3 / TRANSPORTATION ISSUES

THE SINGLE EXISTING ELEMENT of the Space Transportation System (STS) is the Earth-to-orbit vehicle, the Space Shuttle. Passengers can be carried to low-Earth orbit (LEO) with equipment for activities of limited duration. Satellites can be placed into LEO from the payload bay or can be launched beyond LEO using upper stages.

The next element of the STS will be the LEO space station, which will increase the allowed duration of scientific and commercial activities in orbit. Just as the shuttle introduced the concept of reusable space vehicles, the space station will establish permanent human presence as an element of space operations. By the turn of the century, reusable orbit-to-orbit vehicles will be added to the space station. The Orbital Transfer Vehicles (OTV) will be designed to lower the cost of shipment of payloads to deep space through reuse of the upper stage. The space station then will incorporate the function of a freight dock, where cargo is readied for transshipment to destinations beyond.

At the same time, the shuttle will be nearing the end of its operational life; and a second generation of Earth-to-orbit transportation will be readied. Will this second generation consist of an updated shuttle, or will the functions of transporting freight and carrying passengers be explicitly separated to alleviate the crippling expense of launching to LEO? For example, an unmanned launch vehicle based on shuttle propulsion technology could carry many times more cargo for less launch cost than does the shuttle today. However, the cost of developing such a vehicle can be justified only if the projected traffic into LEO is large.

The buildup of a lunar base places demands on the STS that must be reflected in the design of the elements. The first three papers describe the impact of a lunar base and other deep space activities. Woodcock delineates the fundamentals of the orbital mechanics involved in the simple shipment of payloads to the lunar surface and suggests some modes of operation for accomplishing the emplacement of lunar surface facilities. Babb *et al*, examine the role of the LEO space station and define resources that must be made available at that transportation node to support freight operations. Keaton takes a fresh look at the whole infrastructure of the STS from the point of view that a variety of destinations will be desirable in the next century. His elegant analyses of orbital mechanics in the Sun-Earth-Moon system are supported experimentally by the ingenious targeting maneuvers derived by Bob Farquhar for the interception of Comet Giacobini-Zinner by the ICE spacecraft.

The final three papers in this section describe innovative uses of the lunar environment to reduce the operational burden on the terrestrially based STS. Heppenheimer reviews elements of his own work over the past decade on mechanics of launching raw materials and other products from the Moon with large, fixed electromagnetic "guns." Rosenberg discusses the production of silane from lunar materials as a propellant to combine with lunar-produced oxygen in an engine design uniquely suited to the space environment. Finally, Anderson *et al.*, describe a simple lunar surface launching technique for putting small payloads into lunar orbit with solid fuel rockets for scientific investigations and other purposes.

MISSION AND OPERATIONS MODES FOR LUNAR BASING

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Future lunar operations may be directed to permanent or long-term presence on the Moon and supported by a space station as an intermodal transportation complex. Flight mechanics constraints on station-supported lunar operations are described and analyzed, and the implications to space station are presented. Mission modes supportable by the NSTS and its derivatives, and by the space station, are described and compared. Sensitivities of the modes and their uses for lunar operations are described.

INTRODUCTION

A U.S. civil space station is planned for low-Earth orbit by the early to mid 1990's. By the late 1990s it will serve as an intermodal transportation complex, *i.e.*, a spaceport that could, among other things, support advanced lunar operations. The space shuttle and its future derivaties—support by the space station, new technologies for reusable orbit transfer vehicles (ROTVs), and the potential of obtaining propellant oxygen from the Moon itself—all combine to dramatically lower the calculated cost of future lunar operations compared to the "brute force" of the Apollo missions. Where a permanent human presence on the Moon was once viewed as an option only for a heavily funded civil space program, it now appears (as also observed by Paul Keaton and others) as an achievable option within the funding scope of present civil space activities.

Effective use of new mission and operational schemes is not without challenge. The Apollo missions, especially after the requirement for free-return trajectories was removed, were relatively unconstrained. Launch opportunities were frequent, launch windows up to hours in length, and any return trajectory that terminated in a rather large recovery zone in the Pacific Ocean was acceptable.

INFLUENCES OF FLIGHT MECHANICS ON OPERATIONS DESIGN

With an orbiting spaceport, departure to the Moon is constrained to a particular Earth orbit. Further, if oxygen derived from lunar surface materials is to be used by the transportation system, it is important to have an oxygen cache in the lunar vicinity. This necessitates a particular lunar staging point. Although a range of choices (including the L1 and L2 Lagrangian points as well as lunar orbits) is possible, a polar lunar orbit has been selected by this as well as earlier studies, because it permits access to all of the lunar surface and does not constrain base location. Thus lunar base support operations are likely to be constrained to operations from a particular Earth orbit and a particular lunar orbit. If this is to be economical and effective, these constraints must not introduce large performance penalties.

Motion of the Moon

The Moon revolves about the Earth in a nearly circular orbit of approximately 384,000 km average radius, with a period of 27.3 days. Its orbit is inclined 5 degrees to the ecliptic plane. The angular momentum of the Moon in its orbit is torqued by the attraction of the Sun, causing a slow precession of about 18.5 years' period. Therefore, the inclination of the Moon's orbit to Earth's equator varies as shown in Fig. 1., from 18° to 28°.

Synchronized Orbital Motions

The NASA space station orbit is planned as 28.5° inclination. Since this exceeds the Moon's maximum inclination, in-plane transfers from the space station orbit to the Moon will always be possible, *i.e.*, whenever a vector from the Earth to the Moon lies in the space station orbit plane. The frequency of occurrence depends not only on the motion of the Moon, but also on the motion of the space station orbital plane.

Inclined Earth orbits precess in a motion called regression of the line of nodes. This motion is illustrated in Fig. 2. The rate varies with inclination and altitude according to the equation given in the figure and is between six and seven degrees per day (retrograde) for typical space station orbits. As the Moon revolves posigrade at about 13° per day, it passes through the space station orbit plane about every 9 days on the average, permitting

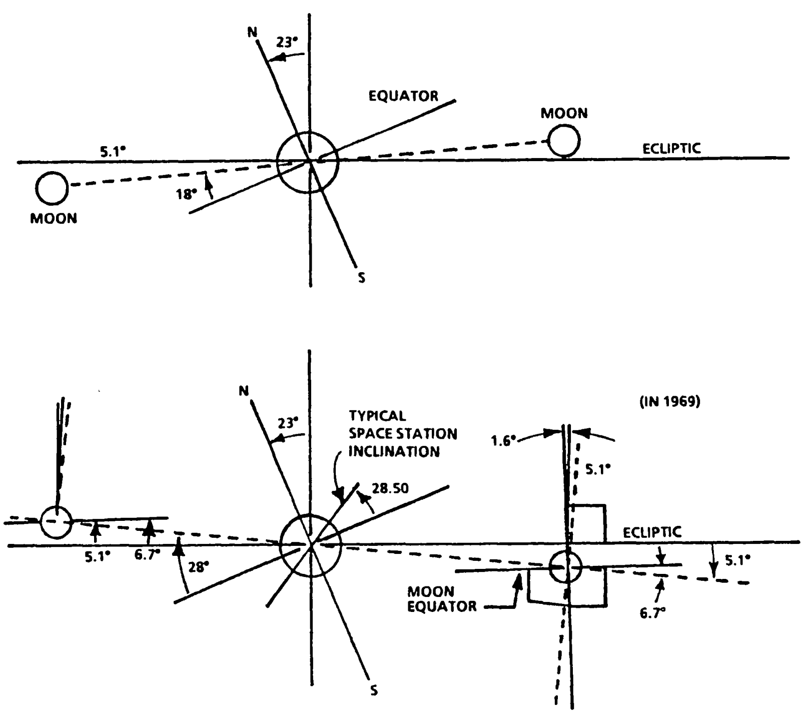

Figure 1. Variation in the Moon's orbital inclination.

