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TO THE MOON FROM A B-52: ROBOTIC LUNAR EXPLORATION USING THE PEGASUS WINGED ROCKET AND BALLISTIC LUNAR CAPTURE

by

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ABSTRACT

A subset of the presently-defined NASA robotic lunar exploration objectives may be achievable with a new mission architecture involving the Pegasus winged rocket, small satellites, and a new class of Earth-Moon trajectories incorporating ballistic lunar capture. Enabling this potentially low-cost method of lunar exploration — perhaps for a few tens of millions of dollars per mission — is the application of the Weak Stability Boundary Theory developed by Belbruno during 1987-89, which leads to ballistic ("maneuverless") Earth-Moon trajectories. On such a path, a spacecraft could be orbited at the Moon for little additional ΔV (<50 m/s for minor trajectory correction maneuvers) beyond that supplied by the Pegasus for the initial Earth departure burn, resulting in a significant propellant savings. (Additional maneuvers would then be required to establish a more useful lunar orbit.) The price for this savings is an extended trip time to the Moon of 3-5 months. This type of trajectory is presently being demonstrated for the first time by the Japanese *Hiten* spacecraft, using an application developed in 1990 by Belbruno and James K. Miller at JPL; it may also be employed for the Japanese *Lunar-A* penetrator mission in 1996.

If conventional Hohmann-like Earth-Moon transfers are employed, present versions of the Pegasus — even if outfitted with a small fourth stage — can deliver only modest-sized spacecraft to the Moon (<50 kg), most likely not big enough to address presently-defined NASA robotic lunar exploration objectives. In contrast, if the ballistic capture technique is employed in conjunction with four-stage versions of Pegasus, an additional 15 to 30 kg or more of spacecraft mass is gained, resulting in 65-80 kg small satellites which may be able to accomplish some meaningful objectives at the Moon, including gravity field determination, magnetospheric studies, and other related fields, particles and waves objectives. Advertised growth versions of the Pegasus combined with recent developments in small-satellite technology may allow for more capable satellites to reach the Moon, perhaps enabling the achievement of more demanding objectives.

In the current tight budgetary climate, this new mission architecture may allow for incremental achievement of some NASA lunar science objectives by enabling significant enhancements in delivered small lunar satellite mass and capability while at the same time reducing the total mission costs for simple lunar missions. This lower-cost way of reaching the Moon may also provide an avenue for pursuing attractive commercial lunar activities and interesting lunar-based small-satellite constellation concepts.

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Background

This paper presents a new concept for robotic lunar missions that the authors feel should be investigated further. The concept combines Pegasus-launched small satellites with Belbruno's concept of Weak-Stability-Boundary (WSB) trajectories (see Refs. 1-4). Belbruno first suggested this idea to Ridenoure on 1991 April 25 in the context of a discussion on the Japanese *Hiten* spacecraft in particular and on applying WSB theory to lunar and asteroid missions in general. (Belbruno had been thinking about this idea privately and while at Pomona College since late 1990.) To support the development of this paper, Ridenoure performed the analysis of potential mission performance for lunar applications based upon typical ΔV savings gained by using WSB trajectories. Belbruno supplied the WSB trajectory characteristics and performance inputs while Fernandez supplied the Pegasus-related performance inputs.

This paper serves only to introduce the Pegasus/WSB idea, and does not attempt to provide definitive analytical results or specific spacecraft designs. However, desirable attributes of the spacecraft that might be used for this mission type can be inferred from this analysis; these attributes are listed near the end of the paper.

Weak-Stability-Boundary Trajectories

It is well known that a transfer from a low earth orbit on a (hypothetical) parabolic trajectory to infinity followed by another (hypothetical) parabolic trajectory back to the Moon is more energy efficient than a standard Hohmann transfer from the Earth directly to the Moon. At infinity, a (hypothetical) zero maneuver will raise periapsis on the incoming parabolic trajectory to the Moon's orbital radius from the Earth. In the two-body case, the biparabolic transfer is not affected by the Sun. This case is also impractical due to the infinite trip time.

The existence of the weak stability boundary (WSB) of the Earth due to the Sun at approximately 1.5 million km from the Earth yields a transfer analogous to the parabolic case, but where the Earth-to-Moon trip time is only about 3-5 months. As described in detail in Refs. 1-3, the idea of this transfer is to leave the Earth and employ one or more lunar swingbys to reach the WSB region — approximately 1.5 million km from Earth — in approximately 2-3 months (see Figure 1, from Ref. 5). One can view this boundary as performing the same type of function as infinity in the parabolic case. Namely, at the Earth-Sun WSB, a nearly-zero maneuver will raise periapsis due to the sensitivity in the region. (This sensitivity is analogous in a sense to the sensitivity a "pop fly" baseball has to wind gusts at the top of its arc.) In general, a small maneuver near the Earth-Sun WSB of <50 m/s will allow a return to a similar WSB region near the Moon — the Earth-Moon WSB region — resulting in an elliptic osculating state over a broad range of possible closest approach altitudes (~100 to 1000 km). This osculating elliptic state will, in general, be unstable; however, a small maneuver of ~10 m/s will stabilize it. One can think of the Earth-Sun WSB as bringing infinity to a finite distance from Earth.

