

COPERNICUS - LUNAR SURFACE MAPPER

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ABSTRACT

The Utah State University (USU) 1991-92 Space Systems Design Team has designed a Lunar Surface Mapper (LSM) to parallel the development of the NASA Office of Exploration lunar initiatives. USU students named the LSM "Copernicus" after the 16th Century Polish astronomer, for whom the large lunar crater on the face of the moon was also named. The top level requirements for the Copernicus LSM are to produce a digital map of the lunar surface with an overall resolution of 12 meters (39.4 ft). It will also identify specified local surface features/areas to be mapped at higher resolutions by follow-on missions. The mapping operation will be conducted from a 300 km (186 mi) lunar-polar orbit. Although the entire surface should be mapped within six months, the spacecraft design lifetime will exceed one year with sufficient propellant planned for orbit maintenance in the anomalous lunar gravity field. The Copernicus LSM is a small satellite capable of reaching lunar orbit following launch on a Conestoga launch vehicle which is capable of placing 410 kg (900 lb) into translunar orbit. Upon orbital insertion, the spacecraft will weigh approximately 233 kg (513 lb). This rather severe mass constraint has insured attention to component/ subsystem size and mass, and prevented "requirements creep". Transmission of data will be via line-of-sight to an earth-based receiving system.

1.0 SYSTEMS

1.1 Introduction

The moon remains the only body in the solar system for which there has been actual sample return. In fact, some of the approximately 840 (381 kg) pounds of lunar material returned is still undergoing investigation at NASA's Johnson Space Center. But, while Apollo and its precursor missions have provided a wealth of information about our nearest neighbor, many fundamental questions remain unanswered. This fact was underscored in late 1990 with the Galileo spacecraft. Galileo, en route to Jupiter, made a flyby of the earth-moon system, and it imaged a region on the far side of the moon that had never been seen before.

Before questions concerning the nature and origin of the moon can be properly addressed, high-resolution lunar data, such as that provided by Galileo, must be collected on a global scale. In the 61 lunar missions that have flown to date, none have provided a global survey, which leaves the current data set limited in two fundamental ways. First, the data is the product of technology that is 25-years-old and is, therefore, very low quality. Second, the data is confined to low latitudes because it was generated by spacecraft which flew in near-equatorial orbits. To complete global assessment of the lunar topography, geochemistry, and surface mineral distribution, the spacecraft must be in a near circular, polar orbit. Such an orbit is necessary because it allows the planet to turn underneath the orbit plane, permitting the spacecraft to pass over the entire surface. It also maintains a constant altitude and, therefore, constant data resolution. In the history of lunar exploration only two U.S. spacecraft have been in polar orbits. These spacecraft were among the five in the Lunar Orbital Program (LOP) from August 1966 to August 1967. However, the orbits of the LOP spacecraft were highly elliptical. Furthermore, the data was analog and had low signal to noise ratios. As a result, the data cannot be enhanced as can be done with the data from today's digital instrumentation.

Scientific interest in the moon can be separated into two broad categories: resources and mapping. Data returned concerning resources will describe the global surface mineral distribution. A high-resolution lunar map will provide the data necessary to answer questions of selenology (called geology on earth) or those processes which shape the surface of the moon. A lunar map is also critical to future surface robotic exploration and establishment of a manned lunar base.

Single spacecraft that address both of these categories, such as the Lunar Observer proposed by the Jet Propulsion Laboratory (JPL), have been studied extensively¹. The Lunar Observer design, with its 13 separate instruments, became very difficult and costly to implement and, as a result, never gained congressional support.

In November of 1991, NASA's Associate Administrator for Exploration, Dr. Michael Griffin, outlined a new approach to lunar missions in his paper "Exploration Program Plan"². This approach reflects a new design

philosophy which has emerged in the space engineering community. By separating diverse, often conflicting, mission requirements, this new design philosophy yields smaller more reliable spacecraft. In addition, the development of such a spacecraft more easily meets the constraints of budget and schedule, making them more acceptable to congress. Dr. Griffin proposed two separate lunar missions: one devoted exclusively to mapping and the other to resources. Griffin also pointed out that these missions should be within the \$100 million class, which is very inexpensive relative to other interplanetary missions.

This year's Space Systems Design Course at Utah State University designed a spacecraft to address the mapping interests in lunar science. A key element in the design was to generate a high-resolution, topographical map of the lunar surface emphasizing the low cost/fast turn around approach outlined above.

Copernicus - Lunar Surface Mapper is named after a 16th century Polish astronomer for whom a large lunar impact crater is also named. The Copernicus project includes the design of each spacecraft subsystem with the goal of incorporating as much detail into the design as possible within the time limitations of the course. This report includes the design of each subsystem, an analysis of the mission operation requirements, and a realistic cost analysis.

