DEEP SPACE MISSIONS FOR SMALL SATELLITES

Paula Butler and Alan Stein

Space Explorations
P.O. Box 77005, Station E
San Francisco, CA 94107

ABSTRACT

Small satellites, with masses well under 100 kg, can perform useful deep space science missions for costs similar to those of low earth orbit and geosynchronous smallsats. Space Explorations has identified several interesting missions for smallsats, with the objective of developing "turnkey" space-science packages which would have a cost to the customer of $10,000,000 and be operable by one or a few people.

We have analyzed one mission, a Lunar Polar Photographic Orbiter, in great detail over the last year. We conclude that it is commercially and scientifically feasible to produce a small lunar satellite which is capable of doing extensive, high-resolution photography of the moon. The LPPO would photograph those portions of the moon which have never been observed, at resolutions of 30 m, and it would remap the entire far southern quadrant of the moon at 100 times the resolution of existing maps.

INTRODUCTION

The purpose of Space Explorations (SpacEx) is to develop and provide scientific and exploratory space missions. What makes this an unusual product is that we intend our market to be non-government and non-Fortune 100 entities, especially including wealthy individuals.

This implies a low total mission cost, preferably under $10,000,000 to the customer, including all associated hardware, software and launch costs. (Obviously, our costs must be even less, if the company is to show a profit.) It also implies highly automated "turn-key" systems.

Governmental space science requires dozens, if not hundreds, of people to support each mission. For individuals to undertake such ventures, the number of personnel supporting it should be no greater than for any high-end 'hobby'.

We have analyzed the feasibility of one such scientific mission in detail -- a lunar polar photographic orbiter. The cost of the mission can be broken down into three major parts-- the satellite itself, the launch costs and the ground receiving station. In this paper, we discuss the satellite and launch costs.

We emphasize that this is a "proof of principle" exercise. The mission and designs we have produced are realistic, but are unlikely to be the final product specifications. For example, we suspect the Orbiter's power requirements are now low enough to let us do away with solar panels, which would reduce the weight and complexity of the spacecraft. This is not reflected in the current design.

The design approach we've taken is mostly one of brute force, out of a sense of conservatism. The final product will certainly be more competently designed than in our studies. We think our analysis is robust; don't take it as gospel.

THE LUNAR POLAR PHOTOGRAPHIC ORBITER MISSION

We selected a lunar photographic orbiter as our first product because the moon is both nearby and interesting. A satellite can enter lunar orbit in a matter of days; the product is more attractive to a customer if they can take 'immediate delivery'. There are frequent launch windows, and the launch requirements are modest.

Furthermore, the moon is still scientifically interesting. Portions of the moon near the southern pole have never been photographed nor mapped. The only photos and maps for much of the southern farside quandrant have very low resolution (as poor as 25 kilometers). This includes some of the most geologically interesting terrain, such as Vallis Planck-- the largest rift valley on the moon.

Exploration is more than mapping. The original Lunar Orbiters photographed earthrise over the lunar horizon and a view across the crater of Copernicus which was called "The Picture of the Century". It's worth remembering that those 'scenic' photos were an accidental side-effect of the design of the Orbiter; very few such pictures were made. The moon still affords as much opportunity for startling and exciting photography as the American West did in the mid-19th century.

The Orbiter will have an operating life of at least one year in a 500-km high polar orbit. It will be equipped with four full-color CCD cameras, capable of producing monitor-quality pictures (approx. 550 by 700 pels). Its 40 mm and 300 mm lenses will provide both wide-field images, showing surface features as small as 200 m, and high-resolution images down to less than 30 m.

The spacecraft is capable of returning up to 70,000 images to earth over its operating life. The actual number depends upon the shooting schedule of the customer. More than enough images will be returned to remap the far southern quadrant at 200-m resolution (a hundred times better than existing maps) and to map the previously-unphotographed southern-polar regions at 30-m resolution.

The spacecraft will be three-axis-controlled; the cameras could be directed to take full-frame photos of the earth, if the owners so desire; the precise pictures they choose to make are up to them. Similarly, the spacecraft could be ordered to reorient itelf so that it photographed panoramic sweeps of the moon.

The pictures will be transmitted by the satellite to the owner's own ground station. A spacecraft with a 1 watt transmitter and a 1/2 meter dish will be able to send over 100 kilobits/sec to a standard 4 meter TV-satellite style receiving dish on earth-- enough data for one image in 20 seconds. Transmissions to and from the Orbiter will be encoded to protect the integrity of the vehicle and the customer's investment.

THE LUNAR ORBITER SPACECRAFT

The most important factor for selecting Orbiter components is weight; it costs nearly $100,000/kilo to send a payload to the moon. Conventional satellite design aims for maximum reliability and function for the minimum weight. This usually produces satellites which cost far more than our total $10,000,0000 price.

We employ a mix of components: where reasonable, we use components which have flight history; this saves money, by reducing the extensive (and expensive!) space-qualifying of new hardware designs. In some subsystems, such as attitude control and power, we use standard spaceworthy aerospace components. In others, such as imaging, we will use off-the-shelf components, subject to our own testing. In a few cases, such as computers, where we have exceptional in-house expertise, we will custom-design and assemble our own hardware.

We prefer redundant systems, especially in the early missions. That is why, for example, our design has four camera/lens systems. Two cameras would do the job; possibly even one, with a mechanism to switch between two lenses. Such a design, while 'cost-effective', would leave many possible single-point failures. We are not performing an experiment; we are selling a product whose performance we must be able to warrant.

