is to give an overview of the communication subsystem of Project Lunar Scout. This section will first examine the general characteristics of the subsystem, then it will examine three budget links for different modes of communication by the spacecraft. And finally it will discuss the hardwares needed to accomplish the communication objectives.

The lunar lander will be communicating with NASA's Deep Space Network (DSN) during the two years duration of the mission. The subsystem will utilized the S-Band at a frequency of 2.3 gigahertz. On the ground station, a 4m parabolic antenna will be used to send and receive signals. The antenna will have an efficiency of .5 and a system noise temperature of 1295K. The small antenna will allowed NASA's DSN to keep a constant track of the lander without using its larger and more important antenna dish. The lunar lander will have the same features discussed above. However, its' system noise temperature is 552K instead of 1295K.

The spacecraft will have three modes of communication. The lander will receive commands from Earth, will send back telemetry, and will send/relay data from the on board experiments to earth. The link budgets for each mode of communication is in Appendix F. The glossary for the abbreviations used in the link budget is in Appendix F (F-1). The first link budget in Appendix F (F-2) shows the requirement for the ground station to send commands to the lunar lander. The transmitter's power is assumed to be 1W for calculating purposes. Miscellaneous loss is at -4dB and the signal is sent at 10 bits per second. The lander will receive the commands using its omni-directional antennas with a gain of 0dB. From the link budget, the amount of power require from the DSN to send information to the lander is 4.53W.

The second link budget is in Appendix F (F-3). Again the transmitter's power is assumed to be 1W for calculating purposes and the signals is sent at 10 bits per second. The spacecraft will use two omni-directional antennas to send telemetry back to earth with a gain of 0dB. There is a -3dB link margin for safety and the initial power requirement is 1.93W. There will be loss to the electronics due to their inefficiency during the transmission and so a factor of 3 was included. Therefore, with loss taken into account, the amount of power needed to send telemetry back to earth is 5.59W.

To send back data from the various scientific payload, the communication subsystem will utilized higher bit rates which require more power. This can be seen in the third link budget in Appendix F (F-4). The variables of the link budget are similar to that of the first two with two exceptions. The data rate of transmission is now at 1000 bits per second and the antenna has a gain of 15dB. The higher bit rate is required to send the data collected from the payload. the antenna had a gain of 15dB due to the fact that it is a feedhorn antenna. The feedhorn antenna is chosen after initial calculations show that using a parabolic antenna was not feasible on the lander. The feedhorn antenna can be seen in Appendix F (F-5). From the link budget, the amount of power required to send data after loss have been taken into account is 19.50W.

The two omni-directional antennas were picked to send telemetry and received command because it has a lower power requirement than that of the feedhorn antenna. The omni-directional antennas only need 5.79W for transmission whereas the feedhorn antenna needed 19.50W for transmission. As far as pointing is concern, the feedhorn will be preset to the earth's altitude as seen on the moon at the desire landing sight. The antenna will have one-degree of freedom to rotate in order to acquire the earth. The feedhorn antenna has a field of view of 23.4 degrees by 25.5 degrees and so acquiring the earth should present no problem. The earth will remain motionless as seen from the moon and so once the feedhorn acquires the earth, the antenna can be set in that position for the duration of the mission.

As mentioned above, the hardwares chosen to meet the communication objectives are two omni-directional antennas for telemetry and command communication and a feedhorn antenna for data transmission. From Larson and Wertz's "Space Mission Analysis and Design" the specific weights and dimensions of the subsystem hardwares are calculated. This is in Appendix F (F-6).

The omni-directional antennas system will have a mass of 18kg which includes the 16kg of two transponders and electronics components for redundancy. And the feedhorn antennas system will have a mass of 20 kg including 16kg for the electronics and 4 kg for the feedhorn. Total volume of the communication subsystem is 18336cm3. The detail volume information and the dimensions of the feedhorn antenna is in Appendix F (F-6).

In conclusion, using two omni-directional antennas and one feedhorn antenna will meet the requirements of the lunar lander. Furthermore, by utilizing the omni-directional antennas for telemetry and command communication instead of the feedhorn will reduced the power usage and makes the communication subsystem more efficient.

5.5 LUNAR SCOUT PROPULSION SYSTEM

5.5.1 PROPULSION SYSTEM REQUIREMENTS:

- LUNAR ORBIT INSERTION FROM 1366.7 km 2869.11 m/s
- RETRO ROCKET BURN AT 200 km LUNAR ALTITUDE 2476.11 m/s
- ATTITUDE CONTROL AND DESCENT DELTA V
- USE "OFF THE SHELF HARDWARE"

5.5.2 PROPULSION SYSTEM CHARACTERISTICS

1. **LUNAR INSERTION DELTA V AT 1366.7 KM ALTITUDE = 2869.11m/s**
   ROCKET:
   THIOKOL STAR 48A LONG NOZZLE
   INITIAL MASS OF SOLID ROCKET AND SPACECRAFT: 3775 kg
   FUEL MASS = 2547.32 kg
   I_{sp} = 283.4s

   DELTA V = 2869.11 m/s

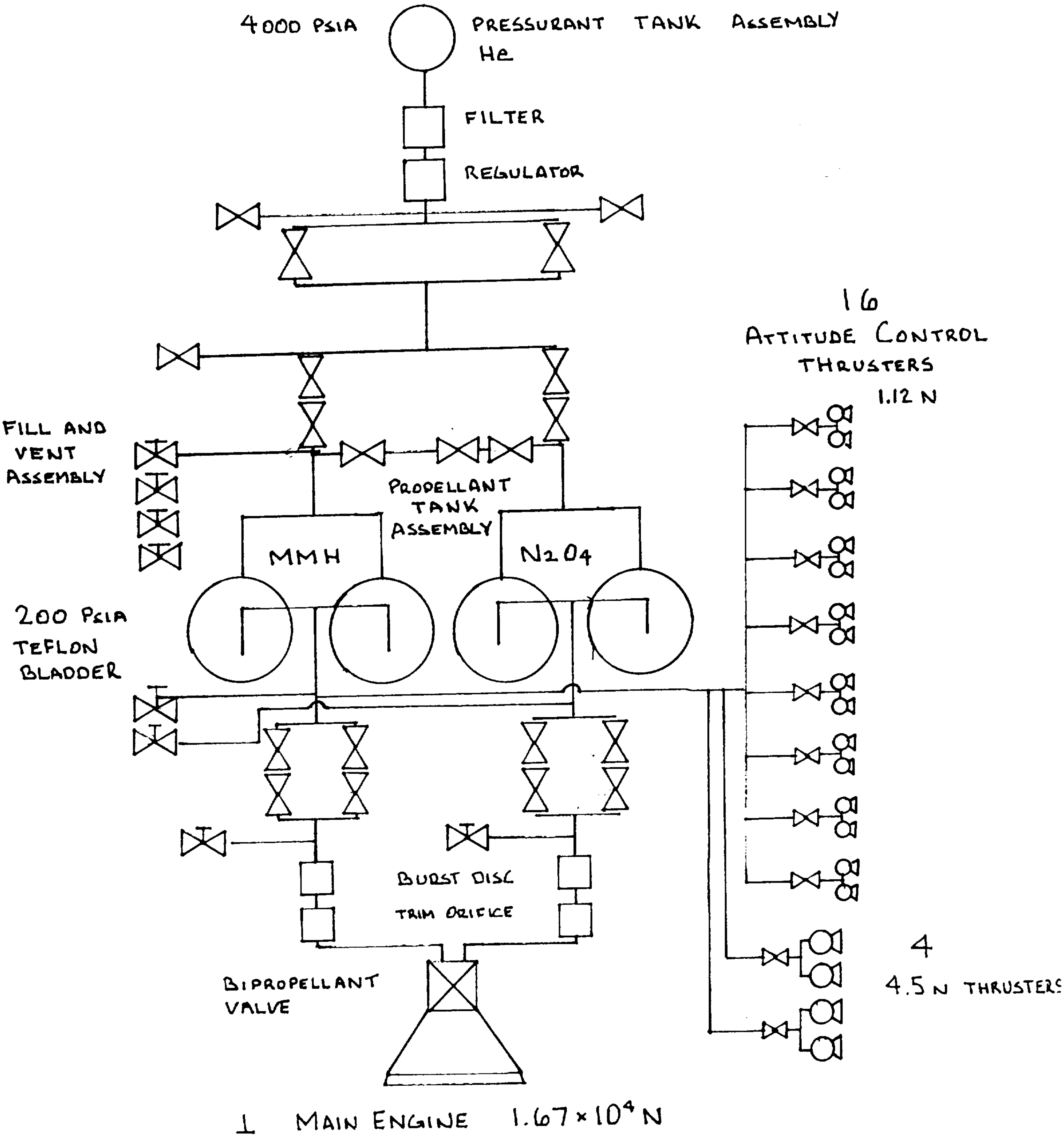
2. RETROGRADE DELTA V AT 200 KM LUNAR ALTITUDE = 2476.11 m/s
   ROCKET:
   ROCKETDYNE XLR-132
   INITAL MASS OF LANDER AND BIPROPELLANT = 1205.91 kg
   MASS OF BIPROPELLANT = 675.36 kg
   I_{sp} = 340s

   DELTA V = 2476.11 m/s

3. DELTA V FOR ATTITUDE CONTROL AND LANDING AT 31.65 km LUNAR ALTITUDE
   THRUSTERS:
   4 - 4.5 N MR-50L ROCKET RESEARCH HYDRAZINE THRUSTERS
   16 - 1.12 N MR-120 ROCKET RESEARCH HYDRAZINE THRUSTERS
   I_{sp} = 229 s - 215 s
   MASS OF LANDER AND REMAINING FUEL = 584.11 kg
   MASS OF REMAINING FUEL = 54.03 kg

   DELTA V = 223.76 m/s

FIGURE 5.1

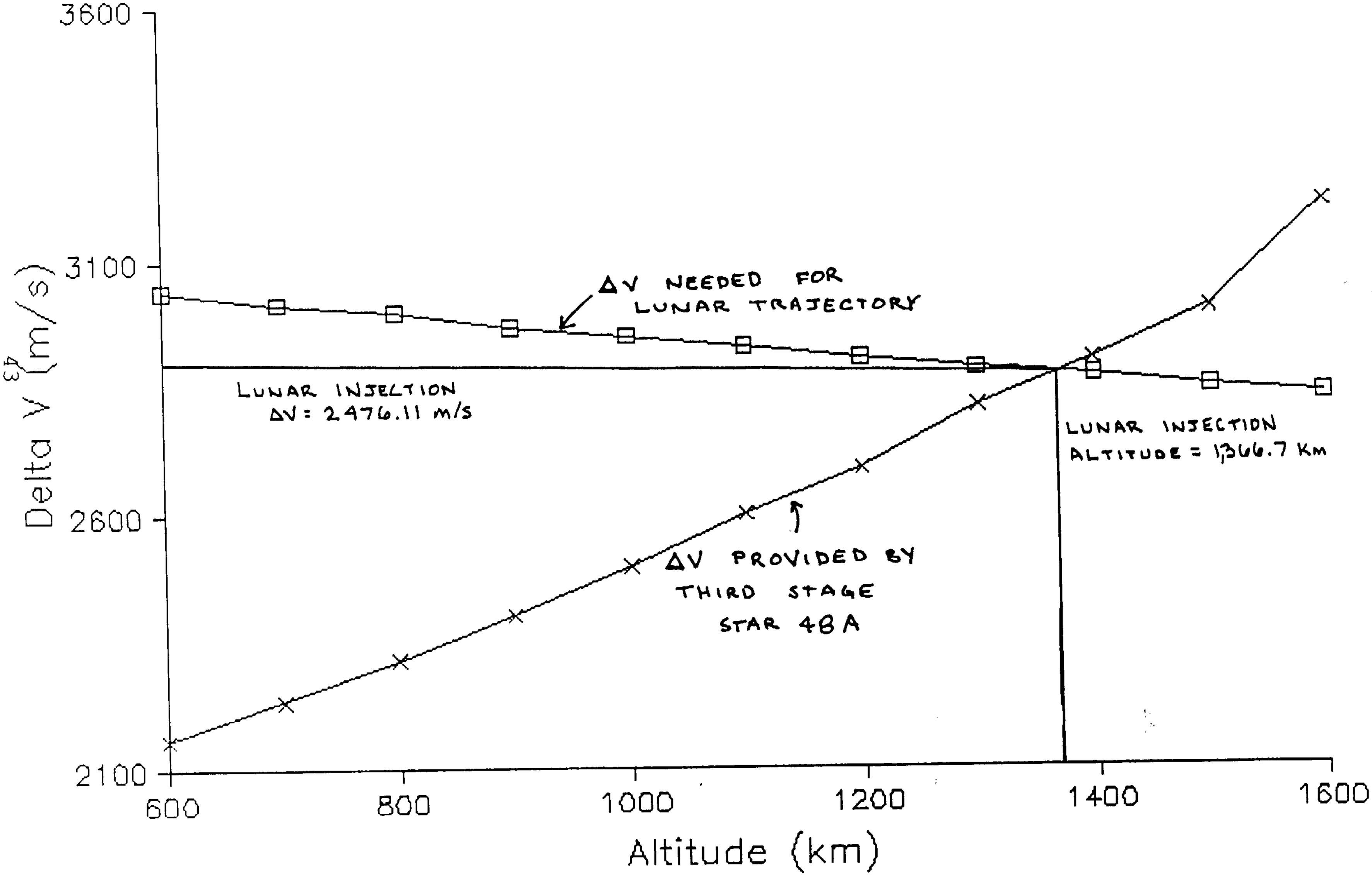


PRESSURE FED PROPULSION SYSTEM

FIGURE 5.2

THIRD STAGE SELECTION

Altitude vs. Delta V.



