ICEBREAKER:

A LUNAR SOUTH POLE EXPLORING ROBOT

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Matthew C. Deans
Alex D. Foessel
Gregory A. Fries
Diana LaBelle
N. Keith Lay
Stewart Moorehead
Ben Shamah
Kimberly J. Shillcutt

Professor: Dr. William Whittaker

The Robotics Institute
Carnegie Mellon University
Pittsburgh PA 15213

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Executive Summary
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Due to the low angles of sunlight at the lunar poles, craters and other depressions in the polar regions can contain areas which are in permanent darkness and are at cryogenic temperatures. Many scientists have theorized that these cold traps could contain large quantities of frozen volatiles such as water and carbon dioxide which have been deposited over billions of years by comets, meteors and solar wind. Recent bistatic radar data from the Clementine mission has yielded results consistent with water ice at the South Pole of the Moon however Earth based observations from the Arecibo Radar Observatory indicate that ice may not exist. Due to the controversy surrounding orbital and Earth based observations, the only way to definitively answer the question of whether ice exists on the Lunar South Pole is *in situ* analysis.

The discovery of water ice and other volatiles on the Moon has many important benefits. First, this would provide a source of rocket fuel which could be used to power rockets to Earth, Mars or beyond, avoiding the high cost of Earth based launches. Secondly, water and carbon dioxide along with nitrogen from ammonia form the essential elements for life and could be used to help support human colonies on the Moon. Thirdly, since these volatiles have been accumulating for billions of years they can provide valuable information about the history of the Moon and cometary impacts. The discovery of volatiles on the Moon would radically change our outlook on the solar system and our ability to explore it.

A surface mission is the only way in which volatile deposits and their composition can be verified. Such a mission would have to survive the Lunar South Pole environment which is significantly different than at the equator. A robot would have to be able to operate in cryogenic temperatures and complete darkness while in the cold traps. However the surface temperature in the sun is quite mild compared to the baking heat of the equatorial regions. Communications with Earth are only possible for two weeks in a month, furthermore the Earth's low elevation allows communications to be obstructed by lunar terrain. The Sun is also very low on the horizon (which causes long shadows). While this low Sun angle creates the cold traps, it also allows for regions with almost permanent light - ideal for charging batteries and travel between cold traps. Further, with the Sun on the horizon, the shaded side of the robot as well as its top offer excellent surfaces for radiation of excess heat since they point to the black of space. Also solar panels do not have to track the Sun as it moves across the sky since it is always on the horizon. In general the pole is an environment which is more favorable to robots than other regions of the Moon.

Icebreaker is a robot design capable of finding ice at the Lunar South Pole. The goals of its mission are to confirm the presence of ice and map its local distribution, determine composition of the ice, determine the existence and nature of stratigraphy and finally to measure the composition of ice to a depth of one meter below the surface. To do this Icebreaker will visit at least two cold traps and take ten samples from each during a two week period during which communications with Earth are present.

An artists conception of Icebreaker is shown above. It is a four wheeled, all wheel drive robot with drive motors mounted inside the wheels. The front wheels of Icebreaker are Ackerman steered and the rear wheels are connected to the chassis through a bogie mechanism. Dust, kicked up by the wheels, is prevented from collecting on the solar panels and optics by fenders.

Due to mass and volume constraints in the Delta II 7925H rocket fairing, Icebreaker will be a combined lander/rover vehicle. This class of vehicles combines the functions of a traditional landing craft and rover into one vehicle. Thruster and fuel tanks are an integral part of Icebreaker and allow it to touchdown on its wheels with very little energy and no disposable shock absorbing structure. Thus Icebreaker will land on the Moon and then drive off in its quest for volatiles.

An inertial measurement unit (IMU) and a pair of star tracker cameras are used to fix the position of Icebreaker on the surface of the Moon, as well as during the flight through space. Radar altimeter and belly camera are used to control the landing. A panospheric camera, mounted on top of the solar panel is used to detect the horizon, as well as to return images to Earth. A pair of forward looking, horizontal baseline stereo cameras return stereo images to a human teleoperator. Strobe lights will be used when ambient light is insufficient for the cameras. A radar sensor is used to detect obstacles and permit the robot to travel autonomously.

To meet the mission goals of determining the composition of volatiles and determining stratigraphy, Icebreaker has a set of scientific instruments. An infrared camera, tuned to water ice, will be used to locate possible deposits of volatiles. After the robot closes on a potential deposit, a cryogenic sampling drill is used to collect a sample. This drill is capable of collecting samples up to a depth of 1m. The drill will then deposit the sample into a Regolith Evolved Gas Analyzer (REGA). The REGA will heat the sample, and using a mass spectrometer, determine the chemical and isotopic composition of the sample. Stratigraphy will be determined by taking and analyzing samples from various known depths.

Icebreaker will return scientific data and images to the Earth over a 10kbps, S-Band radio link. A switched array of limited aperture, high gain antennas, mounted on top of Icebreaker will be used to send and receive data to Earth through NASA's Deep Space Network. As a backup, a small omnidirectional low gain antenna is also included.

While Icebreaker does not have to deal with the extreme heat encountered at the equatorial regions of the Moon, the cryogenic temperatures inside the cold traps make heating an important issue. The main body of Icebreaker will be temperature controlled to preserve the computers, scientific instruments and batteries. This will be done through a series of heat pipes and insulation. Small quantities of RHU's will also be used to generate heat, particularly in isolated regions such as the video cameras.

Power will be generated using solar panels. A solar panel fin is mounted from bow to stern of Icebreaker. This fin is fixed and thus cannot be pointed towards the Sun. Batteries will be used to store electricity, both for journeys into cold traps and for driving when the Sun is not illuminating Icebreaker's solar panels.

The possibility of finding ice deposits within regions of permanent dark adjacent to regions of permanent light makes the Lunar South Pole so valuable to humanity that it warrants a mission of exploration. Icebreaker provides a viable design to successfully complete such a mission. The design rests on the firm foundations of previous NASA research programs. It also makes contributions to advance the current state of the art. The combined lander/rover concept allows current rockets to deliver large scientific payloads to the planets. The use of radar for terrain mapping provides immunity to dust and lighting conditions which could make it the next standard in sensing technology for planetary rovers.

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Chapter 1

Introduction

In 1994, the Clementine lunar orbiter conducted a radar experiment whose results indicate that ice exists near the Lunar South Pole. This ice is postulated to have come from comet and meteorite impacts on the surface, which deposit water ice and other volatiles. These deposits would occur in cold traps--areas perpetually shadowed from sunlight--which are capable of maintaining temperatures potentially as low as 24 K. Initial estimates suggested that the volume of one billion cubic meters, the volume of a small lake, may exist. Any ice deposits occurring in these cold traps will never sublimate, remaining for millions of years.

Experiments conducted by the Arecibo Radar Observatory in Puerto Rico, however, indicate that ice may not exist. Scientists from Arecibo have concluded that the same radar return signals which Clementine scientists claim indicate water ice in permanent shadow also are seen returning from regions which Clementine imagery indicated are lit for extended periods.

1.1 Motivation

Because of the controversy surrounding orbital and Earth based observations, the only way to definitively answer the question of whether ice exists on the Lunar South Pole is *in situ* analysis. Human exploration is possible, as the astronauts of the Apollo missions proved almost thirty years ago, but the mission costs, logistics, and risk preclude manned exploration. Robotic exploration, however, presents a viable solution.

Discovering the extent and composition of these ice deposits would reveal much about the history of the Moon. As the ice deposits accumulate, they may build up stratigraphic layers, providing a historical record of how the Earth-Moon system evolved over time, similar to the rings of a tree or the layers of ice studied by geologists in the Antarctic.

Finding a storehouse of water ice on the Lunar South Pole will change the way human space exploration proceeds. Every kilogram of water brought from the surface of the Earth costs in excess of $50,000, and requires additional fuel, which quickly limits total launch payloads and exaggerates mission costs[21]. Water mined from the lunar surface could be used for manufacturing rocket propellant and building materials, generating power, and establishing a human lunar presence. A permanent base on the Moon could serve as a launching point for further exploration into the solar system and beyond.

1.2 Objectives

The goals of this mission are to conduct the first systematic ground exploration of the lunar south polar region and to confirm the existence of a stratigraphic record of ice deposition in permanently shadowed regions. The objectives are to:

1. confirm the presence of ice and map its local surface distribution
2. determine lunar ice composition
3. determine the existence and nature of a stratigraphic record
4. measure the composition of ice to a depth of a meter below the surface

In this report we will present a description of a working model of the Lunar South Pole environment followed by a full mission concept and the design for Icebreaker, a mobile robot capable of searching for ice on the south pole of the Moon.

1.3 Mission Objectives

The baseline mission for Icebreaker is to visit two cold traps and take ten samples from each. These ten samples are taken from a single bore hole, one meter in depth. One pair of these samples are a centimeter apart, to gain the most information about the stratigraphy. The final set of samples from both cold traps would answer the basic scientific questions Icebreaker sets out to answer.

The baseline mission duration for the exploration of two cold traps is two weeks, during which time Earth communication will be available. After this time, the Earth will pass below the horizon for the next two weeks, and solely autonomous modes will be called for. At the start of this two week period, Icebreaker drives to a potential cold trap near the landing site. Although cold traps can occur outside of craters, the exploration of a small crater provides the baseline description of the mission. Descending into a single cold trap crater is estimated at three hours, which when driving 0.3 m/sec, corresponds to 3.2 km from the rim. Drilling into the ice and processing the samples is estimated to take four hours. The return trip out from the cold trap takes another three hours. Thus, visiting a single cold trap takes ten hours.

The additional time provided by the two week window is a safety factor to ensure success. This time will be used for driving between cold traps, recharging batteries, dealing with complications in driving into the cold trap and taking the samples, and exploring cold traps in which ice is not found. If time is still available after successfully finding ice in two cold traps, additional cold traps can be visited, or additional samples can be taken from the same cold traps. Similarly, after surviving the two week period without Earth communication, Icebreaker could continue to explore the area further.

Chapter 2

Lunar South Pole Environment

The orbits of the Earth, Moon, and Sun give rise to very unique conditions on the Lunar South Pole. The Moon's axes of rotation and revolution are nearly parallel, meaning that the Earth and Sun never rise more than a few degrees above the horizon. This creates the potential for regions of permadark--craters and surface depressions lit for millennia only by earthshine and the light of distant stars. There is no atmosphere to speak of, the surface essentially equivalent to the cold hard vacuum of space. In addition, extreme high temperatures may be achieved for sunlit areas, while permanently shadowed ones may remain close to absolute zero.

2.1 Sun and Earth Visibility

One of the most intriguing characteristics of the Lunar South Pole is the pattern of Sun and Earth visibility. Similar to the Earth's poles, the Sun remains above the horizon at the South Pole for nearly six months at a time, a so-called Lunar Summer. For the other six months of the year, during Lunar Winter, the Sun disappears below the horizon. The angle does not vary much, however, with a maximum and minimum angle of about 1.5 degrees above and below the horizon, respectively. When the sun is visible, the result is a land of very long shadows, allowing even the shallowest indentations in the surface to harbor permanently dark areas.

The Moon always shows the same "face" to Earth. The time between successive star alignments, the so-called sidereal month, lasts 27.3 Earth days. However, at this point the Earth has moved approximately one twelfth of its orbit, which means that from the point of view of the Moon, the Sun has not made a complete cycle. At the end of the synodic month, 2.2 Earth days later, the Sun completes a cycle around the lunar sky. Thus the Lunar day is equal to 29.5 Earth days. The result of the combination of solar elevation and azimuth angles is that the Sun appears to rise slowly upwards in elevation while circling about until it reaches its maximum elevation of about 1.5 degrees, then spiralling downward until it is below the horizon. The Sun will appear to make roughly six cycles in azimuth from sunrise to sunset, during the Lunar summer.

The visibility of Earth from the Lunar South Pole follows a different but similarly intriguing cycle. For an observer at the south pole of the Moon, the Earth rises to its maximum elevation of around 7 degrees above the horizon and falls to 7 degrees below. This cycle lasts the length of the sidereal month, with Earth returning to its starting position every 27.3 days. The Earth remains visible for roughly half of this time, about 13 days. Unlike the solar azimuth angle, however, the Earth does not move appreciably in azimuth since the Moon's orbital and rotational velocity is the same. Thus the Earth simply rises and falls in the same direction all the time. Because the maximum elevation angle of the Earth is larger than that of the Sun, there may exist permanently dark regions which provide Earth visibility. The details explaining the calculation of the Sun and Earth angles are in Appendix A, along with a table listing the maximum elevation angles for the Sun and Earth at various latitudes near the south pole.

2.2 Terrain

In the Sun and Earth visibility study, a spherical model of the Moon is assumed. Natural terrain will of course dramatically influence visibility due to occlusion by lunar surface features such as craters, depressions, and ridges. Unfortunately, detailed topographic maps are not available for the Lunar South Pole region, but it is known that the south pole is in a highland area, similar to the area in which Apollo 16 landed, and a statistical model of the topography can be developed.

2.2.1 Typical Crater Characteristics

Lunar craters caused by meteorite or cometary impacts vary in shape and size, but common statistical relationships exist. For small to medium craters, two basic types occur: simple and complex. Simple craters have diameters of 5-15 km, and are bowl-shaped. Complex craters have diameters of 20-150 km, and are bowl-shaped with a central peak. The typical depths, depth to width ratios, floor diameters and some other characteristics have been determined statistically for these types of craters, and are listed in Appendix A [9].

The information relevant to Icebreaker's mission is the depth to width ratio of craters. For simple craters, this ratio is about 0.2 statistically. For complex craters, the ratio is about 0.12 or smaller, decreasing as the crater diameter increases, becoming about 0.04 for a 100 km crater. The ratio of depth to width is by itself a useful figure of merit for finding candidate locations for ice. Simply stated, any crater with a depth to width ratio greater than 0.03 at the Lunar South Pole will contain a region of permanent shadow.

Typical slopes near the rims of smaller craters are relatively steep, but these statistical values are valid for fresh craters. Eroded craters generally have shallower slopes, which would be easier for Icebreaker to navigate. However, erosion also leads to smaller depth to width ratios, decreasing the likelihood of the existence of cold traps in eroded craters.

2.2.2 The Aitken Basin

Perhaps the largest single terrain feature on the surface of the Moon, the Aitken Basin engulfs the south pole. This is an ancient impact crater 2,500 km in diameter, and up to 12 km deep in places [21]. This enormous feature creates the major topological difference between the Moon's north and south poles. Because of this feature, more of the south pole region is shadowed from the Sun than would otherwise be expected. The total area estimated to be in permanent shadow near the south pole is 15,500 km^2.

2.2.3 Shadows and Cold Traps

Cold traps are areas of extremely low temperature occuring in regions perpetually shadowed from sunlight. Because the maximum elevation angle of the Sun can be calculated, the topography necessary to produce a cold trap can be determined. Figure 2.1 shows the geometry of a cold trap. The arrows indicate the minimum elevation angle required for incident light to reach the crater's interior. Light incident from elevations below this angle will never strike the bottom of the crater.

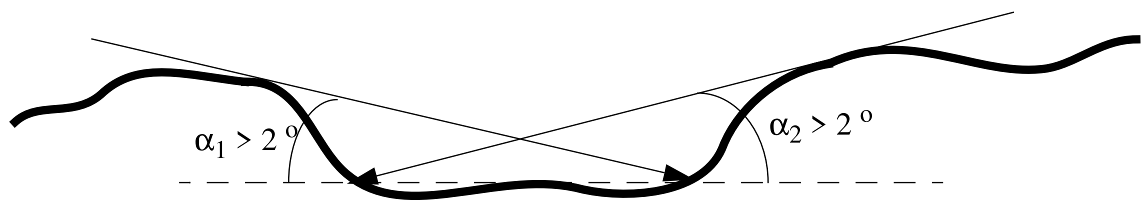


FIGURE 2.1 *Cold Trap Cross Section*

To determine whether a surface depression has the proper geometry for a cold trap, the depth to width ratio must be known. Let α be the angle of incidence of a ray in Figure 2.1 from the top of the rim on one side of a crater to the floor on the opposite side. Then $\tan(\alpha)$ = depth/width. If the maximum solar elevation angle for that location is β, then α must be greater than β for all angles α around the rim of the crater in order for the floor to be permanently shadowed.

2.3 Temperature

The temperature distribution near the Lunar South Pole is quite homogeneous compared to the lunar equator. Due to the low Sun elevation and long periods between sunrise and sunset, the temperature does not vary much, remaining from about -30 C to -50 C in most areas.

Cold traps maintain much lower temperatures, as low as 24 K. Long durations in such regions will require effective thermal insulation and the ability to generate enough heat to survive. Temperature gradients are presumed to exist across cold trap cross-sections, with extreme cold at the center and warmer temperatures near the edges.[19] These cold temperatures allow volatiles to remain permanently on the lunar surface [24]. In addition, the low temperature means that the heat radiated by Icebreaker itself will not cause appreciable sublimation of the ice it is trying to detect. Icebreaker can remain on a single patch of ice in a cold trap for over six hours before sublimating one centimeter of ice (see Appendix B).

2.4 Atmosphere

The Moon has no appreciable atmosphere. The ambient air pressure in regions which are at equilibrium is around 10^{-9} to 10^{-11} Torr. Landing with thrusters activated will introduce gasses at the landing site which may cause a transient increase in atmospheric pressure of a few orders of magnitude, but these gasses will quickly diffuse.

Chapter 3

Mission Overview

3.1 Design Overview

To fulfill the goals described in the Introduction it is proposed that a mobile robot, named Icebreaker, be used. A mobile robot was chosen for several reasons. The mobility of Icebreaker allows it to explore multiple cold traps and collect samples. A manned mission, while also providing the mobility necessary to explore multiple cold traps, is much more expensive. Dropping multiple stationary probes into many cold traps is another option which would permit sample collection from multiple areas without mobility. However, the need to collect stratigraphic information, would make each of these probes quite complex and expensive. Furthermore, Icebreaker is capable of seeking out volatiles whereas the stationary probes would have to land on an ice deposit.

An artists conception of Icebreaker is shown in Figure 3.1. It is a four wheeled, four wheel drive robot. The drive motor for each wheel is mounted inside the wheel. The front wheels of Icebreaker are Ackerman steered. The rear wheels are connected to the chassis through a bogie mechanism. Dust, kicked up by the wheels, is prevented from collecting on the solar panels and optics by fenders over the wheels.

Due to mass and volume constraints in the Delta II 7925H rocket fairing, Icebreaker will be a combined lander/rover vehicle. This class of vehicles combines the functions of a traditional landing craft and rover into one vehicle. Thruster and fuel tanks are an integral part of Icebreaker and allow it to touchdown on its wheels with very little energy and no disposable shock absorbing structure. Thus Icebreaker will land on the Moon and then drive off in its quest for volatiles.

Many sensors are necessary for Icebreaker to successfully complete its goals. An inertial measurement unit (IMU) and a pair of star tracker cameras are used to fix the position of Icebreaker on the surface of the Moon, as well as during the flight through space. Radar altimeter and belly camera are used to control the landing. A panospheric camera, mounted on top of the solar panel is used to detect the horizon, as well as to return images to Earth. A pair of forward looking, horizontal baseline stereo cameras return stereo images to a human teleoperator. Strobe lights will be used when ambient light is insufficient for camera images. A radar sensor is used to detect obstacles and permit the robot to travel autonomously.

To meet the mission goals of determining the composition of volatiles and determining stratigraphy, Icebreaker has a set of scientific instruments. An infrared camera, tuned to water ice, will be used to locate possible deposits of volatiles. After the robot closes on a potential deposit, a cryogenic sample acquisition drill is used to collect a sample. This drill is capable of collecting samples up to a depth of 1m. The drill will then deposit the sample into a Regolith Evolved Gas Analyzer (REGA). The REGA will heat the sample, and using a mass spectrometer, determine the chemical and isotopic composition of the sample. Stratigraphy will be determined by taking and analyzing samples from various known depths.

FIGURE 3.1 *Icebreaker (Artwork by Mark Maxwell)*

Icebreaker will return scientific data and images to the Earth over a 10kbps, S-Band radio link. A switched array of limited aperture, high gain antennas, mounted on top of Icebreaker, will be used to send and receive data to Earth through NASA's Deep Space Network. As a backup, a small omnidirectional low gain antenna is also included.

While Icebreaker does not have to deal with the extreme heat encountered at the equatorial regions of the Moon, the cryogenic temperatures inside the cold traps make heating an important issue. The main body of Icebreaker will be temperature controlled to preserve the computers, scientific instruments and batteries. This will be done through a series of heat pipes and insulation. Small quantities of RHU's will also be used to generate heat, particularly in isolated regions such as the video cameras.

Icebreaker will generate power using solar panels. A solar panel fin is mounted from the bow to stern of Icebreaker. This fin is fixed and thus cannot be pointed towards the Sun. Batteries will be used to store electricity, both for journeys into cold traps and for driving when the Sun is not illuminating Icebreaker's solar panel.

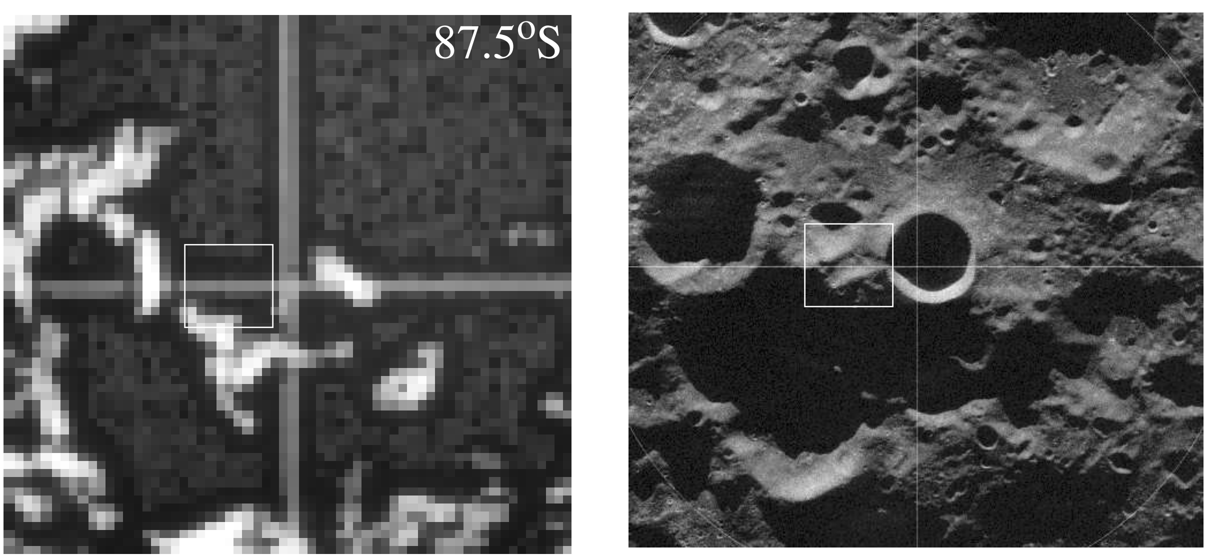
More complete descriptions of Icebreaker’s design can be found in the remainder of this report.

3.2 Landing Site

Icebreaker will be able to land on a specific spot that can be chosen from images as it descends to the lunar surface. A general area, termed the landing site, however has been designated from which this specific spot will be chosen.

The landing site for Icebreaker is an area near the Lunar South Pole. A rectangular area 20 km x 20 km has been selected, ranging from 89.1 S to 89.7 S latitude, and 75 W to 105 W longitude. This area is shown in Figure 3.2, enclosed by the white rectangle. A more specific landing spot, 16 km^2 in size, will be chosen from this area. The area is near valleys and craters with a high potential for cold traps, as well as near ridges that are lit by the sun for extended periods of time. Based on the Clementine images of the south pole, the point at 89.3 S, 105 W is lit by the sun over 80% of the time [21]. Line of sight with Earth is likely for most points in this landing site, allowing communication with Icebreaker. The local topography consists of relatively shallow slopes, with eroded craters.

The left-hand image in Figure 3.2 is a composite of 1500 Clementine images taken over the period of a month during the spring of 1994. This image shows the points that were lit by the Sun at some time during this month. The right-hand image is an Arecibo radar image, showing those points that were visible to Earth when the image was taken. This data corresponds to a time when the Arecibo radar site was approximately six degrees above the lunar horizon at the south pole [22]. As described in Section 2.1, the angle of the Earth above the lunar horizon can be greater than the angle of the Sun above the horizon by as much as five degrees. Therefore, areas which are visible to the Earth in the Arecibo image can be dark in the Clementine image and in fact, can be permanently dark cold traps. A close-up of the Arecibo image landing site is in Appendix A.



Clementine Composite Image

Arecibo Radar Image

FIGURE 3.2 *Lunar South Pole Images*

Chapter 4

Mission Operations

During operation on the lunar surface, Icebreaker will be in one of five operational modes: initial test/shakedown, driving, drilling/scientific analysis, charging and hibernation. Selection of the active mode is based on the visibility of the Sun and Earth.

The environment of the Lunar South Pole region presents several conditions which the robot must deal with during the mission duration. The conditions considered in this document can be divided into lighting conditions and communications availability. They are:

Lighting Conditions

- Lunar Winter - Sun is below the horizon for long period of time
- Lunar Summer - Sun visible for long period of time
- Shadow - darkness caused by terrain blocking the Sun
- Permadark- permanent darkness caused by cold traps

Communications Availability

- Earthnight - Earth is below Moon's horizon for long period of time.
- Earthday - Earth is visible, communications are available
- Earthshadow - Earth is temporarily obscured by terrain blocking line of sight
- Cold Trap Earthshadow - cold trap from which communications are not possible

4.1 Initial Test Mode/Shakedown

This mode is performed immediately after landing. The purpose is to allow mission control to determine the health of all major subsystems. Provided the communication system is functional, Icebreaker will return data on its internal status such as temperature, battery status, solar panel output and estimated position. Camera views of the surroundings will also be returned, providing useful information to the operator about the area. Icebreaker will execute short drives and perform drill tests. It is assumed that pre-planning can guarantee that this mode occurs during Lunar Summer with full communications (Earthday).

4.2 Driving

Icebreaker is in driving mode whenever in motion. A hybrid teleoperation/autonomous control scheme is used whereby a human operator on Earth specifies a series of waypoints [10][25]. Icebreaker will travel to each waypoint autonomously, using its radar sensors to avoid obstacles. The sequence of communications between Icebreaker and Earth are shown in Table 4.1. Due to bandwidth and speed of light limitations the stereo pair of images takes 25 seconds to reach Earth. Icebreaker remains stationary during transmission of the stereo pair and resumes motion when it receives the “GO” signal from Earth. Other data, such as panospheric images and telemetry can be returned while Icebreaker is moving.

TABLE 4.1 *Robot/Operator Interactions*

Icebreaker	Ground Based Operator
Reached Goal waypoint - stop motion	
Send images from forward stereo cameras	
Send telemetry	See Rover’s eye view on 3-D display
Wait for “GO”	Select waypoints
	Give science instructions if necessary
	Can request other images/telemetry
	Give Icebreaker a “GO” signal
Traverse waypoints, autonomously	Monitor progress/telemetry
Arrive at Goal waypoint	

Driving mode assumes Earthday and Lunar Summer conditions are present. Shadow can be distinguished from Lunar Winter (which is very predictable), and most likely can be driven through under battery power. Earthnight will cause the robot to transition to Charging or Hibernation modes.

The likelihood of entering an Earthshadow is high since the Earth is very low on the horizon. While Icebreaker is driving between waypoints Earth communications is not required and Icebreaker will proceed through its waypoints. If Icebreaker is within Earthshadow when it reaches the Goal waypoint, it will continue driving in the same direction (avoiding obstacles) for a fixed distance. If communications are still not present, Icebreaker will then use its stored path, and its tracks as a visual cue, to drive back to the last spot where communications existed.

Entering and exiting cold traps is a somewhat special case of driving. The mission will endeavor to find a cold trap which has Earthday. Upon entering such a cold trap, driving will commence using the waypoint control as before, however lights or strobes will be used as necessary to augment the illumination provided by Earthshine. Unfortunately it is not possible at this time to guarantee that a cold trap with Earthday will exist. This uncertainty means that Icebreaker must be capable of autonomously entering, analyzing, and exiting a cold trap under the Cold Trap Earthshadow conditions. Icebreaker will do this using its terrain radar unit to avoid obstacles. Using the panospheric camera to identify the horizon, it can calculate when an area will be permanently shadowed. This mode of operation will only be used as a last resort if cold traps with Earthday cannot be located, or if ice was not found in any of these traps.

4.3 Drilling/Scientific Analysis

Scientific analysis will occur primarily inside cold traps. Once inside, Icebreaker will use an IR camera tuned to water ice to remotely detect water ice. If no volatiles are visible, Icebreaker will execute a search grid inside the cold trap using the IR camera to look for volatiles. Icebreaker and its human operator, will be aware of battery status and allow sufficient time to exit the cold trap before the batteries become critically low. Once ice is detected, Icebreaker will position the drill and begin taking samples. These samples will be transported from the drill to the Regolith Evolved Gas Analyzer (REGA). The REGA will then determine the composition of the sample. Up to ten samples will be taken from the same bore hole at different depths.