We have assigned estimated weights and prices to the various components of the Orbiter. This, of course, will all be subject to change in the actual design. The data is summarized in Table 1. These prices do not include the cost of assembling the satellite from the subsystems nor the cost of certifying the space-worthiness of the craft. The dry weight does not include the fuel for the ACS system; that is covered in the "Launch and Propulsion " section.

ORBITER SUBSYSTEMS

Propulsion/Attitude Control System

The monopropellant hydrazine propulsion/attitude control system consists of an injection motor, twelve thrusters, a hydrazine tank, pressure transducer, filter, three latch valves, and manifolds and fittings to connect the system. The total dry mass of the system is 10.6 kg, and it will cost $555,000.

The injection motor is used for major midcourse corrections in the translunar trajectory and for braking to achieve lunar orbit. A Rocket Research MR-107 series engine with propellant control valve weighs 0.9 kg and costs approximately $30,000. At an inlet pressure of 450 psia it produces 180 N (40 lbf) of thrust. At 100 psia inlet pressure it produces 50 N (11 lbf).

The thrusters are Hamilton Standard REA-10 with chamber heaters and temperature sensors, weighing 0.29 kg and costing approximately $30,000. At an inlet pressure of 100 psia each delivers 0.40 N (0.09 lbf). In lunar orbit the inlet pressure will be even less, and the thrust will be proportionately smaller.

REA-10 series thrusters have demonstrated over 45,000 cold starts and 357,000 hot starts. We anticipate approximately 50,000 starts during the one-year mission, but about 90% will be minimum-impulse-bit firings too small to produce significant pressure spikes. These firings do not count as cold starts. Heating the chambers reduces the number of cold starts and makes firings more consistent.

A Pressure Systems #80276 tank supplies 27.7 kg of hydrazine to the injection motor and the thrusters (see Table 2 for useage). As fuel is used, the pressurizing nitrogen expands, and the tank pressure drops. This blow-down mode enhances attitude control precision in lunar orbit by reducing the inlet pressure to the thrusters.

The major use of hydrazine in lunar orbit is to orient the spacecraft to take photographs and then return the spacecraft to its nominal attitude. Imparting 1 RPM of rotation about one axis requires about 0.5 g of hydrazine, plus another 0.5 g to stop the rotation.

The worst-case reorientation, 180° at 1 RPM about any two axes, requires 2 gm of hydrazine to orient for the photograph, plus another 2 gm of hydrazine to return to the nominal attitude. Most reorientations

Table 1

WEIGHT AND COST BREAKDOWN FOR POLAR ORBITER

Subsystem / Component	Comp. Weight (kg)	Subsyst. Weight	Comp. Cost ($1000)	Subsyst. Cost
Propulsion/ACS				
1 50-lbf Motor	0.8		$30	
12 Small Thrusters	3.5		$360	
1 30 kg Hydrazine Tank	4.3		$50	
3 Latch Valves	0.6		$75	
1 Pressure Transducer	0.2		$15	
1 Filter	0.2		$15	
Misc. Hardware	1.0		$10	
		10.6		$555 K
Guidance System				
Gas Gyros	0.5		$20	
Star Sensors / CPU	1.0		$15	
		1.5		$35 K
Power System				
Batteries & Conditioning Cir.	7.5		$40	
Solar Array	1.2		$61	
Array Orienting Hdw.	3.0		$25	
		11.7		$126 K
Communications				
Dual Transmitters	1.0		$20	
RF Processing/Oscill.	1.0		$25	
Beacon/Omni-antennae	1.0		$20	
High-Gain Antenna & Pointing Motor	1.0		$40	
		4.0		$105 K
Image Handling & Computers				
4 Camera Systems	5.0		$10	
Image Storage	0.5		$10	
Data Compression & Encoding CPU's	1.0		$25	
Main Bus CPU	1.0		$25	
		7.5		$70 K
Thermal regulation		3.0		$50 K
Carbon Composite Structure and Harnesses		10.0		$50 K
DRY SPACECRAFT TOTALS:------		48.3 kg.		$991 K

will not be worst-case ones. For smaller rotations, or when time permits, we can reorient at less than 1 RPM and use less fuel. Fuel usage can also be minimized by taking a series of photographs (or pairs of photographs, since normally the Orbiter will shoot wide-field and narrow-field images of each target) before returning to the nominal attitude. We estimate the typical hydrazine usage per photographic orientation will be about half the worst-case estimate.

Four and one-half kilograms of hydrazine will easily suffice to take at least 2500 pairs of photographs during a one-year mission. (Any leftover hydrazine from the translunar trajectory corrections will be applied to attitude control.) The number of photographs can be increased many times with suitable planning to economize on fuel, such as making sequences of photographs along the orbital path.

The Orbiter's attitude is controlled about all three axes, but the control accuracy demanded by this mission is much lower, on the average, than that of conventional 3-axis stabilized spacecraft. There are three operating regimes:

The least critical is orientation of the solar array when the spacecraft is neither taking pictures nor communicating with earth. This is a low precision situation -- drifts of 20° have small effects on power collection.

The next most demanding activities are other non-photographic activities. Communication with earth via the high-gain antenna requires a pointing accuracy of ±2.5°, typically for 30-60 minutes. Star sightings to determine attitude require a rotation rate of less than 0.25°/sec for less than a second.

Attitude control must be most precise for photography. At the moment of picture-taking, the orientation of the Orbiter must be known to 0.1°, because of the narrow field of view of the high-resolution cameras. In order to limit image blur to under 30 m, the spacecraft's rotation rate must be under 0.2°/sec, or one rotation in 35 minutes.