5.5.3 LANDER MASS SUMMARY

ITEM	MASS (KG)	COMMENTS
PAYLOAD	**200 KG**	
STRUCTURE	**45 KG**	ESTIMATE
ELECTRONICS SYSTEM INTEGRATION COMPUTER	**10.84 KG**	ESTIMATE
COMMUNICATIONS FEEDHORN ANTENNA DIPOLE ANTENNAS COMMUNICATIONS ELECTRONICS TOTAL COMMUNICATIONS	 4.0 2.0 32.0 **38.00 KG**	ESTIMATE
POWER BATTERIES SOLAR ARRAYS TOTAL POWER	 8.18 33 **41.18 KG**	ESTIMATE
ATTITUDE CONTROL SYSTEM SUN SENSOR VIKING RADAR ALTIMETER STAR TRACKER ATTITUDE CONTROL THRUSTERS TOTAL ATTITUDE CONTROL	 .155 11.13 8.65 12.49 **32.43 KG**	
THERMAL BLANKETS THERMAL CONDUCTOR PLATES TOTAL THERMAL	 11.20 4.0 **15.20 KG**	ESTIMATE
PROPULSION MAIN ENGINE SUPPORT HARDWARE TOTAL PROPULSION	 51.26 96.17 **147.43 KG**	14% OF LIQUID FUEL MASS
TOTAL MASS AT LANDING	**530.08 KG**	

5.5.4 LIFTOFF MASS SUMMARY

TOTAL MASS OF LANDER	**530.08 KG**	
PROPELLANT LIQUID BIPROPELLANT ATTITUDE CONTROL PROPELLANT TOTAL LIQUID PROPELLANT	 621.33 54.03 **675.36 KG**	
THIRD STAGE STAR 48A SOLID ROCKET	**2547.32 KG**	
TOTAL LIFT OFF MASS	**3752.72 KG**	
DELTA 7920 LIFT CAPABILITY TO 1367 KM CIRCULAR ORBIT 3775 KG CONTINGENCY IS 3775 - 3752 = 22.28 KG (.59%)		

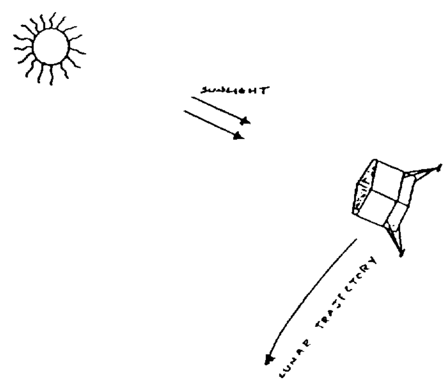
5.6 POWER SUBSYSTEM

5.6.1 GOALS:

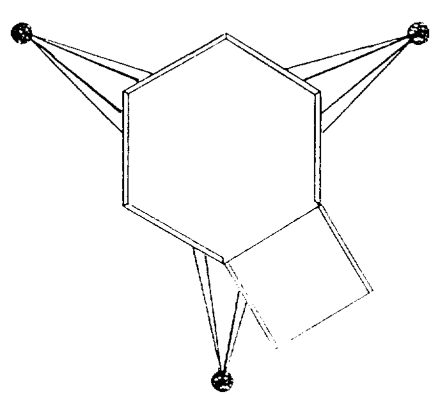
The primary goal of the power subsystem is to provide power to all onboard systems which include the computer, radar, transmitter and attitude control system. However, since the the spacecraft is a lunar transport for a 200 kg payload, we have also added the requirement to supply 500 W of power to the payload. There are two reasons for this addition. First, the lunar lander will still be used upon landing which means that its systems will not be wasted in a one time use. Second, because the lander will supply the power to the payload, the payload is freed of the requirement to supply its own power which allows it to use the mass in other areas. In the long run, money will be saved because there is only one power system instead of two completely independent systems. The final power requirement placed on the lunar lander is also in support of the payload. The lander will supply 10 W of continuous power during the 14 day lunar night. This power will be used to heat the spacecraft and to maintain spacecraft status with ground control. The heat during the night will help protect payloads from freezing during the night which would have an adverse effect on delicate instruments such as the lenses on an astronomical telescope.

5.6.2 CHRONOLOGY:

Prior to launch, the battery onboard the lander will be fully charged. During the course of the 4.5 day flight to the moon from the earth, the battery will provide the power to all systems operating onboard the spacecraft. These will include the transmitter, the attitude control system, and the computer. However, should the battery system fail, the spacecraft will orient itself to point its topside at the sun. This action will allow sunlight to fall upon the solar panels which are closed up.



If this does not generate enough power, a solar panel will be deployed to provide power to the spacecraft. Before landing on the moon, this panel will be jettisoned to prevent it from damaging the spacecraft upon landing.



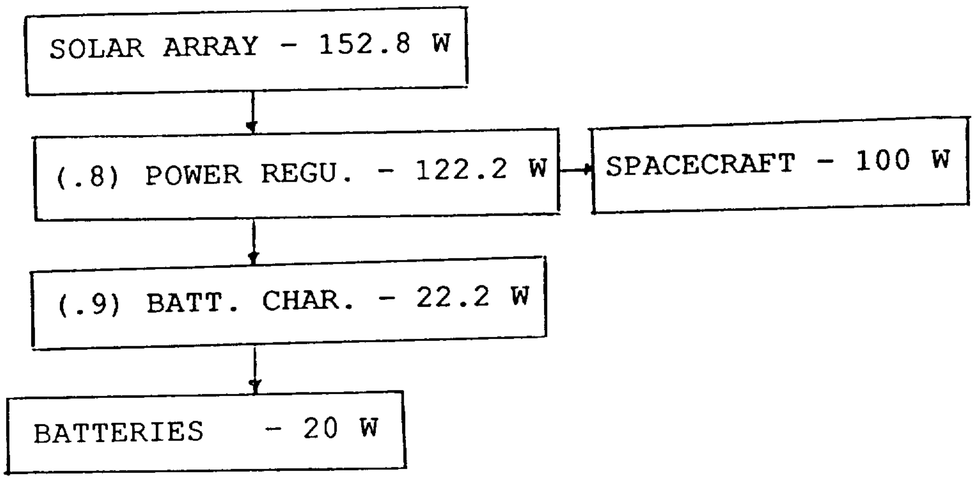
When the spacecraft begins its main retro burn, the radar will be turned on to guide the spacecraft to its landing point.

The radar that will be used will be similiar to the one used on the Viking landers which required approximately 140 W of power. This entire process will last about 2-3 minutes. After touchdown, the spacecraft will shutdown the radar and attitude control systems which are no longer necessary for the mission. Once all of the dust has settled from the impact, the lander will deploy its solar arrays, begin to power up the payload and recharge the battery.

During the spacecraft's 2 year lifetime, it will operate on day/night cycles. During the lunar day (14 earth days), the lander will provide 240 W to 750 W of power depending on the position of the sun in the lunar sky. At night, the lander will shutdown all systems and the battery will provide 10 W of power for heating and health status communications. A power budget is provided in appendix H.

5.6.3 FIRST SOLAR ARRAY DESIGN:

Originally, our design called for 100 W to the payload during the day and 20 W at night. These requirements were simply a guess as to what was required. During the daytime, the solar array would have to provide power to the payload and to charge the battery. Based upon calculations taking into account the minimum amount of sunlight during the day, it was determined that the battery would require 20 W of power during the day. Using a 90% efficiency for the battery charger and an 80% efficiency for the power regulator, the flow chart below shows how the end of life (EOL) power was determined:



Assuming a 7% solar cell degradation, the beginning of life (BOL) power was:

$$BOL = \frac{EOL}{(1 - DEG)} = \frac{152.8\ W}{(1 - .07)} = 164.3\ W$$

To provide this amount of power, a 2 cm x 4 cm silicon solar cell was selected. This cell had a maximum power of 101.7 mW and a minimum power of 71.9 mW calculated at a 45 degree angle. The number of cells required:

$$cells = \frac{BOL}{Pmin} = \frac{164.3\ W}{71.9\ mW} = 2286$$

It was decided that a 28 V bus would be used. Therefore, the cell arrangement would be:

(Vmax = maximum cell voltage)

$$\text{cells in series} = \frac{\text{bus voltage}}{\text{Vmax}} = \frac{28\ \text{V}}{.9\ \text{V}} = 31$$

$$\text{parallel strings} = \frac{2286\ \text{cells}}{31} = 74$$

$$\text{solar cell area} = (\text{cell area})*(\text{cells}) = (.0008\ \text{m})*(2286)=1.83\ \text{m}$$

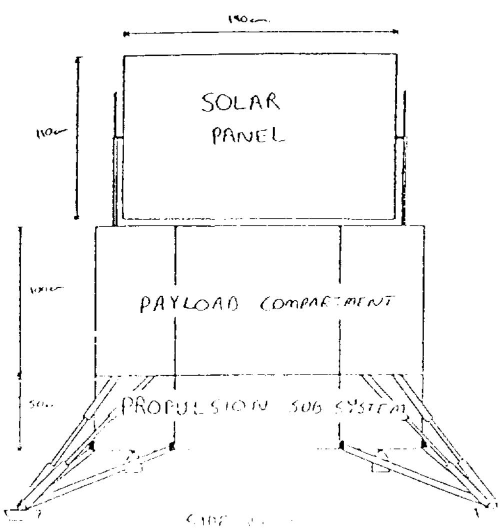
To include cell packing, an area utilization factor (UF) of .85 was used:

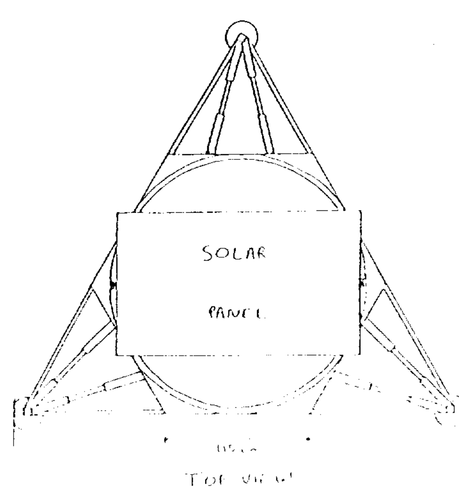
$$\text{solar panel area} = \frac{\text{solar cell area}}{\text{UF}} = \frac{1.83\ \text{m}}{.85} = 2.15\ \text{m}$$

The mass of the array was calculated using an estimate of 5 kg/m :

$$\text{solar panel mass} = \text{solar panel area} * 5\ \text{kg/m}$$
$$= 2.15\ \text{m} * 5\ \text{kg/m} = 10.75\ \text{kg}$$

This solar panel was to track the sun as the sun moved across the lunar sky and was to be mounted as shown:

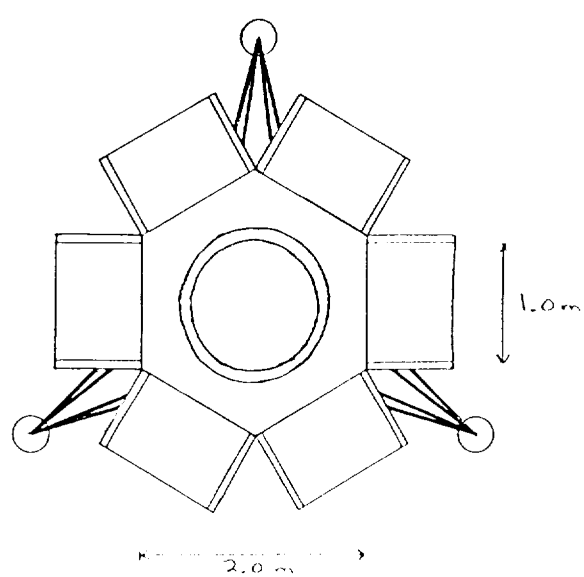




However, it was decided that tracking the sun was too complicated and expensive to design and build. Also, with the solar panel mounted above the lander, it would be more fragile which would make it less likely to survive the launch and the impact on the moon. The panel would also restrict the payload volume and its view of the sky. Given the size of some of the possible payloads, the volume provide between the lander base and panel would clearly not be enough.

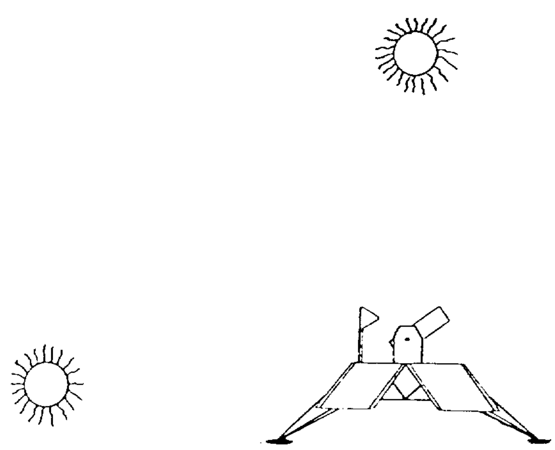
5.6.4 FINAL SOLAR ARRAY DESIGN:

After consulting with Dr. Stan Olendorf, an expert at NASA working on the lunar lander program, the requirements originally set forth were found to be far from what was needed. The payloads would require around 500 W of power, but only 10 W of power would be needed at night. It was obvious that a fixed array would be needed to supply so much power. An RTG was also considered, but the cost of such a power supply was deemed too expensive for this low budget mission. The decision was made to



mount six solar panels as shown below. Each panel would be mounted so it rested on a leg at a 45 degree angle to the horizontal.

With fixed panels, the amount of sunlight falling on them would vary during the day depending on the position of the sun. Therefore, a worse case and best case situation was needed. Looking at the geometry of the lander, it was determined that the worst case would be sunlight falling on two panels (sun on the lunar horizon) and the best case would be sunlight falling on all six (sun at highest point in sky).



Since all calculations will be based on a 45 degree angle, it was not considered in the best/worse cases. Also, the declination of the sun would not be a factor. If the sun did not pass directly over the lander, the sunlight would hit the closer solar panels at less than a 45 degree angle. This cancels the effect of less sunlight falling on the panels on the side of the lander opposite of the sun.

A spreadsheet was set up to perform the calculations to find the solar panel area and mass. It was discovered that to provide

500 W using worse case panel area was nearly impossible due to the required size and mass of the arrays. To fit in the Delta II payload fairing, each individual array would have to be 1.5 m x 1.1 m in area. This would give a total area of 9.9 m , and a mass of nearly 50 kg. Using the same silicon solar cells as before, the calculations showed that the worse case power would be 185 W and best case would be 575 W. This would be enough, but the payload would only be able to operate for a few days each month. Research was then begun to remedy this problem. To further compound the problem, solar array area was to be reduced to 1.1 m x 1.0 m due to a larger retro rocket being required for the landing.

To fix the problem, Gallium Arsenide on Germanium solar cells were used instead of the silicon cells. They cost a little more, but they are more efficient and provide more power which would require a smaller array, defraying some of the cost. The cell chosen to be used was the 44 Ge-90. This cell is a 4 cm x 4 cm cell which would increase the area utilization factor used for the 2 x 4 cell. Overall the solar panel would have a total area of 6.6 m^2. An iteration process using the spreadsheet was then used to find how much power this would provide to the payload. The maximum power provided was calculated to be 750 W and the worst case power was 240 W. The flow chart below is taken from the spreadsheet. It shows how the BOL power was determined. Degradation was reduced to 3% due to the fact that 7% is an estimate for low Earth orbit. Since the lander will not have to deal with an atmosphere continuously, the cells will not degrade as much. Also, the efficiencies are the same as before.

```
                    ---------------------
                    Array:     951.3889 W
                    ---------------------
                             |
                    ---------------------
                    Pwr Reg:  761.1111 W
                    ---------------------
                    |                  |
------------------------               |
S/C:             750 W                 |
------------------------               |
                                       |
                    ---------------------
                    Charger:  11.11111 W
                    ---------------------
                             |
                    ---------------------
                    Batteries       10 W
                    ---------------------
```

Below is the second half of the spreadsheet. The values for the maximum current (Imax) and maximum voltage (Vmax) were taken from the I-V curve that is in appendix H. Starting with the BOL power, it calculates the cell arrangement, solar panel area and power, it calculates the cell arrangement, solar panel area and solar panel mass using the same equations previously used. Due to the larger cell area, the area utilization factor was increased slightly from .85 to .88 which was a reasonable

```
SOLAR ARRAY DESIGN

Cell No.: 44 Ge-90
    Imax:        450 mA          Area:     .0016 Sq Meters
    Vmax:        .85 V
    Pmax:      .3825 W
    Pwor:  .2704658 W   (45 degree angle)

BOL Pwr:   980.8133                      Cells:       3626
Bus Volt:        28 V          Cells in series:         33
                              Parallel Strings:        110

Array Area:
  w/o UF:    5.8016                Util. Factor:        .88
   w/ UF:  6.592727

Array mass:
    ===>   32.96364            Weight estimate:          1 lbf/ft^2
                              Density estimate:          5 kg/m^2
```

estimate. The density estimate of the array was the same because the silicon and gallium arsenide cells weigh about the same.