Due to the risks involved with entering and exiting a cold trap, analysis of all samples will be done in the cold trap. The number of samples taken during each trip to a cold trap will also be maximized. However, due to the limited energy available from batteries, Icebreaker may be required to enter and exit the cold trap multiple times to charge batteries.

The scientific analysis will most likely be performed under Permadark lighting conditions but unusual circumstances may occur, such as a shallow cold trap, which allow the solar array to be lit while the ground is permanently dark. The communications conditions are not as certain. Ideally a cold trap with ice will be found under Earthday conditions. However, by the very nature of cold traps the horizon will be higher than normal to block out the Sun. It is expected that Earthshadows will be more common in Permadark than in lit areas. The distinct possibility exists that all cold traps, with ice, will be in Cold Trap Earthshadow which would require completely autonomous operations of the robot. These cold traps will only be explored as a last resort.

4.4 Charging

The design of the Icebreaker robot includes a fixed vertical solar panel which runs from the bow to the stern of the robot. This solar fin has solar cells on both sides. However, it is not pointable, so it is conceivable that at times the robot will be driving off its battery reserves even while in Lunar Summer conditions. Another load on the battery arises from cold trap operations. It is expected that a cold trap sortie may last 10 hours. This will use 40% of battery capacity.

Since it will not be possible to keep the batteries fully charged during driving and it is desirable to have fully charged batteries before entering a cold trap, the robot will stop and charge at times, most likely just before entering a cold trap and just after exiting a cold trap. To do this, Icebreaker will stop with its solar fin perpendicular to the Sun for maximum power generation. While charging it may be possible to return images and data of the surroundings, depending on power availability.

Charging must be performed in Lunar Summer conditions. While Earthday is not strictly necessary, it is advantageous so that the operator realizes what is occurring. This mode of operation may also occur during the conjuncture of Lunar Summer condition with Earthnight, since the robot is not expected to carry on operations during the long (approx. 14 days) period during which the Earth is below the horizon due to its orbit and not terrain.

4.5 Hibernation

Hibernation is a mode during which operations are halted and power consumption is reduced to a minimum. No communications, scientific analysis, driving, or imaging is performed. This mode of operations is entered when Lunar Winter falls. Since the Sun is no longer visible the robot must survive on batteries alone. The fall of Lunar Winter can not be confused with Shadows or Permadark since it can be predicted by operators on Earth.

Chapter 5

Launch/Landing

This chapter discusses the manner in which Icebreaker will get to the surface of the Moon. More detail, in the form of drawings and figures, can be found in Appendix C.

5.1 Launch

The proposed launch vehicle for the mission is the Delta II 7925H. This vehicle was chosen because it was the largest rocket funded by the Discovery program. The Delta II rocket can mount three different payload fairings - 2.4m, 2.9m and 3.0m. The 2.9m fairing was chosen because it provides almost as much room as the 3.0m, but is capable of launching a larger payload mass. With this configuration, the rocket is capable of launching a 1500kg payload to the Moon.

The Delta II rocket will insert Icebreaker onto a direct descent trajectory to the Moon. This means that Icebreaker will travel directly from Earth to the Lunar South Pole with out entering a lunar orbit. One course correction maneuver will be performed during transit to the Moon. The hydrazine thrusters used for landing (see Section 5.2.1) will perform the correction.

5.2 Landing

The traditional approach to landing an exploration robot on a planetary surface is to employ a separate landing vehicle. The landing craft contains all of the sensors, propulsion and most importantly shock absorbing material required for landing. Often, the landing craft has its own computing and power systems. After landing is complete, the robot leaves the landing craft via ramps which must be deployed. Once the robot has left, the lander can be used as a communications relay station or abandoned.

Icebreaker's mission creates many difficulties in its design. The need to operate for long periods of time in the dark requires large reserves of battery power. The need to enter and exit craters requires a high level of terrainability and stability. Finally the uncertainty of volatile deposits requires Icebreaker to be capable of traversing large distances. All of these factors mean that Icebreaker needs to be a large robot, however with the limited space available in the rocket fairing it is not possible to fit a large rover and a landing vehicle.

To overcome this payload problem a new type of spacecraft, the combined lander/rover, is proposed for Icebreaker. By adding the propulsion and landing sensors to the robot structure, the landing vehicle can be eliminated. Since the robot will not have as much tolerance for shock during landing, control of landing velocity will need to be much more precise than in previous missions. This will be accomplished by the combination of a precision radar altimeter and throttleable hydrazine thrusters. Furthermore, a vision system will be used to select a landing spot with minimal obstructions such as cliffs and rocks. By reducing duplication and using more complicated control, the combined lander/rover allows much larger payloads, such as Icebreaker, to be launched.

5.2.1 Propulsion

Icebreaker will have a total of three propulsion systems to control the landing; initial deceleration, final deceleration and attitude control. The initial deceleration system must remove most of the energy of Icebreaker's descent and does not need to be throttleable. The final deceleration system will remove the remaining energy of descent and is responsible for a zero velocity touchdown. A high level of control is necessary. The attitude control system needs to control the spin and translational position of Icebreaker.

The initial deceleration system consists of a Thiokol STAR 37XFP solid rocket motor (SRM). This provides a simple rocket with a high mass to energy ratio. The SRM is attached to the bottom of Icebreaker during lift-off, flight and initial descent. Once the rocket has exhausted its fuel, it is detached from Icebreaker, to crash into the Moon. An early version of this motor was used for initial deceleration of the Surveyor lunar landers.

The final deceleration system consists of twelve 37lbf hydrazine thrusters. The thrusters are mounted to Icebreaker in four groups of three, at the front, rear, and between the wheels on both sides. Each thruster set is connected directly to a fuel tank, minimizing the length of fuel lines. Each thruster can be turned on and off. By modulating the thrusters, the magnitude and direction of the deceleration can be controlled. This permits control of yaw and pitch as well as vertical velocity.

The attitude control system consists of four sets of three 5lbf hydrazine thrusters mounted directly above the final deceleration thrusters. These thrusters may also be controlled by modulating the output of the thruster. Their main purpose is to remove any element of spin present during descent. They are also capable of moving Icebreaker parallel to the surface of the Moon.

5.2.2 Landing Sequence

A simple three dimensional (x,z,θ) simulation of Icebreaker landing was created. Given the thrusts of each propulsion system and the initial velocity and position relative to the Moon, the simulation calculated the durations of each deceleration period as well as the velocity and position trajectories of Icebreaker during landing.

The results of one simulation run can be found in Figure 5.1. The large free fall period between the SRM and hydrazine decelerations is there to compensate for uncertainties in the SRM firing pattern. Errors in thrust angle and burn length of the SRM significantly impact the velocity at the end of the SRM burn. In the worst case, the free fall period would not exist.

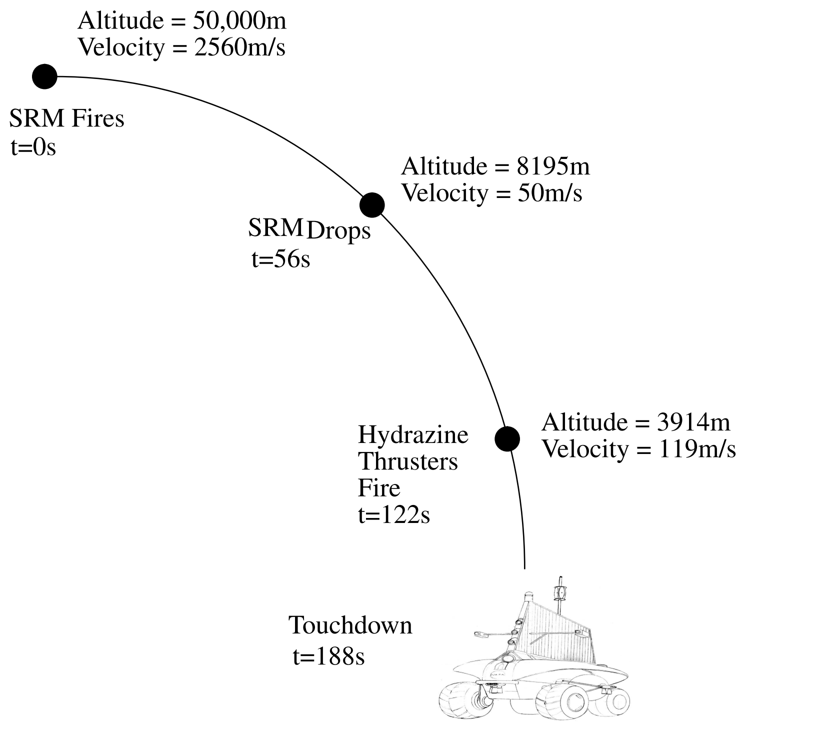


FIGURE 5.1 *Landing Sequence*

The masses of propellant needed for the descent and the mid-course correction are presented in Table 5.1. The table shows that 127 kg of hydrazine fuel is required. To reduce development costs, the fuel tanks from the Mars '98 lander will be employed on Icebreaker. These tanks are spheres, 42cm in diameter. Thus three fuel tanks are required for 127kg of hydrazine (127L). However, to simplify the design and reduce the number of valves and pipes required, four fuel tanks will be used, one mounted beside each thruster pack.

TABLE 5.1 *Landing Fuel Requirements*

Event	Mass of Fuel (kg)
Midcourse Correction (Hydrazine)	34.4
Initial Descent (Solid Fuel)	870.0
Attitude Control (Hydrazine)	25.0
Final Descent (Hydrazine)	67.6

Chapter
6

Locomotion

The purpose of the locomotion system is to transport science equipment to the desired destinations on the Lunar South Pole. In order to satisfy the mission goals and science objectives Icebreaker must exhibit robust locomotion. Over the past 30 years several vehicles have explored the Moon. The knowledge gained from these expeditions must be applied to new lunar surface vehicles. As the locomotion performance of a vehicle is improved the science return is increased. Mission flexibility will also be affected as a more diverse choice of landing sites are accepted because the vehicle can traverse a wider range of terrain features.

This chapter explores the fundamental trends in a parametric locomotion study. Much of the analysis presented is based on the work of M. G. Bekker [3]. The details of the analysis can be found in Appendix D.

6.1 Configuration Requirements

In order to scale Icebreaker's locomotion system several constraints must be considered:

- travel on lunar regolith at an average speed of 30 cm/s
- travel on slopes up to 20 degrees
- travel over obstacles 20 cm high

A parametric mobility study is shown in the appendix. Using the mobility study and the configuration requirements Icebreaker was sized to have a wheel base of 1.4 m x 1.4 m. The wheel diameter was chosen to be 0.6m and a wheel width of 0.4m. Each wheel is independently driven. The primary steering mode was chosen to be front wheel ackerman steering. This configuration allows Icebreaker to travel over flat terrain with a power draw of less than 70 Watts.

6.2 Wheels and Propulsion

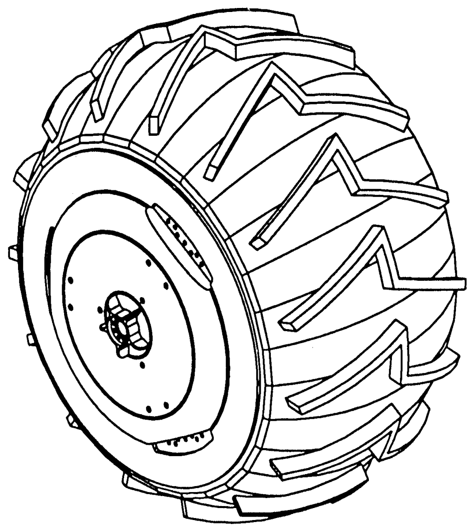


FIGURE 6.1 *Nomad's wheel*

Icebreaker's propulsion is designed after Nomad [26]. Icebreaker has independent four wheel drive, with the propulsion located inside the wheel hub. This design utilizes the large amount of space inside the wheels. The wheel drive motors are not linked to the steering or suspension systems, thus eliminating any mechanical transmission of power and associated inefficiencies. Wheel motions are individually controlled for steering velocity differences or torque differences on rough terrain. The wheel hubs are attached to a vertical plate which can pivot about a stationary crossmember which acts as an axle between the front or rear wheel pairs (Figure 6.3 on page 22). The wheel hub contains the drive motor which drives a larger gear attached to the moving portion of the wheel. The stationary and rotating portions of the wheel are separated by a bearing. The wheels are a solid lightweight composite material in the shape of an inflated tire (Figure 6.1). Grousers increase traction and reduce wheel slippage on steep slopes.

6.3 Steering and Suspension

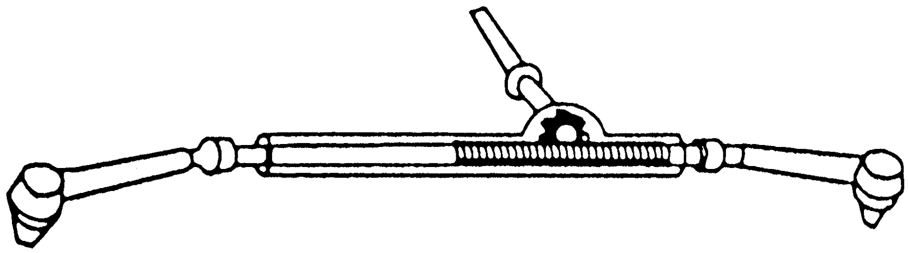


FIGURE 6.2 *Rack-and-pinion linkage*

The basic requirements of the steering were that it be simple to control, have a low parts count, would not require an overly complicated support structure, could support a simple suspension system, and be very stable. Skid steering causes large losses of energy through the skidding motion. Since power efficiency is critical, this steering mode was not considered viable. Articulated frame steering has the advantage that the vehicle is more maneuverable than with Ackerman steering, however this type of steering requires complicated frame design and is less stable on steep slopes and when turning, thus lowering the maximum payload. Single axle Ackerman steering allows a rigid frame design. It can also accommodate a very simple suspension system, such as a floating axle, or bogie suspension, without increasing the complexity of the support structure. Double Ackerman would require more complicated frame design and a higher parts count as well as more complicated control issues. The reference [5] discusses steering modes in more detail. Single Ackerman steering was chosen for Icebreaker (Figure 6.3). Rack-and-pinion, shown in Figure 6.2, is the preferred actuation scheme because it is the least complex and can be easily adapted to vehicles with minimal frames[7].

The suspension system designed for Icebreaker is a single pivot point at the rear axle, or a bogie. The bogie motion is limited by hard stops to a twist angle of +/- 40 degrees. The body of Icebreaker is rigidly attached to the frame from the front axle rearward to just before the pivoting rear axle. A disadvantage of this simple suspension is that there is no body averaging design, so the frame will move through large angles when the rover is moving over an obstacle. An advantage is that the frame can be immobilized more easily in the rocket fairing than with a more complicated suspension.

6.4 Frame and Chassis

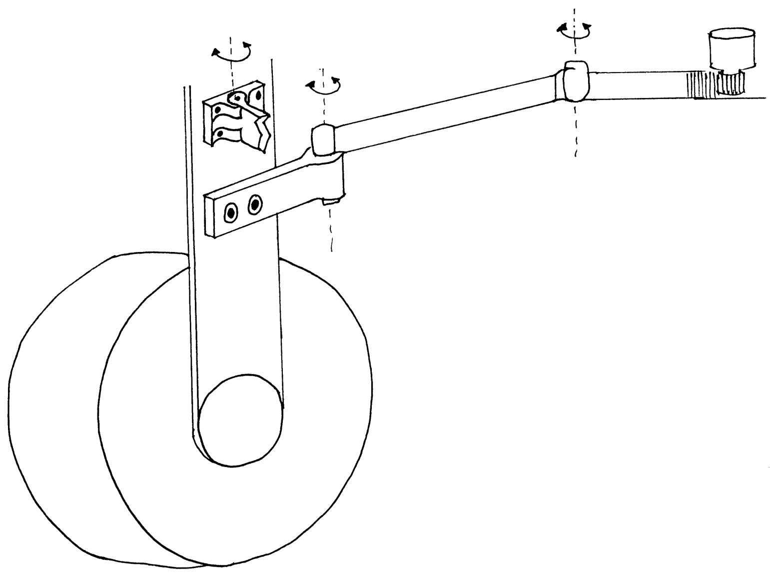


FIGURE 6.3 *Wheel axle and steering arrangement*

The frame is designed to allow a 40 cm ground clearance. With a wheel radius of 30 cm, a single wheel axis through both sets of wheels is not sufficient. The wheels are suspended from a crossmember by a vertical plate that attaches to the wheel hub (Figure 6.3). On the front pair of wheels, these plates also pivot at the ends of the crossmember allowing for steering. The rear wheels have a stiff connection to the rear crossmember. The front and rear crossmembers are attached to a central transverse member with a pivot point at the rear crossmember for suspension. The chassis is built onto the front crossmember and the central member (before the pivot point).

The chassis houses components such as computing, batteries, drill, and some of the sensing, in a thermally isolated section in order to ease thermal control of these components. There is a stiff vertical structure in the center to support the drill and the solar panels. The outside body, which is separate from the thermal enclosure, is a fender that protects the solar panels and upper body from debris kicked-up by the wheels during locomotion. It is not clear whether the fenders are sufficient to shield the sensors and solar array from dust and debris during landing.

Chapter

7

Sensors

Icebreaker will need sensors to land, traverse the lunar surface, remotely detect ice, and perform sampling. This chapter discusses the sensors proposed for these tasks. Appendix E presents a detailed description of the rationale and design of the scanning millimeter wave radar unit being proposed for terrain mapping during lunar surface traversal.

7.1 Landing

Icebreaker's mission includes flying to a general landing area, choosing a landing location, servoing to the landing location and touching down gently on the lunar surface. See Chapter 5 for a description of the landing sequence. A radar altimeter, CCD camera, inertial measurement unit (IMU), and star tracker camera will be used for landing. In addition to these sensors, Icebreaker's position estimate will be augmented using the tracking ability of the Deep Space Network (DSN) to estimate position. This will be done by ground based operations and course corrections will be sent to the spacecraft to achieve a desired trajectory.

7.1.1 Radar Altimeter

The unit that is baselined for Icebreaker's design is the Honeywell Landing Radar being designed into the Mars 98 Lander. This is a 4 beam radar which uses Doppler shift to obtain velocity with respect to the surface. This device measures velocity to within 4 percent and altitude to within 2 percent.

7.1.2 Belly Camera

This is a standard CCD camera with the ability to focus at infinity and perhaps the ability to focus at shorter ranges as the spacecraft nears the surface of the Moon.

7.1.3 Inertial Measurement Unit

Two Inertial Measurement Units (IMUs), one for redundancy, are included in the design to provide incremental position updates. The Litton IMU consists of a three axis fiber optic gyroscope and silicon accelerometer. The specifications of the Litton IMU are the following [13]:

Litton Model No.: LN-200

- Size: 3.5" diameter x 3.2" height
- Weight: 0.71 kg
- Power: 10 W Run (+/- 5, +/- 15 V.D.C.)
- Performance (1 sigma):
    - Gyro 1 deg/hr bias
    - Accelerometer 500 microGs bias
- Environmental:
    - Temp -54 C to 71 C
    - Vibration 15 Grms (20 Hz to 2000 Hz)
    - Shock 1500 G's
    - Hermetic sealed
- I/O: Flexible data rates and scaling, anti-alias filters, RS-422 output.

One issue affecting the design of the IMU that is unique to this mission is the fact that the IMU will be required to function within both a flying and driving vehicle. The dynamics of the flight spacecraft will be different than the dynamics present in the surface traversal application. This could present a range or resolution difference required when reporting angular rates. For instance, in the Clementine mission, the IMU was altered from its commercial off the shelf configuration to report rates of +/- 360 deg/sec. The vehicle dynamics on the lunar surface need to be analyzed to determine if the same range and resolution will suffice in this case.

7.1.4 Star Tracker Camera

The Icebreaker design calls for two star tracker cameras such as the two flown on the Clementine mission to be used to provide an inertial reference for the spacecraft by comparison of star field with an on-board star map [17]. Specifications:

- Mass: 0.29 kg
- Size: 12 x 12 x 14 cm
- Power: 4.5 W
- Field of View: 29 x 43 degrees
- Pixel Format: 384 x 576
- Structural Requirements for Clementine mission were listed as:
    - steady-state - 100 g's each axis
    - random vibration - 14 g rms, 60 s, tested a qualification unit to 19.8 g rms, 60 sec.
    - pyro-shock - 84 g's, peak accel, no testing done, analysis only
- Temperature Limits:
    - CCD min: -15.8 C max: 27.1 C
    - lens min: -26.1 C max: 30.8 C

- Voltage, steady-state, + 15 (+/- 0.25), - 15 (+/- 0.25), +5 (+0.25/-0.15), -5 (+/- 0.25)
- Direct Sun Viewing: Brief exposures ok, but this is not recommended

The Clementine Star Tracker ran Stellar Compass software on it's R3000 processor for star matching and quaternion generation. The measured rotation (quaternion) accuracy (1 sigma) was 80 microRad x 80 microRad x 400 microRad. Using an assumption that this angular accuracy can be converted to an accuracy in meters on the lunar surface by using the product of the lunar radius and the angle, the absolute accuracy on the lunar surface would range between 130 meters and 695 meters. This accuracy will not allow complete position-based navigation over long periods of time and navigation will require using other sources of information. Position updates from the star tracker system will be similar to positioning updates from single-ended coarse acquisition coded GPS on Earth.

7.2 Lunar Surface Traversal

As described in the mission scenario, Icebreaker must allow several control modes. Sensing must be provided to enable these control modes. Sensors used during surface traversal fall into the following categories:

- Terrain Sensors
- Positioning Sensors
- Remote Ice Detection
- Sampling and Analysis
- Internal Monitoring

7.2.1 Terrain Sensors

Icebreaker's terrain sensor design attempts to achieve simplicity, redundancy, and ruggedness. The terrain sensors on Icebreaker will be a combination of cameras and a scanning radar unit that provides terrain maps for obstacle avoidance and autonomous operation in the situations where autonomy is required. Strobe lights will be provided for viewing images collected when Icebreaker is in shadowed areas. Two strobe lights are mounted between the solar panels in front of Icebreaker in Figure 3.1 and a third light will be mounted on the underside to provide illumination when imaging with the camera provided for drilling. The terrain sensors will include:

- 1 Panospheric Color Camera
- 2 Monochrome CCD cameras directed toward the front, 1 meter baseline
- 1 Scanning radar unit, 94 GHz, Frequency Modulated Continuous Wave (FMCW)

Panospheric Camera

A panospheric camera will provide video to allow Icebreaker's ground operators the ability to view the surrounding terrain and find promising locations for ice and to avoid possible communication dropouts due to the surrounding terrain. This view of the surrounding terrain gives the operator an ability to turn the robot around. The panospheric camera is also expected to be used to image the horizon and determine when Icebreaker has descended (on a slope or in a crater) to a point in which the area surrounding it is in permanent darkness. The panospheric camera baselined for the mission is described in [26].

Stereo Cameras

A stereo pair of CCD cameras will be used to provide the operator with two images which can be viewed by the operator with polarized glasses to visualize depth on the lunar surface. Waypoints can be designated on this three-dimensional image and checked for consistency before they are sent to the vehicle. If desired, a depth map can be computed from these stereo images using computers at the ground station on Earth. Either the computed depth map, or the three dimensional view formed by the operator using polarized glasses may be used for designating waypoints for the vehicle to follow.

The CCD cameras will be monochrome, 640x480 pixel format, auto iris, auto gain control.

Radar

A scanning millimeter wave radar unit is proposed for building terrain maps that can be used for safeguarding Icebreaker during traversal of the paths that are planned using operator designated waypoints. This terrain map may also be used for autonomous navigation if an Earthshadowed cold trap must be entered to complete the mission. Radar has the following design advantages:

- Unaffected by dust particles clinging to the surface of the sensor and dust particles that are thrown into space as the vehicle drives.
- Functions in light or dark.
- Uses simple slow moving actuation.
- Rugged enough to withstand vibration during launching, SRM separation, and landing.
- Less computing required for building a terrain map compared to stereo cameras.

The specifications for this unit are the following [11]:

- Transmission method: Frequency Modulated Continuous Wave (FMCW)
- Frequency: 94 GHz
- Mass: 2.5 kg
- Size: 20 x 20 x 20 cm
- Power: 20 W
- Field of View: 80 x 3 degrees
- Mechanical scanning - requires rotary actuator, sealed unit, ceramic coated bearings with low vapor pressure grease

The terrain mapping radar proposed for Icebreaker uses a single radar unit scanning a 3 degree beam across an 80 degree field of view. The horizontal and vertical field of view of a single scan and the dimensions of the footprint on the ground are shown in Figure 7.1. See Appendix E for more information on this radar.

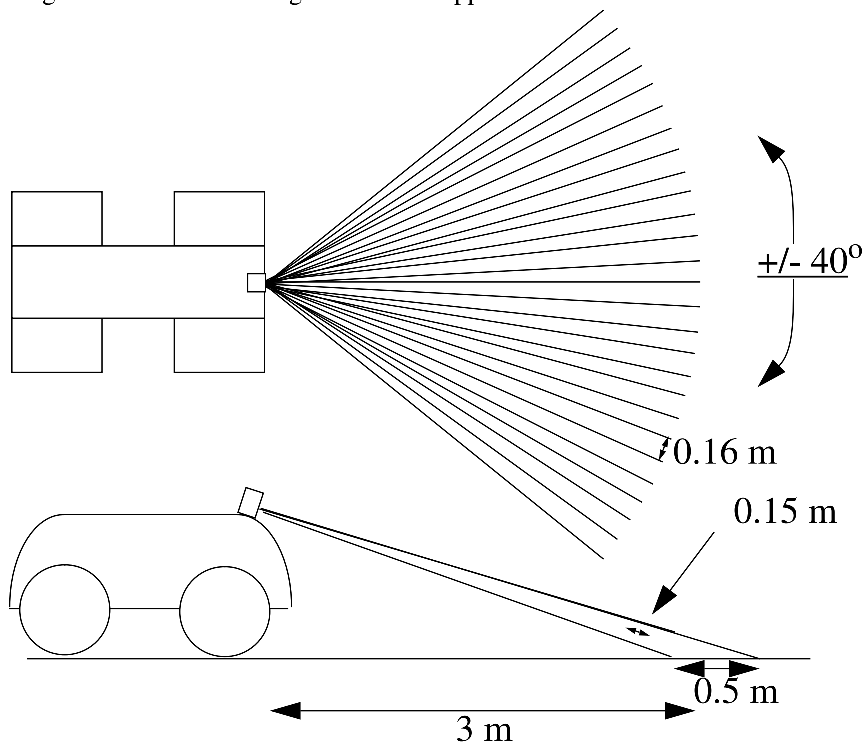


FIGURE 7.1 *Top and side view of scanned radar beam.*

7.2.2 Positioning Sensors

Positioning sensors will be used while landing and while traversing the lunar surface. The IMU, described in Section 7.1.3, and the star tracker camera, described in Section 7.1.4, provide positioning for both landing and driving Icebreaker. Feedback from wheel, steering, and bogie encoders will also be used to calculate Icebreaker's position and orientation.

7.3 Remote Ice Detection

The problem of remotely detecting ice has not been pursued in depth; a preliminary sensor has been baselined for Icebreaker's design. The remote ice sensor is shown in Figure 3.1 on page 8 mounted between the solar panels toward the front of Icebreaker. In the artist's conception, it is the top unit of the three units mounted between the solar panels. Current ideas for this sensor are that it is a CCD camera with a filter that detects electromagnetic energy at a wavelength of 2.7 micrometers. At this wavelength, ice would be dark in the image, while the lunar surface would appear bright in the image[1].

This camera would be similar to the UV-Visible Camera used on the Clementine mission, which used silicon CCD technology and operated in the near ultraviolet and visible region of the spectrum. A six position spectral filter wheel allowed the camera to be used for remote sensing applications [17]. In this mission a single filter tuned to a wavelength of 2.7 micrometers will be used.

The specifications for this sensor on the Clementine mission were:

- Mass: 0.41 kg
- Size: 10.5 x 12 x 16 cm
- Avg. Power: 4.5 W
- Wavelength: 0.3 - 1 micron depending on position of the filter wheel
- Field of View: 4.2 x 5.6 degrees
- Pixel Format: 384 x 288

In order to allow flexibility in including some type of remote ice sensor for the mission, the mass and power limits of this sensor were conservatively estimated to be 2 kg and 15 W.

7.4 Sensing during Sampling

A belly camera and strobe light will be provided for viewing the drilling activity when in a cold trap during an EarthDay. The range map from the radar units, although coarse, may be useful for finding a flat area in which to drill, particularly if the sampling is being done in a Cold Trap Earthshadow.

1. This has not been verified through literature, but was promoted as a possibility during a discussion session on lunar ice, by scientists at the 28th Lunar and Planetary Science Conference, March 17-21, 1997.