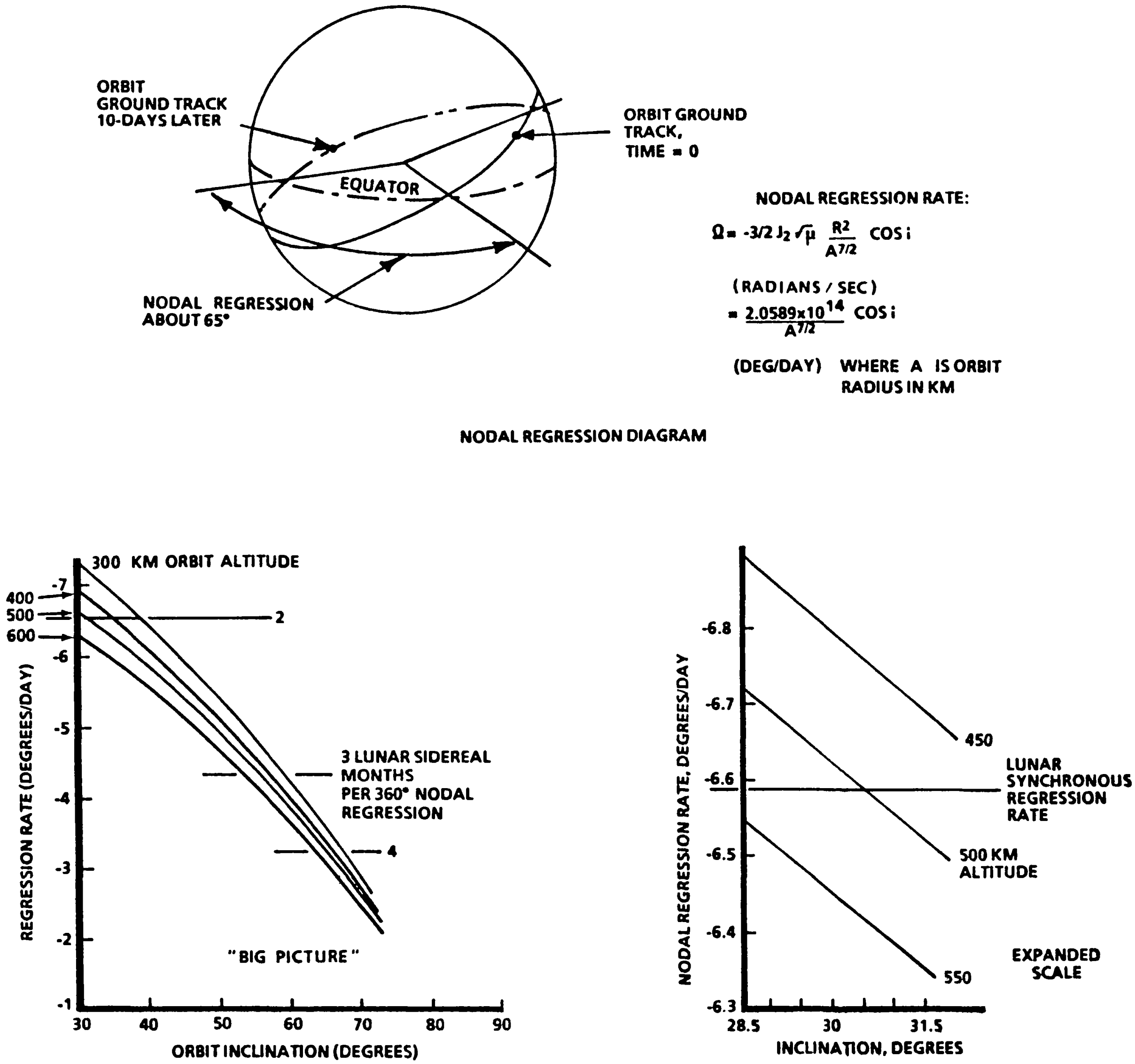

Figure 2. Earth orbit nodal regression.

an in-plane departure from the space station to the Moon. An additional constraint is that the lunar polar orbit needs to be aligned with the incoming and departing transfer vectors so that lunar orbit insertion and departures are at least approximately in-plane. Because of the Moon's motion, the alignment involves a vector sum as illustrated in Fig. 3. The combined motions of the Moon and the space station orbit must be synchronized with the lunar period (lunar month) so that alignments for Earth departure and lunar arrival recur regularly. Synchronism is possible at orbit altitudes and inclinations very close to the NASA space station baseline of 500 km and 28.5°, as also shown in Fig. 2.

For in-plane (minimum ΔV) departure from Earth orbit, the lunar vehicle is constrained to travel to the Moon in the plane representing the space station orbit at the instant

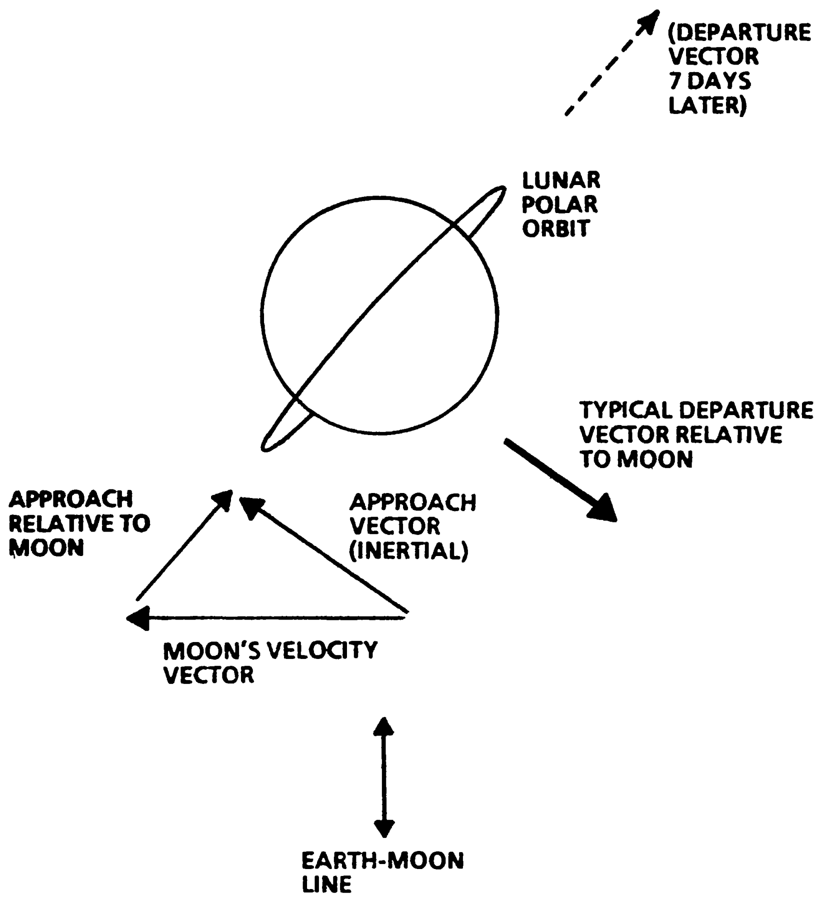

Figure 3. Lunar motion sets the timing for in-plane arrival and departure for lunar polar approach.

of departure (translunar injection, TLI). The plane of the translunar path is nearly "frozen" in inertial space (because of the energy of the transfer orbit), while the space station orbit continues its nodal regression. Earth departure and lunar arrival are timed so that the Moon, traveling in its orbital plane, arrives at the intersection of the two orbital planes at the same time the vehicle does. A similar timing constraint applies to the return trip; the launch to Earth (transEarth injection, TEI) occurs when the Moon crosses the intersection of its orbit plane and the orbit plane that the space station *will be in* at the time of Earth arrival. A typical synchronized round trip is depicted in Fig. 4.

Mission timing may be analyzed by depicting motions in the lunar orbit plane—the Moon moving posigrade at 13.1764° per day, and the intersection line of the lunar and space station orbits traveling retrograde, at half this rate (the slow movement of the lunar orbit plane is neglected). Because the space station orbit is regressing about the Earth's equator and not about a normal to the Moon's orbit, the rate of motion of the intersection line varies, and the dihedral angle between the orbits varies from the sum to the difference of their inclinations. Synchronism occurs whenever the combined angular motions, *i.e.*, displacements, add up to a complete circle, such as $N\pi$ where N is an even integer.

A practical synchronism occurs with N = 4 as illustrated in Fig. 5. In this example, the mission begins when the dihedral angle between the orbit planes is at minimum

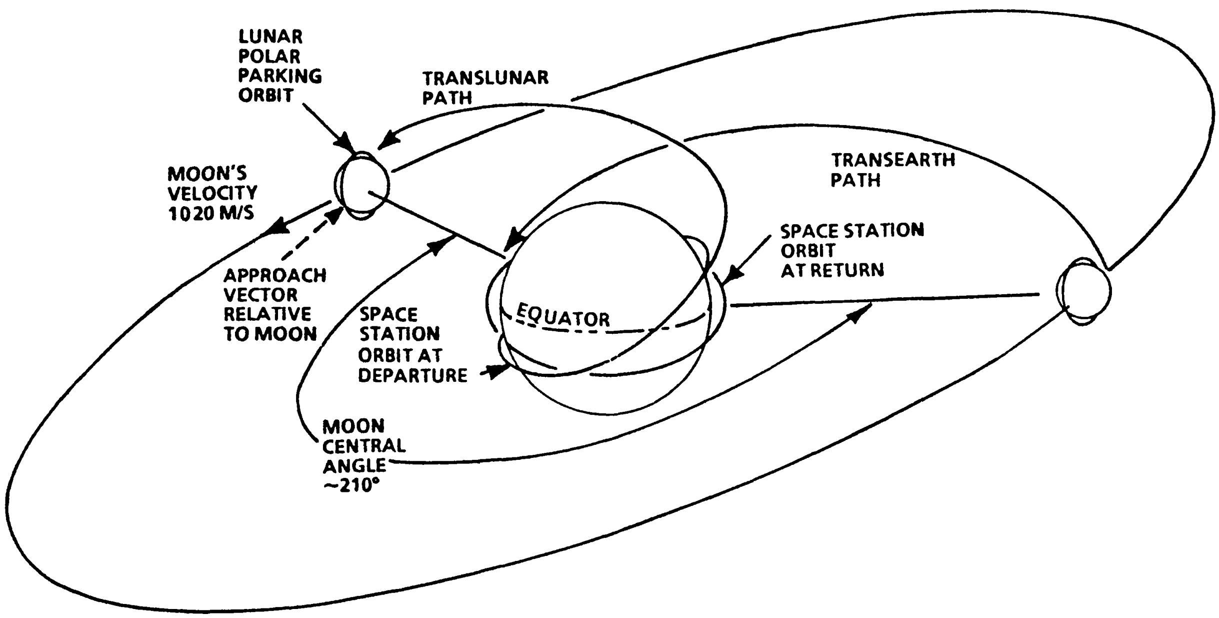

Figure 4. Pictorial of typical synchronized Earth-Moon round trip.

and the rate of motion of the intersection line is therefore maximum. The lunar vehicle paths traverse angles slightly less than π because typical trip times are less than the 5 days needed for a 180° transfer. This example represents the minimum practical mission duration under the constraints that we are dealing with, about three weeks.