5.6.5 BATTERY DESIGN

The purpose of the battery on the lunar lander is twofold. First, it will act as the primary source of power during the flight of the spacecraft from the earth to the moon. Second, it will also be the primary source of power during the lunar night. The largest requirement placed on the battery is the 10 W of continous power during the lunar night. This will determine the size of the battery. The energy required by the battery is:

$$\text{Energy} = 10\ \text{W} * 14\ \text{days} * 24\ \text{hrs/day} = 3360\ \text{W*hrs}$$

Using a 28 V bus, the charge required is:

$$\text{Charge} = \frac{3360\ \text{W*hrs}}{28\ \text{V}} = 120\ \text{A*hrs}$$

The battery will only operate at an 80% depth of discharge. Even though there will only be 27 charge/discharge cycles during the spacecrafts designed lifetime, the emphasis is on constant voltage. After 80%, the voltage begins to drop off. To find the total amount of cells required by the battery, divide by the DOD:

$$\text{Total charge} = \frac{120\ \text{A*hrs}}{.80} = 150\ \text{A*hrs}$$

To provide 150 A*hrs and a 28 V bus, the battery will consist of 22 1.25 V, 7 A*hr Nickel Cadmium cells. Each cell measures 10.05 cm x 5.41 cm x 2.11 cm and has a mass of .31 kg. When finding the total volume and mass of the battery, a 1.2 packaging factor will also be included:

$$\text{Volume} = 10.05 \text{ cm} * 5.41 \text{ cm} * 2.11 \text{ cm} * 22 \text{ cells} * 1.2 = 3029 \text{ cm}^3$$

$$\text{mass} = .31 \text{ kg} * 22 \text{ cells} * 1.2 = 8.184 \text{ kg}$$

The battery will be divided into two parts to help distribute heat. Each part will be 24.5 cm x 5.8 cm x 10.7 cm and will be placed directly beneath the payload so that the power that is lost as heat also keeps the payload warm.

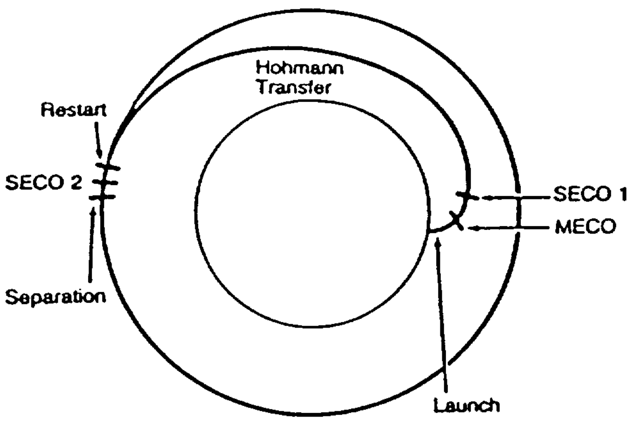
5.6.6 FINAL DESIGN SUMMARY

- -- six solar panels of 1.1 m x 1.0 m
  - total area = 6.6 m^2
  - BOL power = 981 W
  - EOL power = 950 W
  - total mass = 33.0 kg
- -- solar panels made of gallium arsenide on germanium cells
  - 3626 total cells
  - 33 cells in series
  - 110 parallel strings
  - bus voltage = 28 V
- -- power distribution
  - daytime
    - panels provide 240 W to 750 W to payload
    - provide 10 W to charge battery
    - remaining power is lost
  - night
    - panels do not provide any power
    - battery becomes primary power source

6.0

APPENDICES

APPENDIX A



TYPICAL TWO-STAGE MISSION PROFILE

Table 1

EVENT	TIME (SECONDS)
MAIN ENGINE IGNITION	T+ 0
SOLID MOTOR IGNITION (6)	T+ 0
SOLID MOTOR BURNOUT (6)	T+ 63
SOLID MOTOR IGNITION (5)	T+ 66
SOLID MOTOR SEPARATION (6)	T+ 67/68
SOLID MOTOR BURNOUT (3)	T+ 130
SOLID MOTOR SEPARATION	T+ 133
MAIN ENGINE CUT OFF (MECO)	T+ 265
STAGE II IGNITION	T+ 278
FAIRING SEPARATION	T+ 302
SECOND STAGE CUT OFF (SECO)	T+ 620
STAGE II ENGINE RESTART	T+ 3255
SECO II	T+ 3311 (55 MIN)

APPENDIX B

(Orbital Analysis)

An explanation of the supporting calculations for the orbital dynamics of the Lunar Scout compose the appendix. The resulting data printed directly from the spreadsheet used is in Table B-1 on page B-10, and supporting diagrams are also included on pages B-13 to B-16. The main letter headings (ie A. Lunar Insertion) correspond directly with the headings on the spreadsheet, therefore it can be helpful to follow along on the spreadsheet while reading through the information below. A list of variables is also included on pages B-8 to B-9.

A. Lunar Insertion

The insertion altitude is 1366.7 km. At that altitude, the escape and circular orbit velocities are found using:

$$v_{ESC}=\sqrt{2\frac{\mu}{r_1}} \qquad v_{CIRC}=\sqrt{\frac{\mu}{r_1}}$$

where r_1 is the radius from the Earth's center to the insertion altitude and μ = 3.986 10^5 km^3/sec^2. By choosing an insertion velocity (v_1) slightly below the escape velocity, the lander will reach the lunar orbit. The insertion delta v is the difference between the insertion velocity and the circular orbit velocity. The path angle (γ_1) is an input and has a value of 0 .

B. Lunar Trajectory (2 Dimensions)

Using r_1, v_1, and γ_1, the energy (E), angular momentum (h), parameter (p), semi-major axis (a), and eccentricity (e) of the lunar trajectory can be calculated using the following:

$$E=\frac{v_1^2}{2}-\frac{\mu}{r_1} \qquad h=r_1 v_1 \cos\gamma_1 \qquad p=\frac{h^2}{\mu} \qquad a=-\frac{\mu}{2E} \qquad e=\sqrt{1-\frac{p}{a}}$$

The two dimensional lunar orbit is shown in Figure B-1. At the radius of the lunar orbit, r_2 = 384,400 km. At this radius, the velocity and path angle can be computed using the following:

$$v_2=\sqrt{2\left(E+\frac{\mu}{r_2}\right)} \qquad \cos\gamma_2=\frac{h}{r_2 v_2}$$

By finding the true anomalies at the insertion point (ν_1) and at the radius of the lunar orbit (ν_2), the transit angle (ϕ_f) between these two points can be found by taking the difference between the true anomalies. The true anomalies are found using:

$$\cos\nu=\frac{p-r}{re}$$

Next, the eccentric anomalies (E_1 and E_2) at r_1 and r_2 are determined with the following formula:

$$\cos E=\frac{e+\cos\nu}{1+e\cos\nu}$$

These values can be applied to find the time of flight (t_f) in seconds by doing the following computation:

$$t_f=\sqrt{\frac{a^3}{\mu}}\left[(E_2-e\sin E_2)-(E_1-e\sin E_1)\right]$$

C. Lunar Trajectory (3rd Dimension)

The third dimension to the two-body problem needs several parameters to be given. These are the launch latitude (β), the launch azimuth (ψ), the maximum declination of the moon (δ_m), and the transit angle from the launch point to the insertion altitude (ϕ_c). The three dimensional lunar trajectory is shown in Figure B-2. ϕ_c was estimated from Delta II rocket data by finding the arc that the Delta passed through from launch at the Eastern Test Range (β = 28.5 and ψ = 95) until it reached the Earth's equatorial plane. This was done using spherical trig to find the hypotenuse of a right spherical triangle using:

$$\phi_c=\tan^{-1}\left[\frac{\tan\beta}{\cos(180-\psi)}\right]$$

The total transit angle (ϕ_t) is merely the sum of ϕ_f and ϕ_c. Next the instantaneous declination of the Moon (δ) is calculated from:

$$\sin\delta=\sin\beta\cos\phi_t+\cos\beta\sin\phi_t\cos\psi$$

The angles eta (η) and alpha (α) must be calculated in order to find the intersection angle (i) between the Moon's orbital plane and the trajectory plane of the spacecraft. The following formulas are used:

$$\eta=\sin^{-1}\left(\frac{\cos\beta\sin\psi}{\cos\delta}\right) \qquad \alpha=\sin^{-1}\left(\frac{\cos\delta_m}{\cos\delta}\right) \qquad i=180-(\eta+\alpha)$$

The radius of lunar impact (r_i) is the difference between r_2 and the radius of the Moon (r_m) which is 1738 km. The velocity and path angle at r_i (v_i and γ_i) are calculated in the same method as v_2 and γ_2 were. Putting all of these variables together, it is possible to calculate the longitude and latitude of the lunar landing site (λ_m and β_m) with the following:

$$\tan\lambda_m = \frac{v_m - v_1\cos\gamma_1\cos i}{v_1\sin\gamma_1} \qquad \tan\beta_m = \frac{v_1\cos\gamma_1\sin i}{\sqrt{(v_1\sin\gamma_1)^2 + (v_m - v_1\cos\gamma_1\cos i)^2}}$$

where v_m is the velocity of the Moon relative to the Earth. v_m is equal to 1.018 km/s. In calculating this estimated landing site, it is assumed that the lunar trajectory passes through the center of the Moon, as shown in Figure B-3.

D. Patch Conic

The first thing needed for the patch conic approximation is the radius relative to the Earth at which the Lunar Scout enters the Moon's sphere of influence. The only method of doing this is to iterate along the lander's orbital trajectory. This iteration is listed in Table B-2 and is diagramed in Figure B-4.

To carry out the iteration, first a point is chosen along the lunar trajectory at a given radius from the Earth (R_1). The radius to the orbital path of the Moon is R_2. Using p, a, e, and γ_2 of the lunar trajectory, the values of v, ϕ, E (eccentric anomaly), and t_p are calculated between R_1 and R_2 using the same equations as in Section B. During the time t_p, not only does the lander approach point R_2, but the Moon does as well, through an angle E. Angle E is calculated by the following method:

$$E = t_p\left[\frac{R_2^{\,3}}{\mu}\right]^{-\frac{1}{2}}$$

The angle θ is then found to be the difference between E and ϕ.

Now, assuming that the spacecraft and the Moon travel along straight line paths during time t_p, a triangle is formed with an included angle which we can estimate to be γ_2. Using the angles and distances known above, the sides of the triangle (M, L, and S) can be solved for with the following formulas:

$$M=\sqrt{2R_2^2-2R_2^2\cos E} \qquad L=\sqrt{R_1^2+R_2^2-2R_1R_2\cos\phi} \qquad S=\sqrt{L^2+M^2-2LM\cos\gamma}$$

Side S of the triangle is the radius from the moon that the lander is at for a given point along the lunar trajectory. Thus by varying R_1, a point will be reached when S = 66,300 km, which is precisely the radius of the lunar sphere of influence. That given point at R_1 is the patch point.

At the patch point, the velocity and path angle (v_p and γ_p) of the Scout are found with respect to the Earth, in the same method as in Section B. A detailed diagram of the patch point is in Figure B-5. The angle θ is taken directly from the results of the iteration. With this information, the relative velocity (v_{REL}) between the spacecraft and the Moon can be calculated from:

$$v_{REL}=\sqrt{v_p^2+v_m^2-2v_pv_m\cos(\gamma_p-\theta)}$$

where again v_m is the velocity of the Moon relative to the Earth. The angle λ, taken from the moon, is the angle between the Earth and the patch point. ε is the angle between the velocity vector v_{REL} and the nadir line to the Moon from the patch point. λ and ε

are found in the following method:

$$\lambda = \sin^{-1}\left(\frac{\sin\theta R_1}{S}\right) \qquad \epsilon = \sin{-1}\left[\frac{v_m}{v_{REL}}\cos\lambda - \frac{v_P}{v_{REL}}\cos(\lambda+\theta-\gamma_P)\right]$$

Now it is possible to calculate the parameters of the trajectory with respect to the Moon and its gravitational field, in which $\mu_m = 4.9029\ 10^3\ km^3/sec^2$. These parameters are the energy (E_m), angular momentum (h_m), parameter (p_m), and eccentricity (e_m), and they are calculated in the following manner:

$$E_m = \frac{v_{REL}^2}{2} - \frac{\mu_m}{S} \qquad h_m = S v_{REL} \sin\epsilon \qquad p_m = \frac{h_m^2}{\mu_m} \qquad e_m = \sqrt{1 + 2E_m \frac{h_m^2}{\mu_m^2}}$$

E. Retrograde Burn

The most important factor in the retrograde burn is to insure that enough delta v will actually be provided to prevent the lander from literally crashing into the Moon. An initial estimate for the delta v was found by calculating the velocity that the Lunar Scout would attain if it were to actually reach the lunar surface with no retrograde burn whatsoever. This velocity was found using:

$$\Delta v_{RETRO} = \sqrt{2\left(E_m + \frac{\mu_m}{r_m}\right)}$$

where the radius of the Moon r_m is 1738 km. The Δv_{RETRO} has a value of 2.476 km/sec. The burn is initiated at an altitude of 200 km above the lunar surface. At this radius from the Moon (r_{RETRO}), the velocity of the lander is found from:

$$v_{RETRO}=\sqrt{2\left(E_m+\frac{\mu_m}{r_{RETRO}}\right)}$$

Subtracting the Δv_{RETRO} from v_{RETRO} results in the velocity of the Scout at burn completion. By finding the average velocity between the initiation and completion of the retro burn and multiplying this value by a given burn time, the distance of the burn can be found. Now it is possible to find the altitude at which the main retrograde burn is complete.