1.2 The Copernicus Mission Requirements

A list of top-level requirements for the Copernicus project was generated during fall quarter 1991. These requirements were the result of both science objectives in NASA's Office of Exploration and a great deal of research performed by the students of the Space Systems Design class. The Copernicus top-level requirements are as follows:

- Map the lunar surface to a global resolution of 12 meters (39.5 ft);
- Produce stereo images using digital mapping techniques;
- Operate in a circular, polar orbit maintained at an altitude of 300 kilometers (186 miles);
- Transmit the data to earth via line-of-sight communications;
- Complete the mission within one year; and
- Achieve lunar orbit following launch on a Conestoga launch vehicle.

One major modification in the top-level requirements occurred during the course of this project. The initial resolution requirement was to map the lunar surface globally to 5 meters (16.4 ft.) and maintain the capability to map specific sites to 1 meter (3.28 ft). After completing a preliminary design and showing the resulting spacecraft capabilities this requirement was reexamined. A detailed

discussion of the spacecraft capabilities, under both resolution requirements is given in the next section. However, the reduction in resolution does not significantly effect the science return. As mentioned, the main interest in a high-resolution, topographical map is selenology. And selenological features such as cratering, faulting, and volcanism simply do not have much diversity below 12 meters (39.4 ft).

Launch on a Conestoga vehicle represents the most constraining requirements: weight and volume. The Conestoga launch vehicle, which can deliver 410 kg or 900 pounds (lb) to a translunar trajectory, is smaller than those typically considered for lunar missions. The Conestoga was chosen for the Copernicus project because it represents the smallest reasonable launch vehicle possible for a lunar mission. Smaller launch vehicles, such as the Pegasus, have been studied for lunar missions, but they rely on very constraining translunar trajectories³. These trajectories are designed to save weight by reducing the fuel required to achieve lunar orbit but take over 100 days to complete. With a one year mission life, such trajectories were considered far too confining. The next step up in launch vehicle performance is the Delta II which can deliver approximately 1,300 lb (590 kg) to a translunar trajectory and has a substantially larger payload envelope. While the Conestoga is more limited in terms of both weight and volume and does not have the flight history of the Delta II, it will be a much less expensive launch. Furthermore, baselining the Copernicus spacecraft for a Conestoga launch leaves plenty of margin should other options become necessary.

1.3 The Copernicus Spacecraft

Figure 1.1 shows an isometric view and specifications for the Copernicus spacecraft with the original resolution requirement. Important features to note in this preliminary design are the dual-gimballed, high-gain parabolic antenna and the deployable solar panels. A high-gain antenna was required to support the extremely high data rate of 31 megabits per second (Mbps). This data rate, resulting directly from the resolution requirement of 5 meters (16.4 ft), is actually a factor of greater than 100 higher than any previous planetary mission. While the hardware required to process this data rate does exist, it is not yet space qualified. As a result, high development costs would be expected. Another result of the 5 meter (16.4 ft) resolution requirement was the need for deployable solar panels. These panels, in addition to the body-mounted cells, were required to meet the power needs that could not be met by the body-mounted solar cells alone.

Figure 1.2 shows an isometric view and specifications of the Copernicus spacecraft in its current configuration. Note that the antennas and solar cells are now completely body-

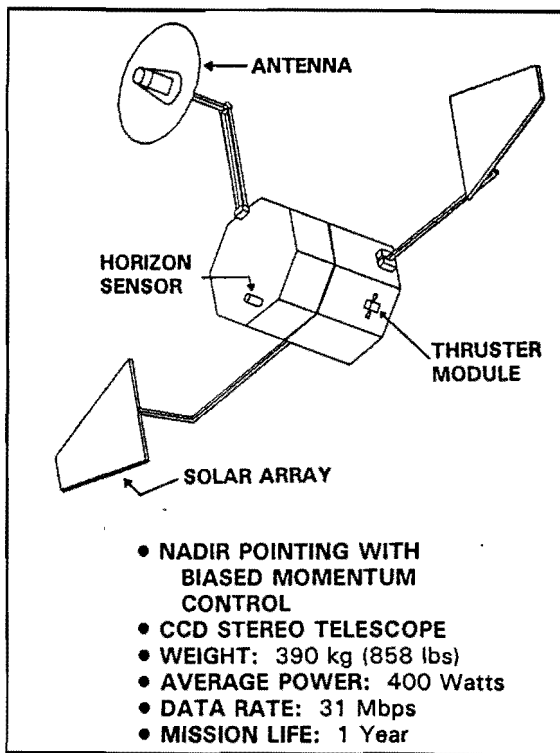


Figure 1.1 Preliminary Design of Copernicus.