Attitude control precision is governed by the minimum impulse bit of the thrusters. REA-10 thrusters have a minimum impulse bit of 0.001 lbf-sec (0.004 N-sec) at 100 psia inlet pressure for 7 msec on. The spacecraft uses a pair of thrusters for a manuever, but the inlet pressure will be about half as much (50 to 75 psia). At half the inlet pressure about half as much fuel will be delivered during 7 msec. The pair of thrusters will torque the spacecraft with a minimum impulse of about 0.004 N-sec, changing the spacecraft rotation rate by 0.03°/sec, or 0.3 rotation/hour per axis. The summed rotation about all three axes could be 0.05°/sec, limiting ground resolution to 8 m.

A minimum controllable rotation rate of 0.05°/sec means that thrusters must be fired every minute or so to keep the high-gain antenna pointing at earth (to ±2.5°) during transmission. This will occupy as much as 100,000 minutes over a year. It will consume up to 0.5 kg of hydrazine and demand perhaps 30,000 minimum-impulse firings of each thruster over the mission lifetime.

Solar radiation pressure on the high-gain dish also rotates the spacecraft. (The solar panels are arranged symmetrically, to avoid this.) Torque from the antenna could be balanced several ways; however, even without such compensation, the torque from the high-gain antenna is not a serious problem for attitude control. Radiation pressure on a mesh antenna produces a maximum possible angular rate of 0.018°/sec, which is smaller than the minimum controllable rotation rate.

Guidance

The spacecraft attitude is periodically determined by star sensors. Between sightings, it is computed digitally from rotation rate data from gas gyros.

A typical star sensor uses a Pulnix TM-540R CCD camera head with a Cosmicar 12 mm f/1.4 lens. The TM-540R sensor contains 510 x 525 pixels and provides a 30° x 40° field of view (pixel size is 0.075°). The sensor weighs barely over 100 gm, is about 6 cm long, and costs well under $1000. A 1/4 sec exposure time records stars down to fourth magnitude. The images are pattern-matched to the 600 brightest stars, by a dedicated computer running a matching algorithm to a ROM-based star map. Four stars suffice to determine the pointing vector of the sensor and the orientation of the spacecraft about that vector. A star sight can be taken with a spacecraft rotation rate of up to 0.3°/sec. The spacecraft thrusters are precise enough to reduce the rotation rate to one-tenth of this amount.

Near the galactic plane, there is an average of one star per 35 square degrees of sky, but this drops to one star per 150 square degrees at the galactic poles. On the average, a camera with a 30° x 40° field of view would see eight stars, even at the poles. Because the stars are not evenly distributed, there is a small chance (1-2%) of a sensor seeing blank sky.

Three sensors mounted 120° apart around the spacecraft almost always provide at least two sensors with fields of view unobstructed by the moon, which subtends 102° from an altitude of 500 km. There is a 10% likelihood that one of these two star sensors will be blinded by the sun's glare. Pointing a sensor at the sun does not damage the sensor, but it does prevent it from taking a sighting. Hence, there is a chance of a tenth of 1% or so that neither sensor will get a fix when the Orbiter is in picture-taking orientation. In that rare case, the Orbiter continues to operate on the basis of the attitude computed from rotation rate data since the previous star sight, until spacecraft rotation allows a sight. Alternately, a rotation about any axis by 45° will allow at least one sensor to obtain a fix.

The rotation rate of the spacecraft is measured by Humphrey RT09 two-axis rate transducers ("gas gyros"). Gas gyros work by pumping a stream of helium between two heated wires. The wires are part of a resistance bridge circuit, which is balanced when the transducer is not rotating. When the gyro rotates, more gas flows over the heated wire on the lagging side of the turn, while less flows over the leading wire. The bridge circuit detects the resulting changes in the resistance and produces an output voltage proportional to the magnitude (and direction) of the input rotation rate. The RT09 has two pairs of heated wires and measures rotation rates about two perpendicular axes.

The RT09 is inherently rugged and long-lived. The helium is pumped by a bimorphic crystal driven by an oscillator circuit. The eventual failure mode is the slow leakage of helium; rated service life is 10,000 hours, but failure typically takes much longer. Shelf life is at least 5 years. The gyros are rated for 100 g's of shock and cannot be tumbled or damaged by high rates of turn. The bare RT09 measures 2.5 x 2.5 x 5.1 cm, weighs about 70 grams, and costs about $1700. Prepackaged in a case with electronics, it is 6-7 cm on a side, weighs about 0.3 kg (most of which is the case), and costs $2700. Two of these can be arranged to provide measurements about all three axes. A spare set of gas gyros will be carried as backup.

The transducer response is linear with a deviation of no more than 1% of full scale. Since the deviation depends primarily on the gas transit time, the deviation is generally less as the input rotation rate decreases. The full scale rotation rate is determined by the gas velocity, which is controlled by the power supply voltage of the oscillator circuit. The small amount of null drift with time and temperature is minimized by compensations in the electronics. The minimum threshold or resolution is dictated by thermal and electronic noise. The threshold resolution with the standard supplied electronics is 0.005°/sec. For some reasonable improvements in the electronics, we can reduce the threshold by a factor of two to 0.0025°/sec.