List of Variables

r_1	radius to insertion altitude
v_{ESC}	escape velocity at r_1
v_{CIRC}	circular orbit velocity at r_1
μ	Earth's gravitational parameter
v_1	insertion velocity
γ_1	path angle at r_1
E	energy of lunar trajectory
h	angular momentum of lunar trajectory
p	parameter of lunar trajectory
a	semi-major axis of lunar trajectory
e	eccentricity of lunar trajectory
r_2	radius of lunar orbit
γ_2	path angle at r_2
ν_1	true anomaly at r_1
ν_2	true anomaly at r_2
ϕ_f	transit angle (insertion to lunar orbit)
E_1	eccentric anomaly at r_1
E_2	eccentric anomaly at r_2
t_f	time of flight
β	launch latitude
ψ	launch azimuth
δ_m	maximum declination of Moon
ϕ_c	transit angle (launch point to insertion)
ϕ_t	total transit angle
δ	instantaneous declination of Moon
i	intersection angle of Moon's orbital plane and trajectory plane

r_i	radius of lunar impact
r_m	radius of Moon
v_i	velocity at r_i
γ_i	path angle at r_i
λ_m	longitude of lunar landing site
β_m	latitude of lunar landing site
v_m	velocity of Moon relative to Earth
R_1	variable radius of iteration (radius to patch point)
R_2	radius to orbital path of Moon for iteration
ϕ	transit angle (patch point to radius of lunar orbit)
t_p	time of flight from patch point to radius of lunar orbit
E	angle Moon tracks while spacecraft tracks angle ϕ
θ	angle between spacecraft at patch point and Moon
M	distance Moon travels during t_p
L	distance spacecraft travels during t_p
S	radius of lunar sphere of influence
v_p	velocity at patch point (R_1)
γ_p	path angle at patch point (R_1)
v_{REL}	relative velocity between spacecraft and Moon
λ	angle between Earth and patch point centered at Moon
ε	angle between v_{REL} and nadir to Moon at patch point
μ_m	gravitational parameter of Moon
E_m	energy of trajectory relative to Moon
h_m	angular momentum of trajectory relative to Moon
p_m	parameter of trajectory relative to Moon
e_m	eccentricity of trajectory relative to Moon
Δv_{RETRO}	delta v of retrograde burn
r_{RETRO}	radius from Moon of retro burn initiation
v_{RETRO}	velocity of spacecraft at r_{RETRO}

TABLE B-1

Orbital Mechanics Spreadsheet

A. Lunar Insertion

1	Insertion Altitude (input)	1366.7	km
2	Insertion Radius	7744.7	km
3	Escape Velocity @ A2	10.14568	km/s
4	Circular Orbit Velocity @ A2	7.174082	km/s
5	Insertion Velocity (input)	10.046	km/s
6	Insertion Delta V	2.871918	km/s

B. Lunar Trajectory (2 Dimensions)

1	Path Angle @ A2 (input)	0	deg
2	Energy	-1.00640	km^2/s^2
3	Angular Momentum	77803.26	km^2/s
4	Parameter	15186.52	km
5	Semi-major Axis	198033.1	km
6	Eccentricity	.9608919	
7	Radius of Lunar Orbit	384400	km
8	Velocity @ B7	.2471579	km/s
9	Path Angle @ B7	.6112751	rad
10	Cos of True Anomaly @ A2	1	
11	Cos of True Anomaly @ B7	-.999585	
12	Transit Angle (Insertion - B7)	178.3488	deg
13	Eccentric Anomaly @ A2	0	rad
14	Eccentric Anomaly @ B7	2.938222	rad
15	Flight Time	383041.5	s
16	Flight Time	106.4004	hr

C. Lunar Trajectory (3rd Dimension)

1	Launch Latitude (input)	28.5	deg
2	Launch Azimuth (input)	90	deg
3	Maximum Declination of Moon (input)	28.5	deg
4	Transit Angle (Launch - A2) (input)	89.2916	deg
5	Transit Angle (Total)	267.6404	deg
6	Instantaneous Dec. of Moon	-1.12564	deg
7	Eta	1.073733	deg
8	Alpha	1.073733	deg
9	Intersection Angle	56.95925	deg
10	Radius of Lunar Impact	382662	km
11	Velocity @ C10	.2655303	km/s
12	Path Angle @ C10	40.02917	deg
13	Longitude of Lunar Landing	79.33801	deg W of nadir
14	Latitude of Lunar Landing	10.46148	deg N of nadir

TABLE B-1
(continued)

D. Patch Conic

1	Radius of Patch Point	362284	km
2	Velocity @ D1	.4332312	km/s
3	Path Angle @ D1	1.052143	rad
4	Radius (Lunar Sphere of Influence)	66298.94	km
5	Theta	.1804655	rad
6	Lamda	1.374479	rad
7	Epsilon	-.225367	rad
8	Relative Velocity (S/C - Moon)	.8101666	km/s
9	Energy	.2542340	km^2/s^2
10	Angular Momentum	-12003.0	km^2/s
11	Parameter	29385.05	km
12	Eccentricity	2.011833	

E. Retrograde Burn

1	Altitude of Retrograde Burn (input)	200	km
2	Radius (from moon) of Burn	1938	km
3	Velocity at Burn Initiation	2.359701	km/s
4	Velocity at Burn Completion	-.116299	km/s
5	Delta V of Retrograde Burn (input)	2.476	km/s
6	Average Velocity During Burn	1.121701	km/s
7	Time of Burn (input)	134.89	s
8	Distance of Burn	151.3063	km
9	Altitude at Burn Completion	48.69374	km

TABLE B-2

Iteration: Finding Radius of Patch Point

Variable	Control	Run 1	Run 2
Radius 1 (input)	384400	362284	362283
Radius 2	384400	384400	384400
Parameter	15186.52	15186.52	15186.52
Semi-Major Axis	198033.1	198033.1	198033.1
Eccentricity	.9608919	.9608919	.9608919
Path Angle	.6112751	.6112751	.6112751
True Anomaly (nu) 1	3.112775	3.065087	3.065086
True Anomaly (nu) 2	3.112775	3.112775	3.112775
Transit Angle (phi)	0	.0476874	.0476890
Eccentric Anomaly 1	2.938222	2.612307	2.612296
Eccentric Anomaly 2	2.938222	2.938222	2.938222
Time of Transit	0	86125.61	86128.27
Angle E	0	.2281529	.2281599
Angle Theta	0	.1804655	.1804709
Distance M	.0078125	87511.86	87514.55
Distance L	.0078125	28385.76	28386.89
Distance S	.0047016	66298.94	66300.81

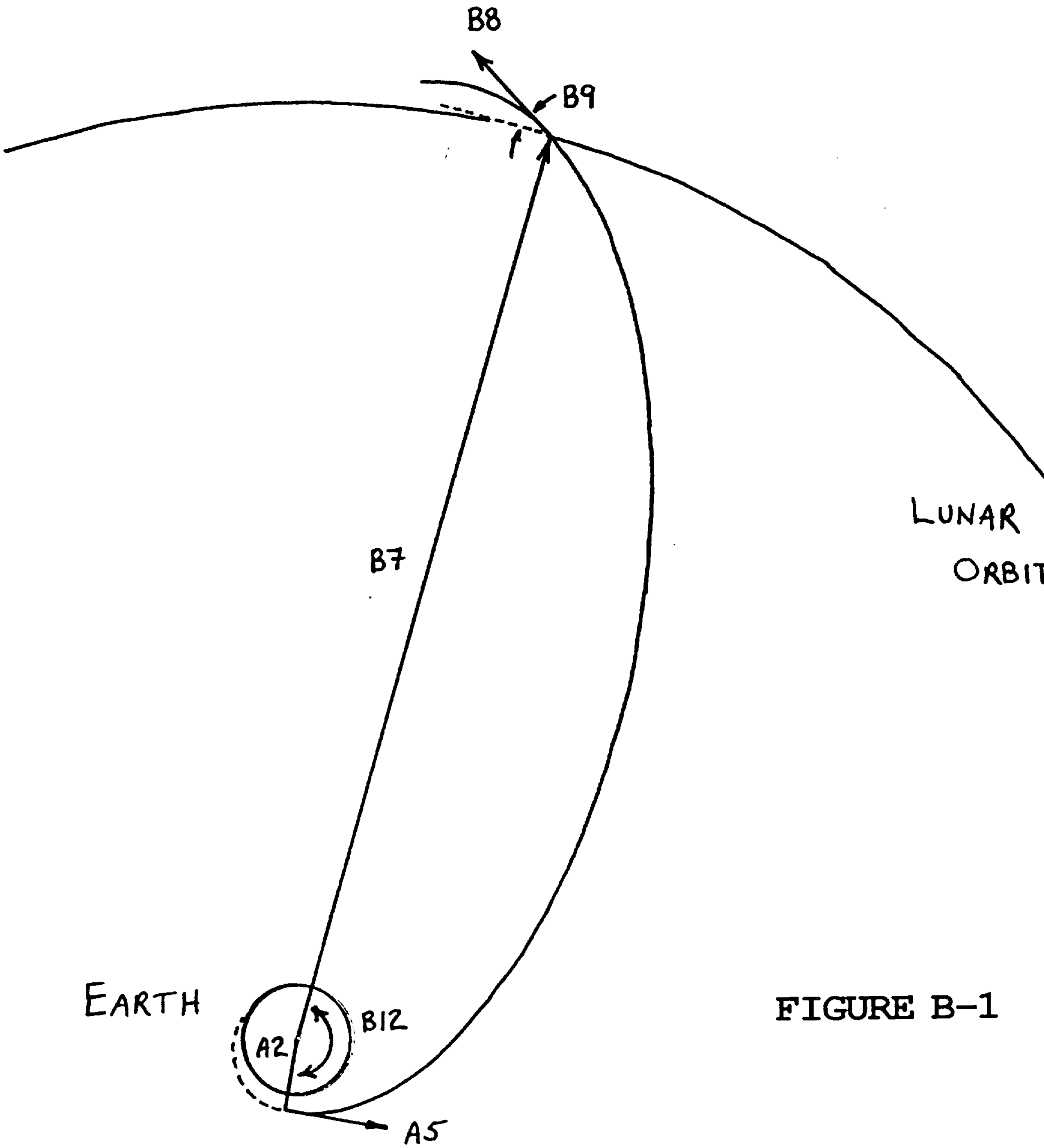
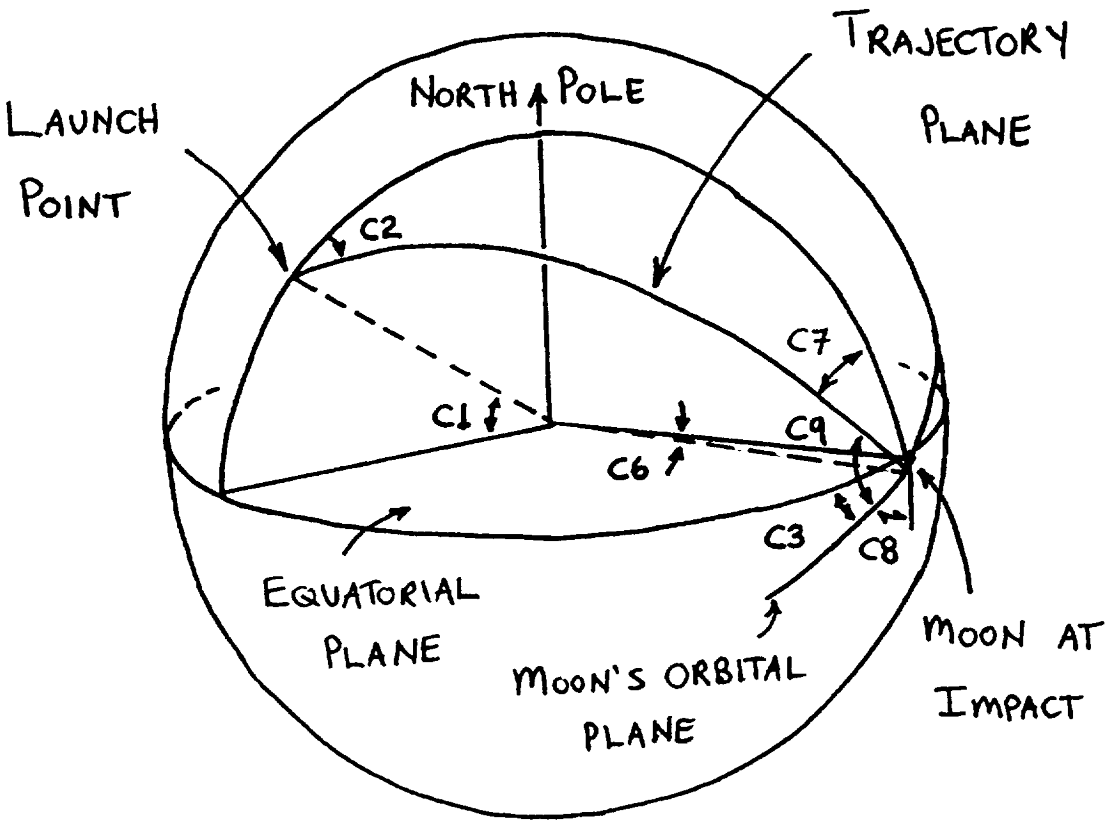
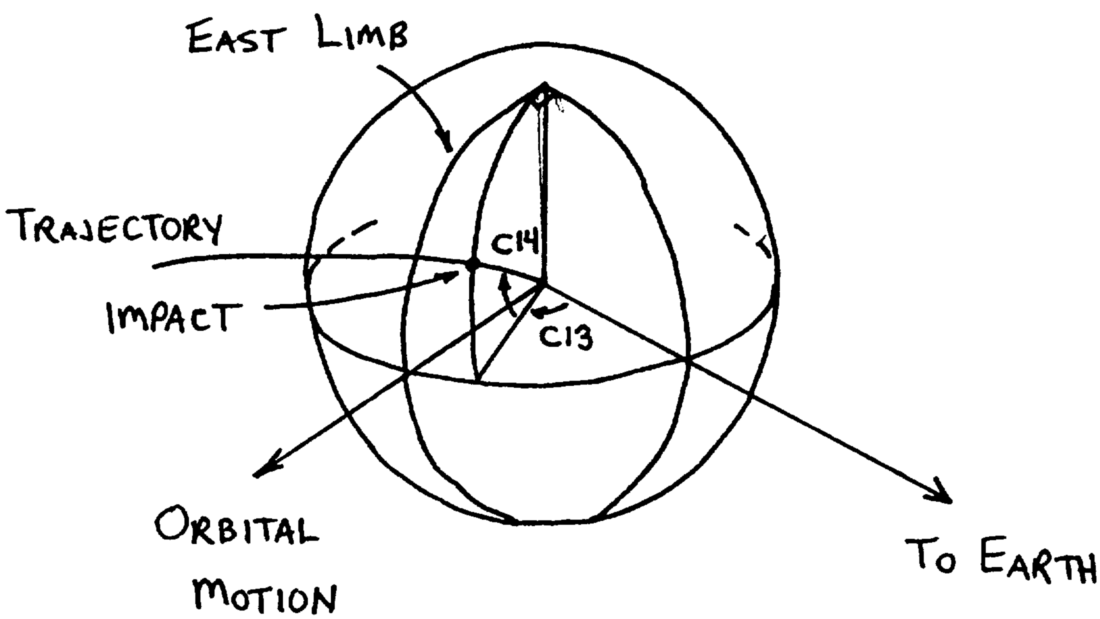