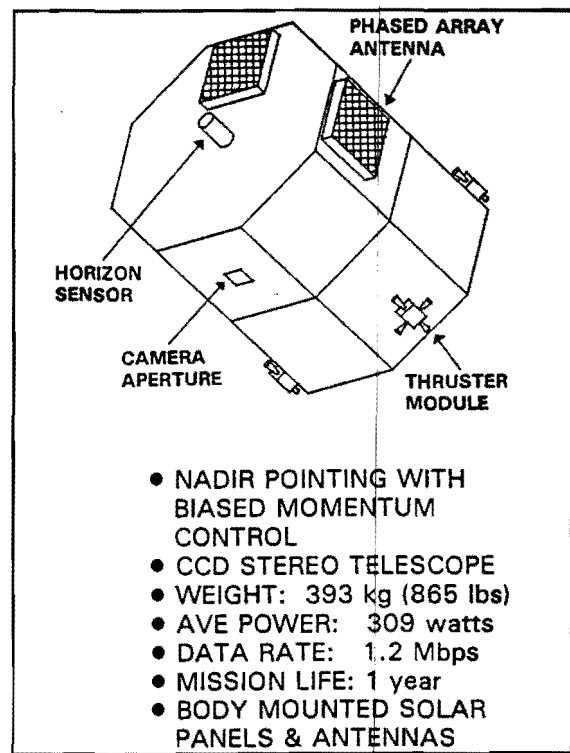


Figure 1.2 Final Design of Copernicus.

mounted. All of the changes shown are a direct result of relaxing the resolution requirement. At a 12 meter (39.4 ft) resolution, the data rate was reduced to 1.2 Mbps, allowing the use of phased array antennas which use electronic steering and are, therefore, much more reliable. Also, the average power was reduced nearly 25%, eliminating the need for deployable panels. This final configuration of Copernicus is much more reliable and much less expensive than the preliminary design.

1.4 Mission Operations

1.4.1 Mapping Mechanics. Because large shadows in the image data are undesirable, mapping can only be performed at a beta angle of 10° to 40° . The beta angle is measured from an incoming light ray to nadir, or the line joining the axis of the telescope to the center of the moon. Because the spacecraft will be in a polar orbit, the beta angle will change continuously. An optimum operational sequence where the spacecraft maps according to latitude and, therefore, according to beta angle, is highly desirable for data storage and transmission. However, the development of such a strategy requires a model that simultaneously predicts the position of the earth, moon, sun, and spacecraft. Such a model was beyond the scope of this project. Consequently, a scenario was assumed where the spacecraft takes data during an entire half orbit, mapping

from the north pole to the south pole. While such a strategy would never be used during the real mission, this assumption results in a spacecraft that is more capable than necessary and ensures that the problem has been bounded. If a "pole-to-pole" strategy were possible, a map could be completed in roughly one month (27.3 days), the time it takes the lunar surface to rotate under the orbit plane. In reality, the time to complete the map is roughly six months. A conservative one year mission life was assumed for Copernicus, allowing for losses in data during transmission, etc.

1.4.2 Swath Overlap. A swath overlap of 25% was assumed for the post-mission reconstruction of the map. After the data has been collected on earth, scientists will begin the process of piecing together the global map. This process includes visually lining up consecutive image swaths, which is impossible without overlap. Previous planetary mapping missions have used swath overlap's in the range of 15 to 25%. The latter was chosen to be conservative.

1.4.3 Data Transmission. During the preliminary design phase of Copernicus, an effort was made to avoid using NASA's Deep Space Network (DSN) for a reception. The reason for this was the high demands that will be placed on the DSN during the time frame that Copernicus would fly. Other spacecraft that will require tracking during the Copernicus flight include Magellan, Galileo, Mars

Observer, and Ulysses. In fact, it had been assumed that other large antennas around the world could be rented or that three 10-meter dishes would have to be built. This assumption was shown to be too confining, however, because the communications subsystem design is very dependent on a particular ground station. Because of this and a later commitment by the DSN to provide 10 hours of coverage per day using the 26 meter receiving antennas, the DSN was incorporated into the design.

2.0 Payload

2.1 Push-Broom Design

The camera is a push-broom type which means it has linear charged coupled device (CCD) arrays placed across track or perpendicular to the spacecraft's orbit path. Every time the satellite passes over 12 m (39.4 ft) on the ground, the arrays will be sampled, or in essence a picture will be taken. Each picture is 12 m (39.4 ft) by 48 km (29.8 mi). These pictures will then be placed side by side to create a map of the lunar surface. The satellite will use two arrays or cameras, one facing 10° forward and the other facing 10° aft. This will provide two views of the surface which will be used to create a topographical or three-dimensional map.

2.2 Stereo Imaging

The Copernicus Lunar Surface Mapper is loosely based upon the French satellite SPOT^{4,5}. The first satellite of this series was launched in 1986. SPOT's mission is to map the surface of the earth. It employs a unique approach to stereo imaging. Instead of looking straight down on the object to be mapped, it has two identical stereo telescopes, each built in at an angle. When the satellite is in position A as shown in Figure 2.1, it views the surface at a given angle. As it moves to position B, it sees the same point on the surface at a different angle. Therefore, every portion of the surface is mapped from two different angles, creating a stereo image pair for each portion of the mapped surface^{4,5}. These stereo image pairs can be sent back to earth and reconstructed, creating an exact three-dimensional representation of the object⁶.