The most demanding operation is orienting the Orbiter for photography, which can take up to a minute to execute. During that maneuver, the linear error could amount to as much as 1.8°, while the noiselike error would accumulate to perhaps 0.1°. This leaves a possible pointing error of nearly 2°, so it will be necessary to take a second star sighting after reorienting the Orbiter for photography. If the pointing error is large enough to demand a second orienting maneuver, that one can be executed to much better than 0.1° total error in the final orientation.

Power

Power requirements in lunar orbit fall into three categories: housekeeping functions, communication with earth, and sunlit operations.

Housekeeping functions demand a maximum of 31-38W, plus 80 W for momentary use. Vital functions use 26 W: 10 W for the star sensors and their dedicated computer, 5 W for the gas gyroscopes and their electronics, 1 W for the radio beacon, 5 W for the main central processing unit (CPU), and 5 W for subsystem CPU's. Optional attitude control thruster heating draws 5-12 W. The thruster control valves draw a total of up to 80 W during firings of a few seconds at most. The maximum energy usage per 158-minute orbit is 82-100 W-hr for housekeeping functions.

Down-link communication with earth requires 8 W, of which 2 W are for the data encryption, 5 W are for the RF transmitter, and 1 W is for the antenna motor. We allow for 1 hour of transmission to earth per orbit, for a total of 8 W-hr per orbit.

The amount of time per orbit that the spacecraft spends in the moon's shadow varies from none to 44 minutes in a 500 km-high circular orbit. For 72% of the year, the orbital plane is such that the spacecraft never enters shadow. (A 250 km-high orbit would have a 133-minute period, of which a maximum of 45 minutes would be in eclipse.)

The maximum sustained power usage in shadow is 48 W, allowing for harness and power conditioning circuitry losses. (When the thruster heaters are not used, power consumption is no more than 35 W.) It requires 35 W-hr of battery capacity to supply 48 W during the worst-case 44 minutes of eclipse. Using a conservative 50% maximum discharge level, the Orbiter needs 70 W-hr or 56 A-hr (at 1.25V) of capacity. Various combinations of Gates space-qualified Ni-Cd cells, weighing about 2.5 kg and costing about $25,000, can furnish this capacity. An additional half kilogram of hardware is needed to contain the cells.

About 45 W-hr are required to fully recharge the battery after it has been discharged by 50%. With at least 114 minutes in sunlight, the maximum average power needed to recharge the battery is 24 W. In order to minimize system load, battery charging and photography do not take place at the same time. Depending on the photography schedule, the recharging power delivered to the batteries will be between 24 and 45 W (a charge rate of 0.35 to 0.7 C).

When the Orbiter is in sunlight, 29-41 W may be needed beyond the aforementioned demands. Photography will almost always occur when the spacecraft is in sunlight, because only then is the visible lunar surface illuminated. Photography requires 27-39 W: the CCD cameras draw 12-24 W, and the imaging computers draw 15 W. We allow for 50 minutes of photography per orbit, which requires 22-32 W-hr per orbit. The solar panel orienting motors require 2 W. A total of 26-38 W-hr per orbit may be used for sunlit functions.

The maximum peak sunlight demand is 60-79 W for housekeeping functions, photography, and repositioning the solar panels. An additional 80 W may be momentarily needed for thruster firings. Battery charging does not occur at times of peak demand. Communication via the high-gain antenna does not occur during photography, because the spacecraft pointing requirements conflict.

The maximum average sunlight load is 68 W for housekeeping functions, 50 minutes of photography, solar panel repositioning, communication to earth, and battery charging. Battery charging may use 19 W-hr to replace up to 15 W-hr drained during 44 minutes of transmission to earth during eclipse and up to 11 W of power deficit during the photography. Allowing for harness and power conditioning losses, the solar cells must supply a maximum average of 70 W.

Sovonics Solar Systems makes photovoltaic solar arrays based on thin-film amorphous silicon alloy for aerospace uses. The Sovonics UL-200 is a 200 W, 3.24 m^2 array blanket weighing 800 gm. The deployment and support hardware weighs 500 gm. The amorphous cells are subject to 15% degradation in the first few months due to sunlight, but the cells are more tolerant of ionizing radiation than crystalline ones. The many interconnections in the Sovonics array help it tolerate punctures from meteorites.

The array blanket supplies 60 W/m^2 at 35°C. After allowing for 20% degradation in one year, array misalignment of up to 20° from normal incidence, and variations in the solar flux with distance of the spacecraft from the sun, 1.62 m^2 are required to guarantee 70 W.

The Orbiter's solar array is divided into symmetric panels to balance solar torques. Each panel will contain 0.81 m^2 of array blanket weighing 0.2 kg and a deployment/support mechanism weighing 0.3 kg. The two panels for the Orbiter will cost $60,000.

The Orbiter will have some solar cells on the body of the spacecraft to generate power before the array panels are deployed. The spacecraft can operate on less than 34 W, until it reaches the moon. Conventional black solar cells produce 210 W/m^2, so 0.16 m^2 could power the spacecraft with normally incident sunlight, or about 0.25 m^2 could power it at up to 45° incidence. A quarter square-meter of conventional black cells weighs about 0.2 kg. The spacecraft can operate on battery power alone for two hours with 50% discharge depth.

We may eliminate solar panels entirely. At normal incidence, 0.40 m^2 of black cells will supply at least 70 W after a year's degradation (13%). This area is less than the cross-sectional area of the Orbiter. If enough cells can be mounted on all sides, the Orbiter's orientation would not be constrained by power generation needs. The mass and cost of the cells would depend on the area actually covered, but in all likelihood would be substantially less than that of the solar panels with their orienting hardware.