Chapter

8

Sampling and Science Instruments

This chapter details the scientific goals of Icebreaker's mission and the design and operation of the sampling drill and chemical analysis instrumentation.

8.1 Sampling and Science Requirements

The sampling and science requirements are:

- Verify existence of water ice and other ices
- Analyze volatiles in surface and subsurface lunar samples
- Measure stratigraphic information in lunar samples

The number one priority of this mission is the verification of the existence of water ice and other ices in the lunar polar regions and the detection and measurement of stratigraphy in the ice/regolith layers. Lunar surface and sub-surface samples may consist of water ice as well as other ices such as carbon dioxide, methanol and ethanol. In addition, these ices are most likely mixed with layers of regolith and lunar dust. Verifying the existence of water and other ices requires *in situ* chemical analysis of surface and subsurface samples.

Stratigraphic information about these different layers of material, and different percentages of compounds at varying depths can be determined on a rough scale by analyzing samples at different depths. The amount and accuracy of stratigraphic information that can be extracted from the samples depends on the method of drilling, the spacing between samples and the level of chemical analysis of lunar samples. In order to obtain acceptable coarse stratigraphic information the following requirements must be met:

- samples taken up to a depth of one meter
- ten samples in one meter
- minimum depth between one pair of samples is one centimeter
- samples must not be contaminated (by drill or in ovens)

8.2 Drilling and Sampling

Icebreaker's drilling and sampling system is modelled after a prototype for the Champollion mission to analyze surface and sub-surface samples on a comet (Figure 8.1)[8]. It is designed to obtain uncontaminated samples at depths up to one meter and transfer samples to an oven for analysis. The Sample Acquisition and Transfer Mechanism (SATM) requires further development. The drill must be lengthened in order to obtain a one meter drill stroke (currently, SATM has a 0.8 meter stroke). In addition, the drill-to-science instrument docking mechanism requires modification to interface with REGA, the unit housing the oven, mass spectrometer and other scientific analysis components. The drill housing is fully enclosed on Icebreaker in order to keep the temperature of the components above 100K.

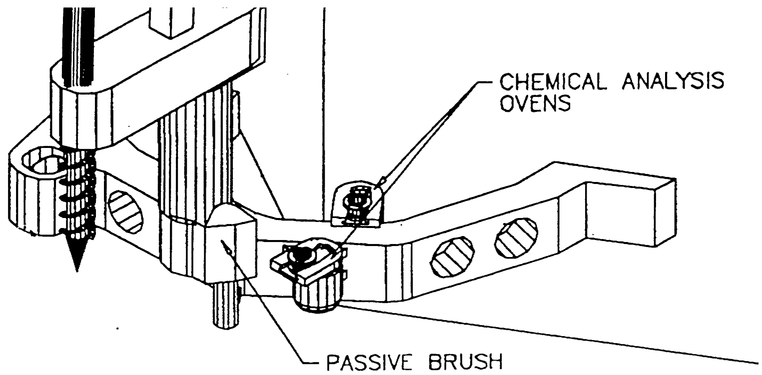


FIGURE 8.1 *SATM and ovens*

8.2.1 Sample Acquisition

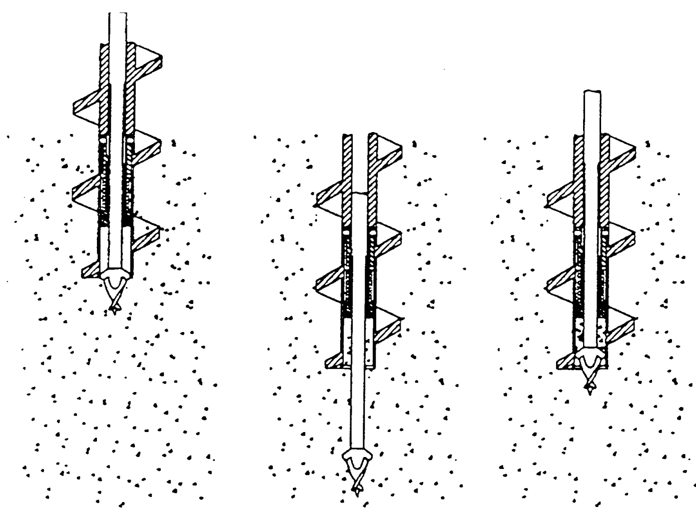


FIGURE 8.2 *Sample Acquisition*

The drill is a 'double auger' design. A drill stem, which is centered inside a main auger housing, contains a secondary auger with a custom drill bit to chisel material. The secondary auger is actuated downward with a lead screw based thrust drive train and rotated with a separate actuator. The primary auger has its own actuation. The drill stem centered inside the main auger housing creates an open annulus between housing and drill stem where samples can be captured (Figure 8.2). When the main auger is not rotating, and the inner drill stem rotates downward, material is chiseled out and transported to the sample area by the flights of the drill tip. When the drill stem is retracted, the sample is packed and sealed into the sample area. The main auger is reversed to bring the sample back to the surface for analysis. The chemical analysis instrumentation will be co-engineered with the drilling system to provide the most efficient means of transferring a sample to science instruments.

8.2.2 Sample Transfer

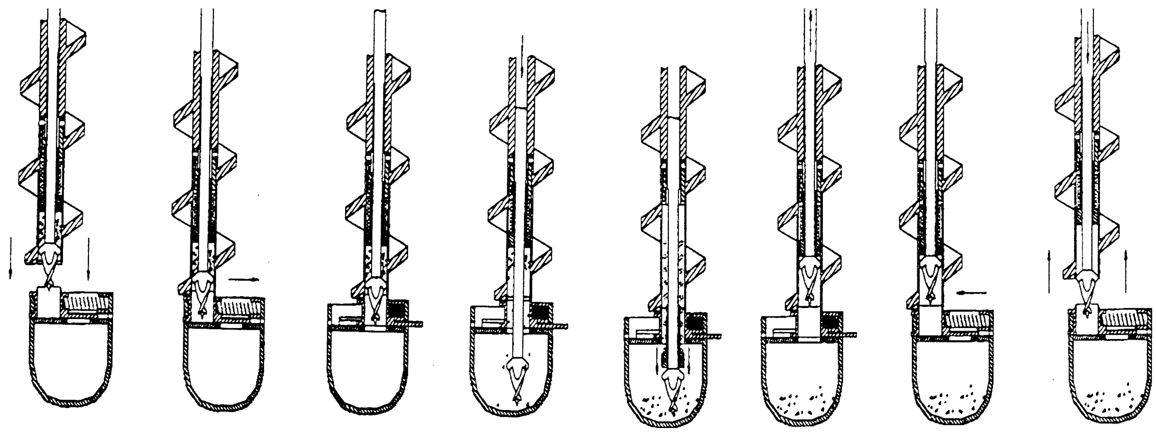


FIGURE 8.3 *Sample Transfer to Science Instruments*

The drill motions provide all actuation necessary to transfer a sample to an oven, from positioning itself over the oven to opening and closing the spring-loaded oven door. The actuation sequence is shown in Figure 8.3. The drill-to-oven docking mechanism prevents contamination or loss of the sample during transfer. The drill mechanically ejects a sample into REGA. There, it is heated to release volatiles and then the Mass Spectrometer determines relative quantities of various compounds in the sample. The drill flights and the sample area inside the drill are cleaned with wipers and passive brushes so that the next sample is not contaminated.

8.3 Sample Analysis

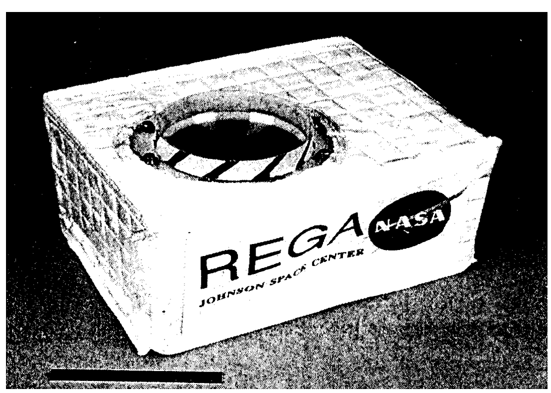


FIGURE 8.4 *REGA*

Figure 8.4 shows the Regolith Evolved Gas Analyzer (REGA) [15]. It is designed to measure volatiles. REGA consists of five subsystems: a programmable furnace to incrementally heat samples to increasing temperatures; a magnetic sector mass spectrometer which detects and measures gases that evolve during heating; tanks of pressurized gas which can be used to measure reactions of materials with different gasses; and a mechanism for dumping the sample after it is analyzed. Figure 8.5 shows the components of REGA.

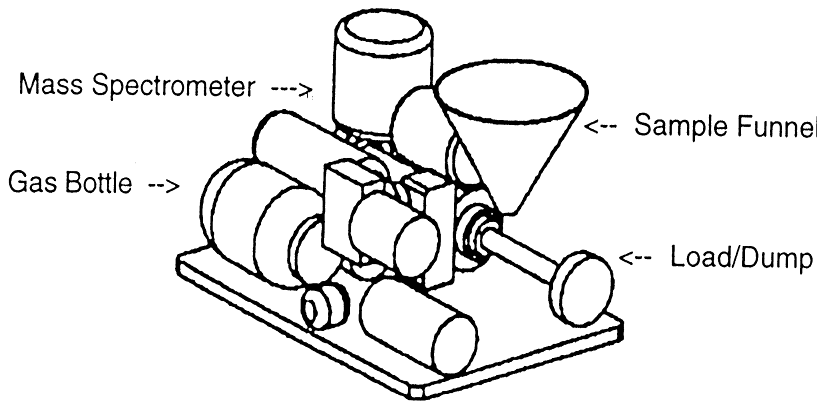


FIGURE 8.5 *REGA's components*

REGA's oven is capable of heating samples to the temperatures required to evolve water and other volatiles of interest.The oven can operate at temperatures up to 1100 degrees Celsius, but it is estimated that samples will require a much lower temperature to be volatilized--probably around room temperature.

The mass spectrometer operates as follows. As gasses are evolved from a sample, the vapors are ionized in an ionization chamber. Then these ions are accelerated through a magnetic field. The curved path that the ions follow through the magnetic field depends on their charge-to-mass ratio. For example, H+ ions follow a path with higher curvature than He+ ions. Three collectors count the number of ions passing through three specific paths. These paths represent three ranges of ion masses. It is then possible to determine the amounts of low, mid, and high mass ions in a certain evolved gas. This information is used to determine the compound from which the ions were evolved. As the oven slowly heats samples, each gas is evolved as its vaporization point is reached, thus allowing a single gas to be analyzed at one time. An embedded microprocessor controls all the processes of the oven and mass spectrometer and stores and sends data as needed.

Chapter 9

Communications

The communication subsystem on Icebreaker provides a functional path for the exchange of command and telemetry data between Icebreaker and the surface of Earth via radio frequency forward and return communication links. It also provides a tracking link via the Deep Space Network (DSN) to provide information on vehicle position and velocity during transit. The link consists of a ground station and the Icebreaker on-board communication system.

Telemetry is sent from Icebreaker to DSN at 10 kbps. The downlink includes an error-correcting coding scheme consisting of a 223/255 Reed Solomon code and a r=1/6, k=15 convolutional code. Command uplink is 2 kbps, with no compression. The link design calls for a bit error rate (BER) for downlink of 10E-5 and uplink BER of 10E-6.

9.1 Ground Stations

Because of the nature of this mission and the supporting agency and program, there are few options to consider for the ground station. The ground side of the communication link is provided by DSN 34 m antenna stations. DSN is a network of ground based antennae supported by NASA Telecommunications and Mission Operations Directorate (TMOD).

DSN consists of three sites: Madrid, Spain; Canberra, Australia; and Goldstone, CA, USA. For nearly 24 hours of each day, one of these three sites has the Moon in its field of view. Therefore, DSN will provide line of sight communication nearly 24 hours a day provided the DSN site which is pointed toward the Moon is also visible from Icebreaker's point of view.

The carrier frequency is not yet determined, since it must be allocated by DSN before construction and launch in order to ensure that each DSN client has a unique carrier. The carrier frequency will however be within a known range: uplink carrier in the range 2.025-2.120 GHz and downlink in the range 2.2-2.3 GHz (S-Band).

9.2 Icebreaker Communications

The Icebreaker communication subsystem consists of a transponder, an antenna array, and the RF assembly.

9.2.1 Transponder

The Cincinnati Electronics TTC-306 is a space-qualified S-band transponder which is currently readily available commercially. The transponder includes a command demodulator and a transmitter/modulator. All necessary power conditioning and telemetry interface circuitry is incorporated. A coherent mode of operation is provided for turnaround ranging, where the uplink carrier is used as a reference to transmit a coherent carrier back to DSN for ranging and tracking. Output RF power is 5W with total power to the transponder at 38W. This transponder has a mass of 4 kg.

A single transponder could cause a single point failure, so the communications design includes a redundant transponder. This does not affect the power budget since nominally only the primary transponder will be activated. In the event of a transponder failure, the backup transponder is activated and the defective one is powered down. The redundant transponder requires the same amount of space as the primary and adds an additional 4 kg of mass.

9.2.2 Antenna Array

The antenna array consists of a switched array of limited aperture, high gain antennas (HGA) and an omnidirectional low gain antenna (LGA).

The HGA ring consists of 11.0 dB planar microstrip patch antennas with a beam pattern of 70^{o} in azimuth (normal +/- 35^{o}) and 40^{o} in elevation (nominally 4^{o} above horizontal +/- 20^{o}). The elevation angle is nominally set to half of the maximum Earth angle to maximize the average case performance. With the Earth at an elevation of 4^{o}, the field of view in elevation for the HGA matches the locomotion requirement for terrain slope of 20^{o}, so that Icebreaker can communicate while traversing the worst case slope. The antennae are designed for circularly polarized radiation, which is used by DSN. There are six such antennae mounted in a ring spaced 60^{o} apart so that the fields of view overlap by 20^{o} degrees between antennae ensuring that at least one antenna can see the Earth any time it is above the horizon. The high gain antennas are switched on and off depending on which one has the Earth in its field of view. Figure 9.1 shows the array of six planar antennae from the top and from the side. This antenna array provides full coverage without mechanical pointing.

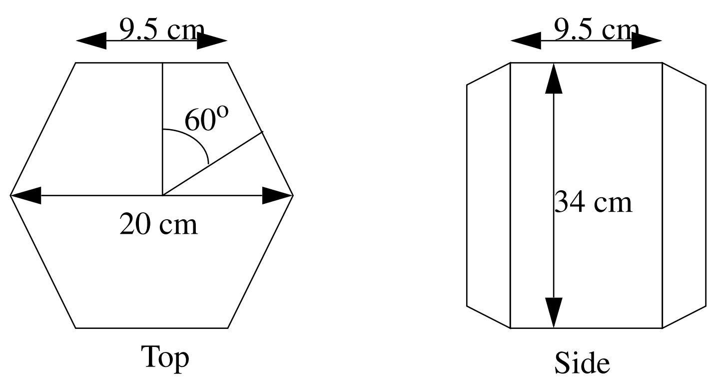


FIGURE 9.1 *Antenna array geometry*

The LGA is a 2.0 dB quadrafilar helical antenna, which is composed of four helical wires in a 25 cm long, 3 cm diameter cylindrical shape. The LGA beam pattern covers the entire azimuth with a half power beamwidth in elevation of 50^o (horizontal $+/-25^o$), again circularly polarized. The low gain antenna is switched off unless for some reason the high gain antenna fails. If the HGA fails, then the LGA is used at a reduced data rate.

The HGA array is covered by a cylindrical radome to prevent any damage by potential projectile impact on landing and micrometeorite impacts after landing. This radome is S-band transparent. The HGA is mounted on a mast above the top rear of the solar panel. The LGA is also covered in a radome and is mounted directly above the HGA.

9.2.3 RF Assembly

The RF assembly provides all necessary functionality between the transponder and the antenna array. The RF assembly contains a sampling receiver which performs RF signal sampling and monitoring functions, and an embedded controller which determines which antenna to activate and switches the signal path to that antenna. A functional block diagram of the RF assembly appears in Figure 9.2.

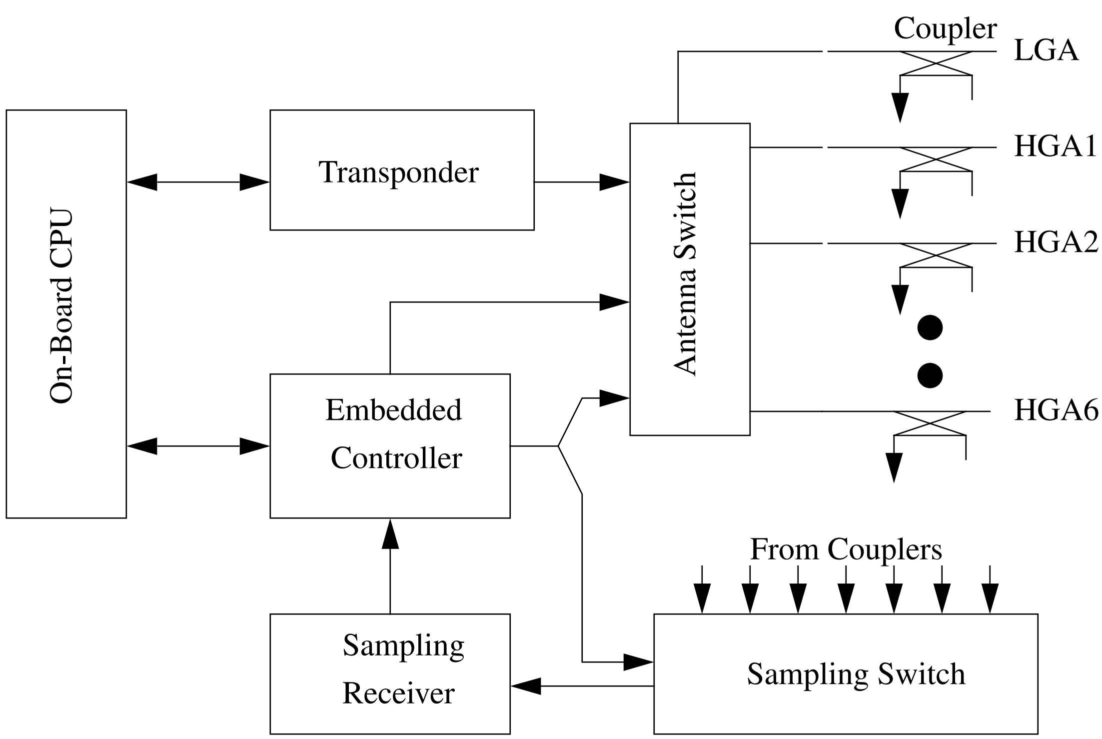


FIGURE 9.2 *RF Assembly*

9.3 Link Analysis

A brief discussion of the link analysis and a rough link budget are presented here. For a detailed discussion and thorough link analysis, see Appendix F.

In order to evaluate the link design, the Consultative Committee for Space Data Systems (CCSDS) *Link Design Control Table* was used[14]. This 6-page spreadsheet and associated macro functions are published by the Telecommunications and Mission Operations Directorate for use in developing communications architectures for missions such as this one. The entire 6-page spreadsheet appears in Appendix F.

Table 9.1 provides a condensed version of the HGA link analysis with only the most salient figures. In the table, dB refers to Decibels, y dB = 10 $\text{Log}_{10}(x)$, and dBW refers to Decibel Watts, y dBW = 10 $\text{Log}_{10}(x \text{ W})$. Table 9.2 provides a condensed version for the LGA link analysis. The only changes between the link budget for the high gain downlink and low gain downlink are in the antenna gains and the data rates.

TABLE 9.1 *High Gain Antenna Link Budget (Downlink)*

Link Parameter	Design Value	Comments
Data Rate	10 kbps	Not for real time live video
Transmitted RF Power	6.99 dBW	5 W RF out from 38 W DC in
Transmit Antenna Gain	11.0 dB	Limited aperture antenna
Total Losses	-217.2 dB	Transmitter, path, and receiver losses
Receiving Antenna Gain	56.9 dB	DSN 34BWG1, Block V
Received Power	-143 dBW	
Received E_b/N_o	8.15 dB	Ratio of energy per bit to noise level
Required E_b/N_o	1.47 dB	r=1/6 Convolutional and (223,255) Reed Solomon
Data Performance Margin	6.69	Factor of 4 margin in signal strength

.

TABLE 9.2 *Low Gain Antenna Link Budget (Downlink)*

Link Parameter	Design Value	Comments
Data Rate	1 kbps	Reduced from HGA value
Transmitted RF Power	6.99 dBW	5 W RF out from 38 W DC in
Transmit Antenna Gain	2.0 dB	Beam is omnidirectional in azimuth
Total Losses	-217.2 dB	Transmitter, path, and receiver losses
Receiving Antenna Gain	56.9 dB	DSN 34BWG1, Block V
Received Power	-143 dBW	
Received E_b/N_o	8.15 dB	Ratio of energy per bit to noise level
Required E_b/N_o	1.47 dB	r=1/6 Convolutional and (223,255) Reed Solomon
Data Performance Margin	7.63 dB	Factor of 5 margin in signal strength

9.4 Unresolved Issues

There is an open question in consideration of communications requirements. Due to low Earth viewing angles, there may be multipath effects where communication signals take paths which follow direct line of sight and paths which reflect energy off of the lunar surface. This can cause destructive interference and degrade signal to noise performance. The multipath effects are difficult to quantify, however. It is assumed that the 6 dB or more of margin in the design will be enough to overcome the multipath problem.

An additional issue that has not been investigated in detail is whether the Earth will remain in the field of view of the HGA as the rover traverses obstacles on the lunar surface. For example, if Earth is centered in the HGA's vertical field-of-view at a 4 degree elevation and is located directly in front or behind Icebreaker, an obstacle of 0.4 meters under the front or rear wheels would cause the Earth to be at the edge of the +/- 20 degree field-of-view for the HGA. The vertical field of view of the HGA was selected based on the maximum traversable slope of Icebreaker. One solution to this issue is to use Icebreaker's positioning system to determine when a slope of greater than 20 degrees is being traversed and have Icebreaker back off this obstacle. The terrain maps, built as Icebreaker traverses waypoints, could also be used to plan paths around obstacles that are above these minimum height requirements. The main task requiring high-bandwidth communication, the transmission of stereo images, will occur when Icebreaker is stopped, allowing it to be done when Earth is definitely in the field-of-view for high-bandwidth communication.

Chapter 10

Power

The power system on Icebreaker provides a source of electric power that meets the mission requirements in the Lunar South Pole environment. It consists on three main components: solar panels for generation, batteries for storage and power electronics to control the energy flows.

10.1 Solar Arrays

Icebreaker‘s solar array provides the only source of electrical power. Its design addresses two main issues: cell technology and geometry.

10.1.1 Cell technology

Gallium arsenide and silicon solar cells have both been used in previous space missions. Gallium arsenide cell technology converts sunlight to electricity more efficiently and is more resistant to radiation decay. However these cells cost 2-5 times more than silicon, weigh 2.2 times as much and are not as mature as silicon technology.

Since Icebreaker requires a large amount of electricity and since space inside the Delta rocket fairing is at a premium, gallium arsenide cells were chosen for Icebreaker's solar array. Using an incident power of 1380 W/m^2 on the surface of the Moon and 18% efficiency for the solar cells, Icebreaker's solar panel will have an output of 248 W/m^2. Considering both power requirements and space limitations, a 1.5 m^2 solar array generating 372.6 W is specified.

10.1.2 Geometric configuration

The Sun is never more than 2 degrees above the horizon at the Lunar South Pole. To be perpendicular to incident light the solar array should be vertical. Three different solar panel designs were examined: fixed cylindrical, pointable flat surface and a fixed flat surface.

A fixed cylindrical solar array would encircle the body of Icebreaker with solar cells much like a communications satellite. The advantage of this strategy is that a you have full power generation regardless of your heading with respect to the Sun. No actuation is needed to point the solar panel at the Sun. However, because the efficiency of solar cells decreases quickly as the incidence angle of Sunlight decreases from 90 degrees, the effective area of the solar panel is quite small compared to the diameter of the cylinder. Thus a very large cylinder is required to generate sufficient power. This creates mass problems as well as volume inside the Delta rocket fairing.

A pointable flat solar panel eliminates the need for the excess solar cells of the cylindrical design by actively keeping the solar panel perpendicular to the Sun. This method is certainly more efficient in terms of mass and volume than the cylinder. However, the large size of the solar array makes it difficult to move such that it is not obstructed by the drill stem, communications antenna and various cameras and sensors mounted on Icebreaker. The motor required to turn the array also introduces a failure point which could leave the solar panel in an unfavorable position.

The final solar panel design examined was a fixed solar panel. A flat solar panel fixed in position means that Icebreaker may have to operate off its batteries even during Lunar Summer conditions since incident light may not be sufficiently close to perpendicular to generate sufficient power. This will require Icebreaker to cease operations at times to recharge. However this simplifies the design of the solar array and makes dual use of the large battery reserves which are required for cold trap investigations. Power can still be generated, even if the desired driving direction has the solar array parallel to incident sunlight, by tacking. Like a sailboat sailing into the wind, Icebreaker could traverse a zig-zag trajectory to allow some solar energy generation.

The fixed solar panel design was chosen. The 1.5 m^2 array is mounted from the bow to the stern of Icebreaker. While this solution does limit Icebreaker's performance it will not affect its ability to complete its mission in any way, and will make full use of the battery power available.

10.2 Battery

Battery requirements are driven by the amount of power needed for Icebreaker to enter a cold trap, collect samples and exit. Given the expected duration of the entire mission, the batteries should be capable of over a hundred charge/discharge cycles without losing their recharge capacity. This is called the cycle lifetime feature.

10.2.1 Battery cell technology

While primary (non-rechargeable) cells provide higher energy densities than secondary (rechargeable) cells, the duration of the mission suggests that secondary cells should be used. Several different types of secondary cells were investigated.

NiCd and NiH_2 Batteries

Two common choices in battery technology are nickel cadmium (NiCd) and nickel hydrogen (NiH_2) batteries. NiCd batteries have been used in space applications for more than 20 years. NiH_2 batteries require each cell to be individually contained in its own pressure vessel to maintain the cell's hydrogen gas at a pressure of 6.2 MPa. NiH_2 batteries are more voluminous than NiCd, but offer substantially longer cycle (charge/discharge) lifetimes, greater depth of discharge and improved tolerance of overcharging. Energy densities in both types tends to be low.

AgZn Batteries

Silver zinc (AgZn) battery technology has a greater energy density (Wh/kg) than NiCd or NiH_2. AgZn batteries have been used in both military and space applications. They have fewer recharge cycles (150-200) than NiCd or NiH_2 but enough for this mission. Given that large energy storage is needed for cold trap sorties and mass is limited, Icebreaker will use AgZn batteries.

10.2.2 Battery requirements

The battery pack on Icebreaker, has to deliver all of the energy for a complete cold trap sortie. It is estimated that a typical cold trap sortie will require 3 hours to enter the cold trap, 4 hours for sample collection and analysis and 3 hours to exit for a total of 10 hours. Driving requires 274 W and sample acquisition 192 W. Therefore the batteries must be capable of providing 2412 Wh.

Given the uncertainty of the necessary sortie configuration, a safety factor of 3 is appropriate to cope with different cold trap terrains. Therefore, the battery pack is intended to provide 7236 Wh. The energy density of the AgZn batteries is 150 Wh/Kg. Therefore, the total amount of batteries necessary is approximately 48 Kg.

10.3 Power electronics

The power electronics provide the means to optimally use the energy available in the batteries or solar array to supply stable voltages to the subsystems of Icebreaker. It has a solar panel regulator, battery charge supervisor, voltage solid state converters to provide the required voltages (+28, +12, -12, +5, -5), and solid state switches to control the power supply of single elements.

The power electronics is closely linked to computing to achieve a proper distribution of power. The battery pack variables (charge, recharge or use rates, battery temperature), the solar panel variables (voltage, currents) and each independent subsystem power usage, are monitored and transmitted over a serial line to computing, as part of the internal Icebreaker state. Also, a command message is transmitted back from computing to execute power management tasks, as dropping loads, or adding them according to the mission priorities.

Chapter
11

Thermal Design

The operational environment consists of large temperature variations-from 130˚C while Icebreaker bakes in the sun to -250˚C in the cryogenic cold traps. Icebreaker must be able to tolerate these large temperature changes and regulate it's internal temperature. The electronic components must be kept within temperature bounds of -40 to 50˚C. CCD chips for video cameras should be kept below 30˚C to avoid degradation of image quality due to dark current.

11.1 Paths for energy transfer

There are two modes of heat transfer on the lunar surface: conduction and radiation.