This transfer technique is referred to as the "WSB technique", or "ballistic capture". Cases have been identified for Earth-Moon transfers where no deterministic maneuvers are needed between Earth departure and lunar orbit; only small Trajectory Correction Maneuvers (TCMs) of <50 m/s are required for navigation, particularly for targeting the lunar swingby(s). In other words, this technique has the potential of eliminating deterministic maneuvers during the Earth-to-Moon phase and the large Lunar Orbit Insertion burn (LOI), at the expense of a longer trip time (3-5 months) compared to the direct Hohmann path (3-5 days). The benefits of the WSB technique have yet to be fully characterized, but investigations to date suggest that a ΔV savings of 25-40% is consistently achievable. Whether a spacecraft is sent on a Hohmann or WSB path to the Moon, additional manuevers of up to ~650 m/s total ΔV are required once lunar orbit is achieved to place the spacecraft into a more useful lunar orbit (e.g., 100-300 km altitude, circular, polar).

One exciting aspect of this concept is that the WSB technique is now being demonstrated for the first time by the Japanese *Hiten* spacecraft. Hiten was launched in January 1990 and ejected the small

Hagoromo subsatellite later that year. Hagoromo subsequently inserted into lunar orbit, though its transmitter had failed before the event. Hiten was then commanded — per the prelaunch mission plan — to execute multiple lunar swingbys in 1990 and Earth aerobraking demonstrations in early 1991. Then in early 1991, Hiten was placed on a WSB trajectory to the Moon; the prototype of this trajectory was supplied to the Japanese in 1990 by Belbruno and James K. Miller at JPL. Hiten will reach the Moon about October 2 this year after a 5-month WSB transfer. Without the WSB technique, Hiten would have been unable to successfully orbit the Moon due to insufficient propellant reserves. (It is notable that the Japanese learned about and added the WSB transfer to the Hiten mission plan well after launch.) Additional details about this application of the WSB theory are found in Refs. 2 and 6-8.

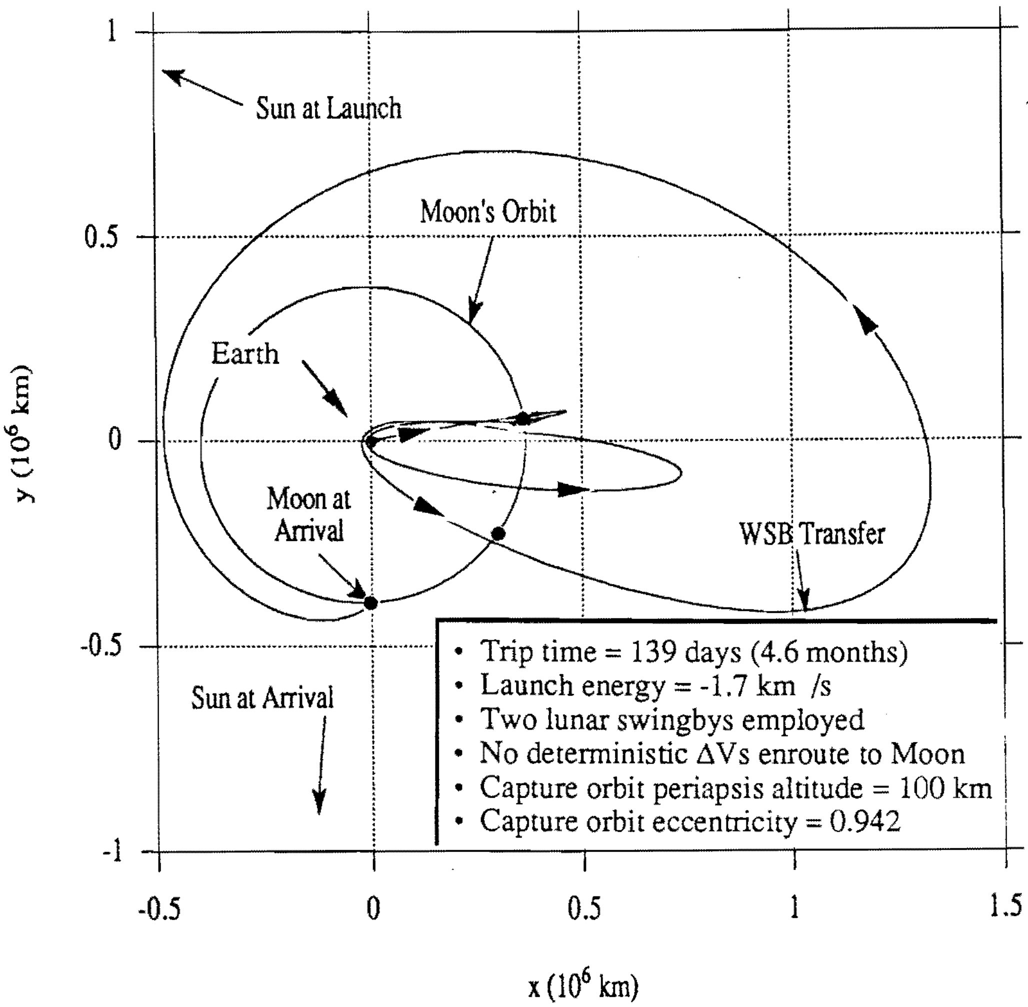

Figure 1. Typical Weak Stability Boundary Earth-Moon trajectory. (View looking normal to Moon's orbit plane; two lunar swingbys are employed to initiate WSB transfer.)

Estimated Mission Performance Using Small Satellites and the Pegasus

For illustrating the Pegasus/WSB mission architecture, the (reasonable) assumption is made that the WSB technique eliminates the translunar deterministic ΔVs and LOI. Furthermore, it is assumed that the ΔV needed for mid-course spin-axis adjustments and TCMs is $\leq$50 m/s and that lunar orbit sustenance ΔVs could be as high as 1 m/s per day in a 100-km altitude circular polar orbit and 0.2 m/s for 300-km altitude circular polar orbit (see Ref. 9). Finally, it is assumed that a desirable mission life in lunar orbit is on the order of a year. Thus, four key issues need to be addressed:

1) What is the interplanetary performance for the Pegasus, expressed in terms of C_3?