FIGURE B-1



TRAJECTORY PLANE
LAUNCH POINT
NORTH POLE
C2
C7
C1
C9
C6
C3
C8
EQUATORIAL PLANE
MOON'S ORBITAL PLANE
MOON AT IMPACT

FIGURE B-2



EAST LIMB
TRAJECTORY
C14
IMPACT
C13
ORBITAL MOTION
TO EARTH

FIGURE B-3

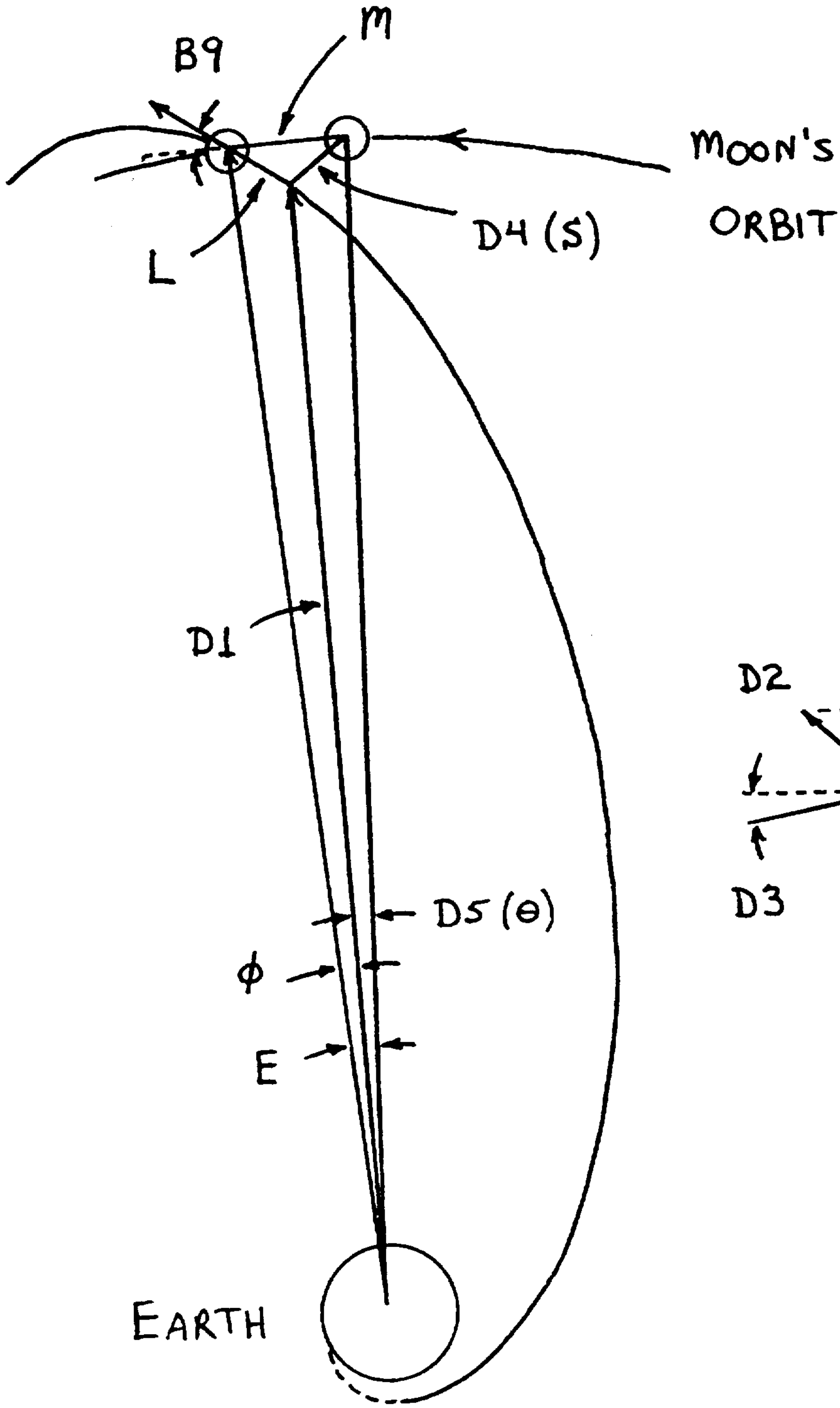


FIGURE B-4

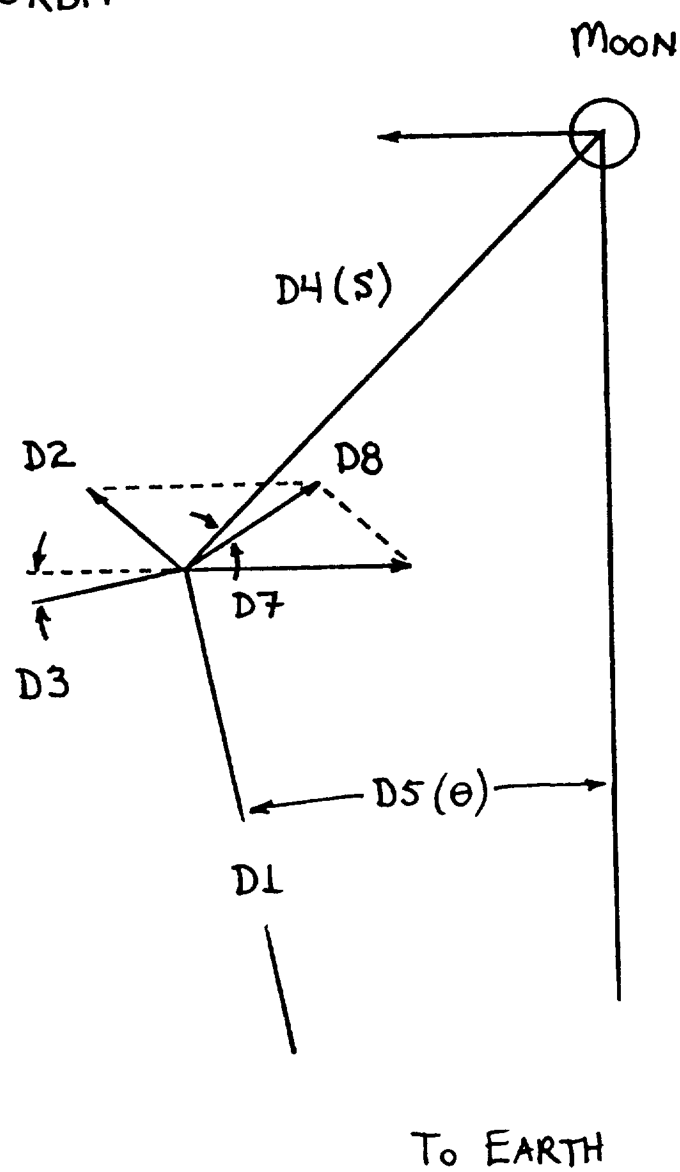


FIGURE B-5

References

Bate, Roger R., Mueller, and White. Fundamentals of Astrodynamics. New York: Dover Publications, Inc., 1971.

Koelle, Heinz Hermann. Handbook of Astronautical Engineering. New York: McGraw-Hill, Inc., 1961.

APPENDIX C
(Structure)

The material specifications were taken from the Mil-Hdbk 5C, p. 3-120 and from Space Mission Analysis and Design, p. 394.

6061-T6 Aluminum:

Structural Tubing	- Specifications WW-T-700/6 Tube, drawn Thickness 0.025 - 0.5 in.
Attachment Ring	- Specifications QQ-A-225/8 Special shapes < 8 in. thick
Density	- 2712.6 kg/m^3
Young's Modulus,E	- 67 * 10^9 N/m^2
Poisson's Ratio,ν	- 0.33

Equations for critical buckling stress and natural frequencies taken from Space Mission Analysis and Design by Wertz and Larson. From p. 404, the equation used to determine the critical buckling stress for a the aluminum tubing members in compression is:

$$\sigma_{cr} = \frac{k\pi^2 E}{12(1-\nu^2)}\left(\frac{t}{b}\right)^2$$

For the 6061-T6 Al tubing:

k = 200 (from Fig. 11-38, p. 405)
t = 0.003175 m.
b = 1.15 m.

$$\sigma_{cr} = \frac{200\pi^2(200N/m^2)}{12[1-(0.33)^2]}\left(\frac{0.003175m.}{1.15m.}\right) = 94.28\ MPa$$

The maximum stress for the worst case when the lander will touchdown on one landing pad is:

$$\sigma = \frac{P}{A} = \frac{m\frac{g}{6}}{A} = \frac{(530\ kg.)(\frac{9.81\ m/s^2}{6})}{2.85 \cdot 10^{-4}\ m^2} = 3.04\ MPa$$

The equations for the natural frequencies are found on p. 409. and are as follows:

$$f_{nat_{axial}} = 0.250\sqrt{\frac{AE}{ML}} \qquad f_{nat_{lateral}} = 0.560\sqrt{\frac{EI}{ML^3}}$$

For the Lunar Scout the axial frequency is:

$$f_{nat_{axial}} = 0.250\sqrt{\frac{(2.85 \cdot 10^{-4}\ m^2)(67 \cdot 10^9\ N/m^2)}{(4.14\ kg)(1.15\ m)}} = 500\ Hz.$$

The lateral frequency is:

$$f_{nat_{lateral}} = 0.560\sqrt{\frac{(67 \cdot 10^9\ N/m^2)(1.14 \cdot 10^{-3}\ m^4)}{(4.14\ kg)(1.15\ m)^3}} = 1951\ Hz.$$

The mass determination equations are as follows -

Leg tubing and vertical support members:

$$m = (13.14\ m)[\frac{\pi}{4}[(0.03175\ m)^2 - (0.0254\ m)^2]](2712.6\ kg/m^3) = 10$$

Horizontal and diagonal support members:

$$m = (20.46\ m)[\frac{\pi}{4}[(0.0254\ m)^2 - (0.01905\ m)^2]](2712.6\ kg/m^3) = 12$$

Shocks are approximately 3.33 kg. each from Surveyor

Three aluminum flex-core landing pads:

$$m = 3\left[\frac{\pi}{4}(0.275\ m)^2(0.05\ m)(112.36\ kg/m^3)\right] = 1.00\ kg.$$

Aluminum honeycomb panel under payload:

$$m = (4.6\ m^2)(0.01905\ m)(41.74\ kg/m^3) = 3.65\ kg.$$

Payload attachment ring:

$$A_{ring} = \frac{\pi}{4}[(1.3\ m)^2 - (1.1\ m)^2] - \frac{\pi}{4}(0.085\ m)^2 = 0.3713\ m^2$$

$$m = (0.3713\ m^2)(0.00625\ m)(2712.6\ kg/m^3) = 6.29\ kg.$$

References:

In addition to the above reference books, the following people from NASA's Goddard Space Flight Center were consulted for the project:

Dr. Steve Paddack

Dr. Stan Ollendorf

Mr. Scot Gordon

Much thanks goes out to these people, especially Dr. Paddack who coordinated and was our liason for many friday afternoon trips to NASA Goddard to get information and do research.

APPENDIX D1

THERMAL DESIGN CALCULATIONS

ELECTRONICS $0°C - 40°C$
BATTERIES $5°C - 20°C$
FUEL TANKS $7°C - 35°C$

LUNAR DAY $117°C$
EARTH ECLIPSE $-93°C$
NIGHT $-183°C$
709 hr DAY/NIGHT CYCLE

$$A \alpha_s Q_s = A \varepsilon_{IR} \sigma T^4$$

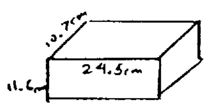
$$T = \left(\frac{A_{sun} \alpha_s Q + POWER_{INT}}{A_{RAD} \varepsilon \sigma} \right)^{1/4} =$$

BATTERIES AND ELECTRONICS ARE SHIELDED FROM SUNLIGHT 100% OF THE 2 YR. PERIOD

5 W CAN BE DISSIPATED FOR HEATING EACH COMPONENT

BATTERIES

RADIATING SURFACE AREA



$$\text{RADIATING SURFACE} = (10.7)(24.5) + 2(10.7)(11.6) + 2(24.5)(11.6)$$
$$= \frac{1078.8\,cm^2}{(100\,cm)^2} = .1079\,m^2$$

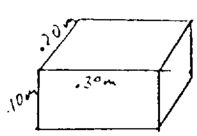
$$T = \left(\frac{P_{INT}}{(.1079\,m^2)(.035)(5.67 \times 10^{-8}\,\frac{W}{m^2 °K^4})} \right)^{1/4}$$

$\varepsilon_{eff} = .035$
FROM FIG 11-27 (LARSON)

$$P_{INT} = T^4 (.1079)m^2 (.035)(5.67 \times 10^{-8}\,\frac{W}{m^2K^4})$$

$T = 5°C$	$P_{INT} = 1.279\,W$
$T = 20°C$	$P_{INT} = 1.578\,W$

ELECTRONICS



ESTIMATE

$$\text{RADIATING SURFACE} = 2(.1)(.2)m^2 + (.2)(.3)m^2 + 2(.3)(.1)m^2$$
$$= 0.160\,m^2$$

$$P_{INT} = T^4 (.160)m^2 (.035)(5.67 \times 10^{-8}\,\frac{W}{m^2K^4})$$

$T = 0°C$	$P_{IN} = 1.76\,W$
$T = 40°C$	$P_{IN} = 3.04\,W$

SOURCES:

LARSON

AN INTRO TO THE STUDY OF THE MOON, ZDENEK KOPAL 1966, GORDON AND BREACH, 150 FIFTH AVE NY, NY 10011 p 343.

THERMO CALCULATIONS FOR A SINGLE SOLAR PANEL:

ASSUMPTIONS:

- Worst case: no incidence angle
- Moon albedo coefficient (a) = .2
- Solar flux only falls on solar cell side of panel.
- Moon albedo only acts on bottom side of solar panel.

Area of a single solar panel is $1.1 m^2$.
Area of fins on back of panel is:

$$A_{fin} = 15 \text{ fins} \times .02 \text{ m} \times 1 \text{ m} = 0.3 \text{ m}^2$$

$\therefore A_{TOP} = 1.1 m^2$, $A_{back} = 1.4 m^2$ (DIAGRAM ON NEXT PAGE)

THERMO EQUATION:

$$\text{POWER RADIATED} = \text{POWER ABSORBED} + \text{POWER } \cancel{\text{DISSAPATED}}^{\,0}$$

NONE IN PANEL

$$P_{rad,sc} + P_{rad,back} = P_{abs,sc} + P_{abs,back}$$

SEE TABLE BELOW FOR CONSTANTS

$$\epsilon_{sc}\sigma T^4 A_{TOP} + \epsilon_w \sigma T^4 A_{back} = A_{TOP}\alpha_s Q_s + A_{back}\alpha_w Q_m$$

$$(.83)(\sigma T^4)(1.1 m^2) + (.9)(\sigma T^4)(1.4 m^2) = (1.1 m^2)(.87)(1353 \text{ W/m}^2) + (1.4 m^2)(.2)(270.6 \text{ W/m}^2)$$

$$(2.173 m^2)\sigma T^4 = 1370.59 \text{ W}$$

$$T^4 = \frac{1370.59 \text{ W}}{(2.173 m^2)(5.67 \times 10^{-8} \frac{W}{m^2 \cdot K^4})}$$

$$T = 324.8 \text{ K} = 51.8 \text{ °C}$$

CALCULATED MAX OPERATING TEMPERATURE:

$$T = 51.8 \text{ °C}$$

	α	ϵ
Solar cells	.87	.83
white paint	.2	.9

Solar flux: $Q = 1353 \text{ W/m}^2$

Moon albedo: $Q_m = aQ_s = (.2)(1353)$

$Q_m = 270.6 \text{ W/m}^2$

$\sigma = 5.67 \times 10^{-8} \frac{W}{m^2 \cdot K^4}$ (Stephan-Boltzmann)