2.3 Detector

A charge coupled device (CCD) will be used to detect the incoming light from the moon. A CCD captures and converts light energy into an electrical signal.

A 12 m (39.4 ft) resolution equates to mapping 12 square meters (39.4 ft²) per 1 picture element (pixel) of the CCD array. Therefore, it was necessary that the CCD array be a minimum of 4,000 pixels long. Utilizing the push-broom technique, a linear CCD array with a high output rate was

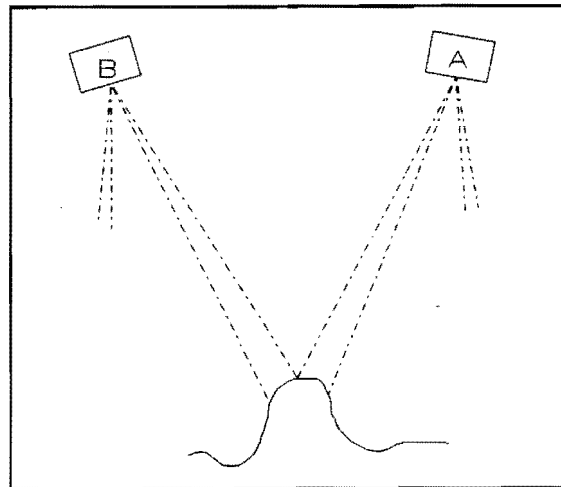


Figure 2.1 Stereo Image Pairs.

the best choice. The following factors influenced the selection of the Loral Fairchild CCD191 6,000 x 1 linear array⁷.

- Detector Size: The CCD191 exceeds the 4,000 pixel requirement.
- Reliability: The CCD191 is space qualified.
- Light sensitivity: The CCD191 satisfies the need for a high dynamic range and responsivity necessary for lunar mapping.
- Output rate: Each CCD array has 2 parallel taps; each tap outputs 3,000 pixels in serial, yielding an approximate 4 MHz output rate⁸.

Since only 4,000 pixels are necessary to map a full 48 km (29.8 mi) swath, the data from 2,000 pixels (1,000 on each end of the array) will be discarded.

3.0 DATA MANAGEMENT AND STORAGE

Data management requirements were to design a system that uses existing technology and components in order to accomplish the following:

- Data Processing
 - Upload data from payload
 - Process data
 - Download data to telemetry
- Housekeeping
 - Remote maintenance
 - Onboard upkeep
 - Status monitoring

In the event that a component did not exist to complete a functioning subsystem, a design study was performed.

Special considerations were taken and consulting with professional engineers was done to assure that our design could be accomplished through research and development.

3.1 Data Processing

For adequate scientific return, a 15 m (49.2 ft) resolution would be sufficient. However, due to the capability of image compression and accessible mass storage, a 12 m (39.4 ft) resolution was selected with a criteria of a 25% swath overlap. The data management subsystem was designed to meet the mission's top level requirements, including the 300 km (186 mi) orbit, 12 m (39.4 ft) resolution, and 25% swath overlap.

3.1.1 Data Processing Requirements. The design requirements for data processing are set by the mission's orbital altitude and resolution criteria.

Orbital Altitude:	300 km (186.5 mi)
Orbital Period:	2.293 hrs (137.6 min)
1/2 Orbital Period:	1.146 hrs (68.8 min)
Orbital Velocity:	1323 m/s (0.882 mi/s)
Relative Ground Speed:	110.25 m/s (0.0685 mi/s)

Resolution: 12 m (39.4 ft) with a 25% overlap

Bytes/Pixel:	1 byte
Pixels/Array:	4,000
# of Arrays:	2
CCD Camera Data Rate:	882 Kbytes/sec
1/2 Orbit Data In:	3.64 Gbytes
Compression Ratio:	10:1
Mapping Data:	364 Mbytes
Map Encoding (20%):	72.8 Kbytes (5.824 Mbits)
Telemetry Encoding (20%):	72.8 Kbytes (5.824 Mbits)
Stored Data:	509.6 Mbytes (4.077 Gbits)
Communications Data Rate:	1.2 Mbps
Download Time:	3,397.5 sec (56.6 min)

a map encoding margin of 20 percent was suggested as a minimum amount of encoding used for digital mapping⁹. The telemetry encoding margin was a requirement given from the communications subsystem for transmitting data back to the earth via phased array antennas.