2) How much spacecraft dry mass is delivered to the Moon once the desired mission C_3 and ΔV are met?

3) Given the spacecraft dry mass and inferred propulsion subsystem size, is there any mass left over for a science payload?

4) Given a science mission potential, can the entire payload fit into the Pegasus shroud?

C_3 Performance. Interplanetary performance for launch vehicles is usually specified in terms of the parameter describing launch energy, C_3. The magnitude of C_3 is equal to the square of the Earth departure asymptotic velocity, V-infinity. Since missions to the Moon do not require Earth escape, C_3 values for lunar missions are typically negative, e.g., -2 km^2/s^2. The WSB technique can be employed over a range of perhaps -0.5 to -2.0 km^2/s^2.

Due to the effects of staging efficiency and the mass of the Pegasus third stage and avionics, C_3 performance is greatly enhanced by integrating a small, spinning upper stage with the Pegasus payload; this module thus can be viewed as a fourth 'kick' stage. (The straight three-stage Pegasus has virtually no capability to send a payload to the Moon.) For the examples given in this paper, solid-rocket kick stages are assumed. The spacecraft/kick stage combination is first launched into a low Earth orbit (LEO), spun up by the Pegasus Nitrogen cold gas reaction control subsystem, and separated from the burned out Pegasus third stage and avionics. From this park orbit, the solid motor is then ignited — either by time delay ordnance or by spacecraft command — to provide the needed ΔV for the WSB transfer. For this analysis, two kick stage options were considered as typical examples: the STAR 26B and the STAR 20A, both manufactured by the Thiokol Corporation (formerly Morton Thiokol, Inc.). STAR motor specifications were taken from Ref. 10.

Figure 2 shows Pegasus/kick stage performance as a function of required C_3, while Table 1 summarizes the key WSB and STAR motor characteristics and assumptions that were used in the analysis. This analysis reveals that the 'useful' mass injected by the Pegasus on the example WSB trajectory (having a C_3 of -2 km^2/s^2) is 72 and 80 kg, respectively, for the STAR 26B and STAR 20A.

The Pegasus performance estimates include a total allocation of 11.4 kg (25 lb) for Pegasus-mounted STAR motor integration hardware, a Pegasus-mounted payload separation subsystem, and spacecraft-mounted STAR motor integration hardware. The spacecraft-mounted hardware items — expected to be <5 kg — would continue on to the Moon with the spacecraft as 'useless' mass. So the <u>total</u> spacecraft masses sent to the Moon in these two cases are about 77 and 85 kg, respectively.

Because the Pegasus avionics is left behind upon staging the Pegasus third stage, it is assumed that the payload is responsible for controlling spin-axis adjustments and minor TCMs during the WSB transfer phase. Spin rate control may not be a requirement. Preliminary calculations indicate that 3-sigma along-track ΔV dispersions will be on the order of 18 m/s for the Pegasus third stage burn and 15 m/s for the kick motor; both are considered tolerable. There may also be some minor, yet tolerable, out-of-plane dispersions caused by spin-balance error and motor burn asymmetries.

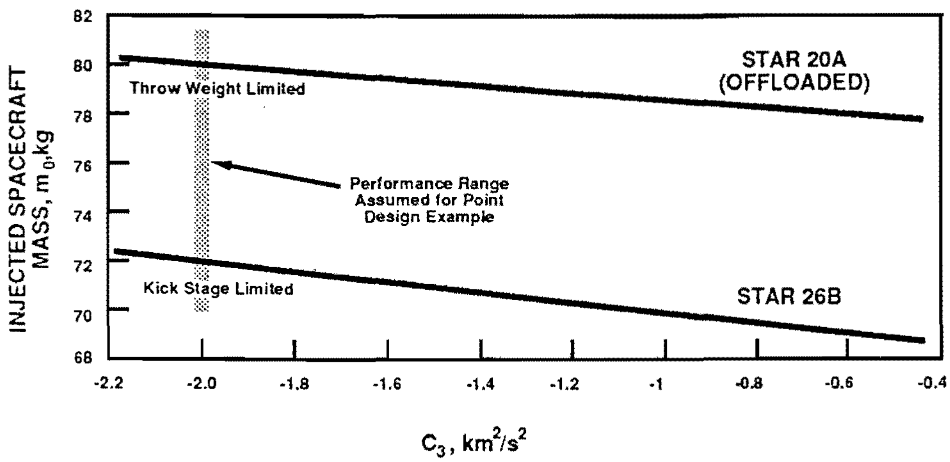

Figure 2. Pegasus/Kick Stage Performance vs. Required C_3 for WSB Transfer

Table 1. Key WSB and STAR Motor Characteristics and Assumptions Used in the Examples.

Type of Example	Kick-stage Limited	Throw-weight Limited
Kick Stage Used	STAR 26B	STAR 20A
Reference WSB Characteristics:		
C_3, km^2/s^2	-2	-2
Park orbit altitude, km	222	222
, nmi	120	120
Inclination, deg.	28.5	28.5
STAR Motor Characteristics:		
Loaded mass, kg	261	246
, lb	575	542
Burnout mass, kg	23	26
, lb	50	58
Effective Isp, sec	271.1	291.9
Motor Case	Titanium	Filament-wound, fiberglass-epoxy
Motor Nozzle	Silica-phenolic, 17.8:1	Carbon-phenolic, 50:1
Motor Diameter, cm	66	50
, in	26	20
Motor Length, cm	84	1.5 m
, in	33	58

For the STAR 26B the park orbit altitude has been optimized for maximum performance, i.e., as the payload gets lighter, the altitude is increased to match Pegasus performance. This is thus a "kick-stage-limited" example. The second payload kick stage option shown in Figure 2 is an offloaded STAR 20A. In this example the park orbit altitude is held fixed and the normal motor propellant load is offloaded (by 14% in the reference example) to meet the Pegasus LEO performance capability per the assumed conditions. This example is thus "throw-weight limited".