11.1.1 Conduction

The conduction through a surface can be modeled by Fourier's Law where the heat flux is proportional to the temperature gradient. The proportionality constant k is called the thermal conductivity of the material that the heat is being transported through. Typical material conductivities can range from 389 W/m/K for copper to 0.02-0.03 W/m/K for various polymer foams. Fourier's Law is shown in the equation below.

$$q''(x) = -k\frac{dT}{dx}$$

11.1.2 Radiation

The loss of energy through radiation from a surface is shown by the equation below. In this equation the emissivity, ε, is a measure of how efficient the surface emits compared to an ideal radiator. Value for emissivity range from 1, black body, down to 0 ideal. σ is the Stefan-Boltzmann constant and T_s, $T_{surroundings}$ are the temperatures of the surfaces and the surroundings. The vacuum of space is equivalent to an ambient temperature of 3K.

$$q'' = \varepsilon\sigma(T_s^4 - T_{surroundings}^4)$$

The energy transfer to a system through absorption is proportional to the incident energy, G. The proportionality constant α is called the total hemispherical absorptivity which is the average for both direction and wavelength. Values for α range between 0 an1 which is ideal.

$$q'' = \alpha G$$

11.2 Mission Scenarios

The mission can be broken up into three thermal scenarios: hibernation, cold trap operations and Lunar summer operations.

11.2.1 Hibernation

During hibernation energy can be lost through conduction, at contact points with the lunar soil, and through radiation from the surface of the Icebreaker, see Figure 11.1. The heat sources are the electronics or computing (if any are working) and Radiative Heating Units (RHUs).

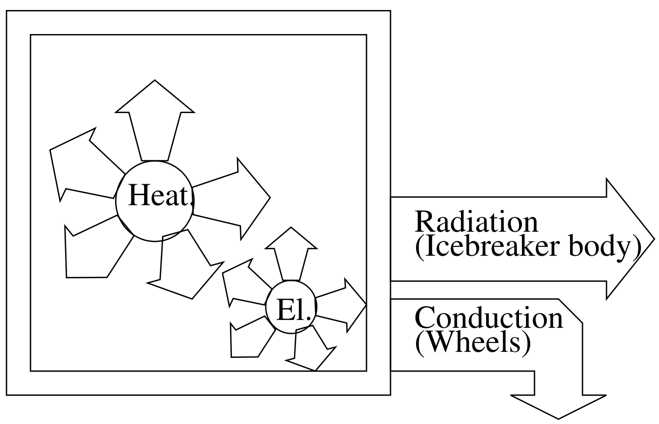


FIGURE 11.1 *Heat flow diagram for hibernation.*

11.2.2 Cold Trap Operations

During operations in a cold trap energy can be lost through conduction, at contact points with the lunar soil, and through radiation from the surface of the Icebreaker, see Figure 11.2. The heat sources are the electronics, computing and sensors used to sense and analyze the contents of the cold trap. Heat will also be generated from locomotion if Icebreaker is moving.

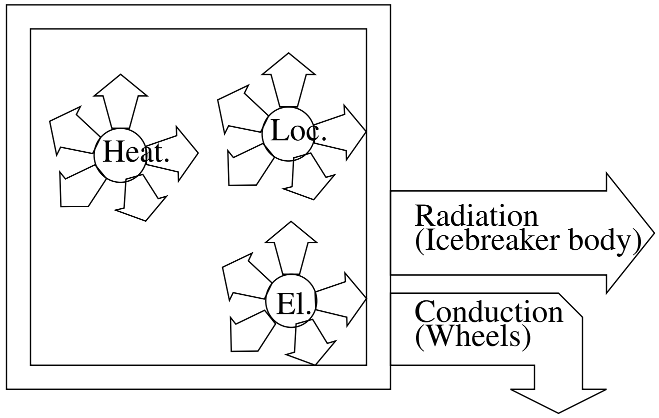


FIGURE 11.2 *Heat flow diagram for cold trap operations.*

11.2.3 Lunar Summer Operations

While Icebreaker is in sunlight, energy is gained through absorption of solar radiation, see Figure 11.3. Depending on local condition energy may be gained or lost through conduction at contact points with the lunar soil. Additional heat will be generated by electronics and locomotion if Icebreaker is moving. Since the temperature could rise due to the high solar energy absorption, radiators are used to provide cooling.

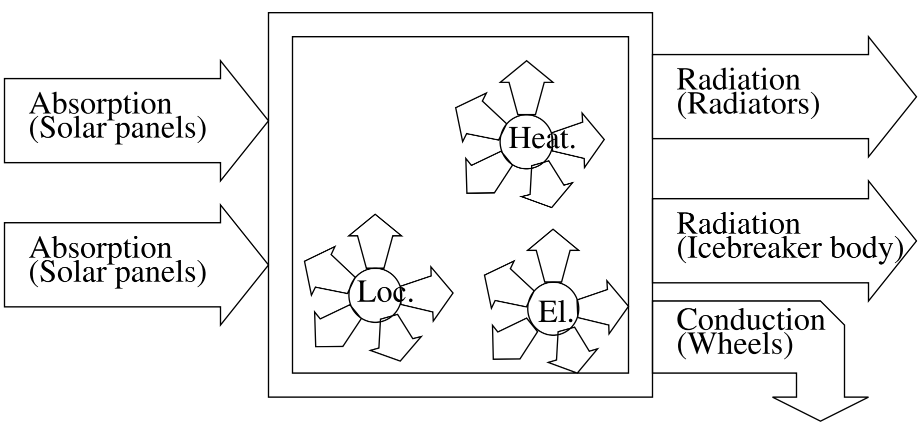


FIGURE 11.3 *Heat flow diagram for lunar summer operations.*

11.3 Thermal Components

Many specialized materials and components have been developed for thermal regulation in the extreme conditions of space. The specialized techniques and materials that are called for in the design of Icebreaker are heat pipes to transport heat, surface coatings which provide desired absorption/reflection characteristics, RHUs, radiators, and multilayer reflective insulation to prevent leakage of heat energy across thermal gradients.

11.3.1 Heat Pipes

Heat pipes are tubes composed of material with high thermal conductivity so that they may conduct energy efficiently when attached to a device. Within these primarily metal containers is a liquid, also of high thermal conductivity, which transfers the energy away from the walls through both conductive and convective modes. Thus the effects of a local energy source may be spread over a larger volume, making it's effects less severe. These systems may be employed at local internal energy sources where the energy must be quickly dissipated in order to prevent local temperatures from rising above limits. Heat pipes can be fabricated in a variety of shapes to provide even temperatures for various components and enclosures. A typical heat pipe of 1.27cm in diameter has a capacity of 5080 Watt-cm. Yet another idea is to restrict the flow of the fluid within the pipes, thus during operations certain channels of heat flow may be shut off and turned on. Such a design will facilitate the intelligent heat regulation of Icebreaker, enabling it to deal with all the outlined scenarios and transitions between these scenarios.

11.3.2 Surface Coatings

Paints and coatings with desirable emittance and absorptance parameters can be used for effective thermal regulation by radiative effects. Generally, high emittance is desirable so that internally generated heat can be radiated effectively and absorptivity is desirable to minimize the effects of variations in solar heating. White paints, epoxies, and ceramic coatings are effective for thermal radiators when exposed to solar radiation due to the combination of a low solar absorptivity below 0.2 and a thermal emissivity above 0.8. There are also materials with a similar ratio of emissivity to absorptivity such as metallized teflon and optical solar reflector (OSR) surfaces, which have greater cost and weight.

RHUs and Radiators

These two components will serve as the main heat source and heat sink of each thermally controlled region within Icebreaker. As the major source of heat, each RHU is attached to the heat pipe network. If the contents of the thermally controlled region drop below the median operation temperature, energy will flow through the heat pipes and heat them. Otherwise any excess energy will be dissipated via the radiator array as shown in Figure 11.4.

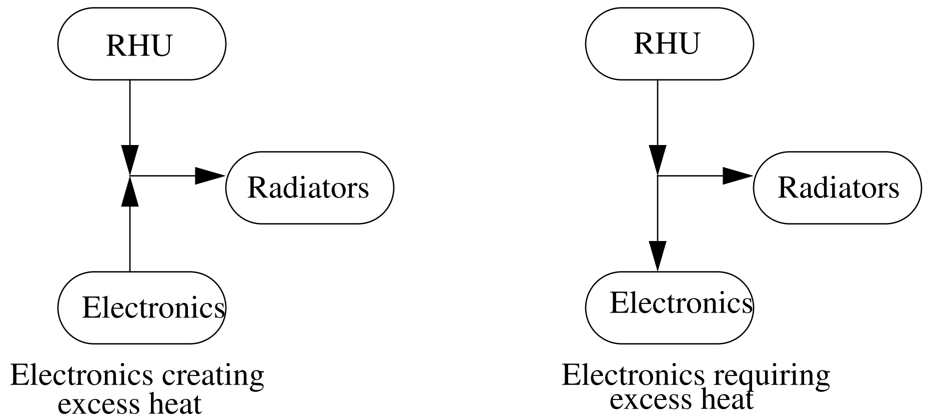


FIGURE 11.4 *Typical operation scenario for temperature controlled regions.*

Thus the heat pipes will be specified according to the energy balance of each thermally controlled region. In addition the radiators will always point into deep space, taking advantage of the low sun angles.

11.3.3 Multilayer Reflective Insulation

This insulation is made of thin layers of metallic foil separated by insulating oxides or fabric in a vacuum. Thus there is little or no physical conduct, reducing if not eliminating the consideration of conductive energy loss. The transfer of energy across this material is primarily by thermal radiation from one layer to the next. The performance of this insulation type depends heavily on the details of its configuration. Seams, attaching points, and folds are all paths for energy leakage through the insulation.

11.4 Thermal Configuration

Icebreaker's thermal design calls for RHUs to provide heat. In the lunar environment described earlier in this paper, with low Sun angles at the Lunar South Pole, Icebreaker can easily exhaust energy to prevent overheating. The two main considerations are how to regulate this exhaust and where to exhaust it. Icebreaker will employ a passive regulation of energy through heat pipes and radiators.The most desirable direction to position a radiator is normal to the surface of the Moon. By taking advantage of the low sun angles, a radiator may be placed on top of Icebreaker and have a nearly constant dissipation into black space, thus dissipating any excess heat from the combined RHU output and other thermal insulation described above.

Chapter 12

Computing

Computing provides the processing, storage, and control capabilities to operate Icebreaker and manage the information flow. The design must be space qualified, meet power requirements and development time and cost constraints. For these reasons, the design calls for off the shelf components that are compatible with the VME bus standard. All components are housed in one chassis. The boards satisfy the requirements for processing, memory, communications, image processing and compression, sensor data processing, data storage, internal communications and control. There is no unit redundancy, but the boards are overspecified for the temperature, vibration and shock requirements, thus providing a reliable system for the duration of the mission.

12.1 Task identification

Icebreaker‘s computing must allow for the following tasks to be accomplished:

- Control actuators in the locomotion subsystem
- Control actuators in the science instruments
- Control propulsion and landing subsystem
- Interpret sensor data, build terrain maps
- Compress imagery, store history data
- Control communications to Earth
- Control operation modes and states of Icebreaker
- Perform position and orientation estimation

12.2 Locomotion control

The computing unit provides processing power to have control loops to servo the actuators on the wheels and steering. Icebreaker has motors in each of the four wheels, and each is controlled to achieve a fair and optimal distribution of the torque to drive over very rough terrain. The steering actuator is controlled to achieve the proper turning radius for path following. The locomotion subsystem uses almost half of the power budgeted for Icebreaker. Also, the fact that it is not possible to always drive with the solar panels perpendicular to the sun, calls for optimal control of the locomotion subsystem, full supervision of the current and voltage for each motor, and tight control of the amplifiers. The locomotion computing is required to generate commands for the amplifiers, and read encoder and electrical information to close the control loops. This task calls for interfacing circuitry in the form of analog input/output channels and analog-to-digital and digital-to-analog conversion boards. Processing power is also necessary to perform the control task described.

12.3 Science instruments control

The drill has its own controller so the computing interfaces with this unit, sending high level commands, and gathering the information resulting from the experiments. The computing has to acquire images taken with the passive sensor, and analyze the data to assess a probability of ice in a coldtrap, before driving into it. This sensor provides color images. The histogram and color spectrum information that is necessary to perform the task of identifying areas that are promising for finding ice must be computed from these color images. The interface for this sensor is the common image acquisition hardware used for the rest of the imaging sensors. Analog and digitial output channels will be used to control the strobe light necessary to acquire images in the dark.

12.4 Imagery

Icebreaker's design calls for eight cameras: a panospheric camera, two belly cameras, two stereo cameras, a camera for remotely detecting ice, and two star tracker cameras. The computing provides the digital interface for acquiring the data from each of these cameras, and transferring it to the image processing engine. See Figure 12.1. The interface consists of a framegrabber board that can store the monochrome images coming from the stereo pair, the high resolution color frames of the panospheric camera, the monochrome images from the belly camera or the passive ice sensor and the high resolution images from the star tracker cameras.

Depending on the operating mode of Icebreaker, the acquisition board takes data from only one source, or multiplexes the images coming from more than one. Communication data rates and the reduced speed of Icebreaker allow for low acquisition rates, both for the panospheric camera and the stereo pair. The sources are multiplexed if necessary, depending on the mode of operation. In case Icebreaker is landing, high rate acquisition is required for the belly camera, and no multiplexing occurs.

Once a panospheric or stereo pair frame is acquired on the framegrabber, it is transferred to the digital signal processor board. This board has two digital signal processors that run code to compress the images into suitable messages to be sent to Earth or stored in memory. If the image is coming from the passive ice sensor, the data is processed on board, and also compressed to be sent to Earth.

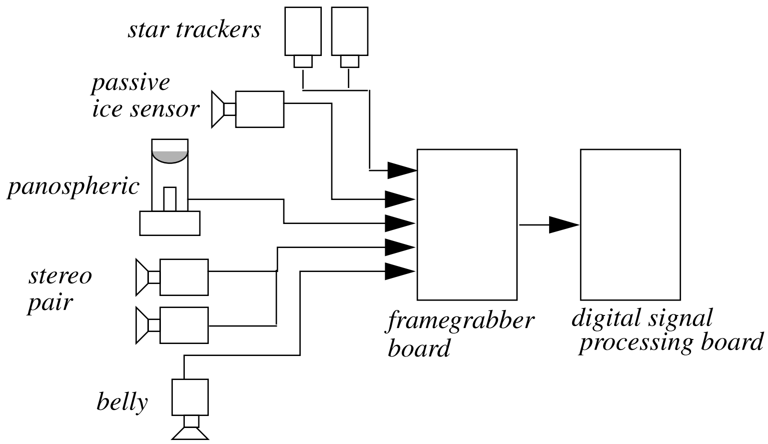


FIGURE 12.1 *Visual sources and their path to processing*

When Icebreaker is landing, the whole set of acquisition-processing is dedicated to getting and analyzing the visual information for landing site detection and tracking. This information is also compressed and sent to Earth, to let mission operators choose a site before Icebreaker transitions to autonomous landing mode. Icebreaker logs data of this process while it is happening.

12.5 Scanning radar and other sensors

Icebreaker's design specifies radar for terrain mapping. Radar data is acquired with an analog port, and is processed to allow terrain mapping, suitable for autonomous and assisted driving.There is also a serial port for initiating a scan and for checking status. The data is sent to the digital signal processor board to perform fast Fourier transforms on the raw data from the radar unit. Appendix E discusses the processing required to build this terrain map.

Other sensors used for position estimation include two inertial measurement units, that transmit data packets via an RS-422 serial link and wheel, steering, and bogie encoders which provide information via digital signals.

12.6 Communications

Icebreaker's preferred mode of operation is with communications to Earth available. It must transmit packets that contain telemetry data, compressed imagery, and receive compressed messages with commands. To send and receive all this information, a full message encoding and compression has to be performed. The data transfer is monitored to and from the communication unit. This data transfer occurs through a synchronous serial channel, with high reliability and transfer rate. See Chapter 9 for more details.

12.7 Command and control

Icebreaker receives commands from Earth and validates these commands with respect to the current state. Command and control software chooses the appropriate modes and calls lower level planning and execution modules which accomplish the desired actions. These actions are dictated by commands from operators on Earth or by the situations encountered on the lunar surface. Icebreaker's environment defines different modes of operation according to Sun and Earth line of sight. Chapter 3, "Mission Operations", includes more information about the operational modes and states.

12.8 Position and orientation estimation

Attitude data derived from the star tracker camera images, terrain maps from the radar unit, odometry, and data from the inertial navigation unit is used to estimate the position and orientation of Icebreaker. Given the stochastic properties of the measurements, an optimal way to fuse the data coming from the sensors is to use a Kalman filter. The output is the position of Icebreaker on the local maps, plus overall position on the lunar surface.

12.9 Board and chassis configuration

The computing design calls for one hermetically sealed chassis containing bus and boards that conform to the VME bus standard. The power supplies are part of the power subsystem and are located within the power electronic unit. The dimensions of the chassis are 200x350x125 mm., and the weight of the chassis plus the backplane and the boards is less than 15 Kg.[6]. Six electronic boards will be used to address the requirements for each task mentioned above. These will be: 1) main processor board, 2) digital signal processor board, 3) framegrabber board, 4) analog input/output board, 5) digital input/output board and 6) memory board. See Figure 12.2.

The main processor board runs the operating system and all the control, servo and logic processes. It has a Motorola 68040 processor, with programmable read only memory to store the operating system and application programs. It also has random access memory to run those applications, and serial channels to communicate with other on board devices (such as the drill controller). This board is also the bus master board, providing access to the rest of the boards as memory locations mapped to the local memory across the bus lines.

The digital signal processor board has four connected digital signal processors, dedicated to the analysis and compression of images, both during the landing process and normal operation. This board has excellent performance when dealing with imaging data, that comes mainly from the framegrabber board through a local bus joining them. It also has a set of memory blocks, designed to give superior performance to the data processing.

The analog input/output board provides an interface for the analog signals of Icebreaker. The output signals produced are the velocity commands for the wheels and the steering servos.

The digital input/output board reads the values of the encoders found on the wheels and steering, the control signals of the power subsystem, the status signals of the science instruments (except the drill controller, which communicates serially). With the outputs, it controls the science instruments, the sensors, the lights, power management and enables the other subsystems.

The memory board has space of random access memory (1 Gigabyte) to hold information gathered during periods without communications. Scientific data and compressed images are stored here for later transmission to Earth. This memory would have a separate power line in addition to normal power lines that would keep memory from being destroyed even in the event of hibernation or low power situations.

Finally, the frame grabber board multiplexes the data coming from the onboard cameras. It receives image frames and transmits them through the local bus to the digital signal processor board, for their analysis and compression.

Each of the VME bus boards has heat conduction bars that allow the transfer of the heat from the integrated circuits, to the chassis frame, which is mounted to a plate whose temperature is controlled by the thermal subsystem. The total power dissipation capacity of the conduction bars and chassis exceeds by a factor of two the maximum of 50W of power consumption of the computing subsystem.

The electronic enclosure receives power from the power subsystem (+12V, -12V, +5V). There are power management devices to selectively supply power to the boards in a priority fashion.

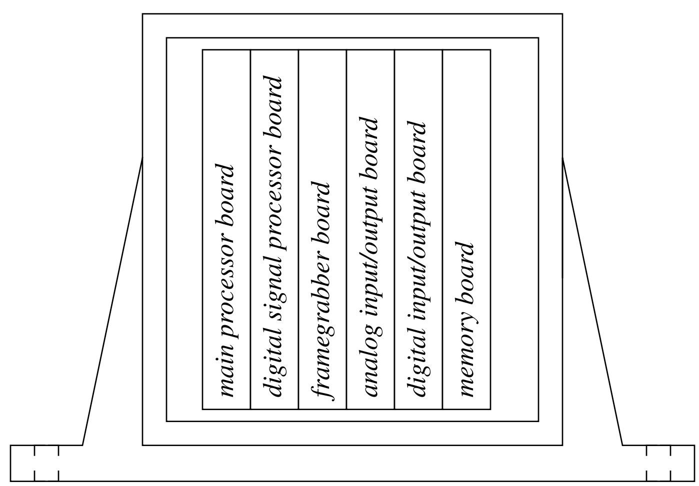


FIGURE 12.2 *Board distribution in the VME chassis*

12.10 Processing Budget

The processing budget justifies the processing power required by the design. The tasks are part of the main tasks assigned to computing, and MIPS (millions of instructions per second) is used to characterize each task. The main processor board has the processing power to handle all tasks, except those of image analysis and compression, and terrain map building. Those are handled by the digital signal processing board.

TABLE 12.1 *Processing power budget*

MIPS	TASK
5	Command and control
10	Kalman filtering and Pose estimation
5	Arc following and driving
10	Terrain safeguarding and path tracking
150	Image analysis compression
5	Data encryption, encoding, packaging and telemetry
5	Science and Misc. Tasks: DSN tracking, Drill control, image processing

Appendix A

Lunar Environment

A.1 Sun and Earth Angles

A.1.1 Saros Cycle

The Sun, Earth, and Moon interact gravitationally in a complex pattern. Each body's orbit about the other two follows Kepler's laws of orbital mechanics, but the interaction of all three bodies creates complications. In order to determine at what angle sunlight hits the surface of the Moon, or at what angle the Earth is above the lunar horizon, the position of the Moon with respect to the Sun and Earth needs to be known. Newton devoted much time to determining accurate predictions for the Moon's position, but complained that thinking about the problem "made his head ache and kept him awake so often that he would think of it no more" [18]. We have come to understand his complaint.

Fortunately, certain patterns can be determined, and simplifications can be made to determine orbital parameters accurately enough for our purposes. Various cycles occur, the longest of which is known as the Saros. This cycle lasts 18 years plus 10 or 11 days (depending on how many leap years intervene). The orbital configuration of the three bodies repeats itself almost exactly after this period of time. Therefore, if the position of the Moon is known for an approximately 18 year period of time, the maximum angles of the Sun and Earth above the lunar horizon for all time can be determined.

The reason for this 18 year cycle can be explained by Figure A.1. The Earth's orbit about the Sun occurs in a plane called the ecliptic. In the ecliptic coordinate system, the direction from the center of the Earth to the point labeled v remains constant, and points in the direction of the vernal equinox. When the Earth is at this point in its orbit, the vernal equinox, commonly known as the first day of spring, occurs. The Moon's orbit about the Earth occurs in a different plane, shown by the plane labeled "lunar orbit.". These two planes intersect along a line called the line of nodes. There are two nodes, each at a point where this line intersects the orbital path of the Moon, one where the Moon is passing beneath the ecliptic (the descending node) and one where the Moon is passing above the ecliptic (the ascending node). The descending node is labeled N in this figure. The arc length between v and N, along the ecliptic, changes over time, however, with N circling counterclockwise around the ecliptic, as viewed from above. The time it takes for N to return to its original position is the 18 years of the Saros cycle.

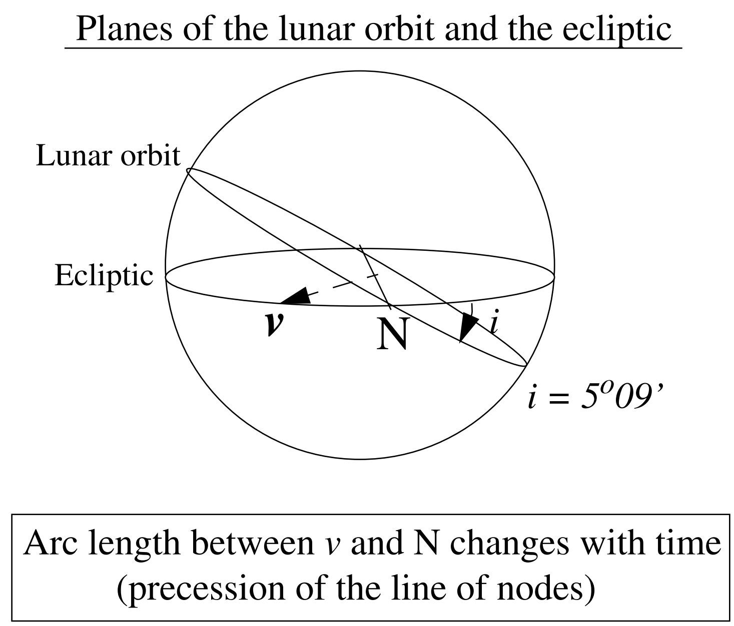


FIGURE A.1 *Planes of the lunar orbit and the ecliptic*

To determine the Moon's position for those 18 years, portions of a program called SatTrack [4], developed to track various satellites from the Earth, were modified for tracking the Sun and Earth from the Moon. SatTrack gives the positions of the Moon's and Sun's centers as seen from the center of the Earth, for any given date. The accuracy is approximately one second of arc. Expansions of the program were then written to convert the results between various coordinate systems and then compute the resulting angles of the Sun and Earth above the lunar horizon. By running the program for a span of dates covering a complete Saros cycle, the following maximum angles (Table A.1) were computed for various latitudes on the Moon.

Latitude	Sun angle (deg)	Earth angle (deg)
90 S	1.5562	6.6153
89.5 S	2.0495	7.1005
89 S	2.5474	7.5989
88.5 S	3.0453	8.0974
88 S	3.5446	8.5958
87.5 S	4.0439	9.0943
87 S	4.5438	9.5928
86.5 S	5.0438	10.0912

TABLE A.1 *Maximum angles of the Sun and Earth above lunar horizon*

A.1.2 Yearly Cycle

Another cycle of the Sun-Earth-Moon system is a yearly cycle. As the Earth and Moon revolve about the Sun, once per year, the tilt of the Moon's rotation axis with respect to the ecliptic causes the Sun's angle above the lunar surface to vary. Figure A.2 (not to scale) demonstrates why this occurs. On one side of the Sun, the Moon's south pole (marked by an x) is pointed more directly at the Sun, while on the other side of the Sun (half a year later), the pole is pointed away. Thus for 6 months of the year, the south pole remains lit by the Sun, while for the other 6 months, it remains dark, similar to the poles of the Earth. Thus, the elevation of the Sun above the lunar horizon near the south pole varies with a yearly pattern.

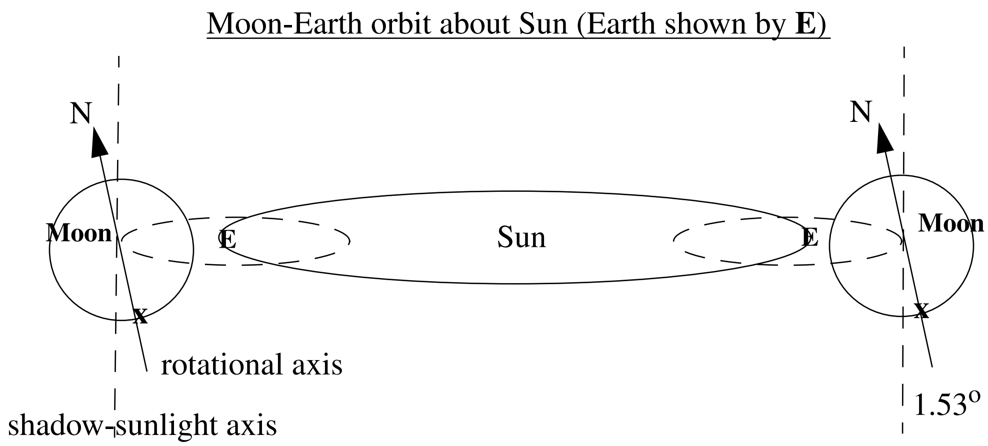


FIGURE A.2 *Moon-Earth orbit about Sun*

A.1.3 Monthly Cycle

A monthly pattern also occurs for both the Earth and Sun angles above the lunar horizon. This occurs because of the Moon's revolution about its axis. For a point not exactly on the south pole, the rotation of the Moon causes that point to inscribe a small circle in space about the south pole (see Figure A.3). The angle at which the sunlight hits that point thus varies with the Moon's rotation period, which is approximately one month, or 28 days. For points further towards the Moon's equator, this results in lunar days and nights, each of an approximately 14 Earth day length.

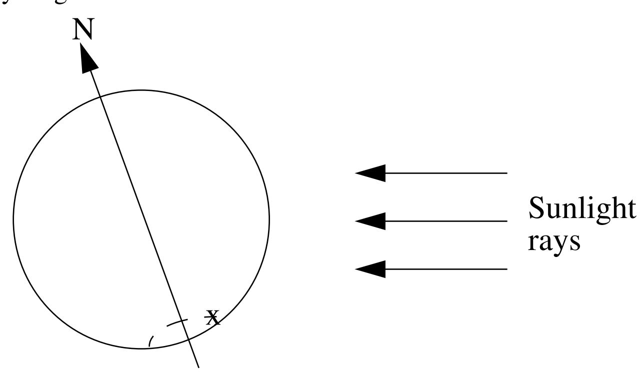


FIGURE A.3 *Monthly rotation of the Moon*

Similarly, the Earth's position above the horizon for a point near the Lunar South Pole also has a monthly cycle. The variation in angle is greater for the Earth than for the Sun, since the Moon's rotational axis is more greatly inclined to its orbit about the Earth than to its orbit about the Sun. This means communication with the Earth is only possible for 14 of every 28 days. The rotation of the Moon about the Earth once a month causes additional variations in the angle of the Earth above the horizon, as do several other characteristics of the Sun-Earth-Moon system. However, the main result for Icebreaker's purposes is that communication with Earth will be available only cyclically.