3.1.2 Compression. Data rate calculations were made assuming a statistically lossless 10:1 data compression. It is presumed that the V.Q. (Vector Quantization) Compression as developed at Utah State University will be used to attain such a compression ratio. Presently, it is proposed that each bit (pixel) x 8 bit (pixel) x 8 bit vector

will be represented by an 8 to 12 bit mean with an approximate 20 bit codebook address. These figures are educated guesses that may well be altered to the scenario best suited for this mission which would be determined in a more detailed manner with development and research¹⁰. One possible concern which arises from the scientific community is how the homogenous nature of the lunar surface can be adequately represented when only 10% of the actual data is being received for reconstruction of the lunar map⁹. At this point, there is every reason to believe that it can be; the nature of V.Q. and its utilization of a codebook would appear very amenable to the processing of a moderately homogenous image¹¹.

3.1.3 Mass Storage. The raw incoming data (without error correction or encoding) presented to the mass storage will be 3.64 Gbits per half orbit. The Solid State Recorder manufactured by Fairchild Space was chosen for this design using the parameters for a 4.5 Gbyte device. This includes a margin for encoding and error correction. The data rates required by this mission fit well within the device specifications. Equipped with its own power supply, chassis, I/O hardware, and error correction, the Solid State Recorder is approximately 30.0 cm x 33.5 cm x 20.0 cm and weighs approximately 23 kg (50.6 lb)¹².

4.0 COMMAND, CONTROL, & COMMUNICATIONS

4.1 Requirements

The design for the command, control, and communications subsystem is based on two main requirements: first, to downlink data from the satellite to the earth at 1.2 Megabits per second (Mbps), and second, to uplink commands from the satellite to the earth at 2 Kilobits per second (Kbps). The data is the signal containing the mapping information collected from the payload. Commands consist of control, handshaking, and emergency communications. To maximize the quality of the mapping pictures, the downlink data must have a minimal amount of erroneous data; thus, minimization of the error probability is an important factor of the required downlink data rate.

The data rates of 1.2 Mbps and 2 Kbps were determined from the amount of data storage provided in the data management subsystem and the amount of time required for a ground station antenna on earth to receive the downloaded data. The Copernicus satellite, which will be using the Deep Space Network (DSN) for communications, is allowed to transmit data to an existing DSN receive antenna for 10 hours per day. This time limitation and the available data memory determined the downlink data rate to be at least 1.2 Mbps. For uplink, however, time for receiving the commands is not such a constraint, and 2 Kbps

is a high enough data rate for the necessary uplink communications.

The design that most appropriately meets these requirements is composed of five phased array antennas, three omnidirectional antennas, and the electronics to control these antennas and data transmitting/receiving. Phased array antennas are relatively flat antennas that consist of multiple elements that electronically steer the transmitting signal. These antennas will be used, under normal conditions, for data downlink. To achieve full steering ability from any satellite position, five phased array antennas are placed on different sides of the satellite structure. The omnidirectional antennas, which are small biconical antennas, will receive commands and, in emergency situations such as when the steered signal loses track of the receive antenna, transmit data. Three of these antennas are placed on opposite sides of the satellite for full steering capability.

4.3 Phased Array Antenna

The phased array antenna that will be used on the Copernicus satellite was chosen to fit design specifications and limitations and to perform effective communications. Each antenna is electrically steerable to an optimal $\pm 45^\circ$ (a maximum of $\pm 60^\circ$) from the perpendicular axis; therefore, five antennas are required so that at least one faces toward the earth at any particular time. To achieve the required data rate of 1.2 Mbps, each antenna has a transmitting gain of 16 dB. The size of each is 41 cm \times 41 cm (16.14 in \times 16.14 in.) and 7.62 cm (3 in.) thick¹³. Each antenna weighs approximately 2.25 to 4.5 kg (5 to 10 lbs) and is made of copper-clad teflon fiberglass and copper-clad epoxy fiberglass.

5.0 SPACECRAFT STRUCTURAL DESIGN

For ease in assembling and testing, a modular design was used. Two modules were utilized, an upper module which houses the electronic components of the satellite and a lower module containing the propulsion systems and momentum wheel.

5.1 Upper Module

The configuration of the electronic components in the upper module is shown in Figure 5.1. To aid in the attitude control of the satellite, the components were placed as symmetric as possible about the geometric center of the satellite. This module may be assembled and tested separately. This decreases the possibility of contamination or damage from the corrosive and carcinogenic hydrazine fuel.

Stringers running down each corner of the spacecraft will be used to connect the components to the satellite structure.