For lunar polar orbits, the staytime in lunar orbit must be at least 13.66 days if out-of-plane descents and ascents are to be avoided. The Moon's rotation is locked to its orbital motion; the Moon rotates under the orbit at one revolution per 27.3 days; any particular landing site passes through the orbit plane twice per revolution. If further constraint is applied, a particular surface site, or to have the surface stay take place in lunar daylight (landing at dawn and liftoff at dusk), a longer orbital stay will usually be needed in waiting for orbital alignment with the lunar terminator. In such a case, a total mission duration on the order of 35 days may be needed (the mission time increases by increments of 13.66 days because a landing site passes under a polar lunar orbit at that interval). In general, one cannot have at the same time a particular lunar orbit, a particular surface site, and daylight surface stay.

Method of Analysis

The key issue for mission mode analysis is whether all these constraints can be accommodated without large performance penalties for propulsive plane changes to achieve the necessary orbital alignments. The problem can be visualized as a large nomograph representing the constraints and their interrelationships. Most of these can be represented in graphical form.

The complete analysis includes 36 interrelationships and is too lengthy to display in this short paper. As shown in Fig. 6, the analysis divides into four sub-problems.

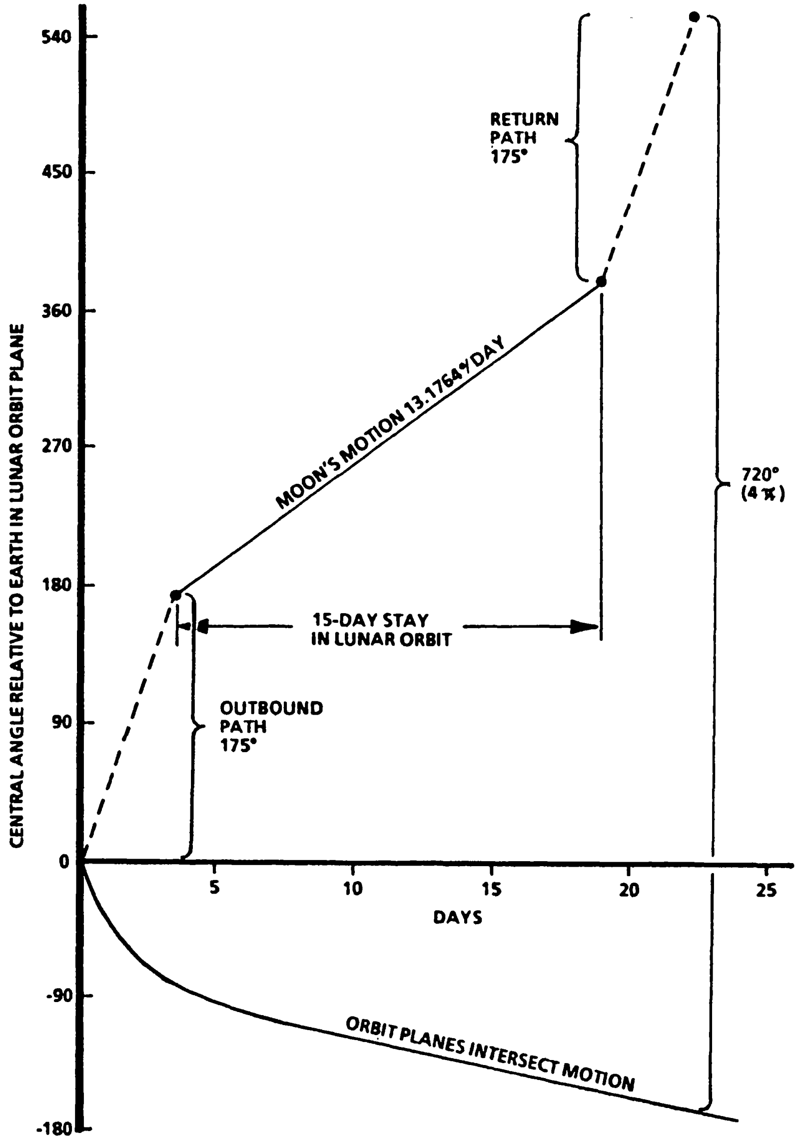

Figure 5. Synchronizing orbital plane and path alignments.

Determination of the outbound paths and ΔVs is also illustrated in Fig. 6. The out-of-plane angle between the arrival path and the Moon's orbit is just the dihedral angle between the space station and lunar orbit planes and is determined by the timing of mission start. TLI ΔV and the geometry and velocity of lunar approach are determined by the outbound transit time and the above out-of-plane angle. The lunar orbit insertion (LOI) ΔV depends on approach velocity relative to the Moon, which is in turn dependent on transit time (the Moon's orbital velocity of about 1020 m/s must also be taken into account), the altitude of the lunar orbit, and the plane change accomplished entering lunar orbit. For this study, lunar orbit altitude was fixed at 100 km. Figure 7 graphs ΔV entering lunar orbit at this altitude.

For this study, plane changes departing from or returning to Earth orbit were ruled out. They are expensive and don't help much to establish the needed synchronisms. (They

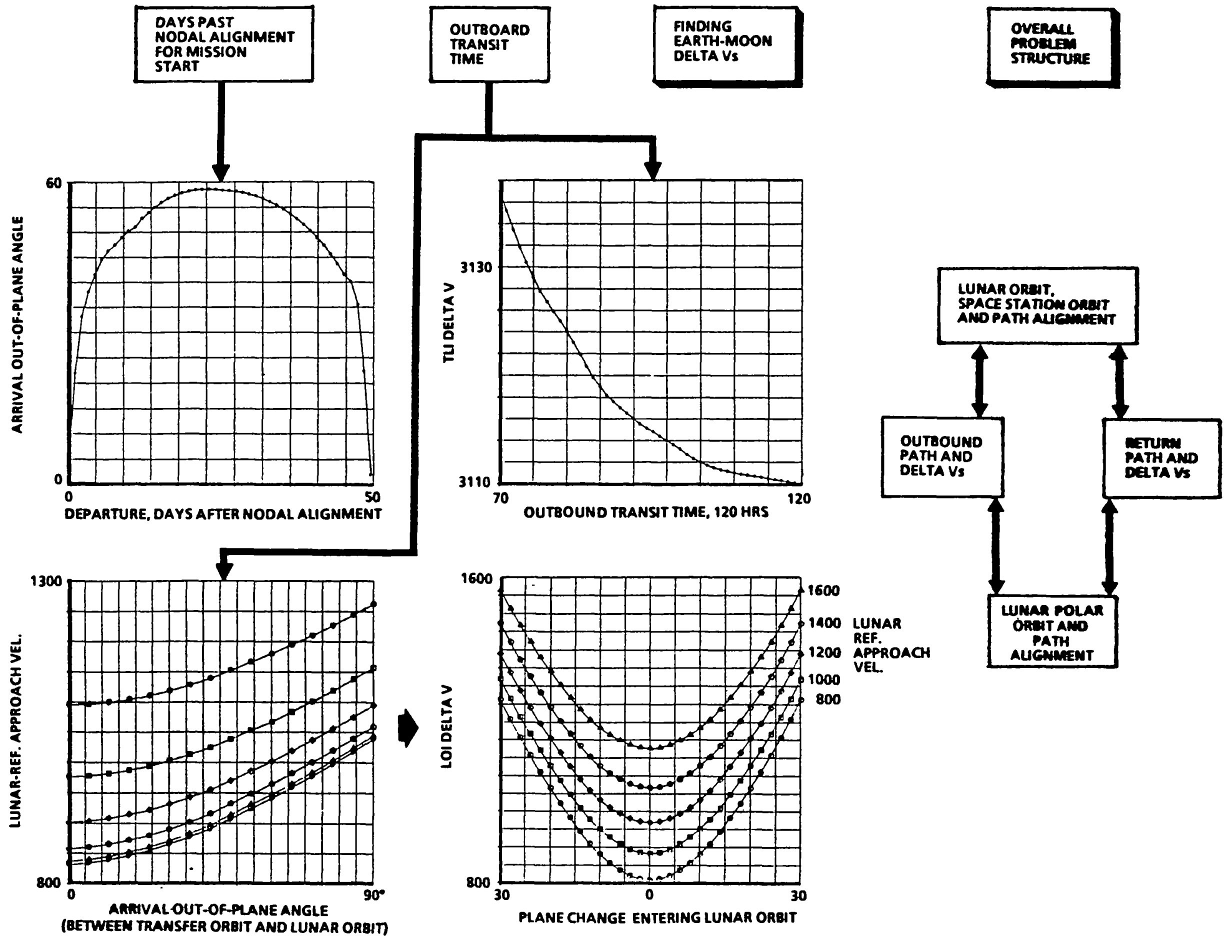

Figure 6. Problem structure with a partial example of detailed structure.

may help by relaxing timing constraints.) Satisfaction of lunar orbit, space station orbit, and path alignments sets conditions for satisfying other constraints. Fixing outbound transit time and mission duration permits iteration on lunar orbit staytime and return transit time to satisfy the constraints. The transit times and staytime set conditions for satifying lunar polar orbit and path alignments. Fixing the plane change entering lunar orbit leaves the plane change departing lunar orbit as a free parameter adjusted to satisfy polar orbit/ path constraints. Outbound transit time, mission duration, and plane change entering lunar orbit are thus *optimization parameters* that can be adjusted to minimize mission ΔV.