For both examples, the final park orbit selection and associated propellant offload (for the STAR 20A) will be a highly mission-specific decision; orbit altitude and geometry, orbit inclination and launch time can all affect the decisions. The examples chosen serve as a conservative estimates of the Pegasus performance in the given configurations.

ΔV Performance. To get a feeling for the satellite's ΔV performance, the following variation of the rocket equation was used (Ref. 11, p. 582):

$$\Delta V = g\, I_{sp} \ln[m_o/(m_o-m_p)]$$

where,

ΔV = ΔV capability of the spacecraft's propulsion subsystem
g = gravitational constant (9.8 m/s^2)
I_{sp} = The specific impulse of the propulsion subsystem in seconds
m_o = injected mass of the spacecraft
m_p = mass of the propellant consumed

Figures 3 and 4 summarize the following discussion. One key indicator of the class of spacecraft that reaches the Moon is its 'dry mass', that which remains after the propellant is consumed. In the rocket equation above, this is the value m_o-m_p. To bracket the cases, a 20-kg range of m_o for each STAR motor option is assumed. In addition, one point design for each STAR option is offered as a typical example of expected performance, per the information supplied in Figure 2.

Assumptions for ΔV are derived from Ref. 9. A desirable ΔV budget for a simple lunar mission is at least 500 m/s, and preferably as high as 1500 m/s. The spacecraft propulsion subsystem must supply about 50 m/s for spin-axis adjustments and TCMs, another 10 m/s or so for initial lunar orbit stabilization after the WSB transfer, 400-800 m/s for orbit circularization, and ~75-400 m/s for one year of orbit sustenance ΔV (for altitudes between 300 and 100 km, respectively). There are many ways to adjust these numbers by altering the mission plan; thus the approach was taken to show a range of ΔVs between 500 and 1500 m/s.

For this analysis, two assumptions for I_{sp} were considered to bracket current technology: 200 sec. (Fig. 3) and 300 sec. (Fig. 4). The former value corresponds to a simple hydrazine system while the latter approaches bipropellant performance. (Solid propulsion was judged impractical for this function and thus was not considered.) To complete this analysis, the desired range of ΔVs were plugged into the rocket equation to generate the amount of propellant used, m_p. Once m_p is known, m_o-m_p follows. We further assume, based upon Lunar Observer study experience, that a dry mass of less than 50 kg is not worth considering for lunar science applications. A capable 50 kg spacecraft could likely be built, but there would be little or no mass left for science instruments.

First we address the STAR 26B case. For an assumed range of m_o of 65-85 kg and $\Delta V > 500$ m/s, Figure 3 (I_{sp} = 200 sec.) shows that dry masses from 50 to 65 kg are delivered to the Moon. For a spacecraft I_{sp} of 300 sec (Fig. 4), the comparable values are 55 and 70 kg, respectively. Direct tradeoffs between dry spacecraft mass and desired ΔV capability (up to ~1000 m/s) are clearly possible. For the STAR 20A example, the assumed range of m_o is 70-90 kg. For hydrazine-like capability (Fig. 3), 55 to 70 kg are delivered, while for bipropellant-like capability (Fig. 4) the values are approximately 60 to 75 kg.