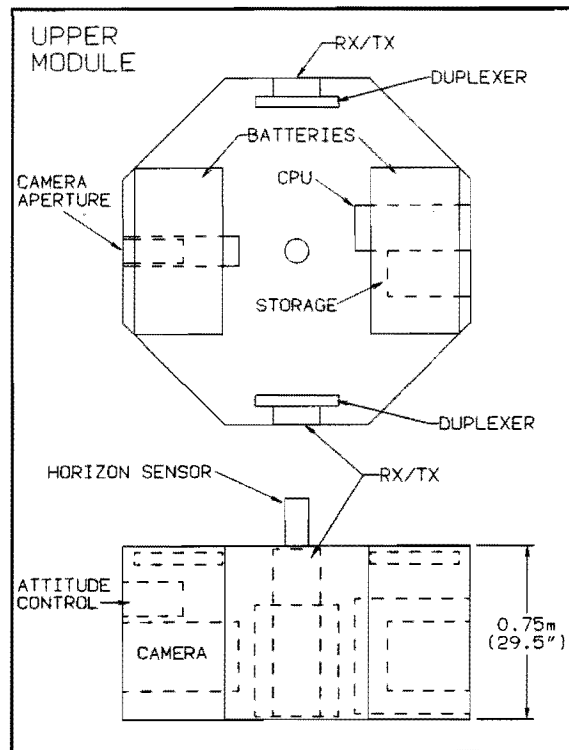


Figure 5.1 Top and Side View of Upper Module Structure of Satellite.

The stringers will be 0.62 meters (24.4 in) apart and extend along the length of both modules. Several components will require mounting arms to connect to the stringers. The other components will utilize arms of aluminum tubing to connect to the stringers.

The aperture of the camera must be protected during translunar flight. A square protective cover 12 cm (4.7 in) wide will be included on the upper module for this purpose. This cover will be secured with explosive bolts and removed after lunar orbital insertion.

5.2 Lower Module

The momentum wheel and all of the fuel elements are housed in the lower module. One phased array antenna will also be connected to the lower module. This will likely be used in testing the upper module and would not be installed on the lower module until final assembly. The lower antenna will be covered during lunar insertion to protect it from the 418 N (100 lb) insertion thruster. After insertion, explosive bolts will be used to separate the cover from the antenna. The lower module of the satellite is shown in Figure 5.2.

The use of body mounted solar cells dictated a satellite structure that was somewhat larger than needed to house the

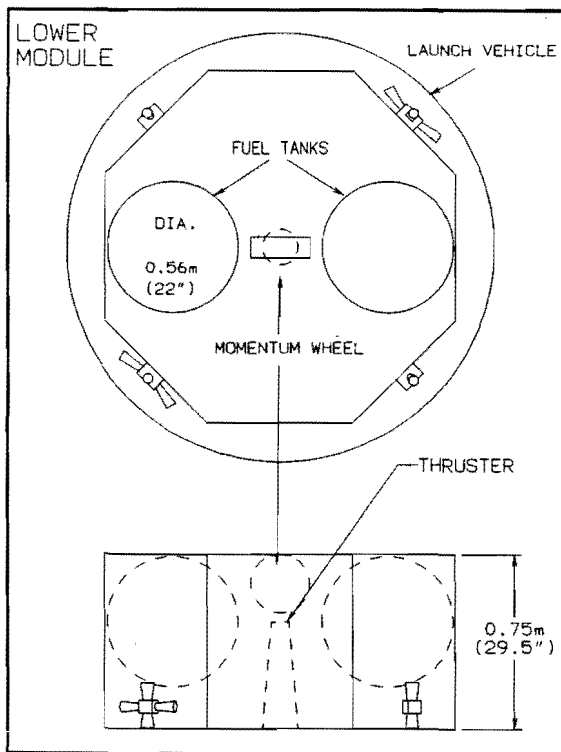


Figure 5.2 Top and Side View of Lower Satellite Module.

components. This allowed for a design where everything could be connected to the structure independently. No two components had to be stacked or connected to each other.

6.0 ATTITUDE DETERMINATION & CONTROL

The attitude control system will provide orientation data of the Lunar Surface Mapper (LSM) in three-dimensional space. It will point and maintain a nominal nadir orientation for mapping the lunar surface. The system will also provide guidance and navigation while the craft is en route to the moon.

6.1 Sensors

The primary attitude sensor will be a Barnes Dual Cone Scanner with Sun Fans (DCS). The DCS is a combination horizon, sun, and moon sensor that also provides altitude information¹⁴. The DCS integrates all of the needed sensors into one unit reducing the error possibility encountered with coordinating multiple sensors.

The DCS with Sun Fans and MANS, designed to function as an autonomous earth sensor for the TAOS mission, flies in late 1992. Earth horizon sensors normally sense the IR signature of the CO₂ band in the atmosphere. This requires

a slight modification for the DCS because of the different IR signature of the moon.

6.2 Translunar Flight

The DCS provides translunar navigation data (2). After translunar insertion, the LSM will be spun about the pitch axis at a rate of 17 rpm (3). This allows the DCS to view the entire sky approximately once every minute. The craft will be spun up using the uncompensated angular momentum of the DCS assisted by the hydrazine thrusters.