The analysis was conducted using a computer code analogous to a modern "spread sheet" code, but having capability to accept graphical (table lookup) relationship descriptions, to internally iterate to resolve interdependencies, and to scan optimization parameters to optimize an objective function, in this case total mission ΔV. Representative results are shown in Fig. 8. (Values shown include 75 m/s margin above the ideal values for finite-burn losses, *etc.*) It was found that adjusting the optimization parameters permitted

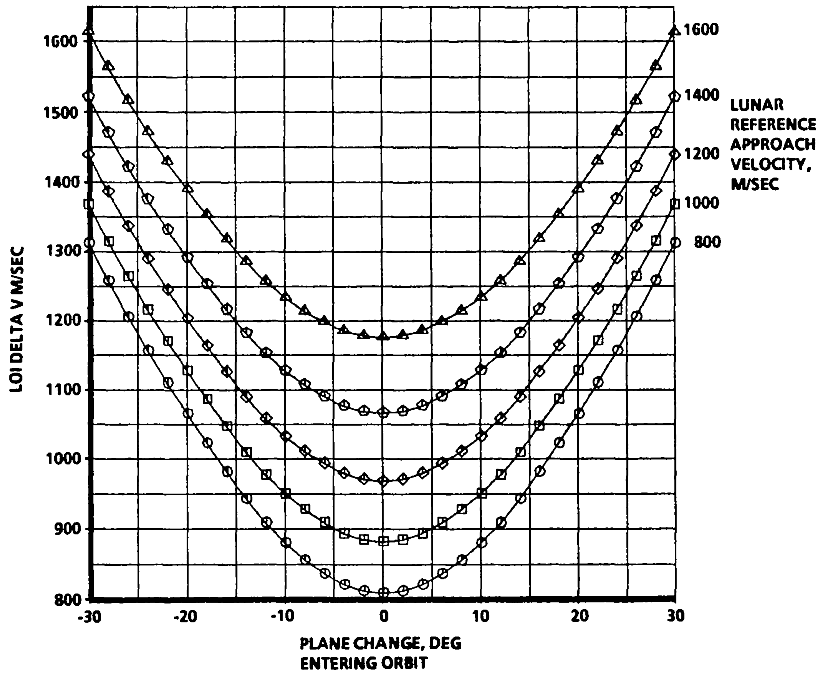

Figure 7. LOI ΔV parametrics.

lunar arrival and departures with very small plane changes, yielding total mission ΔV no greater than typical for an unconstrained mission. The added constraints of the mission can be met by timing and do not impose performance penalties. The timing constraints are, however, stringent; near-optimal missions have windows on the order of a few hours.

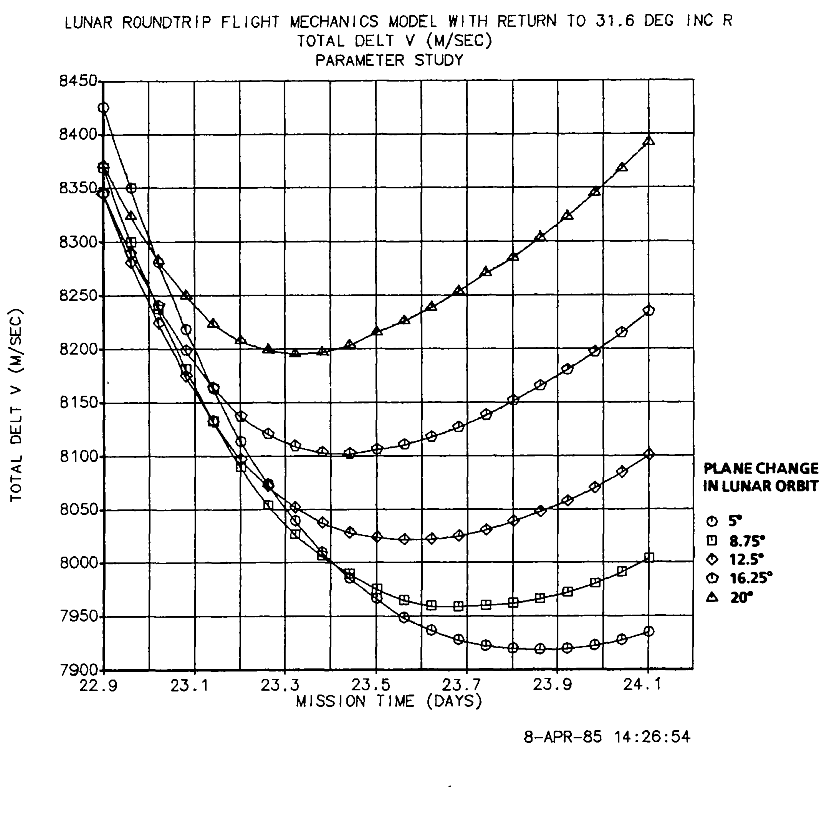

Figure 8. Total mission ΔV (without aeroassist)

Table 1. Delta V Budgets

	Rough	Calculated
TLI	3200 m/s	3139 (includes 75 m/s g loss)
LOI	800 m/s	867
Landing	2100 m/s	2100 (from Apollo)
Ascent	2000 m/s	2000 (from Apollo)
TEI	800 m/s	906
EOI	3200 m/s	3061 (propulsive)
	or 3200 m/s	200 (aeroassisted)
Total	12100 m/s	12073 (propulsive)
	or 9100 m/s	9212 (aeroassisted)

A figure of 915 m/s for LOI/TEI is frequently given in the literature. This value was used in Apollo ΔV budgets constrained to free-return Earth-Moon trajectories; *i.e.,* the lunar vehicle would swing around the Moon and return to Earth without propulsion in the lunar vicinity.

(Note: it is probably the case that very small plane changes leaving or returning to Earth orbit would alleviate the time constraints.)

Mission Modes

Mission modes were screened using the rough ΔV budget presented in Table 1. The rough budget is compared to typical calculated values from the computer analysis. Base buildup and resupply needs are summarized in Tables 2 and 3. It is clear that delivery capabilities of about 13 metric tons of payload (unmanned) for base hardware deliver and 5 tons (manned) for resupply operations are needed. The resupply scenario presumed a lunar base crew of 12, lunar staytime of 164 days, and resupply mission with 4 crew exchanged every 55 days. A conceptual design analysis of the lunar transfer crew cab requirements concluded that an adequate crew cab could be derived from a space station common module and its subsystems and that the mass including crew will be approximately five tons.

Table 2. Base Equipment Delivery Requirements

Flights 1 and 2	Habitat modules
Flights 3 and 4	Laboratory/Work modules
Flight 5	Construction equipment
Flight 6	Nuclear power plant
Flight 7	Solar/Regen fuel cell emergency power supply
Flight 8	Scientific equipment
Flight 9	Mobile explorer
Flight 10	Lunar oxygen production plant

Each flight has cargo capacity of 13 tons (28,500 pounds).

Table 3. Base Resupply Summary

Crew staytime 180 days	
Exchange interval 4 crew/60 days	
Resupply (Per 60 Days)	
Food, water and atmosphere	3.30 Tons
EVA	0.27 Tons
Science	0.22 Tons
Equipment and Subsystems	1.21 Tons
	5.00 Tons (11,000 pounds)
Crew transport module (4 people)—5.0 tons	
Net delivered—10	
Net returned—5	

We selected mission modes that could use orbital transfer stages similar in size and general configuration to the reusable orbit transfer vehicles (ROTVs) now under study for geosynchronous operations, while at the same time providing useful payload for lunar base support and buildup operations. A number of modes were considered, and those shown in Fig. 9 were selected for further analysis. The cargo and independent lunar surface sortie (ILSS) modes are appropriate to base buildup and early operations, as they do not depend in any way on lunar resources. If useful quantities of liquid oxygen can be produced on the lunar surface, the surface-refueling and orbiting lunar station (OLS) modes are attractive. These latter modes are fully reusable; the others are not.

The direct cargo mode offers payload adequate for base buildup with a simple mode that does not depend on lunar resources but that expends a stage. The ILSS mode is the most direct way of providing a crew round trip without lunar resources. The ILSS mode also expends a stage, the lunar lander, and its crew cab. The stage could be recovered, but this would require additional propellant to be delivered to lunar orbit and the propellant to be transfered to the lander after its return from the lunar surface. If the mode were to be used extensively, stage recover might be cost effective.

The surface refuel mode uses the lunar surface rather than an orbit as a staging point, and it is not constrained by lunar orbit selection factors. Its lunar stage is a somewhat more complex design, requiring lunar landing legs and an aerobrake on the same stage. The OLS mode is by far the most efficient as measured by payload versus delivery mass to low-Earth orbit. It does require propellant transfer operations in zero g, as the lander stage must be loaded with hydrogen from Earth and the Earth return stage loaded with lunar oxygen for the TEI and EOI maneuvers.