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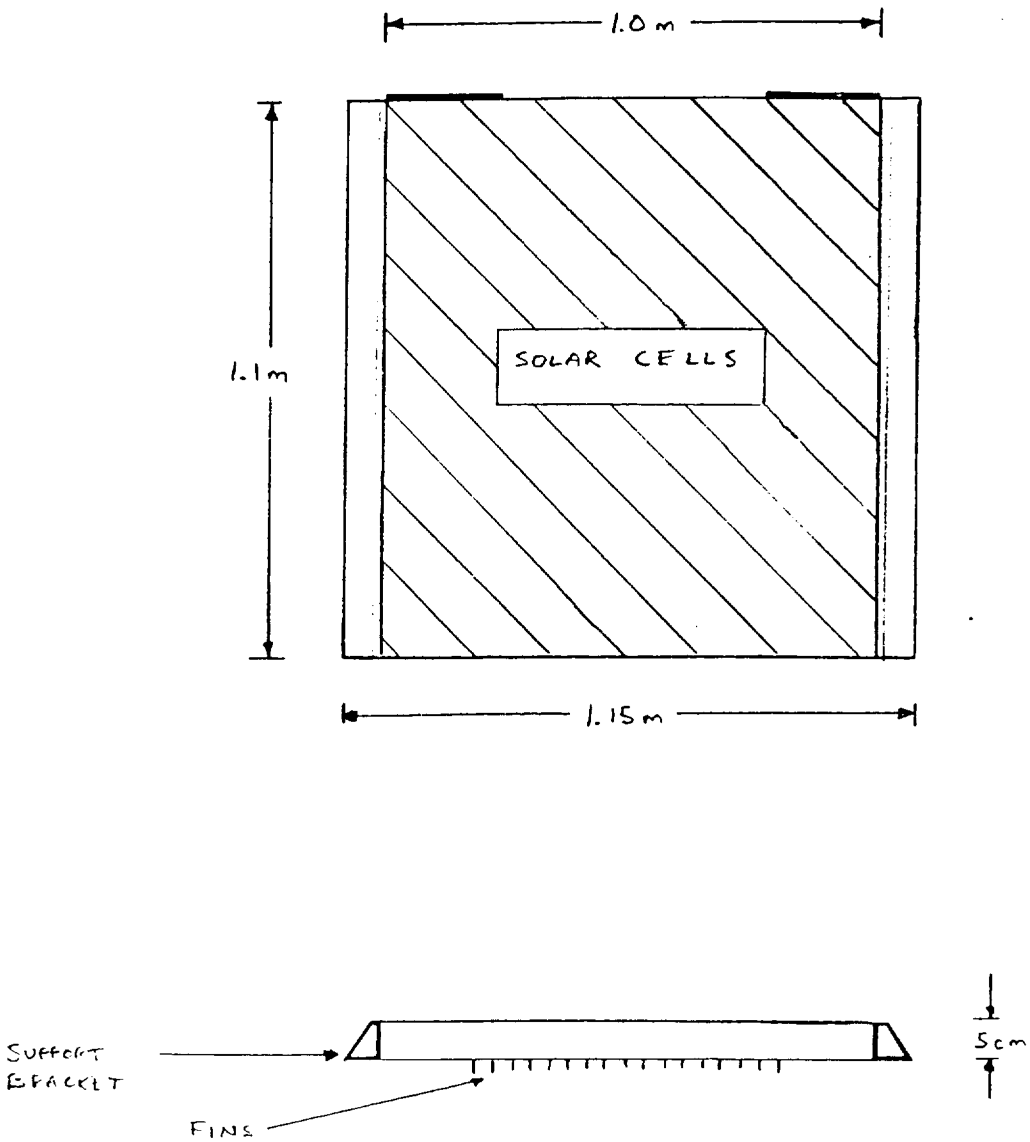
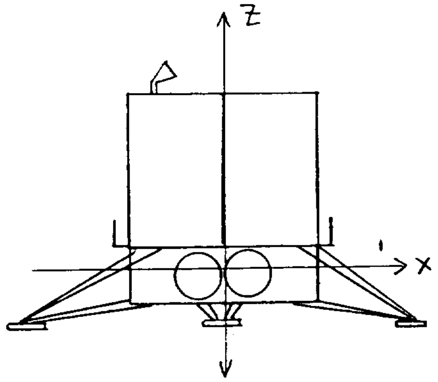
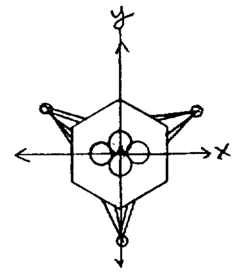
LUNAR SCOUT


1.0 m
1.1 m
SOLAR CELLS
1.15 m
SUPPORT BRACKET
FINS
5 cm

CENTER OF MASS AND MASS MOMENT OF INERTIA CALCULATIONS



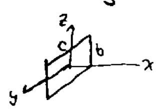


ASSUMPTIONS:

- TANKS ARE SOLID SPHERES
- STRUCTURAL COMP. ARE POINT MASSES
- SOLAR PANELS ARE POINT MASSES
- ELECTRONICS AND THRUSTERS ARE POINT MASSES

$I_{xx} = I_{yy} = I_{zz}$ SPHERE $= \frac{2mr^2}{5}$

THIN RECT. PLANE



$I_x = \frac{(b^2+c^2)}{12}$ $I_y = \frac{mc^2}{12}$ $I_z = \frac{mb^2}{12}$

MASS OF PROPULSION SUPPORT EQUIPMENT IS DIVIDED BY 4 AND ADDED TO FUEL TANKS.

$I_{FULL\ TANKS} = \frac{2(\text{FUEL} + \frac{1}{4}\text{ EQUIP})R^2}{5}$ $(\text{FUEL} + \frac{1}{4}\text{ EQUIP}) = 164.8$ $(\text{FUEL} + \frac{1}{4}\text{ OX}) = 246.64$

FUEL $= \frac{2(127.91 + 36.86)(.3126)^2}{5}$ OXIDIZER $= \frac{2(209.78+36.86)(.3126)^2}{5}$

$I_{F_F} = 6.441$ Kg m^2 $I_{F_O} = 9.64$ Kg m^2

$I_{EMPTY} = \frac{2(36.86)(.3126)^2}{5} = 1.44$ Kg m^2

PAYLOAD: $\frac{2mr^2}{5} = \frac{2(200)(.3)^2}{5}$ ESTIMATE AVG PAYLOAD RADIUS

FULL TANK MASS MOMENT OF INERTIA

A	B	C	D	E	F	G
ITEM		MASS (KG)	X (M)	Y (M)	Z (M)	M*X (kg-m)
FUEL TANK		164.88	.3126	.3126	.4472	51.54149
FUEL TANK		164.88	.3126	.3126	-.4472	51.54149
OX TANK		246.64	.3126	.3126	.4472	77.09966
OX TANK		246.64	.3126	.3126	-.4472	77.09966
ELECTRONICS		25	.8	.8	.836	20
SOLAR ARRAY		33	1.6	1.6	1	52.8
STRUCTURE		45	.5	.5	1	22.5
COMMUNICATIONS		18.184	.8	.8	1.15	14.5472
THERMAL		20.06	.5	.5	.8	10.03
PAYLOAD		200	.9	.9	0	180
ACS		20.06	.8	.8	1.04	16.048
		1184.344				573.2075
			Xc	Yc	Zc	
			.4839873	.4839873	0	

M*Z (kg-m)	Xc (m)	Yc (m)	xbar	ybar	Ixx (kg-m^2)	Iyy (kg-m^2)
73.73434	.4839873	.4839873	-.171387	-.171387	6.44	6.44
-73.7343	.4839873	.4839873	-.171387	-.171387	6.44	6.44
110.2974	.4839873	.4839873	-.171387	-.171387	9.64	9.64
-110.297	.4839873	.4839873	-.171387	-.171387	9.64	9.64
20.9	.4839873	.4839873	.3160127	.3160127	0	0
33	.4839873	.4839873	1.116013	1.116013	0	0
45	.4839873	.4839873	.0160127	.0160127	0	0
20.9116	.4839873	.4839873	.3160127	.3160127	0	0
16.048	.4839873	.4839873	.0160127	.0160127	0	0
0	.4839873	.4839873	.4160127	.4160127	7.2	7.2
20.8624	.4839873	.4839873	.3160127	.3160127	0	0
156.722						

I+mdx^2 (kg-m^2)	I+mdy^2 (kg-m^2)	I+mdz^2 (kg-m^2)
11.28312	11.28312	39.41400
11.28312	11.28312	39.41400
16.88471	16.88471	58.96500
16.88471	16.88471	58.96500
2.496600	2.496600	17.4724
41.10098	41.10098	33
.0115382	.0115382	45
1.815927	1.815927	24.04834
.0051435	.0051435	12.8384
41.81331	41.81331	7.2
2.003272	2.003272	21.69690
145.5824	145.5824	358.0140

EMPTY TANK MASS MOMENT OF INERTIA

A ITEM	B	C MASS (KG)	D X (M)	E Y (M)	F Z (M)	G M*X (kg-m)
FUEL TANK		36.86	.3126	.3126	.4472	11.52244
FUEL TANK		36.86	.3126	.3126	-.4472	11.52244
OX TANK		36.86	.3126	.3126	.4472	11.52244
OX TANK		36.86	.3126	.3126	-.4472	11.52244
ELECTRONICS		25	.8	.8	.836	20
SOLAR ARRAY		33	1.6	1.6	1	52.8
STRUCTURE		45	.5	.5	1	22.5
COMMUNICATIONS		18.184	.8	.8	1.15	14.5472
THERMAL		20.06	.5	.5	.8	10.03
PAYLOAD		200	.9	.9	0	180
ACS		20.06	.8	.8	1.04	16.048
		508.744				362.0149
			Xc	Yc	Zc	
			$\boxed{.7115857}$	$\boxed{.7115857}$	0	

M*Z (kg-m)	Xc (m)	Yc (m)	xbar	ybar	Ixx (kg-m^2)	Iyy (kg-m^2)
16.48379	.7115857	.7115857	-.398986	-.398986	1.44	1.44
-16.4838	.7115857	.7115857	-.398986	-.398986	1.44	1.44
16.48379	.7115857	.7115857	-.398986	-.398986	1.44	1.44
-16.4838	.7115857	.7115857	-.398986	-.398986	1.44	1.44
20.9	.7115857	.7115857	.0884143	.0884143	0	0
33	.7115857	.7115857	.8884143	.8884143	0	0
45	.7115857	.7115857	-.211586	-.211586	0	0
20.9116	.7115857	.7115857	.0884143	.0884143	0	0
16.048	.7115857	.7115857	-.211586	-.211586	0	0
0	.7115857	.7115857	.1884143	.1884143	7.2	7.2
20.8624	.7115857	.7115857	.0884143	.0884143	0	0
156.722						

I+mdx^2 (kg-m^2)	I+mdy^2 (kg-m^2)	I+mdz^2 (kg-m^2)
7.307728	7.307728	8.811552
7.307728	7.307728	8.811552
7.307728	7.307728	8.811552
7.307728	7.307728	8.811552
.1954273	.1954273	17.4724
26.04624	26.04624	33
2.014582	2.014582	45
.1421460	.1421460	24.04834
.8980561	.8980561	12.8384
14.29999	14.29999	7.2
.1568109	.1568109	21.69690
$\boxed{72.98416}$	$\boxed{72.98416}$	$\boxed{196.5022}$

APPENDIX F
(Communication)

ABBREVIATIONS

ANTEF:	antenna efficiency
BIRT:	bit rate
FREQ:	frequency
LMARG:	link margin
MISLS:	miscellaneous loss
NODEN:	noise density
PALS:	path loss
REGN:	receiver gain
REPW:	require power
RETRPW:	require transmitting power
SLTRA:	slant range
SYST:	communication system temperature
RADIA:	antenna diameter
TRGN:	transmitter gain
TRPW:	transmitting power
WAVLN:	wavelength

BUDGET LINK FOR GROUND STATION UPLINK

FREQ:	2.3E+09	Hz	SLTRA:	3.77E+08	m
WAVLN:	0.130435	m	RADIA:	4	m
ANTEF:	0.5		SYST:	1295	K

TRPW:	1	W	0	dBW
MISLS:			-4	dB
REGN:			0	dB
PALS:	1.32E+21		-211.203	dB
TRGN:	4640.899		36.66602	dB
LMARG:			-3	dB

REPW:			-181.537	dB
NODEN:	1.79E-20	W/HzK	-197.479	dB/Hz
BIRT:	10	bps	10	dB
E/No:			5.941384	dB
RETRPW:			4.527532	W

BUDGET LINK FOR TELE/COMM DOWNLINK

FREQ:	2.3E+09	Hz	SLTRA:	3.77E+08	m
WAVLN:	0.130435	m	RADIA:	4	m
ANTEF:	0.5		SYST:	552	K

TRPW:	1	W	0	dBW
MISLS:			-4	dB
TRGN:			0	dB
PALS:	1.32E+21		-211.203	dB
REGN:	4640.899		36.66602	dB
LMARG:			-3	dB

REPW:			-181.537	dB
NODEN:	7.62E-21	W/HzK	-201.182	dB/Hz
BIRT:	10	bps	10	dB
E/No:			9.644691	dB
RETRPW:			1.929882	W
RETRPW WITH LOSS:			5.789647	W

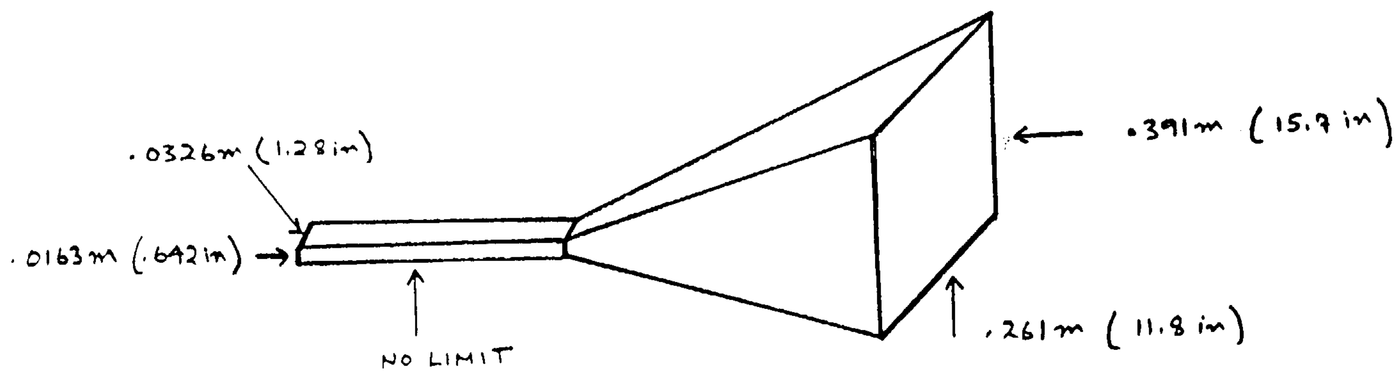
BUDGET LINK FOR DATA DOWNLINK

FREQ:	2.3E+09	Hz	SLTRA:	3.77E+08	m
WAVLN:	0.130435	m	RADIA:	4	m
ANTEF:	0.5		SYST:	552	K

TRPW:	6.5	W	8.129134	dBW
MISLS:			-4	dB
TRGN:			15	dB
PALS:	1.32E+21		-211.203	dB
REGN:	4640.899		36.66602	dB
LMARG:			-3	dB
REPW:			-158.408	dB
NODEN:	7.62E-21	W/HzK	-201.182	dB/Hz
BIRT:	1000	bps	30	dB
E/No:			12.77382	dB
FIELD OF VIEW:			23.4	deg.
			25.5	deg.

TRANS. POWER
WITH LOSS: 19.5 W

DIMENSION OF FEEDHORN ANTENNA



COMMUNICATION SUBSYSTEM DIMENSIONS

TELE/COMM COMMUNICATION SUBSYSTEM (TWO OMNI-DIRECTIONAL ANTENNA)

TOTAL VOLUME = 9168 cm3 (14X33X14cm and 15X30X6cm)
TOTAL MASS = 18 kg.