6.3 Actuators

The attitude control actuators must maintain the position of the LSM to within the pointing requirements as specified in Table 6.1¹⁵. They must also maintain the orbital altitude. A momentum bias system provides attitude control. A single momentum wheel mounted on a double-gimbal platform makes use of the unique gyroscopic properties of a momentum bias system¹⁶. The momentum bias system couples the roll and yaw axes while pitch control is independent. The combination of the momentum wheel on the platform provides three-axis stabilization.

Pitch & Roll	± 0.07 degrees
Yaw	± 0.02 degrees
Altitude	± 500 meters
Sun Az & El	± 0.01 degrees
Earth Az & El	± 0.02 degrees

Table 6.1 Attitude Pointing Requirements.

The momentum wheel is a Type B T-Wheel manufactured by Ithaco¹⁷. Located near the center of gravity of the LSM, mounting is such that the spin axis is parallel with the pitch axis of the LSM. The wheel spins up after the LSM has achieved lunar orbit. The initial spin rate will be such that the camera is always nadir pointing. Variations in the spin rate will be necessary due to solar and gravitational torques on the LSM. Wheel saturation, if it occurs, will be countered through the use of hydrazine thrusters.

7.0 PROPULSION

7.1 Requirements

The functional requirements for the propulsion subsystem include the following:

- Correct for translunar injection errors;
- Insert the spacecraft into a circular, polar orbit around the moon with an altitude of 300 km (186 mi);

- Maintain the mapping orbit at 300 km \pm 10 km (186 mi \pm 6.2 mi) for one year; and
- Provide propulsion for attitude control purposes.

7.1 Mono-Propellant

A mono-propellant or hydrazine system was selected upon completion of a trade study including solid and bi-propellants. A mono-propellant system will be able to accomplish all aspects of the mission with one system. A single pipe delivery system is all that will be required because hydrazine does not need an oxidizer to burn.

7.2 Thrusters

Three different types of burns were needed to satisfy mission requirements. The first type will perform a lunar orbit insertion with a change in velocity of 854 m/s (2801 ft/s). A single 444.8 N (100 lbf) thruster will be used for the insertion burn. Two burns will be performed to insert the spacecraft into a 300 Km (186 mi) orbit. The second type will require the hardware to correct for orbital deterioration or loss of altitude. Four 4.45 N (1 lbf) thrusters will be used to accomplish orbital correction. The third type of burn will be for attitude control or pointing of the spacecraft and desaturation of the momentum wheel (see Section 6.0). Eight 0.89 N (0.2 lbf) thruster plus the previous 4.45 N (1 lbf) thruster, will work together to accomplish this task.

7.3 Tanks

Storage and pressurization of the hydrazine will occur in two titanium tanks that will each be 56.1 cm (22.1 in) in diameter. The tanks will be divided into two parts by a bladder. One side of the bladder will have pressurant in the form of gaseous nitrogen, and the other side will contain hydrazine. The tank pressure will be 2.9 Mpa (420 psi) at launch and blow down to about 861 Kpa (125 psi) at the end of mission life. The titanium tanks were chosen because they are off-the-self components that are slightly oversized for mission parameters.

8.0 POWER

8.1 Functional Requirements

The power subsystem must provide and regulate the power levels of each subsystem. The orbit average power used by the satellite is 268 watts. This is the average power required if an orbital sustenance maneuver (OSM) is not required. If an OSM is required, the orbit power jumps to 378 watts (the orbit peak power). A twenty percent power availability margin is needed for safety reasons. This means that in addition to the 268 watts average power, an

additional 54 watts need to be provided. The safety margin also requires 76 watts of power to be added to the orbit peak power requirement which happens during an OSM. Therefore, there must be 454 watts available for use during any orbit. The orbit average power, including reserves and battery charging, is 309 watts.

8.2 Solar Cells

The efficiency of gallium arsenide is approximately six percent greater than silicon, making it more suitable for use on the limited surface area of the Copernicus spacecraft¹⁸. Another advantage of gallium arsenide over silicon is the ability of gallium arsenide to operate with more efficiency at higher temperatures¹⁹. They are also more resistant to radiation. The initial design included solar panels on the skin of the craft as well as panels located on two deployable arms extending from the body. After further calculations, the deployable arms were eliminated and solar cells were placed on the ends of Copernicus. With this final configuration, there will still be enough power to meet mission requirements.

8.3 Power Storage

Battery requirements needed to meet the demand of 4,000 cycles (charging and discharging) during the life of the mission. For a single battery, this is a difficult requirement to meet. However, using two battery packs cuts the cycle requirement to 2,000 cycles per battery and is much more reasonable. Having two batteries increases the weight; however, it reduces cycling as discussed previously and adds reliability to the spacecraft. Initially, it was thought that nickel-cadmium batteries could not provide power for this many cycles, and nickel-hydrogen batteries seemed to be the answer. Nickel-hydrogen batteries can cycle up to 4,000 times with a 40% depth of discharge and can be custom made to deliver almost any voltage and current requirements²⁰. Further investigation into nickel-cadmium batteries yielded different information than found initially; they were reimplemented into the design for reasons of reliability, cost, and weight.