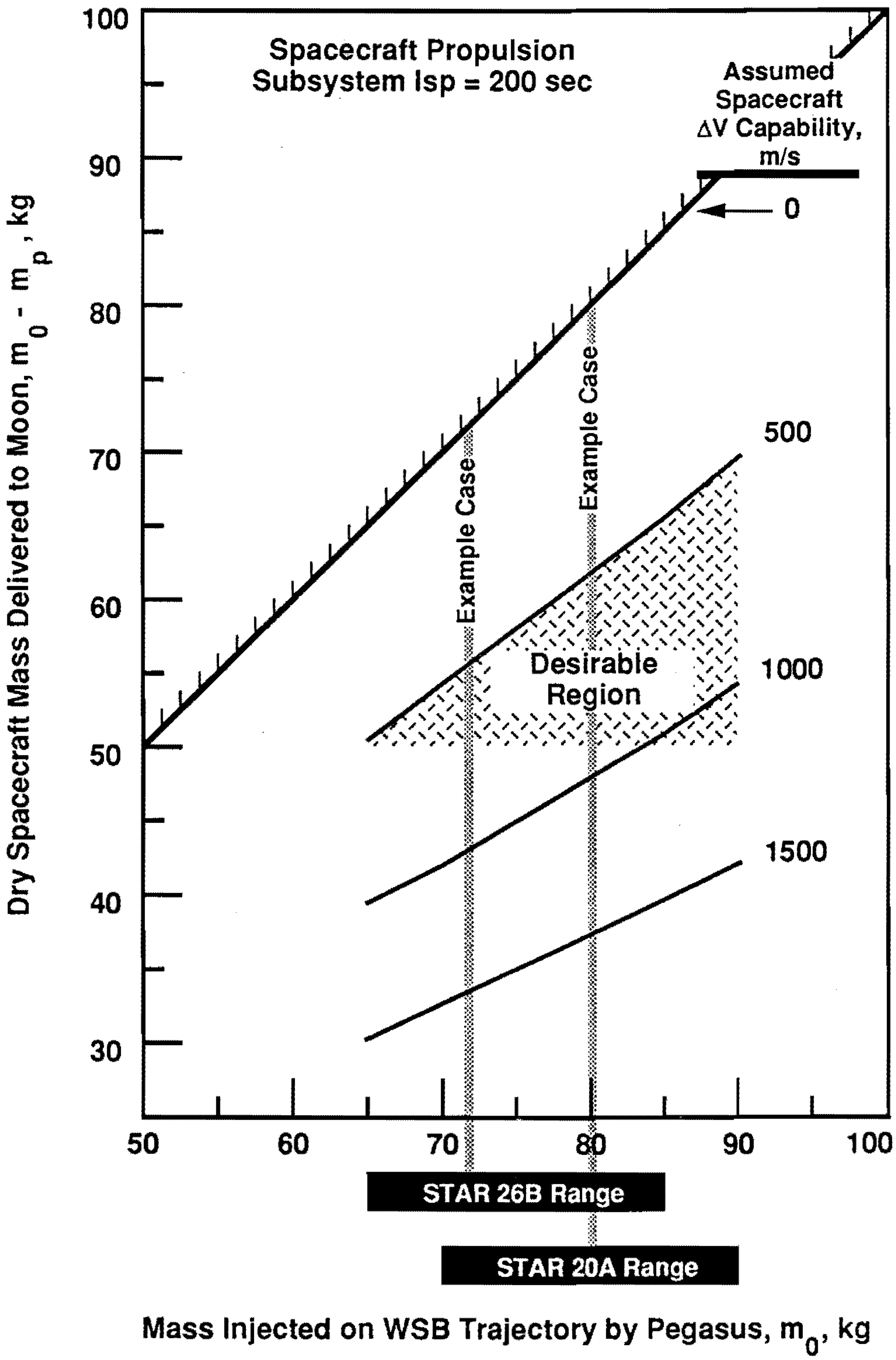

Figure 3. Dry Mass Delivered to Moon — Assumed I_{sp} = 200 sec.

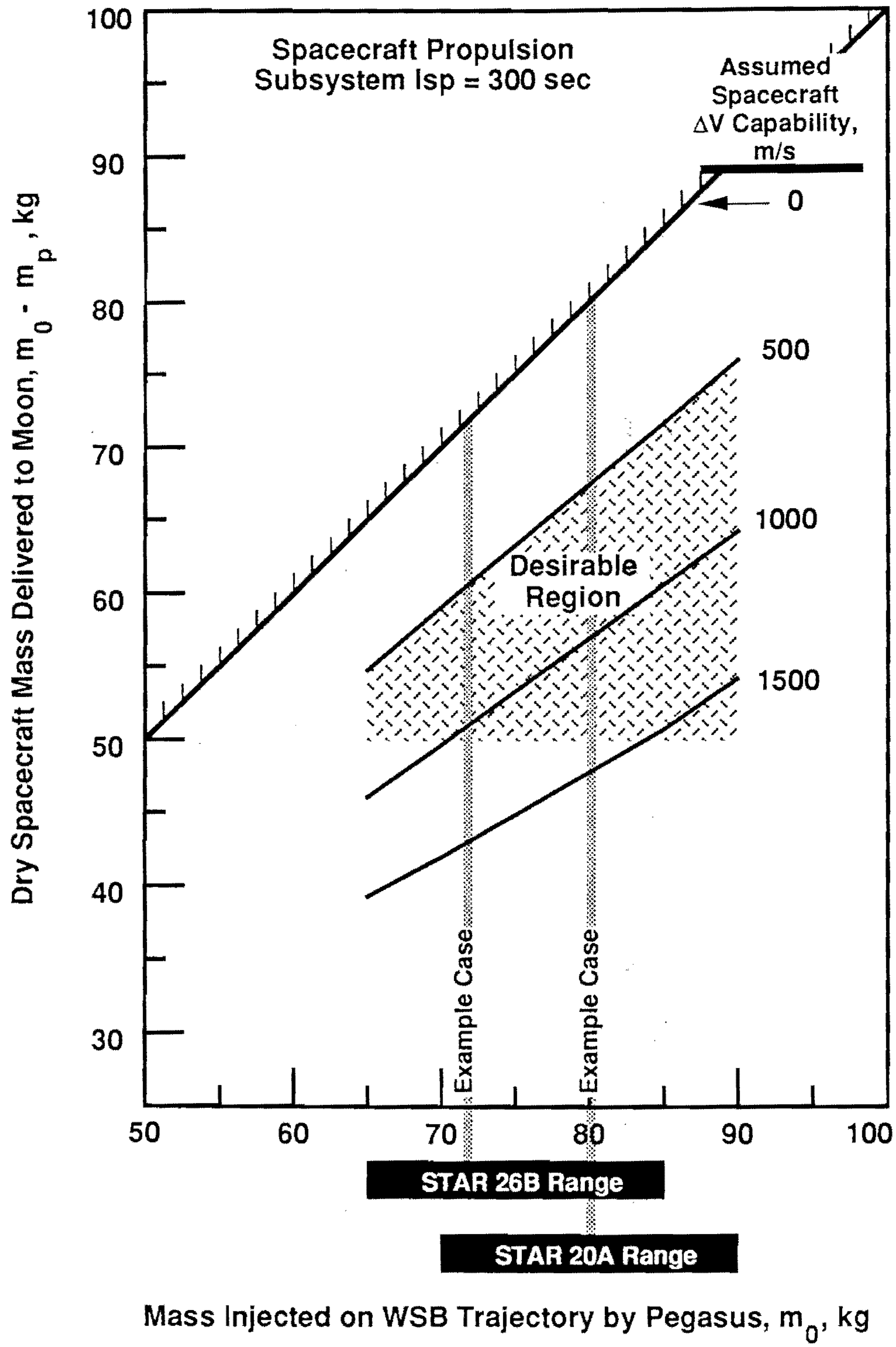

Figure 4. Dry Mass Delivered to Moon — Assumed I_{sp} = 300 sec.