A.1.4 Horizon Angles Graphs

Putting all these cycles together, one can see how the Sun and Earth angles above the lunar horizon vary over time. Figure A.4 shows the angular variations for the period of one year, from January to December 2000, for the location 89.3 S, 105 W, the point referred to in Section 5.1.1. The Sun angles are shown by the dashed line, and one can see both the yearly and monthly cycles. The Earth angles have a greater amplitude, but still show the monthly cycle, appearing and disappearing below the horizon, the dotted line at 0 degrees.

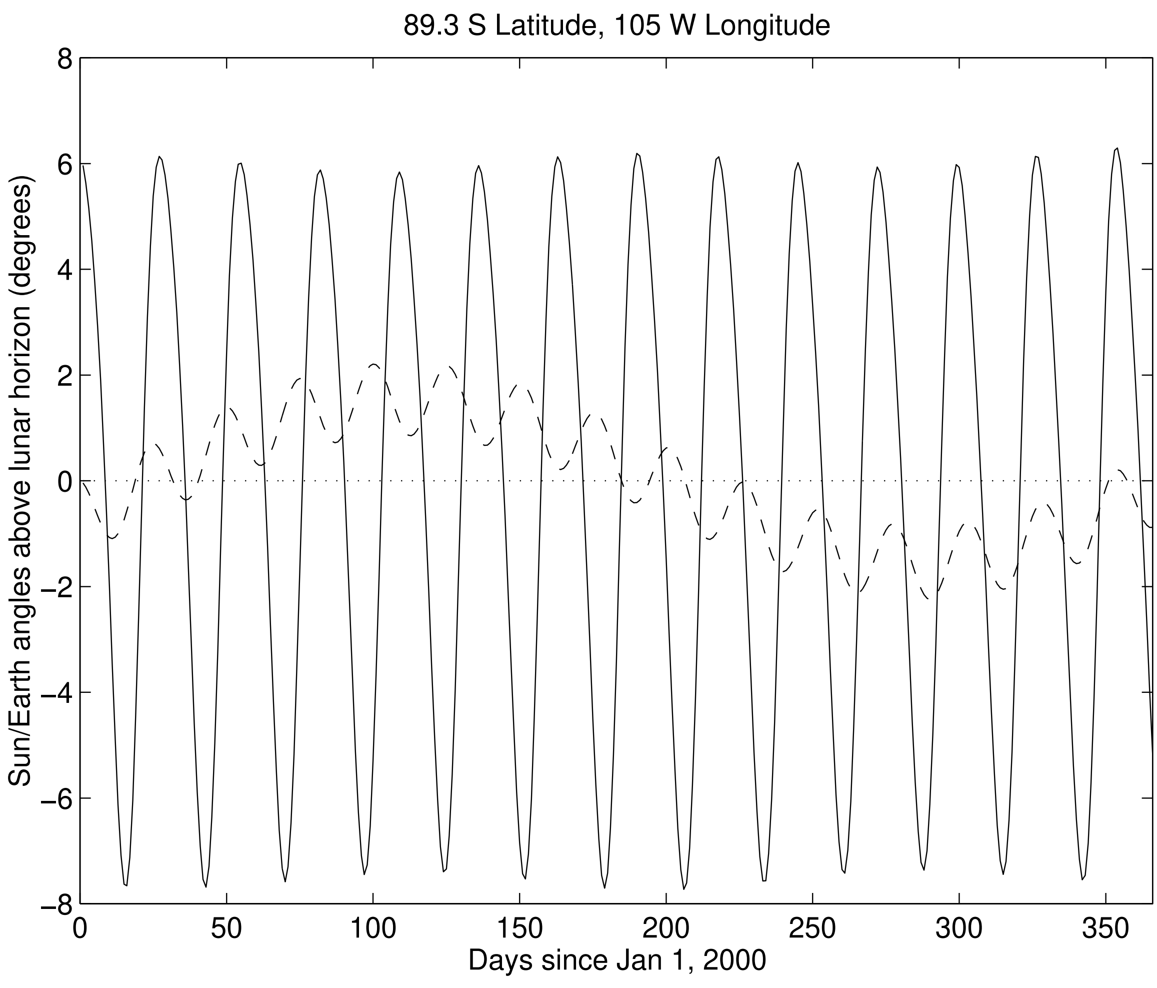


FIGURE A.4 *Year 2000 graph for 89.3 S, 105 W*

Figure A.5 shows this location for an 100 day period surrounding the landing date given in Chapter 1. April 13th is indicated by the vertical line, pinpointing the angles of the Sun and Earth above the horizon at that date.

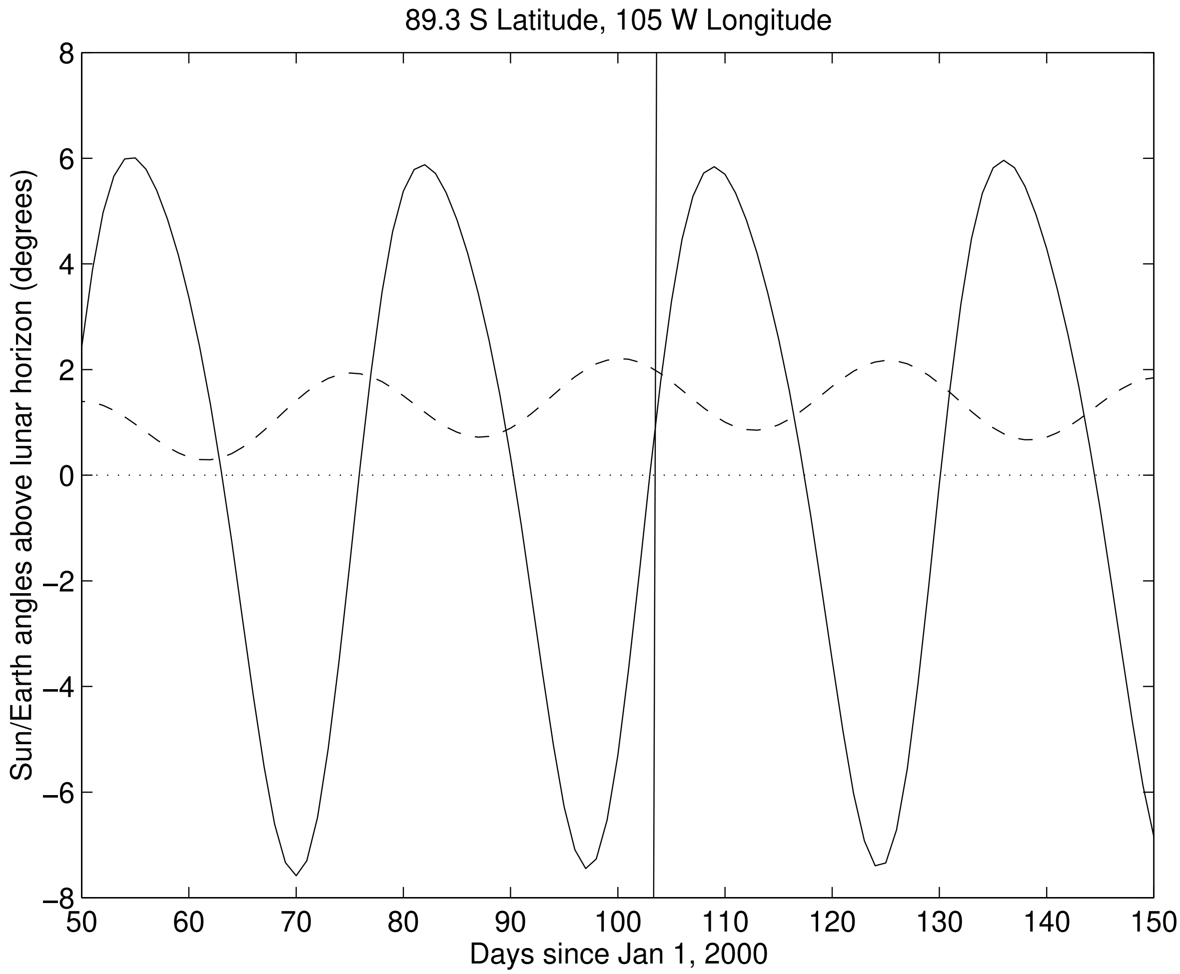


FIGURE A.5 *100 days surrounding landing date*

A.2 Craters

The following tables summarize statistical data concerning small to medium sized lunar craters. The information was taken from the Lunar Sourcebook, p.66 [9]. Figure A.6 and Figure A.7 explain what the various parameters refer to, showing a simplistic drawing of each type of crater. The mathematical details for determining how much of a particular crater can be a cold trap are given below each of the tables. In Section A.2.3, the specific craters near the south pole are described.

A.2.1 Simple Craters

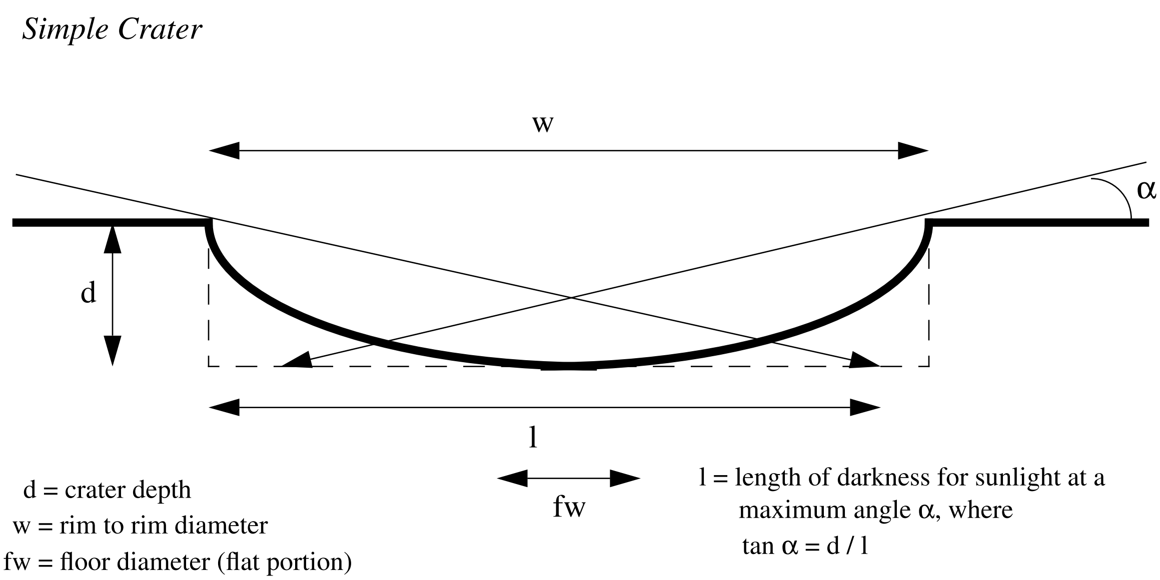


FIGURE A.6 *Simple crater schematic*

Width (rim to rim)	Depth	d/w ratio	Floor Diameter
w	**d**		**fw**
5 km	0.9959 km	0.1992 km	0.5309 km
10 km	2.006 km	0.2006 km	1.805 km
15 km	3.021 km	0.2014 km	3.691 km

TABLE A.2 *Simple craters statistics*

The mathematical details for the existence of cold traps in these small craters are simple. If l, the shadow length for sunlight at an angle α, is greater than the width of the crater, w, then the entire crater is a cold trap. If l is smaller than w, but is still greater than half of the width, a cold trap of size 2l - w exists. The cold trap will be comprised of the central portion of the crater, with the edges near the rims exposed to sunlight from time to time. This assumes a square crater, however. The curvature of the crater floor will cause the actual cold trap size to be slightly smaller, if the sunlight intersects the crater walls.

A.2.2 Complex Craters

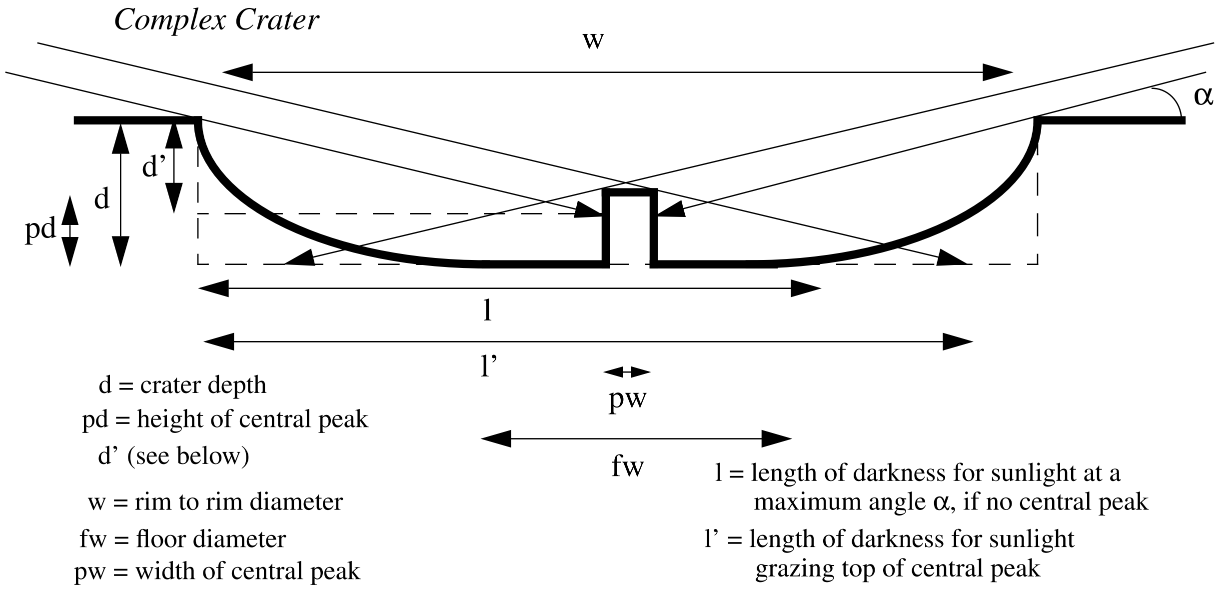


FIGURE A.7 *Complex crater schematic*

Width (rim to rim)	Depth	d/w ratio	Floor Diameter	Central Peak Diameter	Central Peak Height
w	**d**		**fw**	**pw**	**pd**
20 km	2.572 km	0.1286	7.885 km	0.3717 km	0.2147 km
25 km	2.751 km	0.1100	10.42 km	0.4698 km	0.3332 km
30 km	2.906 km	0.09687	13.08 km	0.5690 km	0.4771 km
40 km	3.169 km	0.07922	18.74 km	0.7696 km	0.8406 km
50 km	3.389 km	0.06778	24.77 km	0.9728 km	1.304 km
60 km	3.580 km	0.05967	31.10 km	1.178 km	1.868 km
70 km	3.750 km	0.05357	37.70 km	1.385 km	2.530 km
80 km	3.904 km	0.04880	44.55 km	1.594 km	3.291 km
90 km	4.045 km	0.04494	51.60 km	1.803 km	4.150 km
100 km	4.175 km	0.04175	58.86 km	2.014 km	5.106 km

TABLE A.3 *Complex craters statistics*

There are several cases for which a cold trap might exist in a complex crater. In all the cases described below, the central peak is assumed to be a rectangular projection, of width pw and height pd. This is a considerable simplification, but since central peaks vary in shape from crater to crater, this assumption generalizes the problem enough to apply to all craters.

First, consider the case that the central peak height is less than the crater depth, as shown in Figure A.7.

When the light reaches the point a distance w/2 - pw/2 from the left-most rim, it will be at a depth of d', given by

$$\tan\alpha = \frac{d'}{\left(\frac{w}{2} - \frac{pw}{2}\right)}$$

If (d - pd - pw tan α) < d' < d, this means the light ray has intersected the central peak. In this case, we must consider a second ray of light that just grazes the central peak, and strikes the surface at a distance l' from the left-most crater rim. For this ray, tan α = pd / (l' - w/2 - pw/2). If l' > w, then the entire crater will be a cold trap. If l' < w, some of the crater will be sunlit, but there will still be a cold trap. The size of the cold trap, assuming a "square" crater (the dashed lines in the above diagram), will be 2 x (l' - w/2 - pw/2), centered around the central peak. The central peak itself will be lit, at least partially, in this case. A curved crater will have a smaller cold trap, as with simple craters.

If d' < (d - pd - pw tan α), the light hits the crater after the central peak. This case is just the same as a simple crater.

If d' > d, the light hits the crater before the central peak. The point at which the light just grazing the left-most crater rim hits the crater floor is f, where tan α = d/f. There is also a ray of light that just grazes the central peak, hitting the crater on the right side of the central peak. The point where this light ray hits the crater floor is a distance f' from the base of the central peak, where tan α = pd/f'. If (f + f') > (w/2 - pw/2), a cold trap exists on each side of the central peak. Again assuming a square crater, these cold traps are each of size f + f' - (w/2 - pw/2). The area around the central peak, and the edges of the crater are sunlit at least some of the time.

The second set of cases is where the central peak height is greater than the crater's depth.

Again, consider the distance d', defined as in the equation for tan α above. If d' < d, then the light hits the central peak. Considering the light ray that grazes the edge of the central peak, if d' > pd - d, then some light will hit the right-side crater wall. The entire crater floor will still be a cold trap, but with a curved crater, some of the crater edge will not be permanently dark.

If d' > d, light hits the crater floor before the central peak. Defining f and f' as in the previous case, there will be a cold trap if (f + f') > (w/2 - pw/2). In this case, if f' > (w/2 - pw/2), the cold trap will be of size f, with the only lit spots being the area surrounding the central peak. If f' < (w/2 - pw/2), the cold trap will be of size f + f' - (w/2 - pw/2), where the crater edge is lit in addition to the central area.

A.2.3 South Pole Craters

The area around the south pole under consideration for Icebreaker's mission is not very large, so only smaller craters are present. Four such craters are labeled in Figure A.8, and information about each, along with several other smaller craters, is given in Table A.4. The diameters of these craters were measured from the Arecibo radar image, and the statistical depths for craters of those sizes were determined. The potential cold trap size for each crater was then calculated as described in the sections above.

In addition, the apparent depths of these craters were measured based on the lengths of the shadows they cast in the Arecibo image, since the angle of the incident radar was known. These depths are generally much shallower than the statistical depths, due mostly likely to erosion. Again, the potential cold trap size was calculated for these new depths.

Crater	Statistical measures based on Lunar Sourcebook information		Measures based on shadow lengths of Arecibo image	
location, diameter	depth	cold trap size	depth	cold trap size
A 89.9 S, 45 E 20 km width	2.572 km	entire crater	1.92 km	entire crater
B 89.3 S, 50 W 15 km width	3.021 km	entire crater	0.659 km	14.6 km (central portion)
C 88.3 S, 80 W 30 km width	2.906 km	entire crater	2.63 km	entire crater
D 87.9 S, 40 E 50 km width	3.389 km	35.3 km (central portion)	2.63 km	32.8 km (central portion)
several small craters 88.7 S -89.2 S 5 km width	0.996 km	entire crater	(varies)	entire crater

TABLE A.4 *Lunar south pole craters*

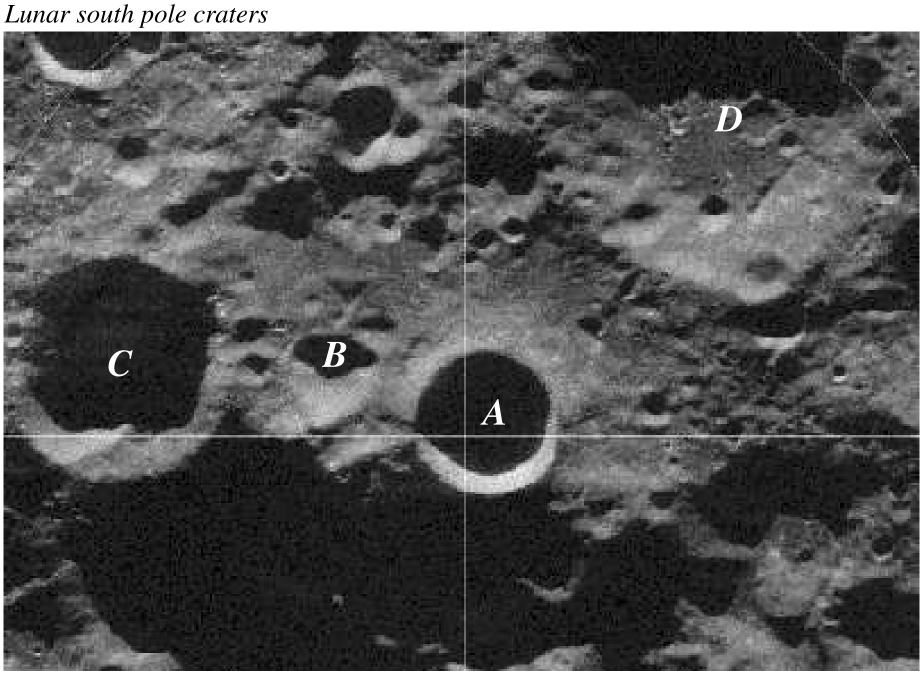
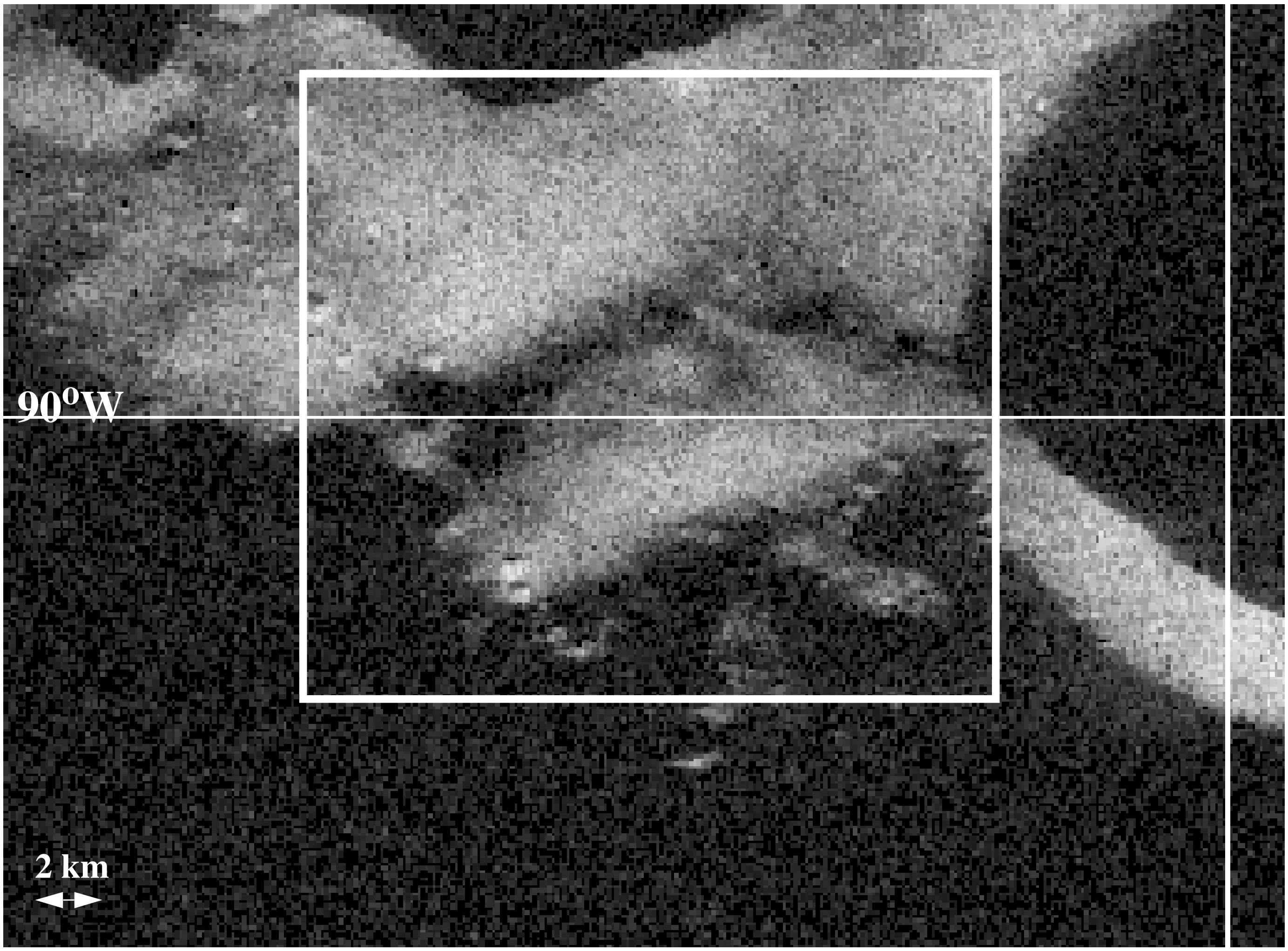


FIGURE A.8 *Lunar south pole craters*

Arecibo Radar Image
close-up view



Arecibo at 6.1° elevation above south pole
150 m resolution

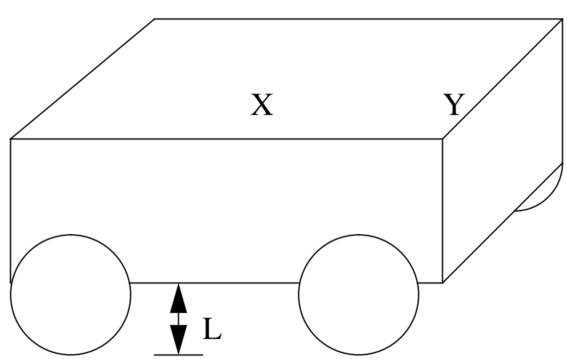
FIGURE A.9 *Close up view of Icebreaker landing area*

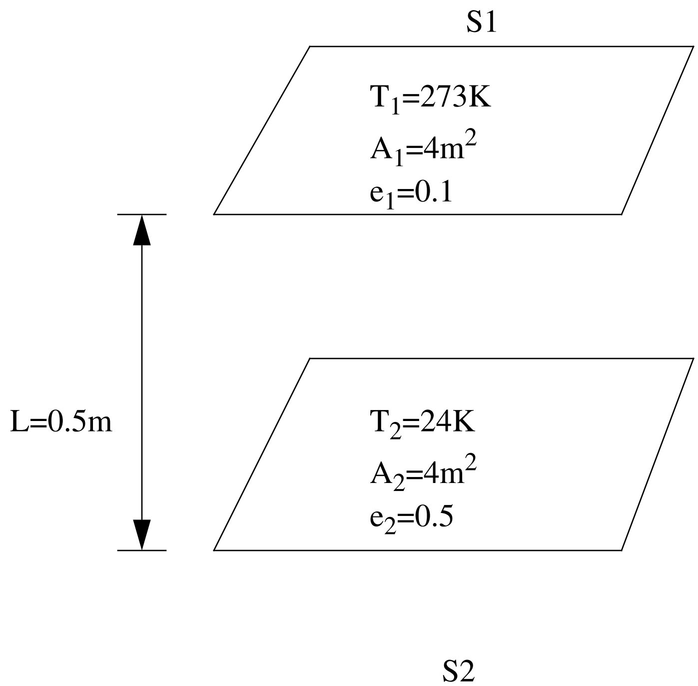
Appendix B

Sublimation of Lunar Ice due to the Presence of Icebreaker

B.1 Model Explanation:

Icebreaker will transmit energy to the lunar surface through radiation from its lower surface, belly. Thus there arises the concern if Icebreaker would sink in a pond of ice which sampling it. To determine how fast Icebreaker will sublime the ice beneath it the following analysis was performed. First Icebreaker was modeled as a rectangle X by Y at a distance L above the lunar surface.





The above diagram is arrived after reducing the first model into the two surfaces and assigning values for parameters. In this diagram S_1 is the rover and S_2 is the lunar surface. Note these are conservative estimates for e, emissivity, and T, temperature. Values for A, exposed projected area, and L, clearance, are based on design of Icebreaker.

To compute the energy transfer from Icebreaker to the lunar surface one must first compute the view factor F_{12}, from geometry $F_{12}=F_{21}$ and $F_{13}=F_{31}=1-F_{12}=F_{23}=F_{32}$.
Using appropriate equations we get that:

$F_{12}=0.6$

Now perform an energy balance to find the rate of energy transfer from S1 to S2

$$q2 = \frac{Eb2 - J2}{(1-\varepsilon 2)/(\varepsilon 2 A2)}$$

$$\frac{Eb2 - J2}{(1-\varepsilon 2)/(\varepsilon 2 A2)} = \frac{J2 - J1}{1/(A2F12)} + \frac{J2 - J3}{1/(A2F23)}$$

We have $J_3=Eb_3=\sigma T_3^4=1.88 \times 10^{-2} W/m^2$.

$$\frac{0.0188 - J2}{1} = \frac{J2 - J1}{1/0.6} + \frac{J2 - 0.0188}{1/0.4}$$

$$0.6J1 - 2J2 = -0.0263$$

Need another relation to get J_2 so use the balance of rover surface to get another equation

$$\frac{Eb1 - J1}{(1-\varepsilon 1)/(\varepsilon 1 A1)} = \frac{J1 - J2}{1/(A1F12)} + \frac{J1 - J3}{1/(A1F13)}$$

We have $Eb_1=\sigma T_1^4=315 W/m^2$.

$$\frac{315 - J1}{0.9/0.1} = \frac{J1 - J2}{1/0.6} + \frac{J1 - 0.0188}{1/0.4}$$

$$0.6J2 - 1.11J1 = -35$$

Solving for J_2 we can get:

J_2=11.3 W/m^2.
so q2=-45.1 W.