A nominal rocket engine specific impulse of 460 s (jet velocity of 4511 m/s) was used to compute stage sizes and propellant quantities for each burn of the mission profiles. Aeroassist was assumed for Earth return. Stage propellant loads cluster around 25 metric tons of hydrogen and oxygen at a mixture ratio of 6. This compares favorably with the 20–25 tons typically chosen for a GEO orbit ROTV. These stages are compatible with

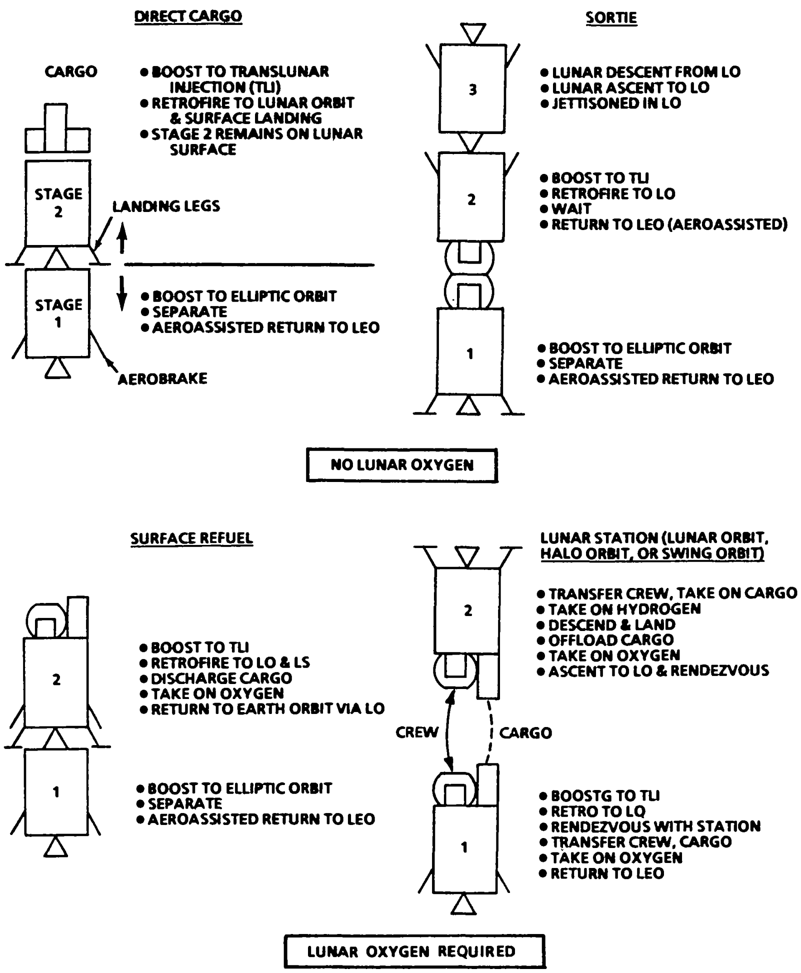

Figure 9. Comparison of lunar mission modes.

space station ROTV accommodation concepts, although the assembled vehicles may not be compatible unless provisions for handling such assemblies are provided. A summary of results from the mode analysis is presented in Table 4. The cost indicators in Table 4 include costs for Earth-to-orbit transportation and use/expenditure of lunar transport hardware but not indirect costs such as mission control operations. Benefits of lunar oxygen and of using a shuttle-derived cargo vehicle to deliver propellant to the space station for lunar operations are apparent.

Mode Performance and Sensitivities

The ILSS mode and three-stage vehicle system might be the first used to validate potential basing sites by manned visits before delivery of base hardware. It might also be used for base resupply operations until lunar oxygen production is validated and operational. Accordingly, performance and sensitivity analyses were made for this mode.

Two crew cabs are used, one for the lunar orbit and one for the lander. This avoids disrupting interfaces between a crew cab and its propulsion stage. For these calculations, the lunar lander and its crew cab were assumed jettisoned in lunar orbit to reduce Earth departure mass.

Table 4. Lunar Mission Mode Performance and Cost Screening Summary

Mode Name	Booster Stage Propellant Load (Metric Tons)	Lunar Orbit State Propellant Load	Lander State Propellant Load	Payload Delivered/ Returned	Propellant from Earth (Metric Tons)	Lunar Oxygen (Metric Tons)	Stages Expended	Crew Cabs Expended	Shuttle Transport Cost at 85m/Flt 26T-Payload)	CLV Transport Cost at 75m/Flt (60T-Payload)	Lunar Transport Vehicle Cost	Total Mode Cost With Shuttle	Total Mode Cost With CLV
Direct cargo	24	Not req'd	25	13/0	50	None	1	0	218	83	55	273	138
ILSS	25	23	12	1/0.5 + Crew round trip	59	None	1	1	236	91	115	351	206
Orbiting lunar station (OLS)	Not req'd	28	20	5/0.5+ Crew round trip	28	18	0	0	106	41	20	126	61
Surface refuel	24	Not req'd	24	5/0.5+ Crew round trip	48	8	0	0	191	75	15	206	90

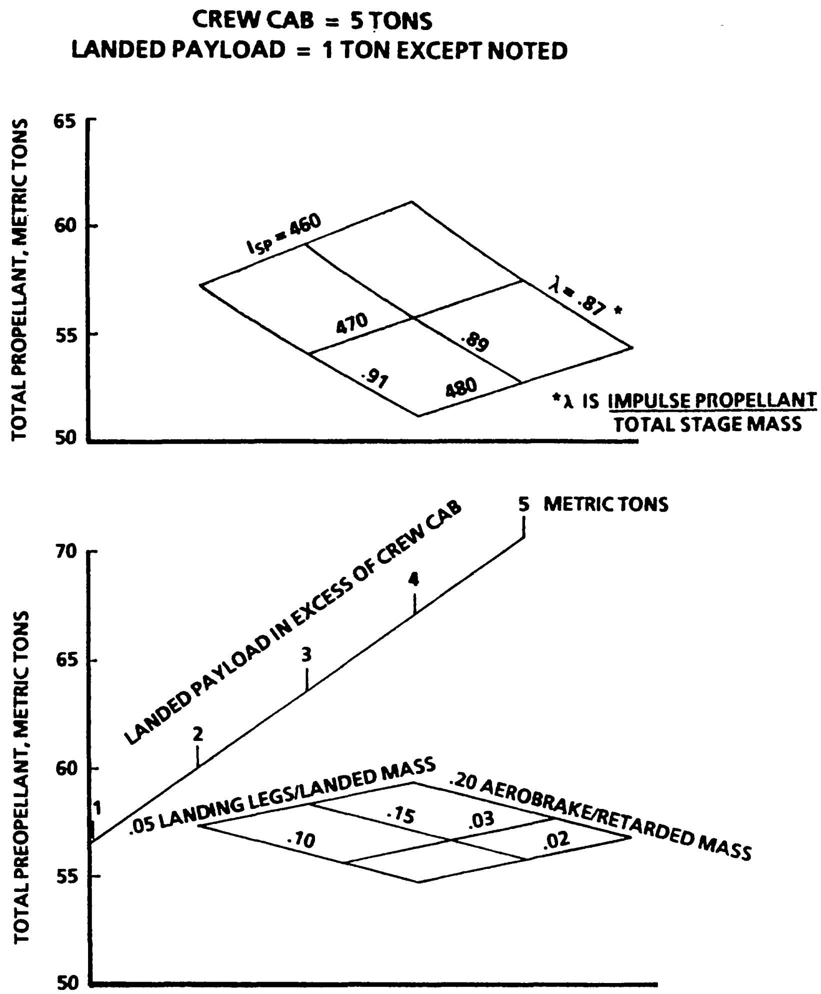

Figure 10. Sensitivities of ILSS modes.

Approximate sensitivities of the aeroassisted ILSS mode are shown in Fig. 10 for delivered payload, stage inert mass (defined in terms of propellant fraction), rocket specific impulse, and landing leg and aerobrake mass fraction. The mode is not highly sensitive; the others are even less so. "Not highly sensitive" means that the total propellant and mass variations are comparable to the uncertainties in design variables; the mode does not "go over a cliff" in the presence of uncertainties in typical preliminary design estimations.

Aeroassisted ILSS mode performance was also calculated by attaching a vehicle-sizing model to the lunar mission ΔV model described earlier. In these coupled models, vehicle performance and sizing calculations are performed for each ΔVs. Results are shown in Fig. 11. The total mass is greater than that shown in Fig. 10 because (1) delivered payload was 5000 kg; (2) accurate ΔVs are slightly higher than the rough values used for screening comparison; and (3) vehicle flight performance reserves were included. Results gave added confidence that the ILSS mode is a good candidate for an early manned return to the Moon.

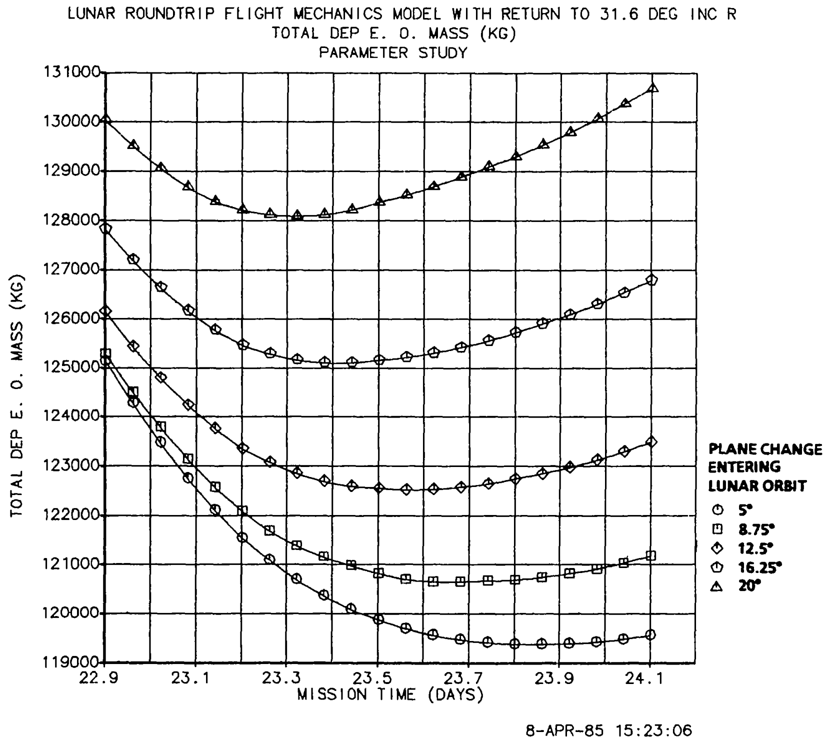

Figure 11. Earth orbit departure mass.