Prospects for Science Mass. From the dry masses quoted above, the dry mass of the spacecraft propulsion subsystem must first be subtracted before an estimate of 'science and engineering' mass can be made. Examination of Figures 3 and 4 will show that the propellant mass required to meet the 500 m/s minimum ΔV target is 15-20 kg for the hydrazine case and 10-15 kg for the bipropellant case. Each additional 500 m/s of ΔV costs another 7-12 kg, depending on where one chooses to be on Figures 3 and 4. Propulsion subsystem hardware (tanks, valves, tubing, thrusters, etc.) to support the storage and use of this propellant could easily amount to 5-15 kg, again depending on the parameters assumed. Clearly a very low-mass subsystem is in order to allow this scenario to bear fruit. For this analysis, we assume that a (dry) propulsion subsystem can be built having a mass of 10 kg.

Thus, for the assumed 20-kg performance ranges of the two STAR options, the usable science and engineering mass becomes approximately 40-60 kg for the hydrazine case and 45-65 kg for the bipropellant case. Given our stated assumption that anything less than 50 kg lacks appeal (i.e., little or no science capability), we conclude that only portions of the assumed STAR performance ranges are applicable.

Spacecraft with hydrazine subsystems appear to be marginal for the STAR 26B (40-55 kg), and only slightly better for the STAR 20A (45-60 kg); the high end of its assumed performance range is all that applies. For spacecraft with bipropellant, the STAR 26B is barely adequate (45-60 kg) while the STAR 20A yields 50-65 kg. The bottom line is that 5 to 15 kg of mass beyond the 50 kg cutoff may be realizable using this mission architecture. For a high-end STAR 20A case with a spacecraft having a bipropellant subsystem, it may be possible to orbit 65-kg of science and engineering mass at the Moon which has about 1000 m/s of ΔV capability — enough to establish a 100- to 200-km circular, polar orbit after the WSB transfer and then sustain it for about a year.

Payload Fairing Volume. Figure 5 shows the advertised baseline Pegasus shroud size with allowable payload dynamic envelope shaded. A hole through the disk-shaped avionics shelf leads aft to thrust support structure and the forward dome of the third stage. The spin-up rocket module is also shown aft, though for certain applications it can be located forward of the avionics shelf.

Figures 6 and 7 show the payload envelope with silhouettes of the installed STAR motors. (Integration hardware is not shown.) For reference, the 38-kg Lunar Observer cylindrical subsatellite concept (Ref. 9) is shown alongside. As originally designed, this 'bare-bones' satellite is about 127 cm in diameter and 37 cm long — a tuna can shape. In these figures, its diameter is reduced by 10% and length increased by 10%; both changes are within the tolerances of the assumptions underlying the conceptual design. The satellite envisioned for the Pegasus/WSB concept would most certainly be bigger than the Lunar Observer subsatellite — as big as the payload fairing and Pegasus performance will allow — and would also require additional subsystems.

The STAR 26B installation suggests that the spacecraft should be mated in-line with the motor, because the motor fills much of the fairing diameter, leaving little volume for any spacecraft hardware around it. For the STAR 20A, however, the small diameter of the motor/nozzle allows it to be submerged into the cavity of the Pegasus avionics structure, leaving only about 81 cm (32 in) extending into the payload fairing and slightly more volume around the motor. This arrangement suggests that the motor might be partially integrated inside the core of the spacecraft.

For either motor option, there is additional volume ahead of the motors to accommodate much of the spacecraft. To first order, there appears to be a reasonable chance of integrating the desired packages into the standard Pegasus fairing.