Communications

We can estimate economically practical data rates by looking at common TV-satellite technology. This analysis, for the 4-6 Ghz communications band, provides a conservative estimate (as we will be operating at higher frequencies), but one which is based on very well-known performance characteristics for common commercial equipment.

At 6 Ghz, a 0.5-m Orbiter dish antenna will transmit a beam about 6° in diameter. This is several times the apparent size of the earth from lunar orbit. The earth station antenna will have a 4-m receiving dish--a common, inexpensive size for private stations. At 6 Ghz, the satellite antenna has a gain of 28 dB and the ground station antenna a gain of 45 dB.

The Orbiter transmits one watt of power; the ground station receives -145 dBW, or about 0.003 picowatt. By comparison, a typical satellite TV station of similar size receives about 4 picowatts and can receive 200 megabits/sec on a 30 Mhz channel with a carrier-to-noise ratio (C/N) of 18 dB for a typical well-designed receiver.

The Polar Orbiter can send proportionately less data: 150 kilobits/sec (kbps) on a 25 Khz channel at 18 dB separation. Digital data may be transmitted with a much poorer C/N ratio-- as low as 10 dB. Even assuming significant transmission losses due to precipitation, inefficient antennae and so forth, we can conservatively expect a data rate over 100 kbps.

Because the Orbiter will be transmitting only 1 watt, solid state amplifiers (SSAs), using gallium arsenide diodes, are the obvious choice for the microwave power amplifiers. SSAs are only 25% efficient, but at the low powers we will be running, this is no problem for the power or thermal control systems. A TWT amplifier typically weighs three times as much as an SSA, including the power conditioning circuitry (for example, a typical 4-Ghz, 10-watt TWT system weighs 2.5 to 3 kg, while the equivalent SSA weighs a kilogram). The Orbiter will carry duplicate transmitters.

What does 100 kbps mean in practical terms? We expect a typical CCD camera image to take up 2 megabits, after compression, giving us a transmission time of 20 seconds for a single image. In 'preview' mode, which will transmit images at half-resolution, pictures can be sent in less than five seconds. The bandwidth is sufficient for delivering at least three full-quality pictures in a minute, greatly exceeding the Orbiter's ability to process and store images.

Should the spacecraft lose its high-gain link with the earth, the omni-antennae could transmit 500 bps at a C/N of 11 dB. Obviously the earth station will have much more transmitting power available to send commands to the spacecraft. Even in the event of a total loss of onboard programming, the customer would be able to 'boot up' the Orbiter from the ground through the omni-antenna in under an hour.

If the main spacecraft antenna fails, it would take 50-60 minutes to transmit a full quality picture at 500 bps. This is half the shortest time (approximately 110 minutes) the satellite is in view of the receiving station when its orbit is occulted by the moon (which occurs for 30% of the lunar month). Hence, 8 to 12 pictures per day could be transmitted via the omni-antennae. That still permits some 4,000 images to be sent over the one year life of the spacecraft. This is sufficient to fulfill minimum scientific objectives of remapping the rear southern quadrant of the moon at 200-m resolution (a 100-fold improvement over existing maps) and mapping the unphotographed south polar region at 30-m resolution.

In reality, the spacecraft lifetime is most limited by the amount of hydrazine available to the attitude control system. Failure of the high-gain antenna eliminates the need to reorient the Orbiter to send images. The more leisurely transmission rate reduces the importance of being able to rapidly reorient the spacecraft. This reduces ACS fuel consumption to the point where a two-year mission would be possible.

Imaging

The high-resolution imaging system is designed so that resolution will be limited by motion blur due to an orbital velocity of 1.5 km/sec. The blur in the direction of travel is 30 meters for 1/50th sec exposure. Compensating for motion-blur adds enough complexity to the spacecraft that we will not build such a capability into the first mission.

The resolution limit is comparable to the best of the original Lunar Orbiter images and better than the worst of them by a factor of 1,000. In the direction perpendicular to the orbit, resolution will be limited by the optics and the pixel size.

Customers will likely make beneficial use of the spacecraft's motion; One can make precise stereographic pairs of photos simply by taking two pictures separated in time by a few seconds. Similarly, to map out a swath of terrain at low resolution, one records pictures at fifty-second intervals, without reorienting the spacecraft.

CCD cameras remain stable over long periods of time and are rugged, lightweight and easily space-qualifiable. While a particular CCD camera/lens combination will distort the picture slightly, displacing features of the image slightly from their true positions, this displacement is constant. Once it is measured for a given camera/lens assembly, any pictures sent by that assembly can be geometrically corrected to within one pixel.

This is invaluable for mapping and for doing 'geology'. The original Lunar Orbiter had very high resolution, but poor accuracy; maps made from those images have the correct features, but they are often in the wrong place by many kilometers. Maps made from the Polar Orbiter images will be accurate enough to allow researchers to do rigorous scientific work.

One high quality CCD camera, the Cohu 1710, weighs about 0.6 Kg, is 6 x 7 x 20 cm, and costs $1650. This camera is rated for 30 g's acceleration, and operates from -10C to 50C while consuming 6W of 12V DC power. The Cohu 1710 is equipped with a 406 kilopel CCD array, with an imaging area of 4.8 by 6.4 mm. The array is configured as 699 color-filtered strips (alternating red, green, and blue), each stripe being 580 pels high. The pels are 9.2 microns wide by 8.4 microns high; hence, each tri-color pixel is 28 microns wide. The camera operates at a 50 Hz frame rate.