Thus spread out over 4m^2 there is 45.1 W transferred from Icebreaker to the lunar surface.

Assumptions:
1) All incident radiation is absorbed by the ice
2) Steady state conditions
3) Surfaces are diffuse and gray
4) Surroundings can be approximated by lunar cold trap model
5) Emissivities and temperatures chosen

HOW THIS HEAT FLUX EFFECTS LUNAR ICE

With the estimation of the energy transmitted to the ice, 45.1 W, the speed at which the ice sublimes must be determined. To estimate this we need the enthalpies of sublimation which are listed below for various potential components of lunar ice.

Enthalpies of Sublimation
CO_2=18.4 kJ/kg
O_2=13.9 kJ/kg
N_2=25.7 kJ/kg
H_2O=333.5 kJ/kg
CH_4=58.7 kJ/kg

For example CO_2 is chosen which has a density equal to 1.5 g/cm^3.

For the rover to sink 1 cm it needs to deliver:

18.4 J/g x 1.5 g/cm^3 x 1 cm x 40000 cm^2=1104kJ

At 45.1 W the rover must sit still for 6.8 hours.

Assumptions
1) Ice is 100% pure, free of rock debris.
2)All energy absorbed is use for sublimation.

Appendix C

Launch/Landing

This appendix contains dimensions and details of the launch and landing segments of the mission.

C.1 Launch Fairing

Icebreaker will launch from the Kennedy Space Center on a Delta II 7925H rocket. This rocket can have one of three different payload fairing sizes. For the Icebreaker, the 2.9m fairing was selected as the best compromise between mass and volume. A diagram of the 2.9m fairing is provided in Figure C.1. The dimensions on the figure are in millimeters above the lines and inches below the line.

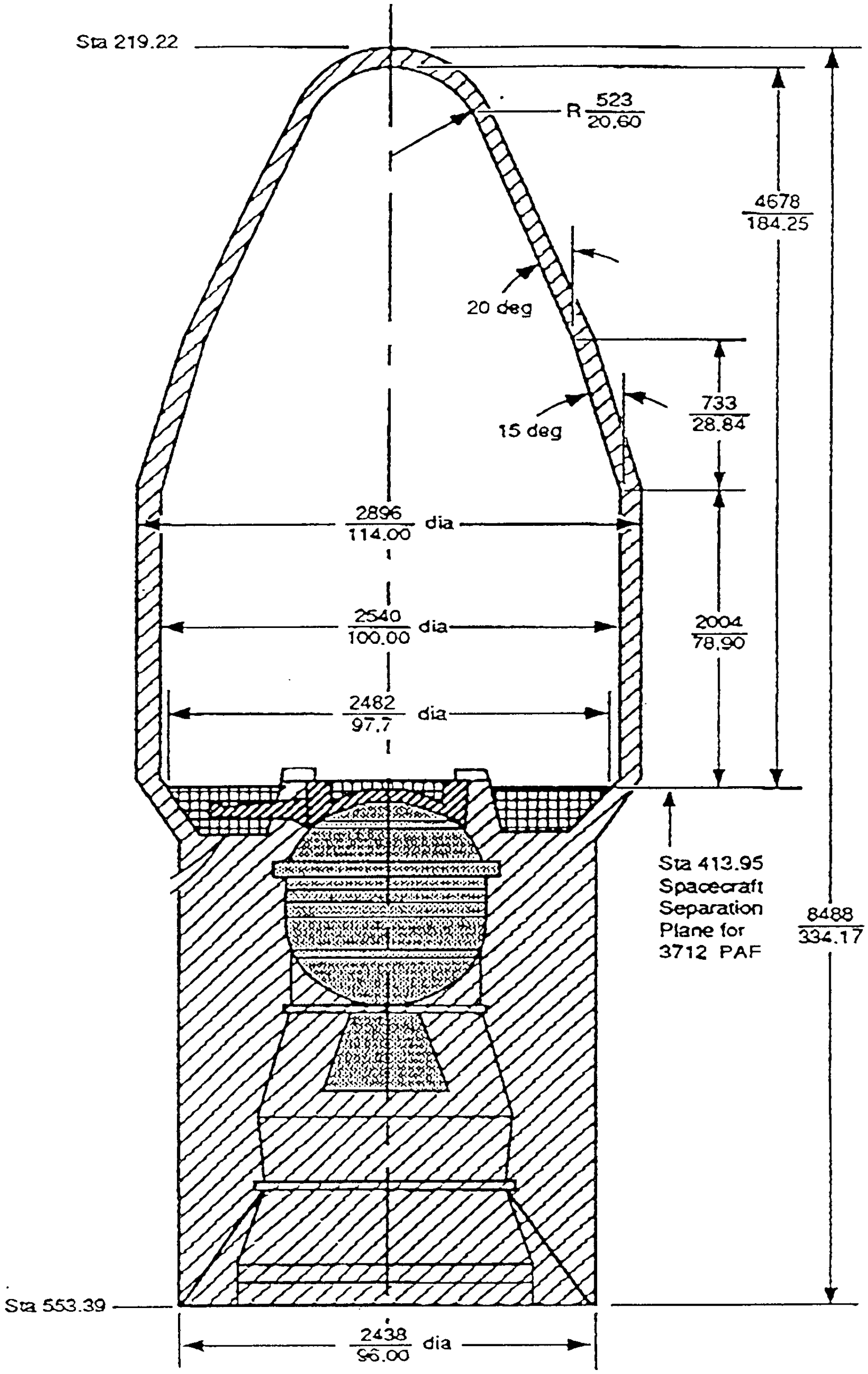


FIGURE C.1 *2.9m Payload Fairing of the Delta II 7925H Rocket*

C.2 Landing Propulsion

The landing propulsion system consists of three parts. The first part, initial deceleration, consists of a Thiokol STAR 37XFP solid rocket motor. This is an off the shelf component and is shown in Figure C.2.

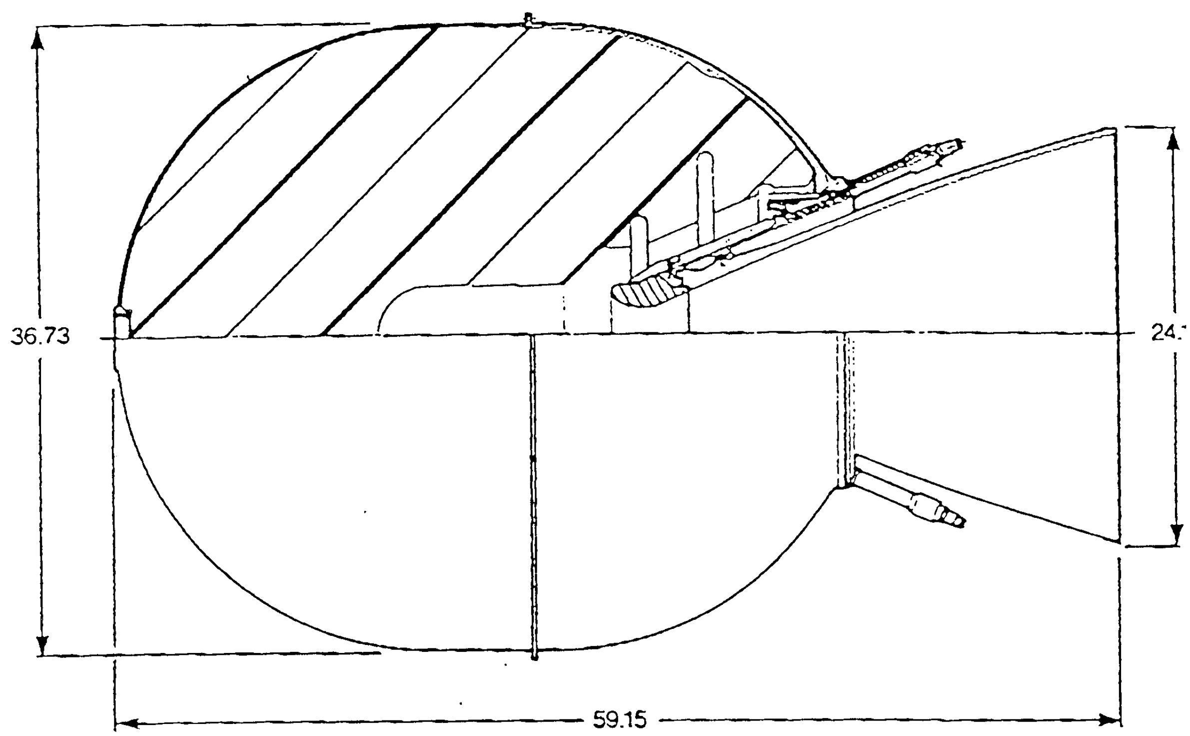


FIGURE C.2 *Thiokol STAR 37XFP Solid Rocket Motor (dimensions are in inches)*

The second propulsion system, final deceleration system, consists of four groups of three hydrazine thrusters. Each thruster in the group is individually controllable. Each thruster group is attached directly to a hydrazine fuel tank. The fuel system is of the blowdown type, where a high pressure gas is used to fill the space left in the hydrazine tank as fuel is consumed in the thrusters. A schematic diagram of the fuel delivery system for one thruster group is provided in Figure C.3. This set-up is duplicated for the other three thruster group, all of which form isolated systems.

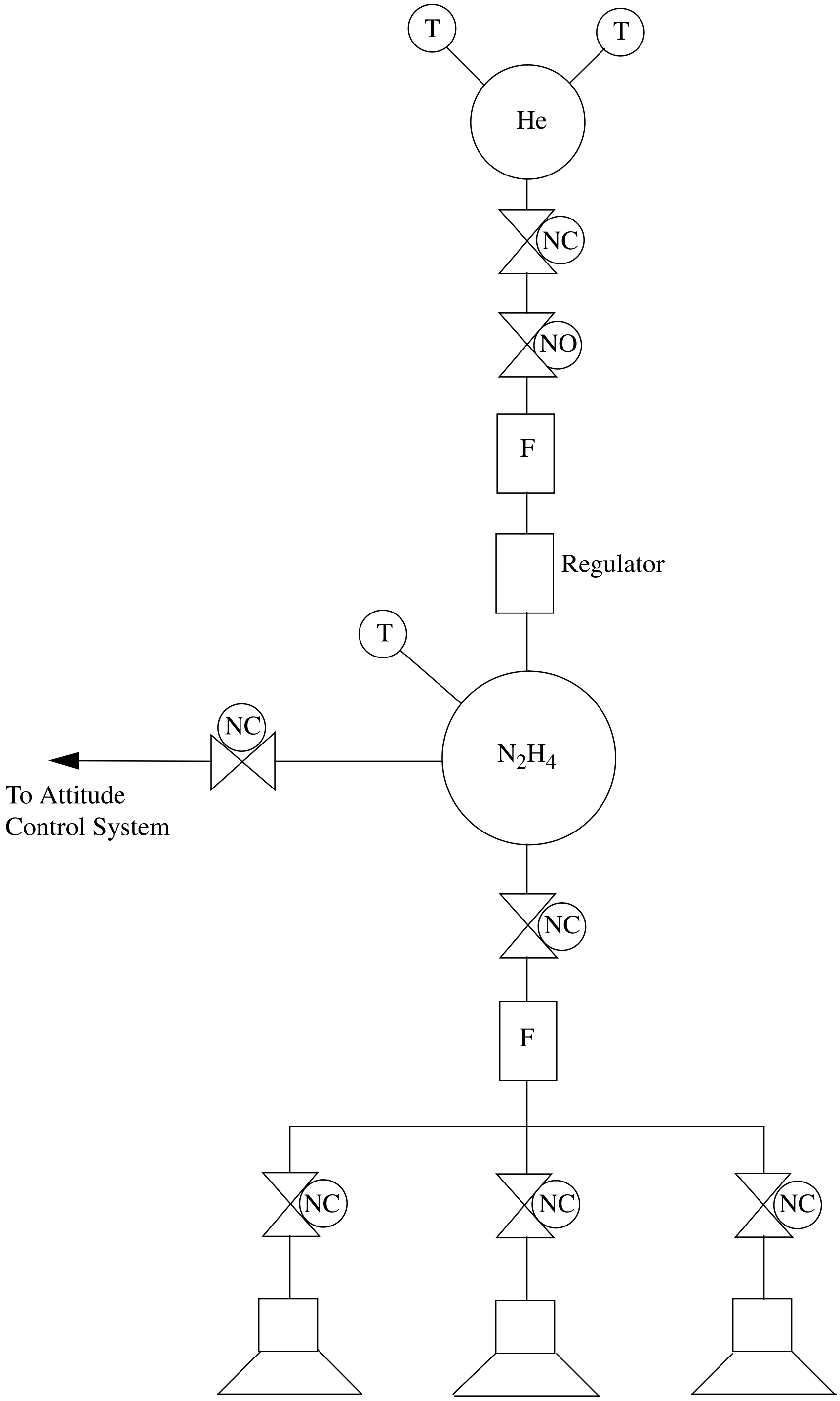


FIGURE C.3 *Hydrazine Fuel Delivery System*

Appendix D

Locomotion

D.1 Mobility

The mobility of a system is based on several vehicle attributes including: wheelbase, number of wheels, dimensions, torque, energy, steering, and suspension. Analysis of locomotion performance can be achieved by studying the drawbar pull and the energy requirements of the locomotion system. Study of these parameters will provide the relationships between the trafficability of the soil, the wheel contact profile, soil thrust, and motion resistance [23].

The drawbar pull provides a measure of the thrust required to create motion in spite of the components of resistance. The drawbar pull can also be used to provide an estimate for the gradeability of the vehicle. The locomotion energy is the energy required to overcome the components of resistance. The components of resistance studied in this analysis include:

- Rolling resistance R_r
- Bulldozing resistance R_b
- Compaction resistance R_c
- Gravitational R_g

The components of resistance are dependent on vehicle parameters as well as soil values. The lunar soil values can be found inTable D.1 [1] [23].

TABLE D.1 *Lunar Soil Parameters*

Parameter	Symbol	Nominal Lunar Value
Cohesive modulus of soil deformation	k_c	140 $kg/m^{(n+1)}$
Frictional modulus of soil deformation	k_ϕ	83033 $kg/m^{(n+2)}$
Soil deformation exponent	n	1.0
Angle of internal friction	ϕ	.698 radians
Coefficient of soil slip	K	.018 m
Soil specific weight	α	2128 kg/m^3
Soil cohesion	c	176 kg/m^2

Given the lunar soil parameters a parametric analysis of mobility was performed as outlined in Systematic Configuration of Robotic Locomotion [1].

Drawbar pull is defined as thrust minus the sum of the resistance components. Given that thrust is defined by:

$$H = \{cA + W_w \cos(\theta) \tan(\phi)\}$$

A = contact area of the tire

θ = slope angle

W_w= normal wheel loading

The dominant component of thrust is the wheel loading W_w . For lunar soils an increase in the weight of the vehicle will increase thrust more than a similar increase in wheel dimensions. An increase in vehicle weight will cause an increase in the sinkage of the vehicle and an increase in the energy required to move the vehicle.

The resistance components are defined by:

$$\sum R = R_r + R_b + R_c + R_g$$

A qualitative assessment was made of the resistance component equations and the sinkage equation. An increase in wheel diameter will directly reduce the resistance due to soil compaction and the amount of sinkage. As the amount of sinkage decreases the resistance due to bulldozing decreases. Therefore, the wheel diameter should be made as large as possible; limited by considerations of mass, volume, and dynamic stability [23]. An increase in the width of the tire will decrease the amount of sinkage of the vehicle. However, the beneficial impact will not be as great as an increase in wheel diameter. Wide wheels do have high flotation characteristics, causing low ground pressure and low sinkage. However, an increase in wheel width will also increase the bulldozing resistance and the mass of the wheel. Figure D.1 and Figure D.2 show the effects of wheel diameter and wheel width versus total drive power.

FIGURE D.1 *Effect of diameter*

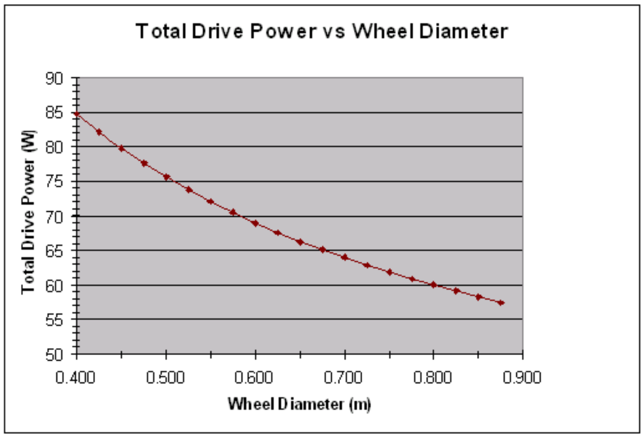


TABLE D.2 *Input Parameters*

Parameter	Value
Total Vehicle Mass	390 kg
Wheel Width	.4 m
Speed	.3 m/s
Driving slope angle	0

FIGURE D.2 *Effect of width*

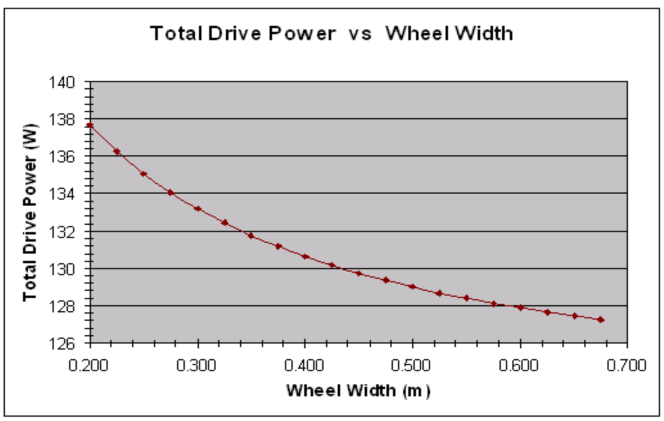


TABLE D.3 *Input Parameters*

Parameter	Value
Total Vehicle Mass	390 kg
Wheel Diameter	.6 m
Speed	.3 m/s
Driving slope angle	0

TABLE D.4 *Locomotion Configuration Worksheet*

Locomotion Configuration Worksheet				
Rover Specifications:				
	English	Units	SI	Units
Total Mass:	859.872	lb	390	kg
Number of Wheels:	4.000		4	
Wheel Diameter:	23.622	in	0.6	m
Wheel Width:	15.748	in	0.4	m
Vehicle length:	78.740	in	2	m
Vehicle max speed:	11.811	in/s	0.3	m/s
Turning reaction time:	1.000	sec	1	sec
Coeff of motion resistance	0.250		0.25	
Coeff of rolling friction:	0.050		0.05	
Contact area length (l'):			0.31299	m
Driving Slope Angle:	0.000	degrees	0	radians

Lunar Soil Model:				
	English		SI	
Surface gravitation (g)	64.304	in/s^2	1.633333	m/s^2
Regolith Sinkage Exp (n)	1.000		1.000	
Cohesive modulus (kc)	0.200	lb/in^(n+1)	140.6026	kg/m^(n+1)
Frictional modulus (kphi)	3.000	lb/in^(n+2)	83033.03	kg/m^(n+2)
Cohesion of regolith (c)	0.251	lbf/in^2	176.4556	kg/m^2
Coeff of slip (K)	0.700	in	0.01778	m
Angle of internal fric (phi)	40.000	degrees	0.698131	radians
Soil specific weight (alpha):	0.080	lb/in^3	2128.495	kg/m^3
Soil Deformation (j):				m

Derived Parameters:			
	SI	Units	Equation (Ref CMU-RI-TR-96-30)
Wheel Load:	159.25	N/wheel	(mass*g_moon/num_wheels)
Sinkage:	0.04405207	m	A2
Soil Thrust (H):	150.97708	N	A5 w/o using j
Compaction (Rc):	32.3629618	N	A9
Bulldozing (Rb):	17.1349343	N	A10
Rolling (Rr):	7.9625	N	A11
Gravitational (Rg):	0	N	A12
Steering (Rs):	0	N	(estimate)
Obstacle Climb (Ro):	0	N	(estimate)
Acceleration (Ra):	0	N	(estimate)
Total Resistance (Rall):	57.4603961	N	A13
Drawbar Pull (DP):	93.5166838	N	A14
Torque (T):	17.2381188	Nm	A15
Drive Power/wheel (Pd):	17.2381188	W/wheel	
Total Drive Power (P):	68.9524754	W	
Max slope:	30.4228416	degrees	

Appendix E

Terrain Mapping Radar

A substantial investigation was performed to design a radar unit capable of meeting the requirements for this mission. This appendix reviews relevant radar design considerations and provides more detail on the scanning radar unit presented in Chapter 7.

E.1 Radar Fundamentals

To achieve a desired beam width, a radar's aperture is governed by the following equation:

$$D = \frac{\lambda}{\Theta}$$

where D is the aperture size, λ is the wavelength of the emitted beam, and θ is the desired beam width. Table E.1 lists common radar frequencies (devices exist to produce radar at these frequencies), and the required aperture (in millimeters) to achieve the given beam widths.

TABLE E.1 *Aperture size for given radar frequency and desired beam width.*

Freq\Beam Width	15°	10°	5°	2°	1°
94 GHz	12 mm	18 mm	37 mm	92 mm	184 mm
77 GHz	15	22	44	112	224
60 GHz	19	29	58	143	286
30 GHz	38	57	114	286	572

Freq\Beam Width	15°	10°	5°	2°	1°
10.5 GHz	109	164	328	819	1,638

In order to get a useful return from the ground, diffuse reflection is required. The Rayleigh criterion states that a surface reflects incident energy diffusely if:

$$h > \frac{\lambda}{8 \sin \alpha}$$

where h is the amplitude of the surface corrugation, λ is the wavelength, and α is the angle between the surface and the ray. The mounting location for Icebreaker's radar unit is 1 meter above the ground as shown in Figure E.1

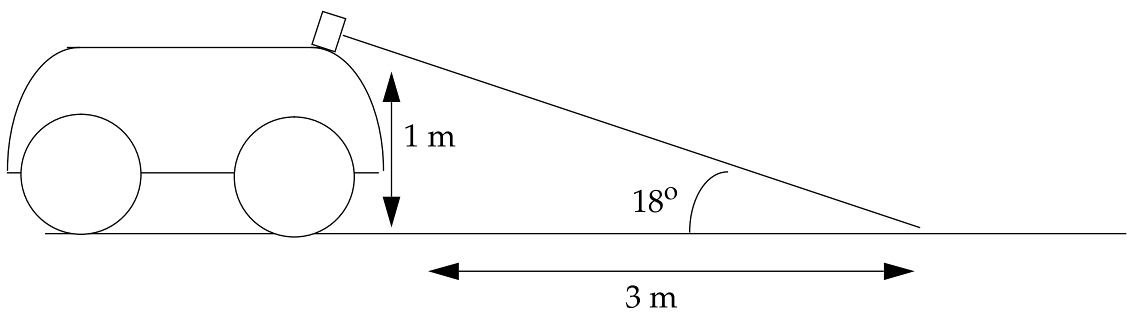


FIGURE E.1 *Radar mounting geometry.*

The amplitude of the surface corrugation for this geometry and the frequencies listed in Table E.1 is given in Table E.2. These values indicate that it is possible that the lunar surface may appear specular to radar with frequencies below 10.5 GHz, but it is almost certain that micrometeorite activity on otherwise smooth rocks or the granularity of the regolith has resulted in enough surface roughness for radar at frequencies above 10.5 GHz to reflect diffusely when the radar is mounted as in Figure E.1.

TABLE E.2 *Required surface roughness at the given frequencies for the geometry shown in Figure E.1.*

Frequency	Surface Corregation (h)
94 GHz	1.3 mm
77 GHz	1.5 mm
60 GHz	2.0 mm
30 GHz	4.0 mm
10.5 GHz	11 mm

Another consideration with radar is that although the beam will be shown as a thin cone this shape is not a true representation of the radar beam. For two reasons, one, the side lobes and edges of the "beam" are really the half-power points which occur on the main lobe with most aperture distributions and two, the beam doesn't disperse in the fan shape until it has traveled into the far field which occurs several aperture diameters from the radar unit. The far field point is given by the equation:

$$R_F = \frac{2D^2}{\lambda}$$

where D is the aperture size and λ is the wavelength. The method used to size the aperture in this design was to choose the far field to begin at the targeted range of 3 meters. This would give the expected footprint at 3 meters. Sensing can be done inside 3 meters (in the near field), although the beam will not necessarily conform to a given beam shape. The size of the beam within three meters from the unit will be between the aperture size and the beam's footprint at 3 meters.

E.2 Proposed Radar Unit for Icebreaker

A scanning millimeter wave radar unit is proposed for building terrain maps that can be used for safeguarding Icebreaker during traversal of the paths that are planned using operator waypoints. The frequency used will be 94 GHz, which is the highest of the frequencies considered. From the table shown above, the higher frequency gives a smaller aperture for a fixed beam size or a smaller beam for a fixed aperture size. Radar components at frequencies higher than 94 GHz are uncommon. Using a transmission frequency of 94 GHz and choosing the far field point at 3 meters requires an aperture size of 6.9 cm. A beam width of approximately 3 degrees can be achieved using this wavelength and aperture size [11].

Figure E.2 illustrates the radar front end for frequency modulated continuous wave (FMCW) radar, which is the chosen method to transmit radar energy in each beam.

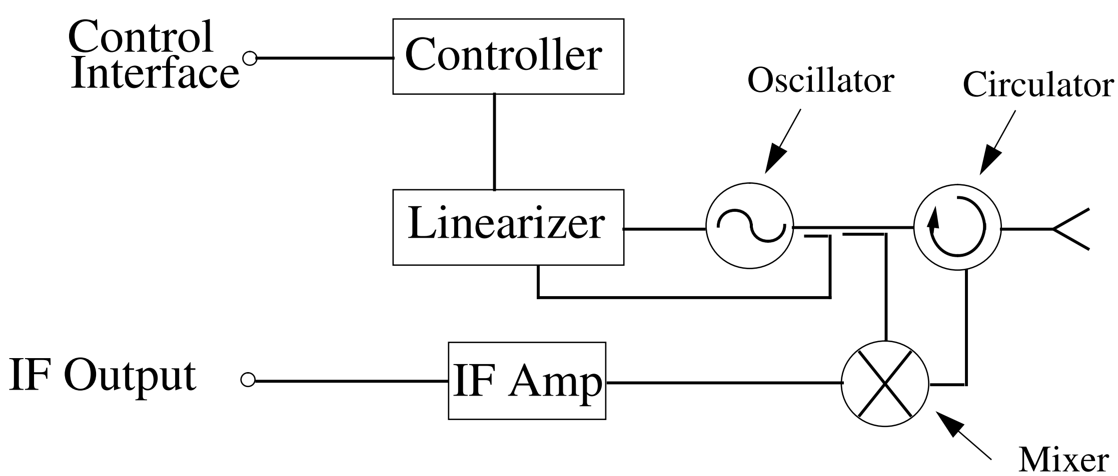


FIGURE E.2 *Block diagram of FMCW radar front end.*

The frequency is modulated around a continuous signal at 94 GHz. An oscillator ramps the frequency. A feedback loop ensures that the frequency ramp is kept linear. A circulator port is used to transmit radio frequency (RF) energy out a wave guide and receive returned RF energy through the same wave guide. A plot of frequency as a function of time for this method is shown in Figure E.3

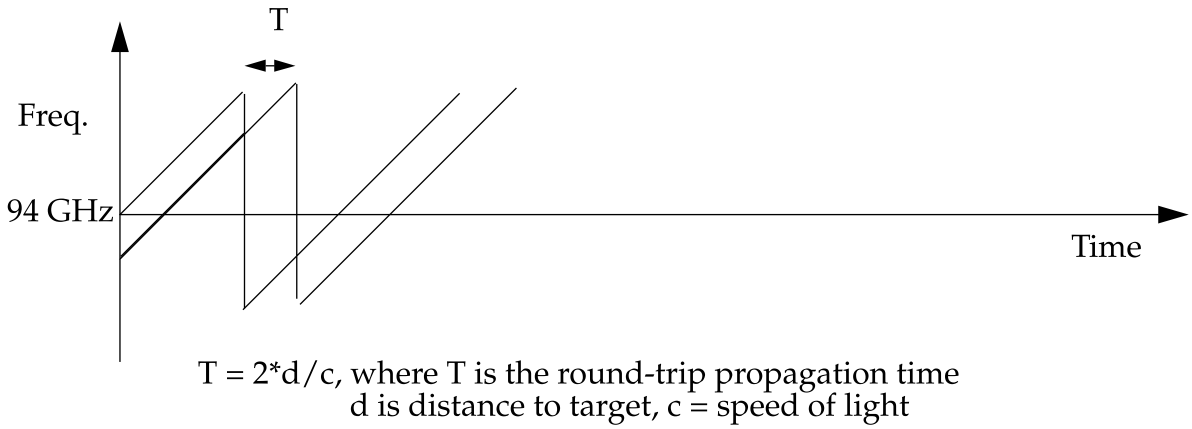


FIGURE E.3 *Frequency as a function of time for FMCW radar.*

The returned RF is sent to a mixer, which mixes the transmitted signal with the received signal to form a beat frequency. This beat frequency is amplified and sent to hardware in the computer enclosure which digitizes the signal and performs a fast fourier transform (FFT) to transform the data to the power domain. This signal is thresholded to extract the range to the return from the lunar surface.The downrange resolution is determined in part by the bandwidth of the modulation. In the configuration specified, the bandwidth is 1 GHz and the downrange resolution is approximately 0.15 meters. If needed, the downrange resolution could be improved by centroiding the return signal as shown in Figure E.4. The arrow is meant to illustrate that centroidal processing of power values above a given threshold might allow better resolution than that given by the FFT.