DATA COMMUNICATION SUBSYSTEM (ONE FEEDHORN ANTENNA)

TOTAL VOLUME = 9168 cm3 (14X33X14cm and 15X30X6cm)
TOTAL MASS = 20 kg.
DIMENSIONS OF FEEDHORN: .391m x .261m

REFERENCES

Larson, W. J. & Wertz, J. R. (1991). Space Mission Analysis and Design. Boston, Kluwer Academic Publishers.

Interview with Mr. Buruniga and Mr. Daniel throughout the course.

PROPELLANT FOR ΔV MANEUVERS

LUNAR INSERTION AT 1366.7 KM   SPACECRAFT MASS = 3775 Kg

SOLID ROCKET MOTOR : STAR 48 A SHORT NOZZLE

$M_{PROPELLANT} = 2427.97$ Kg

$M_{TOT} = 2574.1$ Kg

$I_{sp} = 283.4$ s

$$\Delta V_i = I_{sp}\, g_o \ln \left(\frac{M_{TOT}}{M_{TOT} - M_{PROP}} \right)$$

$$\Delta V_i = (283.4\,s)(9.81\,\tfrac{m}{s^2}) \ln \left(\frac{3775\,kg}{3775\,kg - 2429.99\,kg} \right)$$

$$\boxed{\Delta V_i = 2869.11\ m/s}$$

RETROGRADE BURN

ΔV REQUIRED FOR LUNAR CAPTURE = 2476.11 m/s

LIQUID ROCKET ENGINE : ROCKETDYNE XLR-132

$T = 1.67 \times 10^9$ N   OPERATING LIFE

$I_s = 340$ s   ENGINE MASS = 51.26 Kg

PROPELLANT = N_2O_4 / MMH

$M_F = 530.08$ [FINAL MASS ON LUNAR SURFACE]

$$M_p = M_f \left[e^{\Delta V / 340\, g_0} - 1 \right]$$

$$M_p = (530.08\,g)\left(e^{2476.11\,m/s / 340s \cdot 9.81\,m/s} - 1\right)$$

$M_F' = (583.58)(1.08)$ [ADD 8% FUEL MASS FOR ATTITUDE CONTROL AND LANDING ΔV]

$$M_p' = 576.77\,kg \left(e^{2476.11\,m/s / 340s\ 9.81\,m/s^2} - 1\right)$$

$M_p' = 634.97$ Kg - NOMINAL PROPELLANT LOAD

ADD .5% FOR OFF NOMINAL PERFORMANCE

ADD .25% FOR OFF NOMINAL OPERATIONS

ADD 5% MISSION MARGIN (RESERVES)

ADD 5% LOADING UNCERTANTY

$\boxed{\text{TOTAL PROPELLANT LOAD} = 675.36\ \text{Kg}\ N_2O_4/\text{MMH}}$

$$\frac{O}{F} = \frac{M_{OXID}}{M_{FUEL}} = 1.64 \quad \text{(MIXTURE RATIO)}$$

$$675.36\ kg = x_{kg_f} + x(1.64)\,kg_{ox}$$

$\boxed{255.82\ \text{Kg MMH} \quad 419.54\ \text{Kg}\ N_2O_4}$

TANKS:

255.82 Kg MMH $\left(\frac{1g}{cm^3}\right) = 255{,}800\,cm^3$

2 TANKS = $127{,}900\,cm^3$ ea

VOL = $127{,}900 = 4/3\,\pi R^3$

$\boxed{R = 31.26\,cm}$

419.54 Kg $N_2O_4 \left(\frac{1.64\,g}{cm^3}\right) = 255\,817\,cm^3$

2 TANKS = $127\,909\,cm^3 = \frac{4}{3}\pi R^3$

VOL = $4/3\,\pi R^3$

$\boxed{R = 31.26\,cm}$

ALTITUDE TO BEGIN RETROGRADE ORBIT:

$$t = \frac{I_s m g_o}{T} = \frac{(340\,s)(675.36\,Kg)(9.81\,\frac{kg\,m}{s^2})}{1.67\times10^{9}\,N} = 134.89\,s$$

$V = V_o + at$

$10\,m/s = 2486.11\,\frac{m}{s} + a\,(134.89\,s)$

$a = -18.36\,m/s^2$

$2a\,\Delta x = V^2 - V_o^2$

$2(-18.36\,\frac{m}{s^2})(\Delta x) = -(2486.11\,\frac{m}{s})^2 + (10\,\frac{m}{s})^2$

$$\boxed{\Delta X_{RETRO} = 168.35\,Km}$$

ΔV CAPABILITY OF REMAINING FUEL

$\Delta V = I_s g_o \ln\left(\frac{W_o}{W_o - W_f}\right)$ USING 4 — 4.5 N THRUSTERS

$I_{sp} = 235$

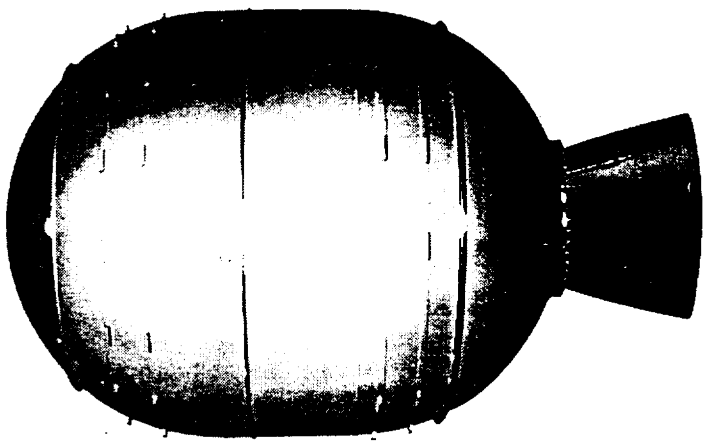
$\Delta V = (235\,s)(9.81\,\frac{m}{s^2}) \ln\left(\frac{584.11}{584.11 - 54.03}\right)$

$\boxed{\Delta V = 223.76\,m/s}$ FOR ATTITUDE CONTROL AND LANDING

Thiokol CORPORATION

TACTICAL OPERATIONS
Elkton Division

★ **STAR 48A**
TE-M-799-1
87.2-KS-17,335
SHORT NOZZLE
UPPER-STAGE MOTOR



The STAR 48A motor is designed to fit within the dimensional envelope of the long-nozzle STAR 48B (TE-M-711-18) and to provide increased payload capability. The design incorporates a longer cylindrical section in the motor case.

MOTOR PERFORMANCE (75°F Vacuum)

Burn Time/Action Time, sec	87.2/88.2
Ignition Delay Time, sec	0.100
Burn Time Average Chamber Pressure, psia	543
Action Time Average Chamber Pressure, psia	541
Maximum Chamber Pressure, psia	607
Total Impulse, lbf-sec	1,528,409
Propellant Specific Impulse, lbf-sec/lbm	285.3
Effective Specific Impulse, lbf-sec/lbm	283.4
Burn Time Average Thrust, lbf	17,335
Action Time Average Thrust, lbf	17,270
Maximum Thrust, lbf	21,140

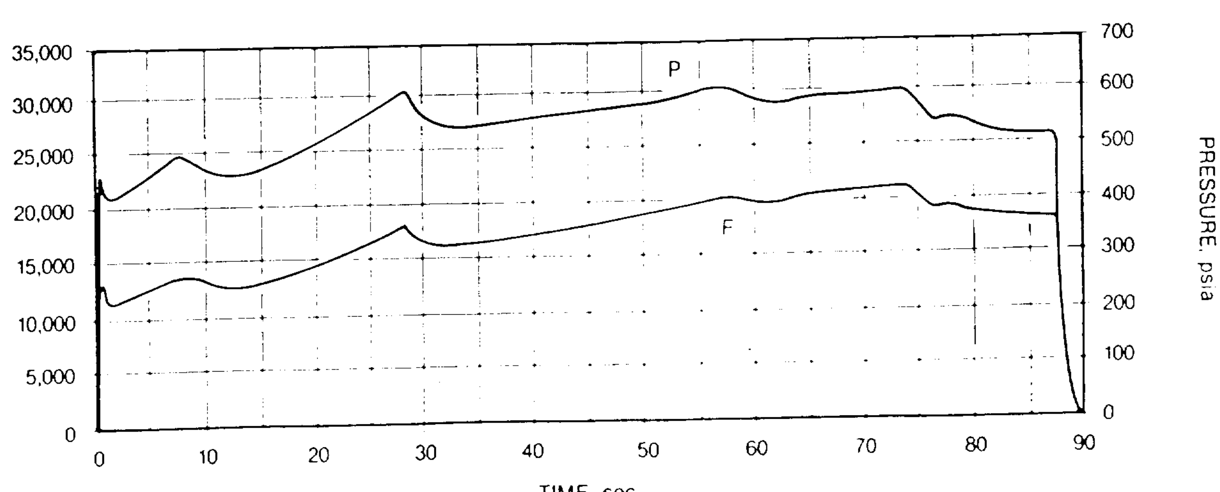
WEIGHTS, lbm

Total Loaded*	5673.7
Propellant (including igniter propellant)	5357.2
Case Assembly	153.6
Nozzle Assembly (excluding igniter propellant)	84.4
Internal Insulation	73.1
Liner	2.7
Miscellaneous	2.7
Total Inert (excluding igniter propellant)	316.5
Burnout*	280.0
Propellant Mass Fraction*	0.944
S&A/ETA	3.7

*Excluding remote S&A/ETA

TEMPERATURE LIMITS

Operation	30 to 100°F
Storage	30 to 100°F

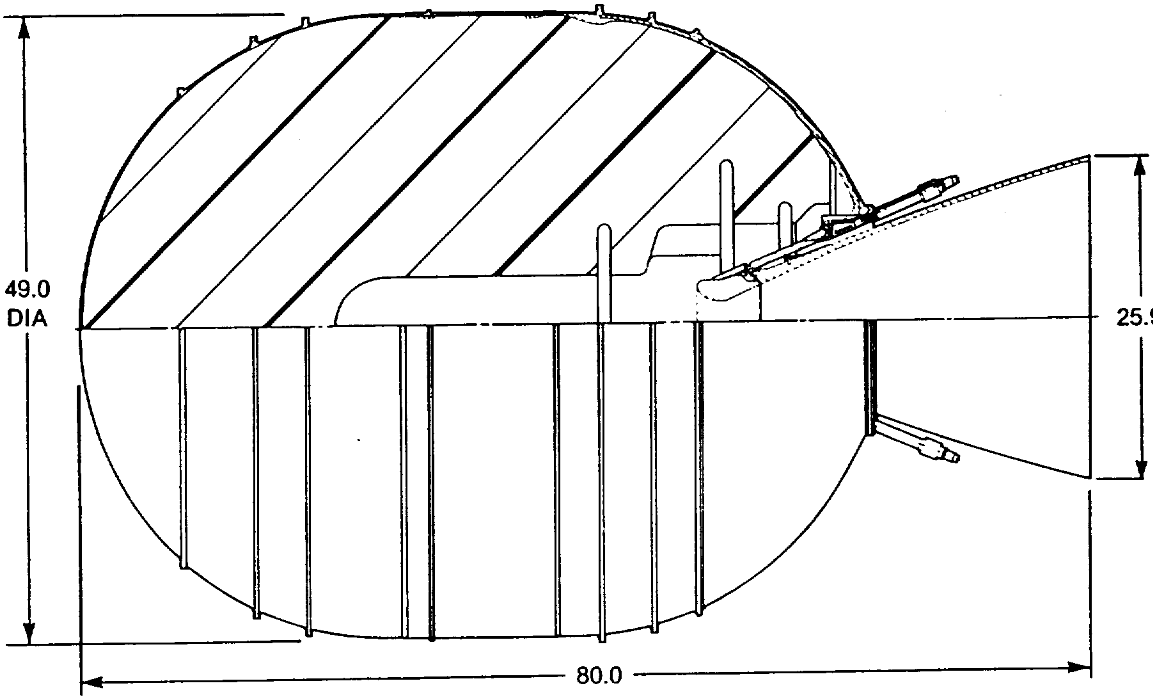


BC13560 4/91

G-2 C-2

★ **STAR 48A**
TE-M-799-1
87.2-KS-17,335
SHORT NOZZLE
UPPER-STAGE MOTOR

Thiokol CORPORATION
TACTICAL OPERATIONS
Elkton Division



CASE

Material	6Al-4V Titanium
Minimum Ultimate Strength, psi	165,000
Minimum Yield Strength, psi	155,000
Hydrostatic Test Pressure, psi	732
Minimum Burst Pressure (70°F), psi	860
Hydrostatic Test Pressure/Maximum Pressure	1.05
Burst Pressure/Maximum Pressure	1.25
Nominal Thickness, in.	0.069

NOZZLE

Exit Cone Material	Carbon-phenolic
Throat Insert Material	3D carbon-carbon
Initial Throat Diameter, in.	4.49
Exit Diameter, in.	25.06
Expansion Ratio, Initial/Average	31.2/27.3
Expansion Cone Half Angles, Exit/Eff, deg	16.5/17.1
Type	Fixed, contoured
Number of Nozzles	1

LINER

Type	TL-H-318
Density, $lbm/in.^3$	0.038

IGNITION TRAIN

Components	S&A/ETA/TBI/flamestick initiator/ toroidal pyrogen
Minimum Firing Current per Detonator, amperes	5.0
Circuit Resistance per Detonator, ohms	1.1
No. of Detonators and TBIs	2

PROPELLANT

Propellant Designation and Formulation	TP-H-3340
AP — 71%	
Al — 18%	
HTPB Binder — 11%	

PROPELLANT CONFIGURATION

Type	Internal-burning, radial-slotted star with head-end web
Web, in.	20.42
Web Fraction, %	84
Sliver Fraction, %	0
Propellant Volume, $in.^3$	81,951
Volumetric Loading Density, %	93.1
Web Average Burning Surface Area, $in.^2$	4,004
Initial Surface to Throat Area Ratio	180.7

PROPELLANT CHARACTERISTICS

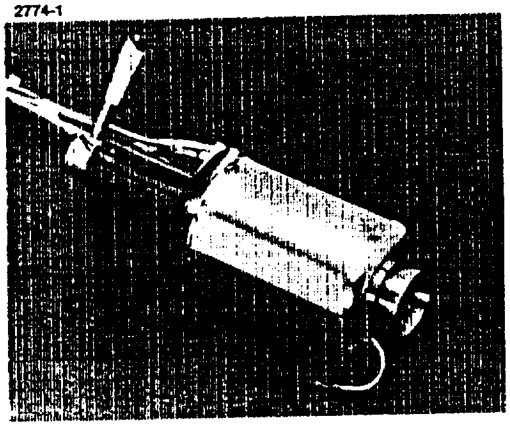
Burn Rate at 500 psia, in./sec, 60°F	0.228
Burn Rate Exponent	0.30
Density, $lbm/in.^3$	0.0651
Temperature Coefficient of Pressure, %/°F	0.10
Characteristic Exhaust Velocity, ft/sec	5,010
Adiabatic Flame Temperature, °F	6,113
Effective Ratio of Specific Heats (Chamber)	1.14
(Nozzle Exit)	1.18