Since individual color pels do not optically overlap, a tri-color pixel contains information from three adjacent regions of the moon. Because of this, color and spatial information can be confused:

Consider a slate grey plain, with a small boulder in it. The sun is low in the sky, so the plain is moderately lit with sunlight. One side of the boulder is in shadow; the other is brightly lit by the sun. If this boulder is about one pixel wide, each tri-color pel in the pixel will see a different part of it. The blue pel may see the shaded side (black), the green one the sunlit side (white) and the red one the plain (grey). If this is interpreted as color information, one concludes that there is a lime-green spot on the moon.

There are ways to untangle the data. The moon does not show extreme variations in color over small regions; the pixel just described is not plausibly a lime-green patch of lunar soil, "green cheese" notwithstanding. Duplicate pictures, taken with the other narrow-field camera will show similar brightness variations, but entirely different color variations, unless the cameras are aligned to each other with subpixel precision. Low-resolution pictures of the same region will give an accurate record of the appropriate hues for the high-resolution pixels, allowing one to extract the high-resolution brightness information from the apparent color value.

Depending upon how the data is interpreted, a spatial pixel may be thought of as being either 9 or 28 microns wide. This increases the useable resolution of the system over what it would be if every pixel were (inappropriately) treated as purely a three-color point.

A single pixel is 8 microns high and 9-28 microns wide. From 500 km, the camera lens must have at least a 300 mm focal length, in order to project 30 meter ground resolution at one "average" pixel-width (18 microns). Depending on data interpretation and the direction of the motion blur with repect to the CCD array, a 300 mm lens will produce an in-camera resolution of between 15 and 45 meters. In order to achieve the highest resolution figure, the lens must resolve at least 100 lp/mm over an imaging area of 4.8 by 6.4 mm, at infinity focus.

For optimum quality, the light level on the CCD should be over 20 lux. The moon receives 50,000 lux at 'noon' and about 10,000 lux when the sun is 10° above the horizon; an f/10 lens will be adequate.

A wide variety of ordinary 35mm camera lenses will meet or exceed these optical requirements. For example, a 300 mm f/4 Pentax lens sells for about $350, weighs 0.7 kg, and measures 13 cm by 9 cm. It produces a field of 0.9° by 1.2°, corresponding to a lunar area of about 8 by 11 km, looking straight down. Such a lens would also be able to take dramatic pictures of the earth, which subtends nearly 2° at lunar distance; earthly features as small as 11 km could be detected.

This lens has a substantially larger aperture than we need, and it covers a field of 24 mm by 36 mm. Optically, it is overkill; it illustrates that we can find optics which substantially exceed our specs. Typical lenses with smaller apertures and imaging areas will weigh less.

The lenses will be in fixed mounts, set for infinity, so thermal expansion and contraction will shift the focus of the lens. The precise magnitude of the shift can only be determined by testing the lens/camera combination, but we can estimate it from the difference in thermal expansion coefficients of the lens barrel and the elements. That difference is 15 ppm/°C for glass vs aluminum, implying that the 300 mm Pentax lens (13 cm long) would shift focus by 2 microns/°C.

Other components of the Orbiter, such as the batteries, should be maintained within a 20 °C temperature range. Over that, the 300 mm lens would shift by ±20 microns. At a working aperture of f/10, that produces image blurs of 2 microns, which is much smaller than a pel.

Two of the CCD cameras will have 'short' lenses with a focal length of about 40 mm and a field of view of about 11° (as measured along the diagonal). A Pentax 40mm f/2.8 lens weighs 0.1 kg, is 6 cm in diameter by 2 in length, and sells for under $50. Although these lenses will only be able to image 200 meter objects, they will permit mapping large areas of about 5,000 sq.km in a single image.

This is important, as the moon is small only in comparison to the earth; it has an area of almost 40,000,000 km^2. Several million km^2

have been mapped only at very low (20 km) resolutions, and some 100,000 km^2 have never been photographed. It will take 1,000 high-resolution photos to make a detailed record of the unphotographed terrain (actually, many more than 1,000, since one needs different lighting angles to bring out different features), a few dozen low-resolution photos will map it out at 200 meter resolution.

The large-area photos also have important pictorial value; one can easily miss the grandeur of the lunar landscape by taking only narrow-field shots. Most major features are much bigger than 10 km on a side; Vallis Planck is many hundreds of km long. Large-area photos will be important for maintaining perspective both literally and figuratively.

Altogether, we expect our four-camera system to weigh under 5 kg, and cost about $10,000. This includes the cost of modifying the lenses for use in space-- removing all lubricants and degreasing the lens thoroughly, and welding or cementing all moving parts, screw heads and retaining rings. It does not include environmental testing costs.

The cameras will be fixed to the Orbiter chassis, and the entire spacecraft will be reoriented to point the cameras at the desired target. The length of the high-resolution lens and camera combination is almost half the expected diameter of the spacecraft. The spacecraft would have to be considerably larger to provide enough space to swing such bulky assemblages around.

Many of the optical requirements can be relaxed considerably if we decide upon an orbit lower than 500 km. For example, in a 250 km orbit, we could use a 200 mm lens for the high resolution pictures. A Pentax 200 mm f/4 lens sells for under $200, weighs only 0.4 kg, and measures about 6 by 11 cm. This, of course, would open up new possibilities for both the cameras and camera-pointing systems.

Computers

By far, the biggest chunk of computing power is dedicated to image processing and compression.