CONCLUSIONS

Contemplation of a permanent manned lunar base in our space future should not be inhibited by the $80-billion costs of the Apollo program in present-year dollars. Few new developments are needed, and these are consistent with current paths of technology evolution. The transportation operations cost of supporting a modest lunar base can evolve to less than a billion per year in 1985 dollars.

Synergistic cost benefits accrue from cryogenic space propulsion, aeroassisted ROTV operations (the benefit for return from the Moon is greater than for return from GEO orbit), the space station as an intermodal transportatoin complex, a heavy-lift shuttle, and oxygen production on the Moon.

Mission designs can accommodate flight mechanics constraints in such a way as to permit repetitive operations between particular Earth and lunar orbits with negligible performance penalties.

Efficient mission modes can use stages in the ROTV-size range; these can be ROTVs or simple derivatives thereof. The space station can provide practical spaceport operations services for lunar operations.

A logical evolution exists from site exploration through buildup to support operations. All can use common transportation hardware and technology. An evolutionary step to very efficient resupply operations through lunar-produced oxygen is a natural beginning for the practical use of extraterrestrial resources.

IMPACT OF LUNAR AND PLANETARY MISSIONS ON THE SPACE STATION

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The report examines the impact of several advanced planetary missions and a populated lunar base on the growth space station. Planetary missions considered include a Mercury Orbiter, a Saturn Orbiter with Multiple Titan Probes, and sample return missions from Mars, the comet Kopff, and the main belt asteroid Ceres. A manned lunar base buildup scenario is defined—from preliminary lunar surveys through 10 years of construction, to the establishment of a permanent 18-person facility with the capability to produce oxygen to be used as propellant. For the lunar base, the space station must hangar at least two OTVs, store 100 metric tons of cryogens, and support an average of 14 OTV launch, return, and refurbishment cycles per year. An average of 630 metric tons per year must be launched to the space station for lunar base support during the 10 years of base construction. Approximately 70% of this cargo from Earth is OTV propellant. An Unmanned Launch Vehicle (ULV) capable of lifting 100 metric tons net useful payload is considered necessary to deliver this propellant. An average launch rate of one shuttle every two months and one ULV every three months to the growth space station will provide the required 630 metric tons per year. Planetary sample return missions require a dedicated quarantine module.

INTRODUCTION

NASA and contractors are now working on the conceptual design of the Low Earth Orbit (LEO) space station. For the Initial Operational Capability (IOC) space station configuration to evolve smoothly, the designers must consider the requirements of the turn of the century "growth" space station. This study, performed by Eagle Engineering, Inc., for the Johnson Space Center (JSC) Solar System Exploration Division, examines the effects of advanced lunar and planetary missions upon the growth space station.

Mass estimates were developed for a lunar base with an emphasis on science, which, in its later phases of development, uses oxygen produced on the Moon in a transportation system sized to land its elements on the lunar surface. A 10-year flight schedule was developed, including weights, propellants, crew size, *etc.* The impact of such an undertaking upon the space station was then estimated.

Similarly, five advanced planetary missions were examined, as were three sample return missions, and two orbiter/probe missions. Weight statements and trajectories for each of the missions were determined. For this effort a NASA-developed standard OTV (Orbit Transfer Vehicle) was used. Propellant loads, configurations, and mission plans were determined. Some of these missions were found to require two OTVs in a two-stage stack. The effect on the space station of conducting these missions was then estimated. The final report of the study (Babb *et al.*, 1981) provides details and the full methodology.

Ground Rules and Assumptions

The OTVs used in this study are based upon a report by Scott *et al.* (1985). They are sized to deliver 9 metric tons one way or 6 metric tons round trip to Geosynchronous Orbit (GEO) using a single stage and to deliver 18 metric tons of lunar surface payload plus a lander to lunar orbit using two OTV stages in tandem. Both OTV stages then return to the space station.

For this study a mature aerobraking technology is assumed. All stages are to use LO_2/LH_2 with the exception of expendable ascent stages, which use storables.

LUNAR MISSIONS

The transportation operations required for buildup and support of a lunar base were examined using the space station and a fleet of large OTVs as the major transportation elements.

Lunar Base Description

The lunar base model examined was supplied by NASA/JSC (B. Roberts, personal communication, 1984) as an example of a research installation with an emphasis on science and some lunar materials utilization. With this base model, lunar oxygen production starts in the fourth year. The oxygen is provided for a Reusable Lunar Lander/Launcher (R-LEM) so that only hydrogen fuel needs to be brought from Earth.

At the end of the first 10 years, the lunar base staffed by 18 permanent personnel includes:

- Five habitability modules
- Five research units: a geochemical laboratory, a chemical/biological laboratory, a geochemical/petrology laboratory, a particle accelerator, and a radio telescope
- Three production plants (preceded by pilot plants): oxygen, ceramics, and metallurgy
- Two shops
- Three power units
- One earthmover/crane
- Three mobility units and trailers

Lunar Base Support Requirements

This study focuses on the base buildup, starting in the year 2005. The eight years preceding the buildup of the base are devoted to unmanned exploration and mapping of the Moon. One landing of a roving vehicle per year is required and could be flown directly from the shuttle with a modified Centaur-class vehicle.

There are 25 base elements that average 17.5 metric tons each. Figure 1 shows delivery of a habitability module. They are delivered over a period of 10 years for a total of 465 metric tons. An additional 233 metric tons of miscellaneous cargo is delivered

Figure 1. Unloading module on lunar surface.

during the manned resupply missions. Launch manifest and lunar mission schedules were derived for the entire 10-year buildup period. Table 1 shows the detailed schedule for the first year (2005).

Lunar Transportation System

The elements of the transportation system are as follows.

1. Aerobraking Orbital Transfer Vehicle (AOTV)—A 49 metric ton gross mass LO_2/LH_2 propulsion stage (42 metric tons of propellants).

2. Expendable Lunar Lander (E-Lander)—A LO_2/LH_2 landing stage with 13.6 metric tons of propellant that will land 17.5 metric tons.

3. OTV Manned Module (OMM)—A 5.5 metric ton orbit-to-orbit reusable crew transport module with four personnel to be carried on the OTV.

4. Lunar Landing Manned Module (LLMM)—A 3.25 metric ton expendable module for temporary life support of four crew members during lunar landing and launching. It is attached to a 7.6 metric ton expendable launcher.

5. Reusable Lunar Lander/Launcher (R-LEM)—A 5 metric ton LO_2/LH_2 single-stage vehicle using lunar-produced O_2 for propellant.

6. Reusable Lunar Landing Manned Module (R-LLMM)—A 5 metric ton, six man, lunar-based crew compartment for the R-LEM. This crew compartment is maintained and stored at the lunar base.

Table 1. Detailed Launch Manifest and Lunar Mission Schedule

Month	Launch No.	Type	Cargo Manifest	Cargo Wt.*	Lunar Flight No.	Flight Type	Space Station Tasks for Flight	LOX/LH2 at Depot*
				Year 2005				
Jan	5-1	SD-ULV	LOX/LH2 propellant supply unit	100				100
Feb	5-2	STS-ACC	E-lander, +base element #1	21	L5-1	Unmanned Delivery	Prepare (Check out & fuel) 2 OTVs and E-lander; Check out cargo; & mate stack; -(2 OTVs, E-lander, and base element)	0
March	5-3	SD-ULV	LOX/LH2 propellant supply unit	100				100
April	5-4	STS-ACC	E-lander, +base element #2	21	L5-2	Unmanned Delivery	Prepare (Check out & fuel) 2 OTVs and E-lander; Check out cargo; & mate stack; -(2 OTVs, E-lander, and base element)	0
May	5-5	SD-ULV	LOX/LH2 propellant supply unit	100				100
	5-6	STS-ACC	E-lander, +base element #3	21	L5-3	Unmanned Delivery	Prepare (Check out & fuel) 2 OTVs and E-lander; Check out cargo; & mate stack; -(2 OTVs, E-lander, and base element)	0
July	5-7	SD-ULV	LOX/LH2 propellant supply unit	100				100
Aug	5-8	STS-ACC	E-lander, +base element #4	21	L5-4	Unmanned Delivery	Prepare (Check out & fuel) 2 OTVs and E-lander; Check out cargo; & mate stack; -(2 OTVs, E-lander, and base element)	0
Sept	5-9	SD-ULV	LOX/LH2 propellant supply unit	100				100
Oct	5-10	STS-ACC	E-lander, +E-LLMM/Ascent, +OMM, +4 crew +2 ton PL	20	L5-5	Manned Sortie	Prepare 2 OTVs and E-lander; Check OMM, E-LLMM/Ascent; mate stack -(2 OTVs, OMM, E-LLMM/Ascent, & E-lander); And transfer crew to OMM	100 0
Dec	5-12	STS-ACC	E-lander, +E-LLMM/Ascent, +OMM, +4 crew +2 ton PL	20	L5-6	MS-(4M +2t for 14 days)	Prepare 2 OTVs and E-lander; Checkout OMM, E-LLMM/Ascent; mate stack -(2 OTVs, OMM, E-LLMM/Ascent, & E-lander); And transfer crew to OMM	100 0