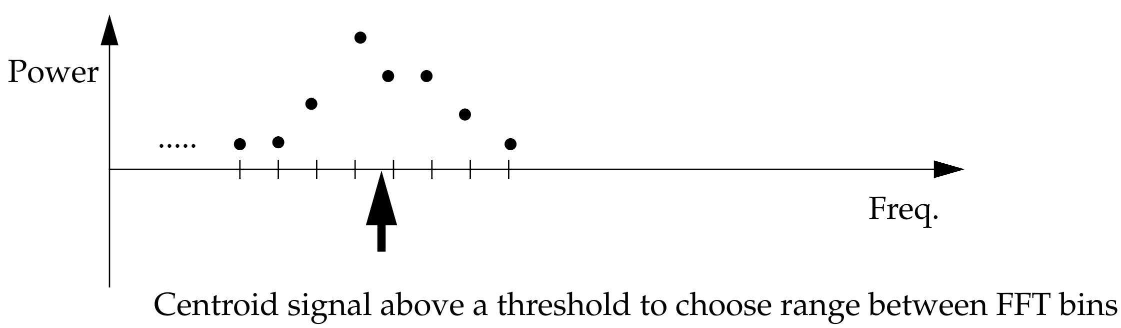


FIGURE E.4 *Plot of radar power as a function of frequency.*

In order to scan the beam, and keep the volume of the unit small, a twist-reflector concept was used [20]. The concept is shown in Figure E.5.

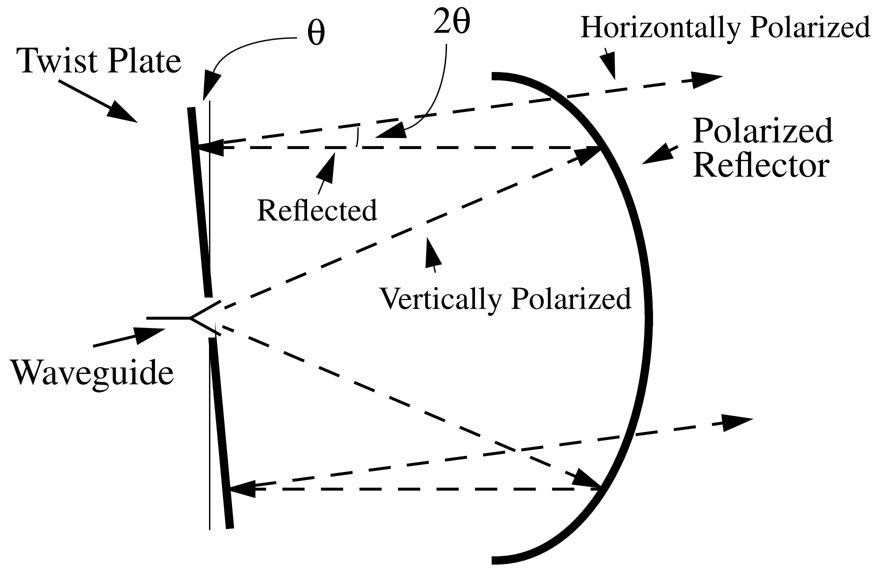


FIGURE E.5 *Geometry of the twist-reflector scanning antenna.*

Radar energy, polarized in a given direction, is emitted from a waveguide and is reflected by a polarized reflector. It then contacts a "twist" plate, which contains grooves that twist the polarization angle of the RF energy by 90 degrees. The RF energy then passes through the polarized reflector and into the region of interest. The shape of the polarized reflector shapes the beam. The radar beam is scanned by scanning the twist plate.

The configuration chosen for Icebreaker is shown in the artist's conception, Figure 3.1. Detailed drawings of this unit are shown in Figure E.6. and Figure E.7

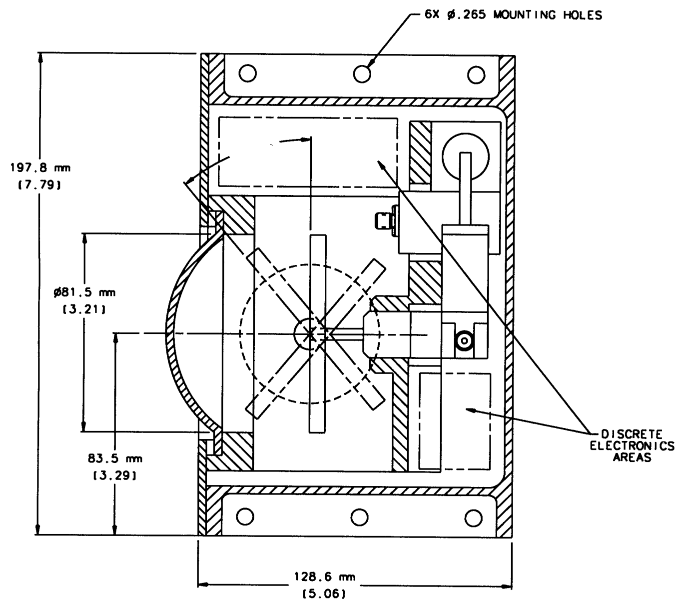


FIGURE E.6 *Top view of proposed scanning millimeter wave radar unit [11].*

The unit in the artist's conception has the same height as the drawings, but is half as wide. The unit could be narrowed to the dimensions shown in the artist conception; the mechanical drawings are the first cut at the volume required for the components needed to accomplish scanning. The height and width of the drawings shown in Figure E.6 and Figure E.7 is about 20 centimeters.

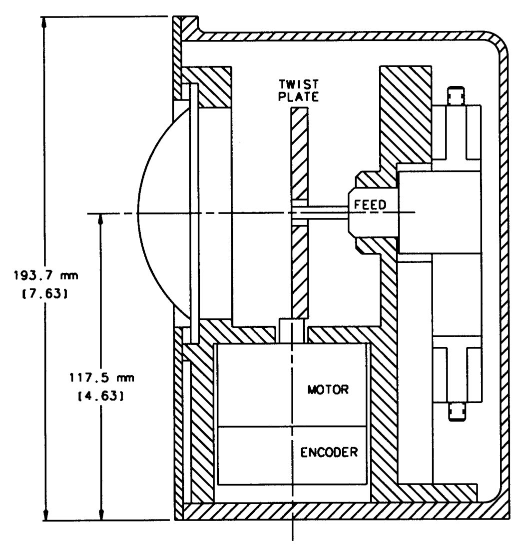


FIGURE E.7 *Side view of proposed scanning millimeter wave radar unit [11].*

The radar will be scanned so that the beams will cover the area in front of the vehicle which the radar may drive into. The footprint of beams on flat ground is shown in Figure E.8. In Figure E.8 the vehicle is moving forward and the sequence shown is first a left to right scan, then a right to left scan with a frame rate of 1.2 Hz. This scanning speed allows the radar to avoid gaps between footprints while traveling at 0.3 m/sec. on flat ground. Terrain maps which will be used by Icebreaker for safeguarding will be built from this range data.

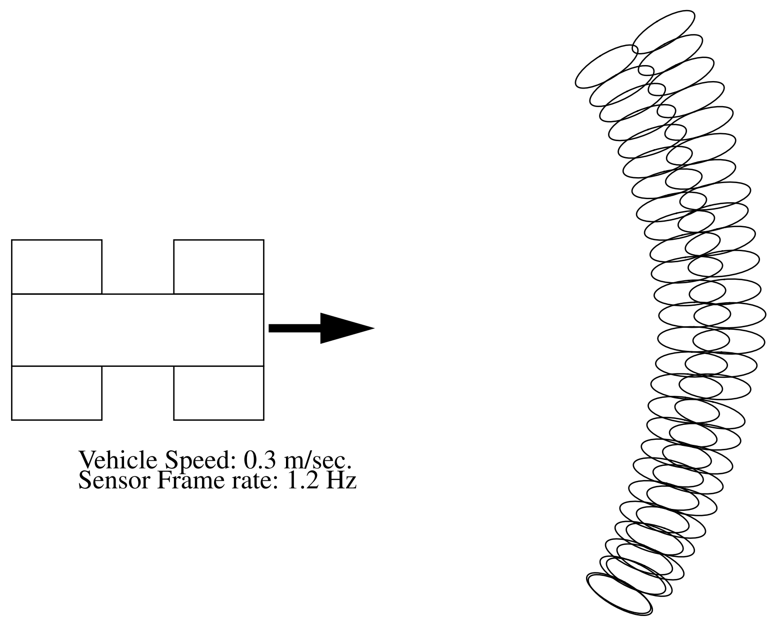


FIGURE E.8 *Sensor footprints on flat ground.*

E.3 Open Issues with Radar

Two open issues with the radar design being proposed for Icebreaker are the size of the side lobes and the amount of radar power that will be returned from the lunar regolith for the given mounting location. The side lobes are a function of the design of the polarizing reflector. If they are large, they could affect the shape of the beam footprint. A lens could be used if needed to focus the radar beam in the given area and reduce the side lobes. In Section E.1, radar was shown to reflect diffusely. The transmitter and receiver must have the dynamic range to be able to detect the radar power returned in the case of this diffuse reflection. Testing to determine the power from a simulated lunar surface would provide the needed information to set the appropriate transceiver dynamic range.

Appendix F

Communications Link Analysis

F.1 Link Requirements

The Space to Earth link is the critical path for telemetry information and science return for this mission. This appendix gives a detailed discussion and itemized link analysis of the communication link from Icebreaker on the South Pole of the Moon to the DSN antennas on Earth.

The High Gain Antenna (HGA) array provides the maximum bandwidth link from Icebreaker to mission control at 10 kbps. This requirement comes from considering the image data collected by the forward looking pair of CCD cameras which the operator uses for teleoperation. At 640x480x8-bit, these two cameras provide about 5 Mb of information. Considering 40:1 compression of data, this will require 123 kb of information. Over a 10 kilobit per second (kbps) link, the compressed image data can be transferred to the Earth in roughly 12.3 seconds, excluding the delay time of 1.28 seconds and the delays in land lines which can exceed 10 seconds.

A Low Gain Antenna (LGA) is designed to provide a backup antenna in the event of an HGA failure. An HGA failure could be caused by electrical failure or by pointing beyond the active antenna field of view. The LGA will require a lower data rate in order to still provide a viable downlink path.

F.2 Communications Link Parameters

The communication system designed for Icebreaker provides a 10 kbps link for telemetry downlink using only 5 W RF output power and an array of six 11.0 dB antennas. These high gain antennae have a beam pattern which is 40 degrees in elevation and 70 in azimuth, which when mounted in a hexagonal pattern provide a full azimuth of coverage and enough elevation coverage to see the Earth in any direction at any elevation even when on terrain which has a 12 degree slope.

If any of the HGA malfunction, or if Icebreaker is on too great a slope, a backup link is provided by the low gain antenna (LGA). The LGA has a beam pattern which is omnidirectional in azimuth and has a 50 degrees beamwidth in elevation. The LGA has an antenna gain of 2.0 dB, which reduces the bandwidth over the LGA link to only about 1 kbps. This is acceptable since the LGA should only be required in extreme cases and at times when operators will likely require longer decision-making times in order to solve whatever problems may be occurring to cause the HGA link to fail.

The following pages contain a detailed analysis for the HGA and LGA full-duplex links. The most important figures to examine are the downlink data performance margin (line 175 on each link analysis). The uplink performance margin in each case far exceeds requirements.

CMU Lunar Ice Discovery | Icebreaker | May 1, 1997.
CMU Discovery 1

CCSDS LINK DESIGN CONTROL TABLE
GENERAL INFORMATION
(Link and Weather Not Combined)

Page 1

1	Owner CCSDS Agency	NASA
2	Name of Mission	CMU Lunar Ice Discovery
3	Name of Spacecraft	Icebreaker
4	Mission Category a. A = Alt.<2,000,000 km b. B = Alt.>2,000,000 km	A (Near Earth Mission)
5	Link Budget Number	High Gain Antenna Array #1
6	Revision No. / Conditions	S/C-E St. Rng= km, B. Rt. = 10 Kbps S/C: Ant Gn = 11.0dBi; Xmt Pow = 5.0 W; Res. Car. DSN : 34BWG1 m; 17 KW; Cmd = 2000 bps; Sim. Rng. TLM Coding: Ch. 1: r = 0.166666666666667 , k = 15 and 223/255 R/S
7	Date	May 1, 1997.
8	File Name	CMU Discovery 1
9	Project Cognizant Person Name: Title: Address: Telephone: Fax Number: Telemail/Telex No:	Matthew Deans The Robotics Institute Carnegie Mellon University Pittsburgh, PA 15213 (412) 268-1860 (412) 268-5571 deano@ri.cmu.edu
10	Network Cognizant Person Name: Address: Telephone: FAX No: Telemail / Telex No:	Gary K. Noreen Jet Propulsion Laboratory 4800 Oak Grove Drive, M/S 303-402 Pasadena, CA. 91109-8099 (818) 354-2699 (818) 393-1692 gary.k.noreen@jpl.nasa.gov

CMU Lunar Ice Discovery Icebreaker May 1, 1997.
CMU Discovery 1

CCSDS LINK DESIGN CONTROL TABLE
COMMUNICATIONS SYSTEM OPERATING CONDITIONS
(Link and Weather Not Combined)

Page 2

EARTH - SPACE - LINK			SPACE - EARTH - LINK		
E/S TRANSMITTING RF CHANNEL:			S/C TRANSMITTING RF CHANNEL:		
1	RF Carrier Modulation		11	RF Carrier Modulation	
	a. Ch 1 Type	PM		a. Ch 1 Type	PM
	b. Ch 1 Format	PSK		b. Ch 1 Format	NRZ-L
	c. Ch 2 Type			c. Ch 2 Type	
	d. Ch 2 Format			d. Ch 2 Format	
E/S TRANSMITTING DATA CHANNEL:			S/C TRANSMITTING DATA CHANNEL:		
2	Baseband Data		12	Baseband Data	
	a. Ch 1 Bit Rate, b/s	2000.00		a. Ch 1 Bit Rate, kb/s	10.00
	b. Ch 1 Bit Err Rate	1.0E-06		b. Ch 1 Bit Err Rate	1.0E-05
	c. Ch 2 Bit Rate, b/s			c. Ch 2 Bit Rate, kb/s	
	d. Ch 2 Bit Err Rate			d. Ch 2 Bit Err Rate	
3	Data Coding		13	Data Coding	
	a. Ch 1 Type			a. Ch 1 Rate	0.17
	b. Ch 1 No. Info Bit			b. Ch 1 Constr Lngth	15
	c. Ch 1 Block Length			c. Ch 1 Concat Code	R/S
				d. Ch 1 Data/Tot Bit	223/255
	d. Ch 2 Type			e. Ch 2 Rate	
	e. Ch 2 No. Info Bit			f. Ch 2 Constr Lngth	
	f. Ch 2 Block Length			g. Ch 2 Concat Code	
				h. Ch 2 Data/Tot Bit	
4	Subcarrier		14	Subcarrier	
	a. Ch 1 Waveform	Sine		a. Ch 1 Waveform	
	b. Ch 1 Frequency	16KHz		b. Ch 1 Frequency	
	c. Ch 1 Mod Type	PSK		c. Ch 1 Mod Type	
	d. Ch 2 Waveform			d. Ch 2 Waveform	
	e. Ch 2 Frequency			e. Ch 2 Frequency	
	f. Ch 2 Mod Type			f. Ch 2 Mod Type	
E/S TRANSMITTING RNG CHANNEL:			S/C - E/S RNG CHANNEL:		
5	a. System Type	DSN	15	a. Code Regenerate	No
	b. Tone/Code Wavfm	Square		b. Coh Ops Reqd	Yes
	c. Highest Frequency	1MHz		c. Reqd Accuracy (m)	1
	d. Lowest Frequency	1KHz		d. Bandwidth T/C 1 (Hz)	1
	e. Total Comp No.	12		e. Bandwidth T/C 2 (Hz)	
EARTH-TO-SPACE PATH PERFORMANCE:			SPACE-TO-EARTH PATH PERFORMANCE:		
6	a. Weather Avail (%)	95.00	16	a. Weather Avail (%)	95.00
	b. S/C Distance (km)	384400		b. S/C Distance (km)	384400

CMU Lunar Ice Discovery Icebreaker May 1, 1997.
CMU Discovery

CCSDS LINK DESIGN CONTROL TABLE
EARTH - SPACE - LINK
INPUT DATA SHEET

Page 3

MISSION AND SPACECRAFT				UNITS	CHANNEL 1 DESIGN VALUE	CHANNEL 1 FAV TOL	CHANNEL 1 ADV TOL	CHANNEL 2 DESIGN VALUE	CHANNEL 2 FAV TOL	CHANNEL 2 ADV TOL
E/S TRANSMITTING RF CARRIER CHANNEL PARAMETERS:										
1	Transmitter Power			dBW	42.30					
2	Transmitter Frequency			MHz	2,073					
3	Ant. Gain	DSN	34BWG1	dBi	55.20	0.00	0.00			
4	Antenna Circuit Loss			dB	0.00					
5	Ant. Pointing Loss			dB	-0.10	0.00	0.00			
E/S TRANSMITTING DATA CHANNEL PARAMETERS:										
6	Information Bit Rate			b/s	2,000					
7	Subcarrier Frequency			kHz	16KHz					
8	Subcarrier Waveform			Sin-Sq	Sine					
9	RF Modulation Index			Rad-pk	1.00					
E/S TRANSMITTING RANGING CHANNEL PARAMETERS:										
10	Simultaneous With Data			Yes-No	Yes	Incomp Mod Indices; Pls xqt CNTRL R at I 131				
11	Ranging Waveform			Sin-Sq	Square					
12	Mod Index Tone/Code			Rad-pk	0.80					
	b. Mod Index Tone/Code 2			Rad-pk						
EARTH - TO - SPACE PATH PARAMETERS:										
13	Topocentric Range			km	384400					
14	Atmospheric Attenuation			dB	-0.16					
15	Ionospheric Loss			dB	-0.10					
16	Antenna Elevation Angle			deg	11.00					
S/C RECEIVING RF CARRIER CHANNEL PARAMETERS:										
17	Antenna Gain (HGA)			dBi	11.00					
18	Polarization Loss			dB	-0.50					
19	Ant. Point Loss			dB	-3.00	0.00	0.00			
20	Antenna Circuit Loss			dB	-1.00					
21	Carrier Circuit Loss			dB	-0.50					
22	Total Noise Temperature			K	640.00	0.00	0.00			
	a. Rcvr Operating Temp			K	600.00	0.00	0.00			
	b. Feed Through Noise			K	0.00	0.00	0.00			
	c. Hot Body Noise			K	40.00	0.00	0.00			
23	Threshold Loop Noise BW			Hz	174.40					
24	Reqd Thrshld SNR in 2Blo			dB	12.00					
S/C RECEIVING DATA CHANNEL PARAMETERS:										
25	Phase Jitter Loss			dB	-0.50					
26	Demod / Detector Loss			dB	-0.50					
27	Waveform Distortion Loss			dB	-0.50					
28	Max Rng Interfer to Data			dB	-20.00					
29	Reqd Data Eb/No			dB	10.52					
S/C RECEIVING RNG CHANNEL PARAMETERS:										
30	Ranging Demodulator Loss			dB	-1.00					
31	Ranging Filter Bandwidth			MHz	1.50					
32	Reqd Tone/Code 1 SNR			dB	0.00					
33	Reqd Tone/Code 2 SNR			dB						

CMU Lunar Ice Discovery — Icebreaker — May 1, 1997.

CMU Discovery — CCSDS LINK DESIGN CONTROL TABLE

SPACE - EARTH - LINK

INPUT DATA SHEET — Page 4

MISSION AND SPACECRAFT		UNITS	CHANNEL 1 DESIGN VALUE	CHANNEL 1 FAV TOL	CHANNEL 1 ADV TOL	CHANNEL 2 DESIGN VALUE	CHANNEL 2 FAV TOL	CHANNEL 2 ADV TOL
S/C TRANSMITTING RF CARRIER CHANNEL PARAMETERS:								
51	Transmitter Power	dBW	6.99	0.00	0.00			
52	Transmitter Frequency	MHz	2,250					
53	Antenna Gain (HGA)	dBi	11.00	0.00	0.00			
54	Antenna Circuit Loss	dB	-1.50	0.80	-0.80			
55	Ant. Point Loss	dB	-3.00	0.00	0.00			
S/C TRANSMITTING DATA CHANNEL PARAMETERS:								
56	Information Bit Rate	kb/s	10					
57	Subcarrier Frequency	kHz						
58	Subcarrier Waveform	Sin-Sq						
59	RF Modulation Index	Rad-pk	1.40	0.05	-0.05	0.80	0.05	-0.05
S/C TRANSMITTING RNG CHANNEL PARAMETERS:								
60	Simultaneous With Data	Yes-No	Yes	Incomp Mod Indices; Pls xqt CNTRL R at I 186				
61	Mod Index Tone/Code	Rad-pk	0.40	0.05	-0.05			
SPACE - TO - EARTH PATH PARAMETERS:								
62	Topocentric Range	km	384400					
63	Atmospheric Attenuation	dB	-0.09	0.00	0.00			
64	Ionospheric Loss	dB	0.00	0.00	0.00			
65	Antenna Elevation Angle	deg	20.00					
E/S RECEIVING RF CARRIER CHANNEL PARAMETERS:								
66	Ant Gain DSN 34BWG1	dBi	56.90	0.00	0.00			
67	Polarization Loss	dB	-0.40	0.05	-0.10			
68	Ant. Pointing Loss	dB	-0.20	0.00	0.00			
69	Antenna Circuit Loss	dB	-1.00	0.00	0.00			
70	Total Noise Temperature	K	166.59	0.00	0.00			
	a. Rcvr Operating Temp	K	41.00	0.00	0.00			
	b. Feed Through Noise	K	0.00	0.00	0.00			
	c. Hot Body Noise	K	120.00	0.00	0.00			
	d. Weather Temp Increase	K	5.59	0.00	0.00			
71	Threshold Loop Noise BW	Hz	10.00	0.00	0.00			
72	Reqd Threshold SNR in 2Blo	dB	12.00	0.00	0.00			
E/S RECEIVING DATA CHANNEL PARAMETERS:								
73	Phase Jitter Loss	dB	-0.25	-0.01	0.01			
74	Demod / Detector Loss	dB	-0.20	0.00	0.00	-0.20	0.00	0.00
75	Waveform Distortion Loss	dB	-0.20	0.00	0.00	-0.20	0.00	0.00
76	Max Rng Interfer to Data	dB	-20.00	0.00	0.00	-20.00	0.00	0.00
77	Reqd Data Eb/No	dB	0.75	0.00	0.00		0.00	0.00
E/S RECEIVING RNG CHANNEL PARAMETER:								
78	Ranging Demodulator Loss	dB	-1.00	0.00	0.00			
79	Reqd Tone/Code 1 Pwr/No	dB-Hz	25.45	0.00	0.00			
80	Reqd Tone/Code 2 Pwr/No	dB-Hz						

CMU Lunar Ice Discovery Icebreaker May 1, 1997.
CMU Discovery 1

CCSDS LINK DESIGN CONTROL TABLE
EARTH - TO - SPACE LINK
LINK COMPUTATIONS
(Link and Weather Not Combined)

Page 5

MISSION AND SPACECRAFT		UNITS	DESIGN VALUE	MEAN VALUE	VARI- ANCE	PDF REF	REMARKS
E/S TRANSMITTING RF CARRIER CHANNEL PERFORMANCE:							
101	Transmitter Power	dBW	42.30	42.30	0.00	TRI	
102	Transmit Ant Gain [Effect]	dB	55.10	55.10	0.00	UNI	
103	Transmitting EIRP	dBW	97.40	97.40	0.00	TRI	
104	Trans Carrier Power	dBW	91.94	91.94	0.00	TRI	
105	Trans Carrier Power/PT	dB	-5.46	-5.46	0.00	TRI	1, 2, 2A
E/S TRANSMITTING DATA CHANNEL PERFORMANCES:							
106	Trans Ch 1 Data Power	dBW	90.14	90.14	0.00	TRI	
107	Trans Ch 1 Data Power/PT	dB	-7.26	-7.26	0.00	TRI	3, 4, 4A
108	Trans Ch 2 Data Power	dBW				TRI	
109	Trans Ch 2 Data Power/PT	dB				TRI	
E/S TRANSMITTING RNG CHANNEL PERFORMANCE:							
110	Tone - Code 1 Power	dBW	92.19	92.19	0.00	TRI	
111	Tone - Code 1 Power/PT	dB	-5.21	-5.21	0.00	TRI	5, 6, 6A
112	Tone - Code 2 Power	dBW				TRI	
113	Tone - Code 2 Power/PT	dB				TRI	
EARTH - TO - SPACE PATH PERFORMANCE:							
114	Free Space Loss	dB	-210.47	-210.47	0.00	TRI	7, 8
115	Atmospheric Attenuation	dB	-0.16	-0.16	0.00	GAU	9
116	Ionospheric Loss	dB	-0.10	-0.10	0.00	GAU	
S/C RECEIVING RF CARRIER CHANNEL PERFORMANCE:							
117	Receiving Ant Gain [Effect]	dBi	6.50	6.50	0.00	UNI	
118	Noise Spectral Density	dBW/Hz	-200.54	-200.54	0.00	GAU	
119	Threshold Loop BW, 2BLo	Hz	463.60	174.40	0.00	TRI	
120	Rcvd Carrier Power	dBW	-112.29	-112.29	0.00	TRI	SMEX S-Band
121	Carrier Performance Margin	dB	**49.09**	**53.34**	**0.00**	TRI	Sim Cmd & Rng
S/C RECEIVING DATA CHANNEL PERFORMANCE (SIMULT RNG):							
122	Ch 1 Data Loss Due to Rng	dB	-0.15			TRI	
123	Rcvd Ch 1 Eb / No	dB	53.45	53.45	0.00	TRI	
124	Reqd Ch 1 Eb / No	dB	12.17	12.17	0.00	TRI	
125	Ch 1 Data Perform Margin	dB	**41.28**	**41.28**	**0.00**	TRI	SMEX S-Band
126	Ch 2 Data Loss Due to Rng	dB				TRI	
127	Rcvd Ch 2 Eb / No	dB				TRI	
128	Reqd Ch 2 Eb / No	dB					
129	Ch 2 Data Perform Margin	dB				TRI	
S/C RECEIVING RNG CHANNEL PERFORMANCE (SIMULT DATA):							
130	Rcvd Code 1 Power / No	dB-Hz	87.51	87.51	0.00	TRI	
131	Rcvd Code 2 Power / No	dB-Hz				TRI	
132	Rcvd Total Rng Power / No	dB-Hz	87.51	87.51	0.00	TRI	
133	Ranging Margin	dB	**25.74**	**25.74**	**0.00**	TRI	SMEX S-Band

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CMU Discovery | CCSDS LINK DESIGN CONTROL TABLE

SPACE - TO - EARTH LINK

LINK COMPUTATIONS

(Link and Weather Not Combined)