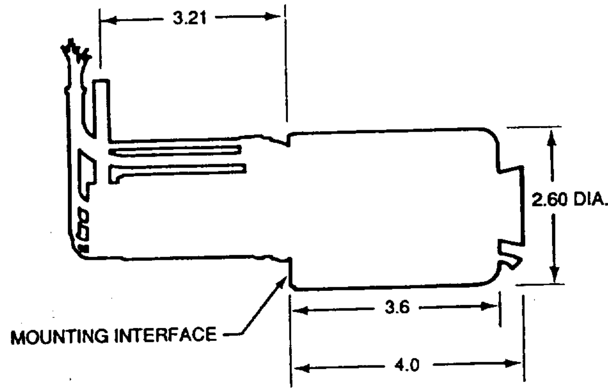
CURRENT STATUS

Development

MR-50L
5-lbf ENGINE

2774-1





Design Characteristics

- ☐ Propellant .... Hydrazine
- ☐ Catalyst .... Shell 405
- ☐ Thrust/Steady State (lbf) .... 5.0—2.2
- ☐ Feed Pressure (psia) .... 245—85
- ☐ Chamber Pressure (psia) .... 106—43
- ☐ Expansion Ratio 40:1
- ☐ Flow Rate (lbm/sec) .... 0.0216—0.0103
- ☐ Valve .... Wright Components Dual Seat
- ☐ Valve Power .... 29 watts max. at 33 vdc & 35°F
- ☐ Weight (lbm) 1.50
- ☐ Engine 0.90
- ☐ Valve 0.60

Demonstrated Performance

- ☐ Specific Impulse (lbf-sec/lbm) .... 225—215
- ☐ Total Impulse (lbf-sec) 11,394
- ☐ Total Pulses 12,300
- ☐ Minimum Impulse Bit (lbf-sec) .... 0.075 @ 245 psia & 22 ms ON
- ☐ Steady-State Firing (sec) 600

Flight Status

Program	Customer/User
Global Positioning Satellite (GPS)	Rockwell/USAF

ROCKET RESEARCH COMPANY
Olin AEROSPACE DIVSION

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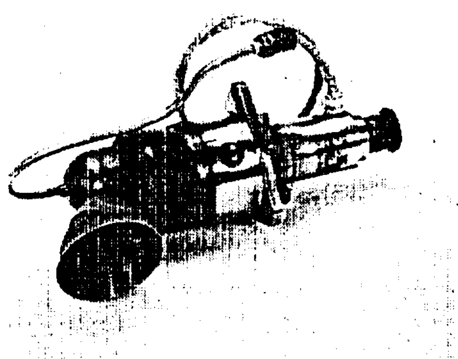
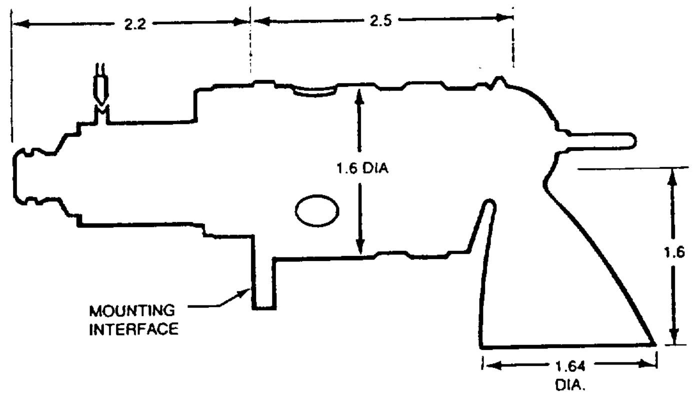
MR-120
20-lbf ENGINE





Design Characteristics

- ☐ Propellant .... Hydrazine
- ☐ Catalyst .... LCH 207/LCH 202
- ☐ Thrust/Steady State (lbf) .... 25 — 9
- ☐ Feed Pressure (psia) .... 355 — 105
- ☐ Chamber Pressure (psia) .... 115 — 45
- ☐ Expansion Ratio 15:1
- ☐ Flow Rate (lbm/sec) .... 0.11 — 0.043
- ☐ Valve .... Moog Single Seat
- ☐ Valve Power .... 45 Watts @ 32 vdc and 40°F
- ☐ Weight (lbm) 0.90
  - Engine 0.63
  - Valve 0.27

Demonstrated Performance

- ☐ Specific Impulse (lbf-sec/lbm) .... 229 — 222
- ☐ Total Impulse (lbf-sec) 8,289
- ☐ Total Pulses 1,911
- ☐ Minimum Impulse Bit (lbf-sec) .... 0.22 @ 325 psia & 11 ms ON
- ☐ Steady-State Firing (sec) 150

Flight Status

Program	Customer/User
Small ICBM	Martin Marietta/USAF

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Olin AEROSPACE DIVSION

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APPENDIX H:
(POWER)

POWER BUDGET

In flight:

-- computer :	50 W	(est)
-- star tracker :	10 W	
-- sun sensors :	0 W	
-- transmitter :	5 W	
-- radar :	135 W	

Lunar surface:

-- payload :	240 W to 750 W
-- battery :	10 W
-- nighttime :	10 W

Gallium Arsenide on Germanium Solar Cells

CELL SPECIFICATIONS

Test Conditions: AMO, 135.3 mW/cm², 28°C

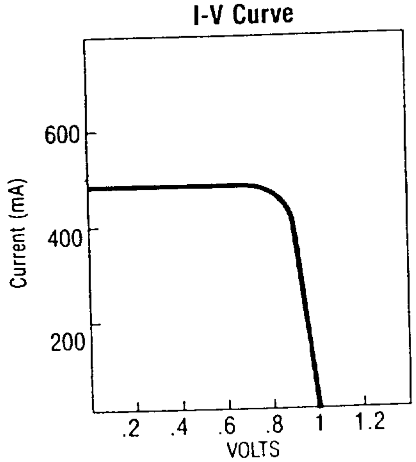
Parameter	Symbol	Unit	22 Ge-200	24 Ge-200	24 Ge-90	44 Ge-200	44 Ge-90
Cell Size	—	cm	2 x 2	2 x 4	2 x 4	4 x 4	4 x 4
Cell Thickness	—	µm	200	200	90	200	90
Efficiency	η	%	18.5	18.5	18.0	18.5	18.0
Open Circuit Voltage	Voc	V	1.000	1.000	1.000	1.000	1.000
Short Circuit Current	Isc	mA	122	244	242	488	484
Optimum Load Voltage	VL	V	0.870	0.870	0.865	0.870	0.865
Load Current	IL	mA	115	230	225	460	450
Maximum Power	Pmax	mW	100	200	195	400	390
Weight	—	gm	0.454	0.908	0.425	1.816	0.850
Solar Absorptance**	αs	—	0.870	0.870	0.870	0.870	0.870

**350 nm filter

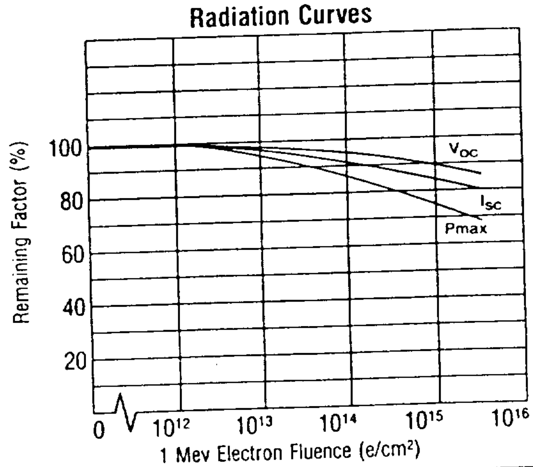
CELL CHARACTERISTICS

Test Conditions: 4 x 4 cm² x 90 µm

I-V Curve



Radiation Curves



Temperature Coefficients (28°C to 60°C)

Parameter	%/°C
Voc	-0.194
Isc	+0.056
VL	-0.232
Pmax	-0.223

Applied Solar Energy Corporation
15251 East Don Julian Road • City of Industry, California 91749
Phone (818) 968-6581 • FAX (818) 336-8694

Gallium Arsenide On Germanium Space Solar Panels

Applied Solar Energy Corporation produces GaAs/Ge panels in a variety of efficiency configurations. The following power and mass information is presented for two efficiency ratings as a guide to assist in determining the optimum panel configuration for your application. For additional information, please contact Applied Solar Energy Corporation at the address and phone numbers listed below.

How To Put ASEC To Work On Your Next Project

Putting the ASEC team to work on your next project is easy. Just give Marketing a call at the phone number listed below and ASEC:

1. Obtains all the preliminary information needed to get started via a simple questionnaire.
2. Develops panel layout drawings.
3. Develops B.O.L. and E.O.L. data.
4. Provides you with data and pricing.

You are cordially invited to give us a call today at:

1-818-968-6581

Power In Watts (W) Per One Square Foot of GaAs/Ge Solar Panel - AMO

Panel Type	Min. Avg. Cell Efficiency at 28°C	Beginning of Life Power (W) Un-irradiated			End of Life Power (W) 1 MeV Electron Fluence 3 x 10^{14} e/cm^2			End of Life Power (W) 1 MeV Electron Fluence 1 x 10^{15} e/cm^2		
		0°C	28°C	60°C	0°C	28°C	60°C	0°C	28°C	60°C
GaAs/Ge Single Junction	18.0%	20.7	19.5	18.1	18.0	17.0	15.7	15.5	14.6	13.5
	18.5%	21.3	20.0	18.6	18.5	17.4	16.2	16.0	15.0	14.0

Mass (gm) of Cell Stack For A One Square Foot GaAs/Ge Solar Panel

Component	Cell	Solder	Interconnect	Adhesive/Coverglass	Coverglass	Adhesive/Substrate	Total
Thickness	3.5 mils 2 x 4 cm	1.5 mils Zone	2 mils Ag Mesh	2 mils Bond Line DC93-500	6 mils CMX	4 mils Bond Line GE RTV-566	19 mils Stack
Mass (gm)	42.5	0.5	3.3	4.4	31.9	12.3	94.9

Note: Panel power and mass calculations are based upon 2 x 4 cm x 3.5 mils cells covered with 6 mils A/R coated CMX Coverglass, 100 cell assemblies per square foot.

ORIGINAL PAGE IS OF POOR QUALITY

Applied Solar Energy Corporation

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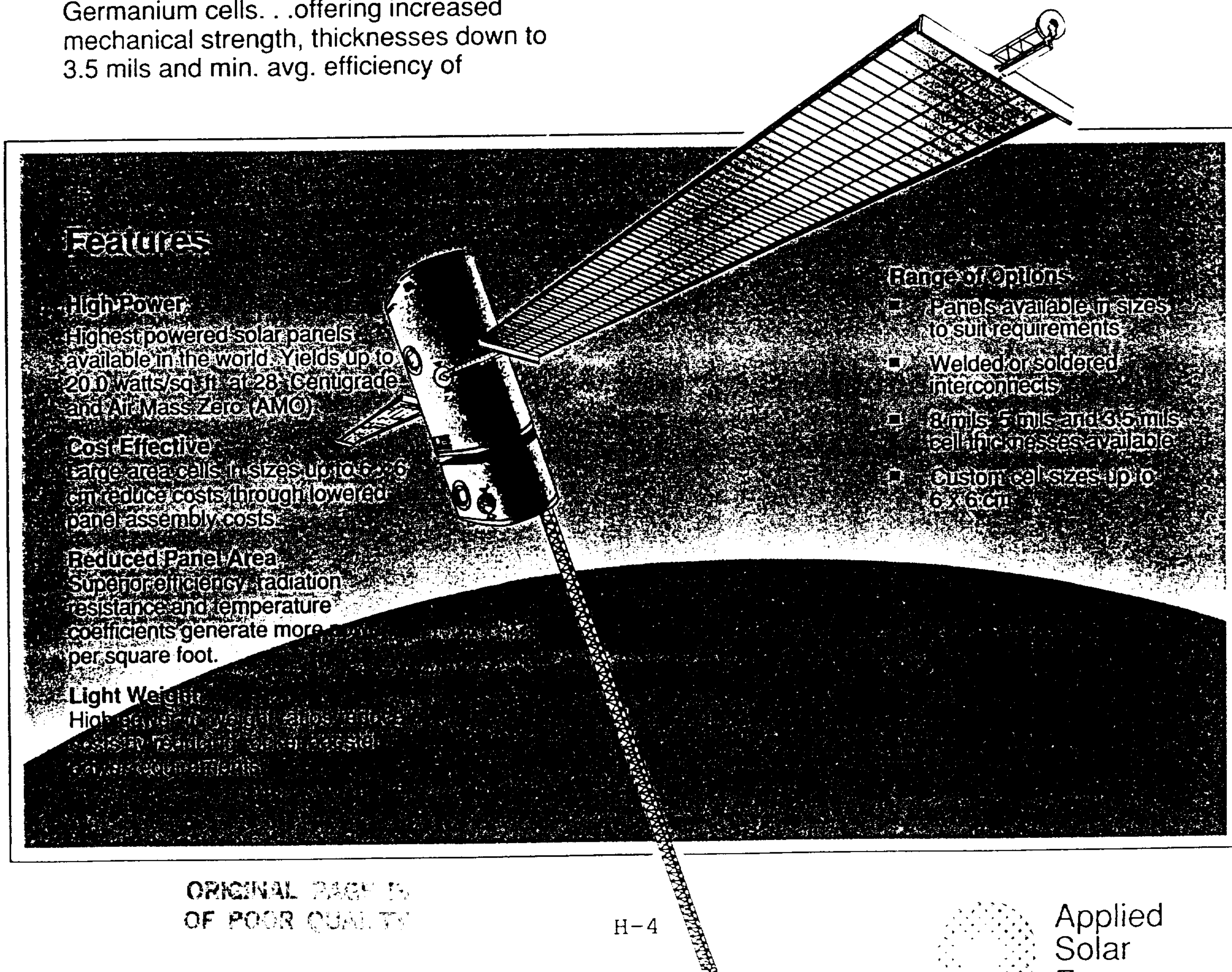
Gallium Arsenide On Germanium Space Solar Panels

In 1983 ASEC pioneered the use of Metal Organic Chemical Vapor Deposition (MOCVD) for the production of Gallium Arsenide (GaAs) solar cells, providing a 50% improvement in power. Today our Gallium Arsenide on Germanium cells. . .offering increased mechanical strength, thicknesses down to 3.5 mils and min. avg. efficiency of 18.5%...are ideal for satellite applications where maximum output power per area is required. Our GaAs facility is the world's largest with the capacity to produce over 25,000 2 x 4 cm cells or over 5KW per month.



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Telephone (818) 968-6581 • FAX (818) 336-8694

REFERENCES FOR POWER SUBSYSTEM

1. Assc. Prof. Walter Daniel, instructor for spacecraft design course.

2. Dr. Jim Severns, instructor for spacecraft design course.

3. Dr. Stephen Paddack, Chief, Advanced Missions Analysis Office, NASA Goddard Space Flight Center.

4. Dr. Stan Olendorf, NASA Goddard Space Flight Center.

5. Dr. John Day, Payload Engineering Section, NASA Goddard Space Flight Center.

6. Dr. Gopalakrishna Rao, Energy Storage Section, NASA Goddard Space Flight Center.

7. EA 463 Spacecraft systems course notes.

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