Year 2006

Jan	6-1	SD-ULV	LOX/LH2 propellant supply unit	100				100
Feb March	6-2	STS-ACC	E-lander, +base element #5	21	L6-1	Unmanned Delivery	Prepare (Check out & fuel) 2 OTVs and E-lander; Check out cargo; & mate stack; -(2 OTVs, E-lander, and base element)	0
April	6-3	SD-ULV	LOX/LH2 propellant supply unit	100				
May June	6-4	STS-ACC	E-lander, +E-LLMM/Ascent, +OMM, +4 crew +2 ton PL + 4 tons of AOTV elements	20	L6-2	MS-(4M +2t for 14 days)	Prepare 2 OTVs and E-lander; Check OMM, E-LLMM/Ascent; mate stack -(2 OTVs, OMM, E-LLMM/Ascent, & E-lander); And transfer crew to OMM	0
July	6-5	SD-ULV	LOX/LH2 propellant supply unit	100				100
Aug Sept	6-6	STS-ACC	E-lander, +base element #6	21	L6-3	Unmanned Delivery	Prepare (Check out & fuel) 2 OTVs and E-lander; Check out cargo; & mate stack; -(2 OTVs, E-lander, and base element)	0
Oct	6-7	SD-ULV	LOX/LH2 propellant supply unit	100				100
Nov	6-8	STS-ACC	E-lander, +E-LLMM/Ascent, +OMM, +4 crew +2 ton PL + 4 tons of AOTV elements	20	L6-4	MS-(4M +2t for 14 days)	Prepare 2 OTVs E-lander; and OMM; Checkout E-LLMM/Ascent; Mate Stack -(2 OTVs, OMM, E-LLMM/Ascent, & E-Lander); And transfer crew to OMM	0

*in million tons

7. Large OMM—An enlarged, 8 metric ton, reusable crew transport module for six personnel, carried on an OTV.

8. H_2Transfer Tank—A 1 metric ton expendable container for carrying 4 metric tons of LH_2 to the lunar surface as fuel for the R-LEM.

9. Orbital Maneuvering Vehicle (OMV)—A small, 3 or 4 metric ton, remotely operated propulsion stage to provide controlled close-in operations at the space station.

Earth Launch Vehicles

The Earth launch vehicles include the following.

1. Space Shuttle (STS)—The space shuttle can launch 25 metric tons of crew and payloads to the space station (about 50% improvement over STS capabilities cited in the recent NASA literature).

2. Shuttle/Aft-Cargo Carrier—Shuttle with a cargo compartment on aft end of the External Tank. This allows the launch of oversized vehicles. It is used for launching E-Landers.

3. Shuttle-Derived Unmanned Launch Vehicle (SDULV)—Unmanned launcher, which was designed using shuttle elements, can deliver 100 metric tons of LO_2/LH_2 to the space station propellant depot. It is used for launching all cryogenic propellants.

Figure 2 shows a pair of OTVs with an E-Lander and an unmanned base element as they debark from the space station orbit. An OMV is shown returning to the space station after having moved the "stack" to a safe distance.

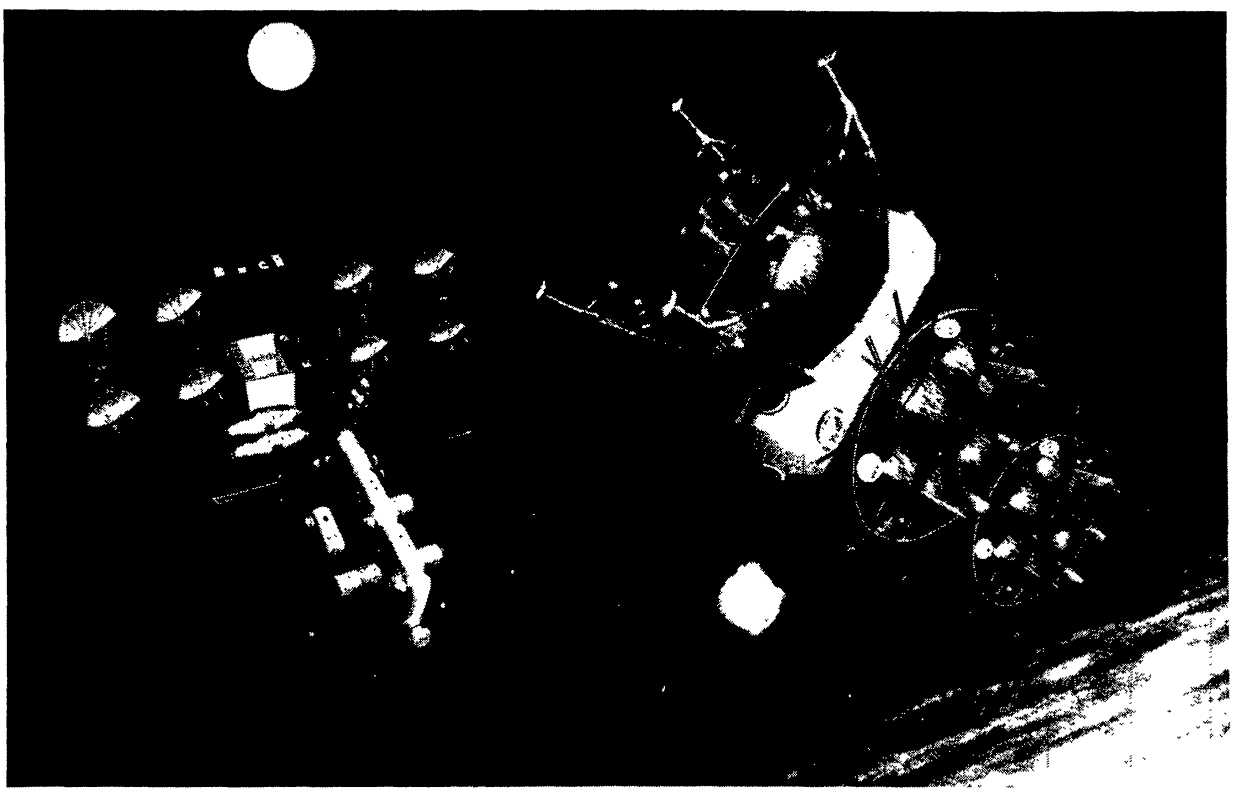

Figure 2. OTV departing space station with lander and module.

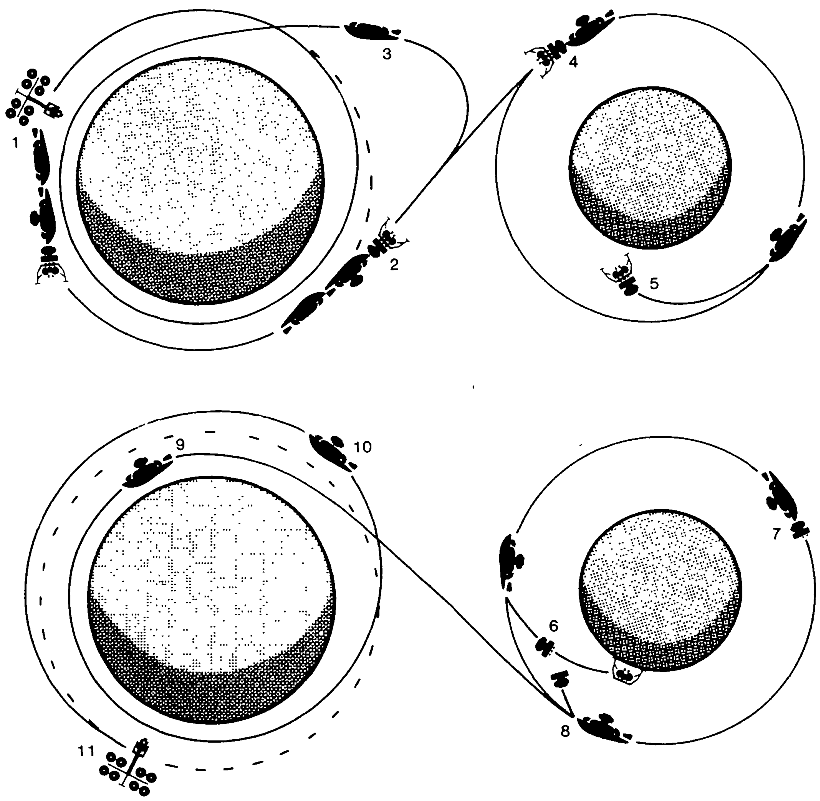

Figure 3. Manned lunar mission scenario. 1-Stack departs space station; 2-trans-lunar injection burn; 3-first stage returns to space station; 4-second stage, lander, and manned module insert into lunar orbit; 5-lander descends; 6-ascent stage departs lunar surface; 7-ascent module rendezvous with second stage; 8-second stage returns to Earth with OMM, ascent module discards; 9-aerobraking; 10-circularization above space station orbit; 11-rendezvous with space station.

Figure 3 depicts the flight scenario for a manned lunar flight. The unmanned flights are similar except that no cargo elements are left in orbit with the OTV, and none return from the lunar surface. The OTV returns to Earth empty.