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MISSION AND SPACECRAFT		UNITS	DESIGN VALUE	MEAN VALUE	VARI-ANCE	PDF REF	REMARKS
S/C TRANSMITTING RF CARRIER CHANNEL PERFORMANCE:							
151	Transmitter Power	dBW	6.99	6.99	0.00	TRI	
152	Transmit Ant Gain [Effect]	dB	6.50	6.50	0.21	UNI	
153	Transmitting EIRP	dBW	13.49	13.49	0.21	TRI	
154	Trans Carrier Power	dBW	13.11	13.10	0.21	TRI	
155	Trans Carrier Power/PT	dB	-0.38	-0.39	0.00	TRI	10 , 11,11A
S/C TRANSMITTING DATA CHANNEL PERFORMANCE:							
156	Trans Ch 1 Data Power	dBW	-2.26	-2.26	0.21	TRI	
157	Trans Ch 1 Data Power/PT	dB	-15.75	-15.75	0.00	TRI	12, 13, 13A
158	Trans Ch 2 Data Power	dBW	No Tlm	No Tlm	No Tlm	TRI	
159	Trans Ch 2 Data Power/PT	dB	No Tlm	No Tlm	No Tlm	TRI	
S/C TRANSMITTING RNG CHANNEL PERFORMANCE:							
160	Tone - Code 1 Power	dBW	-1.17	-1.19	0.40	TRI	
161	Tone - Code 1 Power/PT	dB	-14.66	-14.68	0.19	TRI	14, 15, 15A
162	Tone - Code 2 Power	dBW				TRI	
163	Tone - Code 2 Power/PT	dB				TRI	
SPACE - TO - EARTH PATH PERFORMANCE:							
164	Free Space Loss	dB	-211.18	-211.18	0.00	TRI	16, 17
165	Atmospheric Attenuation	dB	-0.09	-0.09	0.00	GAU	18
166	Ionospheric Loss	dB	0.00	0.00	0.00	GAU	
E/S RECEIVING RF CARRIER CHANNEL PERFORMANCE:							
167	Receiving Ant Gain [Effect]	dBi	55.30	55.27	0.00	UNI	
168	Noise Spectral Density	dBW/Hz	-206.38	-206.38	0.00	GAU	
169	Threshold Loop BW, 2BLo	Hz	10.00	10.00	0.00	TRI	
170	Rcvd Carrier Power	dBW	-142.86	-142.89	0.22	TRI	
171	Carrier Performance Margin	dB	**41.52**	**41.49**	**0.22**	GAU	DSN Block V
E/S RECEIVING DATA CHANNEL PERFORMANCE (SIMULT RNG):							
172	Ch 1 Data Loss Due to Rng	dB	-0.07			TRI	
173	Rcvd Ch 1 Eb / No	dB	8.15	8.13	0.22	TRI	
174	Reqd Ch 1 Eb / No	dB	1.47	1.47	0.00		
175	Ch 1 Data Perform Margin	dB	**6.69**	**6.66**	**0.22**	GAU	DSN Block V
176	Ch 2 Data Loss Due to Rng	dB				TRI	
177	Rcvd Ch 2 Eb / No	dB	No Tlm	No Tlm	No Tlm	TRI	
178	Reqd Ch 2 Eb / No	dB	No Tlm	No Tlm	No Tlm		
179	Ch 2 Data Perform Margin	dB	**No Tlm**	**No Tlm**	**No Tlm**	GAU	
E/S RECEIVING RNG CHANNEL PERFORMANCE (SIMULT DATA):							
180	Rcvd Code 1 Power / No	dB-Hz	48.24	48.20	0.41	TRI	
181	Rcvd Code 2 Power / No	dB-Hz				TRI	
182	Rcvd Total Rng Power / No	dB-Hz	48.24	48.20	0.41	TRI	
183	Ranging Margin	dB	**22.79**	**22.75**	**0.41**	GAU	DSN Block V

CMU Lunar Ice Discovery Icebreaker May 1, 1997.
CMU Discovery 1

CCSDS LINK DESIGN CONTROL TABLE
GENERAL INFORMATION
(Link and Weather Not Combined)

Page 1

1	Owner CCSDS Agency	NASA
2	Name of Mission	CMU Lunar Ice Discovery
3	Name of Spacecraft	Icebreaker
4	Mission Category a. A = Alt.<2,000,000 km b. B = Alt.>2,000,000 km	A (Near Earth Mission)
5	Link Budget Number	Low Gain Antenna Array #1
6	Revision No. / Conditions	S/C-E St. Rng= km, B. Rt. = 1 Kbps S/C: Ant Gn = 2.0dBi; Xmt Pow = 5.0 W; Res. Car. DSN : 34BWG1 m; 17 KW; Cmd = 2000 bps; Sim. Rng. TLM Coding: Ch. 1: r = 0.166666666666667 , k = 15 and 223/255 R/S
7	Date	May 1, 1997.
8	File Name	CMU Discovery 1
9	Project Cognizant Person Name: Title: Address: Telephone: Fax Number: Telemail/Telex No:	Matthew Deans The Robotics Institute Carnegie Mellon University Pittsburgh, PA 15213 (412) 268-1860 (412) 268-5571 deano@ri.cmu.edu
10	Network Cognizant Person Name: Address: Telephone: FAX No: Telemail / Telex No:	Gary K. Noreen Jet Propulsion Laboratory 4800 Oak Grove Drive, M/S 303-402 Pasadena, CA. 91109-8099 (818) 354-2699 (818) 393-1692 gary.k.noreen@jpl.nasa.gov

CMU Lunar Ice Discovery | Icebreaker | May 1, 1997.
CMU Discovery 1

CCSDS LINK DESIGN CONTROL TABLE
COMMUNICATIONS SYSTEM OPERATING CONDITIONS
(Link and Weather Not Combined)

Page 2

EARTH - SPACE - LINK			SPACE - EARTH - LINK		
E/S TRANSMITTING RF CHANNEL:			S/C TRANSMITTING RF CHANNEL:		
1	RF Carrier Modulation		11	RF Carrier Modulation	
	a. Ch 1 Type	PM		a. Ch 1 Type	PM
	b. Ch 1 Format	PSK		b. Ch 1 Format	NRZ-L
	c. Ch 2 Type			c. Ch 2 Type	
	d. Ch 2 Format			d. Ch 2 Format	
E/S TRANSMITTING DATA CHANNEL:			S/C TRANSMITTING DATA CHANNEL:		
2	Baseband Data		12	Baseband Data	
	a. Ch 1 Bit Rate, b/s	2000.00		a. Ch 1 Bit Rate, kb/s	1.00
	b. Ch 1 Bit Err Rate	1.0E-06		b. Ch 1 Bit Err Rate	1.0E-05
	c. Ch 2 Bit Rate, b/s			c. Ch 2 Bit Rate, kb/s	
	d. Ch 2 Bit Err Rate			d. Ch 2 Bit Err Rate	
3	Data Coding		13	Data Coding	
	a. Ch 1 Type			a. Ch 1 Rate	0.17
	b. Ch 1 No. Info Bit			b. Ch 1 Constr Lngth	15
	c. Ch 1 Block Length			c. Ch 1 Concat Code	R/S
				d. Ch 1 Data/Tot Bit	223/255
	d. Ch 2 Type			e. Ch 2 Rate	
	e. Ch 2 No. Info Bit			f. Ch 2 Constr Lngth	
	f. Ch 2 Block Length			g. Ch 2 Concat Code	
				h. Ch 2 Data/Tot Bit	
4	Subcarrier		14	Subcarrier	
	a. Ch 1 Waveform	Sine		a. Ch 1 Waveform	
	b. Ch 1 Frequency	16KHz		b. Ch 1 Frequency	
	c. Ch 1 Mod Type	PSK		c. Ch 1 Mod Type	
	d. Ch 2 Waveform			d. Ch 2 Waveform	
	e. Ch 2 Frequency			e. Ch 2 Frequency	
	f. Ch 2 Mod Type			f. Ch 2 Mod Type	
E/S TRANSMITTING RNG CHANNEL:			S/C - E/S RNG CHANNEL:		
5	a. System Type	DSN	15	a. Code Regenerate	No
	b. Tone/Code Wavfm	Square		b. Coh Ops Reqd	Yes
	c. Highest Frequency	1MHz		c. Reqd Accuracy (m)	1
	d. Lowest Frequency	1KHz		d. Bandwidth T/C 1 (Hz)	1
	e. Total Comp No.	12		e. Bandwidth T/C 2 (Hz)	
EARTH-TO-SPACE PATH PERFORMANCE:			SPACE-TO-EARTH PATH PERFORMANCE:		
6	a. Weather Avail (%)	95.00	16	a. Weather Avail (%)	95.00
	b. S/C Distance (km)	384400		b. S/C Distance (km)	384400

CMU Lunar Ice Discovery | Icebreaker | May 1, 1997.

CMU Discovery

CCSDS LINK DESIGN CONTROL TABLE

EARTH - SPACE - LINK

INPUT DATA SHEET

Page 3

MISSION AND SPACECRAFT				UNITS	CHANNEL 1 DESIGN VALUE	CHANNEL 1 FAV TOL	CHANNEL 1 ADV TOL	CHANNEL 2 DESIGN VALUE	CHANNEL 2 FAV TOL	CHANNEL 2 ADV TOL
E/S TRANSMITTING RF CARRIER CHANNEL PARAMETERS:										
1	Transmitter Power			dBW	42.30					
2	Transmitter Frequency			MHz	2,073					
3	Ant. Gain	DSN	34BWG1	dBi	55.20	0.00	0.00			
4	Antenna Circuit Loss			dB	0.00					
5	Ant. Pointing Loss			dB	-0.10	0.00	0.00			
E/S TRANSMITTING DATA CHANNEL PARAMETERS:										
6	Information Bit Rate			b/s	2,000					
7	Subcarrier Frequency			kHz	16KHz					
8	Subcarrier Waveform			Sin-Sq	Sine					
9	RF Modulation Index			Rad-pk	1.00					
E/S TRANSMITTING RANGING CHANNEL PARAMETERS:										
10	Simultaneous With Data			Yes-No	Yes					
11	Ranging Waveform			Sin-Sq	Square					
12	Mod Index Tone/Code			Rad-pk	0.80					
	b. Mod Index Tone/Code 2			Rad-pk						
EARTH - TO - SPACE PATH PARAMETERS:										
13	Topocentric Range			km	384400					
14	Atmospheric Attenuation			dB	-0.16					
15	Ionospheric Loss			dB	-0.10					
16	Antenna Elevation Angle			deg	11.00					
S/C RECEIVING RF CARRIER CHANNEL PARAMETERS:										
17	Antenna Gain (LGA)			dBi	2.00					
18	Polarization Loss			dB	-0.50					
19	Ant. Point Loss			dB	-3.00	0.00	0.00			
20	Antenna Circuit Loss			dB	-1.00					
21	Carrier Circuit Loss			dB	-0.50					
22	Total Noise Temperature			K	640.00	0.00	0.00			
	a. Rcvr Operating Temp			K	600.00	0.00	0.00			
	b. Feed Through Noise			K	0.00	0.00	0.00			
	c. Hot Body Noise			K	40.00	0.00	0.00			
23	Threshold Loop Noise BW			Hz	174.40					
24	Reqd Thrshld SNR in 2Blo			dB	12.00					
S/C RECEIVING DATA CHANNEL PARAMETERS:										
25	Phase Jitter Loss			dB	-0.50					
26	Demod / Detector Loss			dB	-0.50					
27	Waveform Distortion Loss			dB	-0.50					
28	Max Rng Interfer to Data			dB	-20.00					
29	Reqd Data Eb/No			dB	10.52					
S/C RECEIVING RNG CHANNEL PARAMETERS:										
30	Ranging Demodulator Loss			dB	-1.00					
31	Ranging Filter Bandwidth			MHz	1.50					
32	Reqd Tone/Code 1 SNR			dB	0.00					
33	Reqd Tone/Code 2 SNR			dB						

CMU Lunar Ice Discovery — Icebreaker — May 1, 1997.

CMU Discovery — CCSDS LINK DESIGN CONTROL TABLE

SPACE - EARTH - LINK

INPUT DATA SHEET — Page 4

MISSION AND SPACECRAFT		UNITS	CHANNEL 1			CHANNEL 2		
			DESIGN VALUE	FAV TOL	ADV TOL	DESIGN VALUE	FAV TOL	ADV TOL
S/C TRANSMITTING RF CARRIER CHANNEL PARAMETERS:								
51	Transmitter Power	dBW	6.99	0.00	0.00			
52	Transmitter Frequency	MHz	2,250					
53	Antenna Gain (LGA)	dBi	2.00	0.00	0.00			
54	Antenna Circuit Loss	dB	-1.50	0.80	-0.80			
55	Ant. Point Loss	dB	-3.00	0.00	0.00			
S/C TRANSMITTING DATA CHANNEL PARAMETERS:								
56	Information Bit Rate	kb/s	1					
57	Subcarrier Frequency	kHz						
58	Subcarrier Waveform	Sin-Sq						
59	RF Modulation Index	Rad-pk	1.40	0.05	-0.05	0.80	0.05	-0.05
S/C TRANSMITTING RNG CHANNEL PARAMETERS:								
60	Simultaneous With Data	Yes-No	Yes					
61	Mod Index Tone/Code	Rad-pk	0.40	0.05	-0.05			
SPACE - TO - EARTH PATH PARAMETERS:								
62	Topocentric Range	km	384400					
63	Atmospheric Attenuation	dB	-0.09	0.00	0.00			
64	Ionospheric Loss	dB	0.00	0.00	0.00			
65	Antenna Elevation Angle	deg	20.00					
E/S RECEIVING RF CARRIER CHANNEL PARAMETERS:								
66	Ant Gain DSN 34BWG1	dBi	56.90	0.00	0.00			
67	Polarization Loss	dB	-0.40	0.05	-0.10			
68	Ant. Pointing Loss	dB	-0.20	0.00	0.00			
69	Antenna Circuit Loss	dB	-1.00	0.00	0.00			
70	Total Noise Temperature	K	166.59	0.00	0.00			
	a. Rcvr Operating Temp	K	41.00	0.00	0.00			
	b. Feed Through Noise	K	0.00	0.00	0.00			
	c. Hot Body Noise	K	120.00	0.00	0.00			
	d. Weather Temp Increase	K	5.59	0.00	0.00			
71	Threshold Loop Noise BW	Hz	10.00	0.00	0.00			
72	Reqd Threshold SNR in 2Blo	dB	12.00	0.00	0.00			
E/S RECEIVING DATA CHANNEL PARAMETERS:								
73	Phase Jitter Loss	dB	-0.25	-0.01	0.01			
74	Demod / Detector Loss	dB	-0.20	0.00	0.00	-0.20	0.00	0.00
75	Waveform Distortion Loss	dB	-0.20	0.00	0.00	-0.20	0.00	0.00
76	Max Rng Interfer to Data	dB	-20.00	0.00	0.00	-20.00	0.00	0.00
77	Reqd Data Eb/No	dB	0.75	0.00	0.00		0.00	0.00
E/S RECEIVING RNG CHANNEL PARAMETER:								
78	Ranging Demodulator Loss	dB	-1.00	0.00	0.00			
79	Reqd Tone/Code 1 Pwr/No	dB-Hz	25.45	0.00	0.00			
80	Reqd Tone/Code 2 Pwr/No	dB-Hz						

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CCSDS LINK DESIGN CONTROL TABLE
EARTH - TO - SPACE LINK
LINK COMPUTATIONS
(Link and Weather Not Combined)

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MISSION AND SPACECRAFT		UNITS	DESIGN VALUE	MEAN VALUE	VARI-ANCE	PDF REF	REMARKS
E/S TRANSMITTING RF CARRIER CHANNEL PERFORMANCE:							
101	Transmitter Power	dBW	42.30	42.30	0.00	TRI	
102	Transmit Ant Gain [Effect]	dB	55.10	55.10	0.00	UNI	
103	Transmitting EIRP	dBW	97.40	97.40	0.00	TRI	
104	Trans Carrier Power	dBW	91.94	91.94	0.00	TRI	
105	Trans Carrier Power/PT	dB	-5.46	-5.46	0.00	TRI	1, 2, 2A
E/S TRANSMITTING DATA CHANNEL PERFORMANCES:							
106	Trans Ch 1 Data Power	dBW	90.14	90.14	0.00	TRI	
107	Trans Ch 1 Data Power/PT	dB	-7.26	-7.26	0.00	TRI	3, 4, 4A
108	Trans Ch 2 Data Power	dBW				TRI	
109	Trans Ch 2 Data Power/PT	dB				TRI	
E/S TRANSMITTING RNG CHANNEL PERFORMANCE:							
110	Tone - Code 1 Power	dBW	92.19	92.19	0.00	TRI	
111	Tone - Code 1 Power/PT	dB	-5.21	-5.21	0.00	TRI	5, 6, 6A
112	Tone - Code 2 Power	dBW				TRI	
113	Tone - Code 2 Power/PT	dB				TRI	
EARTH - TO - SPACE PATH PERFORMANCE:							
114	Free Space Loss	dB	-210.47	-210.47	0.00	TRI	7, 8
115	Atmospheric Attenuation	dB	-0.16	-0.16	0.00	GAU	9
116	Ionospheric Loss	dB	-0.10	-0.10	0.00	GAU	
S/C RECEIVING RF CARRIER CHANNEL PERFORMANCE:							
117	Receiving Ant Gain [Effect]	dBi	-2.50	-2.50	0.00	UNI	
118	Noise Spectral Density	dBW/Hz	-200.54	-200.54	0.00	GAU	
119	Threshold Loop BW, 2BLo	Hz	463.60	174.40	0.00	TRI	
120	Rcvd Carrier Power	dBW	-121.29	-121.29	0.00	TRI	SMEX S-Band
121	Carrier Performance Margin	dB	**40.09**	**44.34**	**0.00**	TRI	Sim Cmd & Rng
S/C RECEIVING DATA CHANNEL PERFORMANCE (SIMULT RNG):							
122	Ch 1 Data Loss Due to Rng	dB	-0.15			TRI	
123	Rcvd Ch 1 Eb / No	dB	44.45	44.45	0.00	TRI	
124	Reqd Ch 1 Eb / No	dB	12.17	12.17	0.00	TRI	
125	Ch 1 Data Perform Margin	dB	**32.28**	**32.28**	**0.00**	TRI	SMEX S-Band
126	Ch 2 Data Loss Due to Rng	dB				TRI	
127	Rcvd Ch 2 Eb / No	dB				TRI	
128	Reqd Ch 2 Eb / No	dB					
129	Ch 2 Data Perform Margin	dB				TRI	
S/C RECEIVING RNG CHANNEL PERFORMANCE (SIMULT DATA):							
130	Rcvd Code 1 Power / No	dB-Hz	78.51	78.51	0.00	TRI	
131	Rcvd Code 2 Power / No	dB-Hz				TRI	
132	Rcvd Total Rng Power / No	dB-Hz	78.51	78.51	0.00	TRI	
133	Ranging Margin	dB	**16.74**	**16.74**	**0.00**	TRI	SMEX S-Band

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CMU Discovery

CCSDS LINK DESIGN CONTROL TABLE
SPACE - TO - EARTH LINK
LINK COMPUTATIONS
(Link and Weather Not Combined)

Page 6

MISSION AND SPACECRAFT		UNITS	DESIGN VALUE	MEAN VALUE	VARI-ANCE	PDF REF	REMARKS
S/C TRANSMITTING RF CARRIER CHANNEL PERFORMANCE:							
151	Transmitter Power	dBW	6.99	6.99	0.00	TRI	
152	Transmit Ant Gain [Effect]	dB	-2.50	-2.50	0.21	UNI	
153	Transmitting EIRP	dBW	4.49	4.49	0.21	TRI	
154	Trans Carrier Power	dBW	4.11	4.10	0.21	TRI	
155	Trans Carrier Power/PT	dB	-0.38	-0.39	0.00	TRI	10 , 11,11A
S/C TRANSMITTING DATA CHANNEL PERFORMANCE:							
156	Trans Ch 1 Data Power	dBW	-11.32	-11.32	0.21	TRI	
157	Trans Ch 1 Data Power/PT	dB	-15.81	-15.81	0.00	TRI	12, 13, 13A
158	Trans Ch 2 Data Power	dBW	No Tlm	No Tlm	No Tlm	TRI	
159	Trans Ch 2 Data Power/PT	dB	No Tlm	No Tlm	No Tlm	TRI	
S/C TRANSMITTING RNG CHANNEL PERFORMANCE:							
160	Tone - Code 1 Power	dBW	-10.23	-10.24	0.41	TRI	
161	Tone - Code 1 Power/PT	dB	-14.72	-14.73	0.19	TRI	14, 15, 15A
162	Tone - Code 2 Power	dBW				TRI	
163	Tone - Code 2 Power/PT	dB				TRI	
SPACE - TO - EARTH PATH PERFORMANCE:							
164	Free Space Loss	dB	-211.18	-211.18	0.00	TRI	16, 17
165	Atmospheric Attenuation	dB	-0.09	-0.09	0.00	GAU	18
166	Ionospheric Loss	dB	0.00	0.00	0.00	GAU	
E/S RECEIVING RF CARRIER CHANNEL PERFORMANCE:							
167	Receiving Ant Gain [Effect]	dBi	55.30	55.27	0.00	UNI	
168	Noise Spectral Density	dBW/Hz	-206.38	-206.38	0.00	GAU	
169	Threshold Loop BW, 2BLo	Hz	10.00	10.00	0.00	TRI	
170	Rcvd Carrier Power	dBW	-151.86	-151.89	0.22	TRI	
171	Carrier Performance Margin	dB	**32.52**	**32.49**	**0.22**	GAU	DSN Block V
E/S RECEIVING DATA CHANNEL PERFORMANCE (SIMULT RNG):							
172	Ch 1 Data Loss Due to Rng	dB	-0.07			TRI	
173	Rcvd Ch 1 Eb / No	dB	9.09	9.07	0.22	TRI	
174	Reqd Ch 1 Eb / No	dB	1.47	1.47	0.00		
175	Ch 1 Data Perform Margin	dB	**7.63**	**7.60**	**0.22**	GAU	DSN Block V
176	Ch 2 Data Loss Due to Rng	dB				TRI	
177	Rcvd Ch 2 Eb / No	dB	No Tlm	No Tlm	No Tlm	TRI	
178	Reqd Ch 2 Eb / No	dB	No Tlm	No Tlm	No Tlm		
179	Ch 2 Data Perform Margin	dB	**No Tlm**	**No Tlm**	**No Tlm**	GAU	
E/S RECEIVING RNG CHANNEL PERFORMANCE (SIMULT DATA):							
180	Rcvd Code 1 Power / No	dB-Hz	39.19	39.15	0.41	TRI	
181	Rcvd Code 2 Power / No	dB-Hz				TRI	
182	Rcvd Total Rng Power / No	dB-Hz	39.19	39.15	0.41	TRI	
183	Ranging Margin	dB	**13.73**	**13.69**	**0.41**	GAU	DSN Block V

Appendix G

Detailed Budgets

The following table contains the mass and power requirements of each subsystem on Icebreaker. Power is divided into the requirements for each mode of operation. The rows of the table in bold indicate the totals for that subsystem. The normal text rows below the bold row is a breakdown of how that subsystem's mass and power is used. For example, Science Instruments indicates that a total of 8.7kg is used for scientific instruments. That mass is comprised of 6.0kg for the REGA, 2.2kg for the drill and 0.5kg for the Mossbauer. The numbers in this table are approximations and are meant to provide some idea of power requirements and where mass is being used. Many smaller items, such as cables, have been lumped into larger categories such as Mechanical Structure. It is expected that many of these numbers will change as the design is refined.

TABLE G.1 *Mass & Power Budget*

Categories	Mass (kg)			Power (W)	
		Landing	Driving	Drilling	Charging
Science Instruments	**8.7**			**60**	**3**
REGA	6.0			50	
Drill	2.2			10	
Mossbauer	0.5				3
Locomotion	**110**		**112**		
@0.3m/s up a 10deg slope			100		

TABLE G.1 *Mass & Power Budget*

Categories	Mass (kg)			Power (W)	
		Landing	Driving	Drilling	Charging
Wheel Modules (4)	80				
Steering	30		12		
Sensors	**19**	**37**	**64**	**34**	**19**
Radar Altimeter	2	12			
Belly Cam	1	6			
IMU (2)	2	13	13	13	13
Star Tracker (2)	1	6	6		6
Stereo Cameras (2)	2		12		
Panospheric	2		12		
Strobe Lights (3)	2				
Terrain Radar Unit	3		20		
Drill Cam	1			6	
Encoders (10)	1		1		
Remote Ice Sensing IR camera	2			15	
Thermal System	15	0	0	0	0
Mechanical Structure	90				
Power System	**17**				
Solar Panels	5				
Batteries	50				
Support Electronics	2				
Computing	**15**	**60**	**60**	**60**	**60**
Commun-ications	**5**	**38**	**38**	**38**	**38**

TABLE G.1 *Mass & Power Budget*

Categories	Mass (kg)			Power (W)	
		Landing	Driving	Drilling	Charging
Propulsion	**1134**				
Solid Fuel	870				
SRM	64				
Hydrazine	127				
Liquid Propulsion System	50				
Total	**1413.7kg**	**135W**	**274W**	**192**	**120**

Appendix H

References

1. Apostolopoulos, D., *Systematic Configuration of Robotic Locomotion*, Carnegie Mellon University, technical report, CMU-RI-TR-96-30, 1996.
2. *Astronomical Almanac*, Naval Observatory, U.S. Department of Defense, Washington, D.C., annual.
3. Bekker, M., G., *Theory of Land Locomotion: The Mechanics of Vehicle Mobility*, University of Michigan Press, Ann Arbor, MI, 1956.
4. Bester, Manfred, *SatTrack* [computer program], http://shuttle.nasa.gov/demos/tracksw/sattrack.V3.1.tar.gz.
5. Dudzinski, Piotr, *Design Characteristics of Steering Systems for Mobile Wheeled Earthmoving Equipment*, Journal of Terramechanics, Vol. 26, No. 1, pp. 25-82, 1989.
6. DY4 Systems, Inc. *VME Product Catalog*, Revision 1995.
7. Gillespie, Thomas, *Fundamentals of Vehicle Dynamics*, Society of Automotive Engineers, 1992.
8. Gorevan, Stephen, *A miniature Sample Acquisition and Transfer Mechanism (SATM) for the Champollion Cometary Lander*, Honeybee Robotics Ltd., technical report, 1996.
9. Heiken, G., D.T. Vaniman and B.M. French, *Lunar Sourcebook*, Cambridge University Press, 1991.
10. Kay, J., and C.E. Thorpe, *STRIPE Supervised Telerobotics Using Incremental Polygonal Earth Geometry*, Preceedings of the 3rd International Conference on Intelligent Agents, Pittsburgh, PA, 1993.
11. Koh, Chris, Personal Communication, April, 1997, Millitech Corp., South Deerfield Research Park, Post Office Box 109, South Deerfield, MA 01373-9990, Tel (413) 665-8551.
12. Larson, W. and Wertz, J., ed., *Space Mission Analysis and Design*, Microcosm, Inc. and Kluwer Academic Publishers, 1992.
13. Litton Systems, Inc., *LN-200 Fiber Optic Inertial System*, Proceedings Of The Clementine Program's Engineering and Technology Workshop, Lake Tahoe, NV, July 18-19, 1994.

14. Martin, W., *CCSDS Link Design Control Table: Software User's Manual*, Tech Report, Jet Propulsion Laboratory, Pasedena California, 1995.

15. McKay, David and Carlton Allen, *REGA (Regolith Evolved Gas Analyzer)*, Proceedings 5th International Conference on Space, June, 1996.

16. Meeus, Jean, *Astronomical Algorithms*, Willmann-Bell: Richmond, VA, 1991.

17. Pleasance, L. Lawrence Livermore National Laboratory, *Lightweight Sensor System*, Proceedings Of The Clementine Program's Engineering and Technology Workshop, Lake Tahoe, NV, July 18-19, 1994.

18. Roy, A.E., *Orbital Motion*, Wiley & Sons, New York, 1978.

19. Salvail, James R. and Fraser P. Fanale, *Near-Surface Ice on Mercury and the Moon: A Topographic Thermal Model*, *Icarus*, Vol.111, pp.441-455, 1994.

20. Skolnik, Merrill I., *Introduction to Radar Systems*, 2nd Edition, 1980, McGraw-Hill Publishing Co.

21. Spudis, Paul D., *The Once and Future Moon*, Smithsonian Institution, Wash. D.C., 1996.

22. Stacy, N. J. S., D. B. Campbell and P. G. Ford, *Radar Mapping of the Lunar Poles: A Search for Ice Deposits*, submitted to *Science*.

23. Wallace, B., and Rao, N., *Engineering Elements for Transportation on the Lunar Surface*, Applied Mechanics REview, Vol. 46, N0. 6, June 1993.

24. Watson, Kenneth, Bruce C. Murray and Harrison Brown, *The Behavior of Volatiles on the Lunar Surface*, *Journal of Geophysical Research*, Vol. 66, No. 9, pp.3033-3045, 1961.

25. Wilcox, B., L. Matthies, D. Gennery, B. Cooper, T. Nguyen, T. Litwin, A. Mishkin, and H. Stone, *Robotic Vehicles for Planetary Exploration*, IEEE International Conference on Robotics and Autonmation, Nice, France, 1992.

26. Whittaker, William, Bapna, Deepak, Maimone, Mark, Rollins, Eric. *Atacama Desert Trek: A Planetary Analog Field Experiment*, i-SAIRAS '97, Tokyo, Japan, July 14-16, 1997.