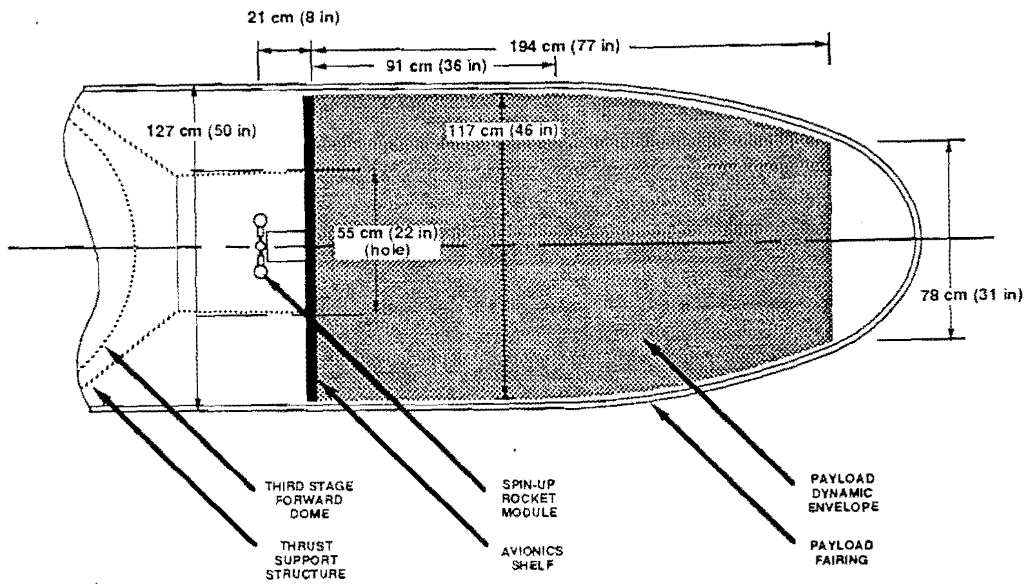

Figure 5. Pegasus Payload Fairing

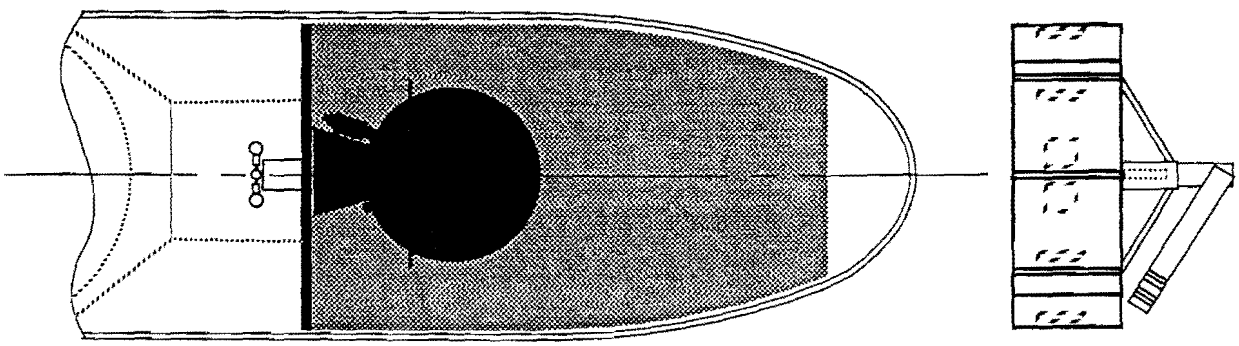

Figure 6. Pegasus with Installed STAR 26B Motor

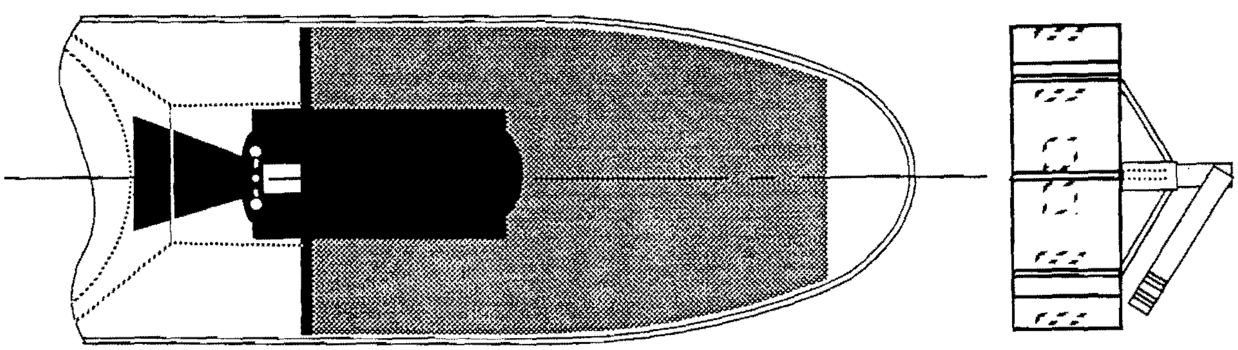

Figure 7. Pegasus with Installed STAR 20A Motor

Spacecraft Attributes

We argue that with some clever thinking, a low-mass spacecraft design, and use of the WSB technique, 50-65 kg of spacecraft science and engineering mass — of which 5 to 15 kg or more is science mass — could be sent to the Moon by the Pegasus. The spacecraft would probably require the following:

- Low-mass technology; standard complement of engineering subsystems
- Bipropellant propulsion; 500-1000 m/s ΔV capability; maneuver control capability
- Spin stabilization; spin-axis control capability (spin rate control perhaps not required)
- Full use of Pegasus payload fairing volume
- Accommodations for 5-15 kg of science instrument(s)
- 1-2 year design life

The final question is: what do we do at the Moon?

Lunar Science Options

The science priorities at the Moon have been well defined over the past twenty years. The most recent listing (see Table 2) has been developed by NASA's Lunar Exploration Science Working Group (LEXSWG). The present list of science instrument candidates (Ref. 9) reveals that some are near the 5-15 kg science mass target defined here. Besides the gravity field measurement system (GMS, implemented on Lunar Observer with the 38 kg subsatellite and 18 kg of main spacecraft hardware), we might consider the Magnetometer (MAG; 3.6 kg), Electron/Ion Mass Spectrometer (EIMS; 15.0 kg), X-ray Spectrometer (XRS; 16.8 kg), Thermal Emission Spectrometer (TES; 15.0 kg), and Ultraviolet Spectrometer (UVS; 8.8 kg).

The GMS seems like a good candidate not only because the Lunar Observer subsatellite is close in size and mass to what we seek, but also because measuring the lunar gravity field is the #2 science priority after Gamma-Ray spectroscopy (see Table 2). Here we do not include the laser altimeter instrument as part of this measurement, as was done for Lunar Observer. Thus, there is the implication of less science for the Pegasus option vs. Lunar Observer. The objective of this experiment would be to collect global gravity field data, with emphasis on far side and high-latitude (>30 deg.) near side coverage. Low-altitude (100-200 km) circular, polar orbits would be most desirable. It would be necessary to have two such satellites for this experiment to ensure the acquisition of lunar far side data, although a single-spacecraft, near-side-only experiment could be conducted and still yield more understanding of the lunar gravity field than we have now.