Each pel requires 10 bits of integer data; and is allotted a 2-byte (16 bit) integer before processing. If straighforward compression algorithms are used, the computing overhead per image can be kept relatively low:

Double-delta encoding demands little more than two subtractions per pel (with some checks for discontinuities)-- what the Orbiter stores and eventually sends to earth are the second-order differences between adjacent pels, rather than their absolute magnitudes. Run-length encoding compresses strings of identical bytes down to tokens, which are expanded in reconstruction. Most of its computational overhead comes from running comparisons and loops.

So long as the image being compressed doesn't have frequent sharp changes in brightness from pel to pel (lunar images fit this criterion), these two schemes, applied along the X and Y axes, can produce a substantial reduction in image size, without losing any data. A conservative estimate is that compressing an image by a combination of these methods will reduce the file size by a factor of two to three and will demand 10 16-bit integer operations and 100 machine instructions per pel. Compressing a single image may consume as much as 4 million integer operations and 40 million instructions, to produce a 0.15 to 0.25 Mbyte file.

(There are algorithms which can compress video data 100-fold, by sacrificing information. These schemes are unsuitable for a research mission, which cannot determine in advance what is extraneous.)

Our performance goal is to process sequential wide-field images as fast as they come in, in order to map out continuous strips of the moon. That means an image must be compressed in slightly under one minute, which demands a throughput of 70,000 integer operations/sec and 700,000 machine instructions/sec (0.7 MIPS). This is modest throughput for today's systems; a high-speed 8086 runs at several tenths of a MIP. Three Harris radiation-hardened 80C86's, one each for the red, green and blue channels, can handle the job (with some coprocessor support for the integer arithmetic), with a total power consumption of well under 1 watt. The image processing system will weigh considerably less than a kg. The total weight (excluding mass storage) with frame grabbers, transmitter ports and similar 'glue' included is about 1 kg.

Static RAM will reduce the overhead of supporting the memory by eliminating refresh and reducing the error rate. Rad-hard components are not required; the system can treat radiation-induced soft errors (which are relatively infrequent) as it would any other parity error. The Orbiter will include radiation-sensing circuits which will shut down most computer functions if there is a serious solar flare; this will keep hard errors low enough to be manageable.

Our goal is to store 50 images before having to download data to earth. One Mbyte of static RAM will consume something under one watt of power and store five compressed images. At $10- $15 per chip, 1 Mbyte currently costs about $400, not counting mounting and assembly. We can minimize weight by using direct surface-mount boards; one Mbyte of assembled RAM can weigh as little as 20 grams. SIMM packages would weigh 50 grams or so per Mbyte. Including scratch and temporary storage, memory sufficient to store 50 images will contain 12 Mbytes of RAM, weigh about 0.5 kg, and consume 12 W.

In total, the image handling computers will weigh about 1.5 kg and cost $35,000. Power consumption will be under 15 watts.

Attitude determination is the second-largest task. The star sensor computer grabs a frame from each of the star sensor CCD cameras and selects one with at least four stars. The CPU rectifies the positions of any stars in the CCD image, to correct for optical distortions. This is done by a lookup table. The CPU then runs a pattern-matching routine to fit the star positions from the CCD image to an internal map of the positions of the 600 brightest stars. Four stars are sufficient to make an unambiguous match.

The matching routine, a straightforward partition-and-discard algorithm, runs in about 0.5 Mbytes of static RAM. It takes approximately one million instructions, worst case, to complete the algorithm. For a positional update every few seconds, a single IBM XT-equivalent computer is sufficient. The computer will weigh substantially less than a kilogram and consume a few watts at worst.

Other Orbiter functions have their own computers; of course, there will also be a central computer to monitor the overall status of the spacecraft. These all have very modest capabilities compared to the two subsystems just described. The secondary subsystem CPU's have such low weight and power requirements that we have not bothered to break them out of the total subsystem budget. The weight and power demand of all these other systems is about a kilogram and 5 W.

The Orbiter system architecture is a low-bandwidth bus, with each subsystem passing messages as demanded by the central CPU. Internal computations of the subsystems proceed more or less autonomously, depending on the task. Star sensing, for example, is a background task which goes on regardless of what operations the spacecraft is engaged in. The attitude CPU makes an updated position available to the bus every few seconds. The ACS computer is semi-autonomous. It handles housekeeping tasks, such as hydrazine pressure monitoring, temperature monitoring and control of the thrusters as ongoing background tasks. Attitude corrections, though, are an on-demand task ordered by the central CPU. The imaging system is almost totally demand-driven; when it isn't processing images, only minimal housekeeping is needed to maintain the integrity of the subsystem.

The main CPU will weigh a kilogram. This is actually very heavy, in proportion to its computing power, but reflects the fact that a lot of peripheral and I/O circuitry 'glues' it to the other subsystems. Power consumption will be five watts.

We have specified computing systems weighing a total of approximately 4 kg and consuming 30 watts. For this, we get about 15 Mbytes of memory and a throughput of a few MIPS.