The nickel-cadmium batteries used in the Copernicus are capable of 2,000 cycles at a 40% depth of discharge, 10 amperes of current, and 35 volts end of life voltage. The weight is approximately seventeen kilograms per battery, costing \$160,000 each²¹. Two battery packs consisting of 14 cells each will provide power storage for Copernicus.

9.0 THERMAL MANAGEMENT

Upon completion of a SINDA thermal analysis, it was determined that the batteries will be the only component that

will get too hot. A radiator is the simplest means of cooling components. An aluminum rod was chosen for the radiator because of its low density and high thermal conductivity. The section of the rod that protrudes from the satellite will be coated with silvered teflon which has a low absorptivity to emissivity ratio.

Another critical area of concern was the hydrazine fuel. The fuel freezes at 271 K. Therefore, heaters will be needed to keep the hydrazine tanks warm. If the tanks are kept relatively warm, the route from the tanks to the thrusters, if adequately insulated, should not be a problem. Results indicated that a 9 watt immersion heater in each hydrazine tank should keep the tanks adequately warm. However, the heaters will not have the ability to heat an abundant amount of fuel in a short period of time. The heaters may even need to be on constantly to guarantee that the temperature of the hydrazine does not drop significantly.

10.0 CONCLUSIONS

10.1 Design Approach

The small satellite philosophy has been stressed throughout the design of Copernicus. A key element of this philosophy is the use of "off-the-shelf" components wherever possible. This approach has many advantages in terms of the cost and complexity of the design. However, one aspect of using off-the-shelf components that has not received attention in this design is component interfacing. Off-the-shelf components in various systems rarely "speak the same language" in terms of power, data rates, etc., and can have a significant impact on the development of any small satellite.

10.2 Launch Vehicle Selection and Mass Reserves

Demonstrating that a lunar mapping mission is possible within the limitations of the Conestoga is a critical outcome of this design effort. As mentioned in the introduction, launch on the Conestoga will result in a substantial cost savings over flying on a Delta II. While it is difficult to accurately estimate launch costs, rough numbers suggest a \$15 million savings. Staying within the volume and mass limitations imposed by this choice of launch vehicle was a constant design challenge. At the conclusion of the preliminary design, a mass reserve of 20% was imposed on the design to ensure that, even with uncertainties in the mass estimation of each subsystem, the Conestoga would still be possible. This reduced the mass limit from 410 kg (900 lb) to 327 kg (720 lb). However, late in the design it became necessary to choose an off-the-shelf launch vehicle interface due to the lack of time for a custom design. A McDonnell Douglas PAF 6306 interface was selected. This interface,

capable of holding a 2600 kg (5512 lb) spacecraft, is far beyond the needs of Copernicus. The PAF 6306 has a mass of 50 kg (110 lb), which eliminated a significant portion of the 20% mass reserve. With a final launch weight of 393 kg (865 lb) Copernicus retains a 4% margin for launch of the Conestoga. However this margin would increase substantially if a lighter, custom interface were designed.

10.3 Power Reserves

As with mass, a 20% reserve was placed on the power design. However, after the resolution requirement was relaxed, the power reserve grew to 30%, even with the elimination of the deployable solar panels.

10.4 Redundancy

The limited use of redundancy was consistent with the low-cost approach to the Copernicus design. In fact, the transponder and bolt cutters on the two deployable shields are the only redundant systems aboard the Copernicus spacecraft. These redundancies eliminate two single-point failures. They have a low mass and relatively inexpensive and, therefore, do not significantly increase the cost or complexity of the design. The lack of redundancy dictates a meticulous fabrication and test program to insure against failure.

10.5 Mission Costs

Two cost models were generated for the Copernicus design. One was a component level cost assessment performed by each subsystem. This hardware model did not include estimates for development, fabrication or assembly. It was based simply on cost estimates provided by the manufacturer of each component. In order to get a better estimate of the overall cost, including development, a test and evaluation version of the Satellite Cost Model developed by The Aerospace Corporation was employed²². This model uses empirical relationships derived from NASA and Department of Defense (DOD) spacecraft that were developed over a fifteen year period ranging from 1963 to 1978. This software allows the user to estimate the research, development, testing, and fabrication costs for all components on the spacecraft. A summary of the software output is shown in Table 10.1.

It should be noted that nonrecurring costs comprise \$62.1 million of the \$94.9 million spacecraft cost. These results agree well with estimates from NASA's Office of Exploration and with the hardware estimate that was performed. However, because this model was generated

from data on larger spacecraft it is likely that the estimate is high.

SPACECRAFT	\$94.9
LAUNCH	\$30.0
TOTAL	\$124.9

Table 10.1 Mission Cost in Millions of 1992 Dollars.

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