MAG is also attractive with one or two satellites. The science objectives include mapping the surface magnetic fields, measuring the magnetic dipole moment of the Moon, and characterizing the size of a possible lunar magnetic core. Such investigations would be greatly augmented by carrying an electron reflectometer (ER) with the MAG instrument. This in fact was the baseline Lunar Observer configuration until recently, and remains the configuration for Mars Observer. (The ER on Mars Observer has a mass of <5 kg, so the MAG/ER set is still within our 5-15 kg target.) This experiment is perhaps most amenable to non-circular lunar orbits (as long as the periapsis is ~50-150 km) as well as higher-altitude orbits. The Lunar Observer study developed a subsatellite concept with magnetometer having a total spacecraft mass of about 50 kg. This may well be within the mass range of the Pegasus, but the question remains whether the additional spacecraft subsystems and MAG booms could be packaged within the available mass/volume constraints.

The EIMS instrument is another likely candidate, since it could likely achieve its main objectives of electron reflectometry and ion mass spectroscopy on a spinning small-satellite platform which has only spin-axis pointing control. The electron reflectometry function is used in conjunction with the MAG experiment, as stated above, while the ion mass spectrometer is used to map global atmospheric distribution, ion species composition, ion source and sink rates, and surface composition via measurement of secondary ions sputtered by the solar wind. Orbit requirements for this instrument are similar to those for MAG.

XRS, TES and UVS require nadir pointing and thus are not the best candidates for a simple small satellite design. Should nadir-pointing capability be feasible on a spacecraft of 50 kg or less, then these instruments deserve consideration.

So, we conclude that at least three possible lunar science missions should be evaluated with the baseline Pegasus/WSB mission architecture in mind: lunar gravity field measurement (using one or two satellites), electron/ion mass spectroscopy (one satellite), and perhaps magnetometer/electron reflectometer studies (one or two satellites). We suggest that this cursory review of science options be followed by more detailed evaluations.

Table 2. Absolute LEXSWG Science Instrument Priorities for Lunar Observer (from Ref. 9)
(Measurement contributions correlate with measurement priorities below)

RANK	INSTRUMENT NAME	ACRONYM	MEASUREMENT CONTRIBUTION
1.	Gamma-Ray Spectrometer	GRS	1
2.	Gravity Measurement System + Laser Altimeter	GMS + LOLA	2
3.	Visible/Infrared Mapping Spectrometer	VIMS	3
4.	Magnetometer #1	MAG1	5
5.	Electron/Ion Mass Spectrometer	EIMS	1, 5, 6
6.	X-Ray Spectrometer	XRS	1
7.	Geodetic Camera (100 m/pixel resolution)	LOIS-3	4
8.	Thermal Emission Spectrometer	TES	3
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9.	Mapping Camera (15 m/pixel resolution)	LOIS-2	4
10.	Ultraviolet Spectrometer	UVS	6
11.	Magnetometer #2	MAG2	5
12.	Microwave Radiometer	MRAD	7
13.	Neutral Mass Spectrometer	NMS	6

* = Limit of strawman payload list for 1991 Lunar Observer baseline concept

MEASUREMENT PRIORITIES
1. Global elemental composition (much higher priority than all others)
2. Global gravity and topography
3. Global mineralogy
4. Global imaging
5. Global magnetics
6. Global atmospherics
7. Global heat flow

Implications

The chief implication of using the Pegasus/WSB concept for lunar science is cost. The WSB trajectory technique eliminates the need for a large LOI burn and allows this mass savings to be translated directly into additional science and engineering mass. This leads to a modest yet useful science payload. The Pegasus launch cost is roughly $10-15M per launch. Adding a small satellite and science payload costing about one to two times this amount plus a year or so of operations costs results in a mission cost of only tens of millions instead of the more typical hundreds of millions.

By design, the aircraft-launched Pegasus offers a wider range of launch parameters (location, timing), which results in more flexibility for mission and trajectory planners. Preliminary calculations from other Pegasus mission studies also suggest that in many cases a launch from inclinations less than 28.5 deg. enhances performance; whether this is valid for lunar missions remains to be shown. Launch timing and the WSB transfer trajectory design will undoubtedly be factors.

Integration and turn-around time is advertised to be relatively short for the Pegasus; such features would be quite useful for conducting dual-launch missions which last only a year or two. Finally, there are several upgraded versions of the Pegasus under study now. These advertised growth versions of the Pegasus combined with recent developments in small-satellite technology may enable the achievement of more demanding objectives such as global elemental mapping, the search for water, surface imaging, and studies of the tenuous lunar atmosphere.

This lower-cost way of reaching the Moon may also provide an avenue for pursuing interesting lunar-based small-satellite constellation concepts (e.g., far side or store/forward communications, navigation, data retrieval) and attractive commercial lunar activities (video, research and development).

All of these facts lead us to conclude that the Pegasus/WSB mission concept deserves a closer and more thorough look. We caution that the understanding of the WSB technique is in its infancy and that useful science and engineering mass at the Moon is strongly driven by the desired orbit parameters there. Careful engineering, science and mission tradeoffs will be required to enable a worthwhile science mission with this mission architecture. Certainly this concept will <u>not</u> achieve all of the lunar science objectives defined for Lunar Observer, but it may achieve some of the simpler goals.

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