To those familiar with spacecraft computers, this will seem outrageously optimistic. The typical spacecraft computer has more in common with an Apple II, in terms of power, weight and throughput, than it does

Table 2

PROPULSION WEIGHT AND COSTS

DRY SPACECRAFT WEIGHT		48.3 kg.	
HYDRAZINE FOR ACS SYSTEM		4.5 kg.	
TOTAL ORBITER WEIGHT		52.8 kg.	
propellant/dry weight ratio (P/D) for lunar injection hydrazine:	0.4		
HYDRAZINE FOR LUNAR INJECTION		21.1 kg.	TOTAL HYDRAZINE
WET WEIGHT		73.9 kg.	27.7 kg
P/D for hydrazine for midcourse corrections:	0.03		
HYDRAZINE FOR MIDCOURSE CORRECT.		2.1 kg.	
TOTAL WET SPACECRAFT WEIGHT		76.0 kg.	
EMPTY STAR 24 SHELL		16.1 kg.	
TOTAL STAR 24 DRY WEIGHT (w/ wet payload)		92.1 kg.	
P/D for solid fuel for transfer orbit injection :	2.07		
SOLID PROPELLANT WEIGHT		190.6 kg.	
INERT FRACTION WEIGHT		2.1 kg.	
STAR 24 + PAYLOAD WEIGHT		284.8 kg.	
LAUNCH CRADLE WEIGHT		15.0 kg.	
TOTAL LAUNCH WEIGHT		299.8 kg.	
Launch cost ($1000's/kg):	$15.6		
TOTAL LAUNCH COST		$4,677	
Cradle Cost		$20	
Star 24 Cost		$500	

with our system. We are proposing to fly one to two orders of magnitude more power, for less weight and power.

Nonetheless, those familiar with current microcomputer hardware will observe that our numbers are, if anything, conservative. Today's laptop computers provide a substantial fraction of a MIP and a megabyte of RAM, on a board which weighs under a kilogram and consumes less than a watt of power.

We do not claim that such a board is, in itself, flightworthy. We merely note that our projected performance is in line with standard consumer computers. Our three computer experts have designed and built everything from single-board controllers through supercomputers, and all have produced spacecraft systems.

LAUNCH AND PROPULSION

There is a common belief that putting a spacecraft into a lunar polar orbit (LPO) is either difficult or energetically much more expensive than going into an equatorial orbit. Both these beliefs are false.

The delta-v (velocity change) required to send a probe from low earth orbit (LEO) to the vicinity of the moon is slightly over 3 km, regardless of what part of the moon the probe passes over. Any trajectory which takes the Orbiter from LEO over one of the lunar poles can be coverted to an LPO. That demands nothing more than applying enough braking thrust, when one reaches the moon, to convert the flyby trajectory into a closed orbit.

The NASA LUNAR FLIGHT HANDBOOK describes a sample mission which goes from a LEO 250 km above the earth to a circular LPO 185 km above the moon. A transtage adds a delta-v of 3.1 km/sec to the LEO orbital velocity, for a final velocity of 10.9 km/sec, which is slightly greater than escape velocity.

When the spacecraft reaches perilune, 185 km above the moon, it is traveling at 2.4 km/sec relative to the lunar surface. Braking thrust reduces this velocity by a delta-v of 0.8 km/sec, leaving the spacecraft with the proper orbital velocity of 1.6 km/sec.

The precise delta-v's needed for leaving LEO and entering lunar orbit vary slightly depending upon the altitude of the LEO, the altitude of the lunar orbit, and the distance between the earth and moon:

If the LEO is lowered from 250 km to 180 km, the delta-v to enter the translunar trajectory increases from 3.106 km/sec to 3.124 km/sec-- an increase of 0.018 km/sec. As the moon's distance from the earth varies from 356,000 to 407,000 km, the delta-v required for transfer decreases or increases by 0.006 km/sec. In a best-case launch, we would have a transfer delta-v of 3.10 km/sec, while the worst case is 3.13 km/sec.

This variation alters the weight of the transfer stage plus payload by about 1%.

If the LPO is raised from 185 km to 500 km, as we have chosen for our mission, then the delta-v to inject into lunar orbit goes from .781 km/sc to .741 km/sec-- a decrease of .040 km/sec. That changes the total weight of the satellite plus LPO injection motor by less than 2%.

Propulsion Requirements

The mass and propulsion needs for all these orbital manuevers are found in Table 2.

The weight of the Orbiter, including ACS hydrazine, is 52.8 kg. Decelerating the spacecraft into LPO will consume 21.1 kg of hydrazine. The lunar payload will weigh 73.9 kg.

A Morton Thiokol Star 24 solid fuel motor will inject the Orbiter into the transfer trajectory. Nominally, it weighs 218 kg, of which 200 kg are propellant, but Morton Thiokol can adjust the propellant load to match the payload mass and delta-v needed. They can predict the propellant weight to better than 0.5%, and the specific impulse is reproduceable to better than 0.6%. (nominally 282.3 sec). It costs approximately $500,000.

The translunar trajectory will likely require correction, due to two sources of error in the injection delta-v. A possible 1° error in pointing when the Star 24 is fired produces an unwanted transverse velocity component of 0.055 km/sec. The delta-v itself may be in error by as much as 0.029 km/sec, due to the variations in the motor. These two near-orthogonal errors produce a worst-case error of 0.062 km/sec in the delta-v. Corrections may require 2.1 kg of hydrazine.

The total payload of the Star 24 is then 76.0 kg. A propellant load of 190.6 kg will produce the requisite delta-v for our mission. This makes the total weight of the Orbiter, the Star 24 motor, and the launch cradle about 300 kg.

At the present time, the Chinese Long March is the most attractive launch vehicle, although the Soviet Union is starting to negotiate favorable arrangements. Of course, we cannot say what price the Chinese will ask of us, until we negotiate with them. The Swedish Space Corporation contracted for a piggyback launch of their 275 kg Mailstar satellite for $4,300,000. We assume the same cost per kg for our satellite, for lack of a better way to estimate the launch charges.

CONCLUSION

Smallsats need not be limited to near-earth missions. Careful study has convinced us they offer much value for specialized deep space missions.