The Earth launch tonnage requirements to the space station from Earth for the lunar base over the 10-year buildup are shown in Fig. 4. The LO_2/LH_2 tonnage requirements shown include the propellant for both the OTV and the lunar lander. Such propellant is only the portion (of the total required) that would be launched by the SD-ULV to the Low Earth Orbit propellant depot.

IMPACT OF THE LUNAR MISSIONS ON THE GROWTH SPACE STATION

The space station must provide propellant storage and transfer facilities (propellant depot), the capability for assembly of the mission stack, facilities for payload checkout and integration into mission stacks, maintenance and checkout of vehicles stored on-orbit (OTVs, OMVs, OMMs), flight control (rendezvous, proximity operations, and docking), personnel billeting, and temporary payload storage.

Hardware required to be added to the growth space station includes:

1. Permanent basing (hangars, storage, and shops) for four OTVs, three OMMs, and two OMVs.

2. Gantrys for preparing mission stacks of up to 40 m length of two OTVs, plus a lunar lander, plus various manned and unmanned lunar cargó elements.

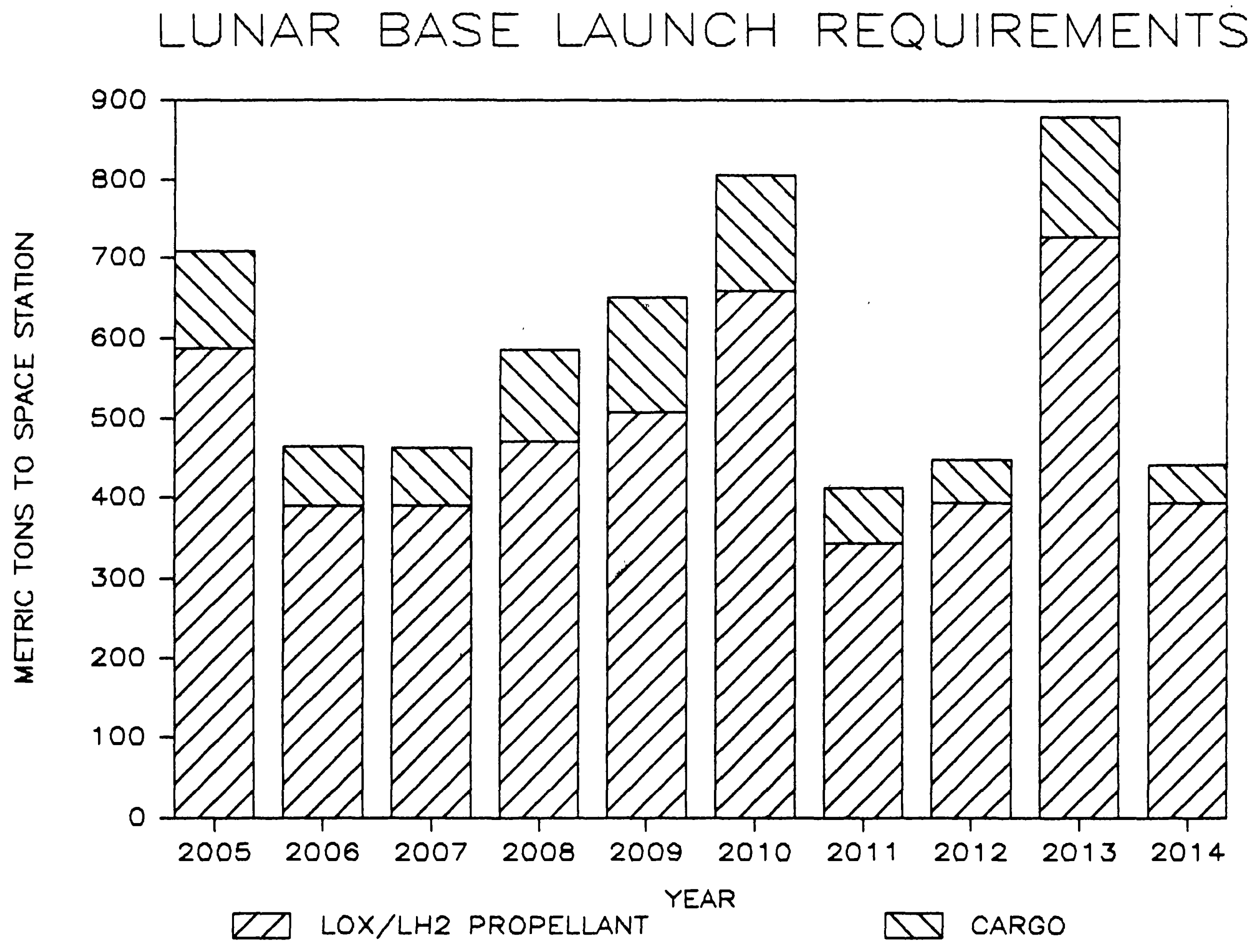

Figure 4. Lunar base launch requirements.

3. A propellant depot for cryogenic LO_2/LH_2 propellant with the capacity of at least two tanker units of 100 metric tons each.

4. A propellant transfer capability to perform a measured propellant transfer from the depot to various vehicles in the mission stack at the assembly docks. A rate of 5 tons per hour is required to complete transfer in one 24-hour period.

5. Temporary storage for lunar vehicles and 20-30 tons of lunar payload.

6. An additional habitat module for housing the additional space station crew and temporary billeting of 4-6 transient lunar base personnel.

7. An estimated 20 kW of continuous *additional* power with appropriate heat rejection. This power budget breaks down into 10 kW for depot cryogenic refrigeration, 5 kW minimum for the extra habitat, and 5 kW or more for gantrys.

Manpower requirements identified at the space station are 14 man-weeks per lunar sortie. Five of these man-weeks will be required for general OTV turnaround and maintenance (Maloney *et al.*, 1983). An estimated five man-weeks will be required for stacking the lander and the manned module and for fueling the lander. It is assumed that an extensive checkout will be performed on the completed stack. One man-week will be dedicated to manned module refurbishment outfitting and checkout. Finally, three man-weeks are estimated for traffic control and OMV operations. These operations include one shuttle arrival for cargo and crew transfer, one ULV arrival for propellant, one ULV propellant tank disposal, one stack departure, and two separate OTV arrivals. These operations require a minimum extra crew complement of two persons and possibly another pair for other required tasks not yet identified.

Since an average of one lunar sortie will be required every eight weeks, two dedicated crew members will probably be required. Less complex unmanned lunar and planetary missions will require less manpower.

All of the required space station capabilities need to be online at the beginning of the lunar base buildup in 2005. Capabilities will have to be developed earlier and procedures learned from geosynchronous mission preparations or some of the more difficult unmanned planetary missions.

A brief sensitivity study showed that reasonable changes in I_{sp} and inert weight do not alter the scale of the operation to first order. An I_{sp} change of 20 seconds combined with a 20% decrease in vehicle inert weights can reduce the weight in Low Earth Orbit (LEO) from 10–20% (mostly LO_2/LH_2 propellant), a reduction of about one unmanned launch of propellant per year.

A brief examination of planetary mission launch from lunar orbit using lunar produced oxygen for propellant found no great advantage over launch from LEO.

PLANETARY MISSIONS

A set of missions was examined for impact to the growth space station. They were chosen to show how the space station with reusable OTVs might enable more ambitious planetary exploration and to see how the use of this infrastructure for planetary exploration would affect the growth space station. This set of missions is an example set and not a proposed addition to the NASA Plantary Exploration Program (1983).

Table 2. Planetary Missions—Performance Summary

	C3 (km/s^2)	Type of OTV*	Payload out of LEO†	LEO Total Departure Mass†	OTV Propellant Load†	Propellant + Payload (Lift Req.)†
Mars Sample Return	9.0	1 Stage Reusable	8.89	44.03	27.76	36.65
Kopff Sample return	80.7	2 Stage, 1st Stage Returns	8.38	92.49	71.51	79.89
Ceres Sample Return	9.9	2 Stage, 1st Stage Returns	43.57	131.59	75.47	119.04
Mercury Orbiter	18.7	1 Stage Reusable	5.63	41.62	28.90	34.53
Titan Probes/Saturn Orbiter	50.5	1 Stage Expendable	6.34	53.54	41.81	48.15

*Isp = 455.4 seconds, all stages have a total propellant capacity of 42 metric tons. A = 3,731 kg, B = .0785. Stages that do not return have the aerobrake removed.
†In metric tons

All of the example missions used the delta V tables and detailed weight statements for spacecraft to the subsystem level that can be found in Babb *et al.* (1981). For the planetary missions considered, Table 2 lists the C3's and gives the weight breakdowns in terms of payload out of LEO, LEO OTV propellant load, and propellant plus payload (life required).

Mars Sample Return (MSR) Mission (November 1996 Launch)

The Mars sample return mission was the most complex one studied. The general mission plan and orbital mechanics data were taken from Bourke *et al.* (1984, 1985) and Feingold *et al.* (1982).

The space-based aerobraked OTV will load 27.76 metric tons of LO_2/LH_2 into its 42 metric ton capacity tanks and return to the space station after releasing the spacecraft. Figure 5 shows the spacecraft within its aeroshell during mating with the OTV. Figure 6 depicts the mission sequence. The returned sample is to be retrieved from Earth orbit with an OMV and brought to the space station.

Table 3 summarizes the impact of the various planetary missions on the space station/OTV system. The major impact is that the Quarantine Module is now required to provide a place to receive the returned samples. Each of the sample return missions is assumed to use the Quarantine Module, which is discussed in more detail following the sections on individual missions.

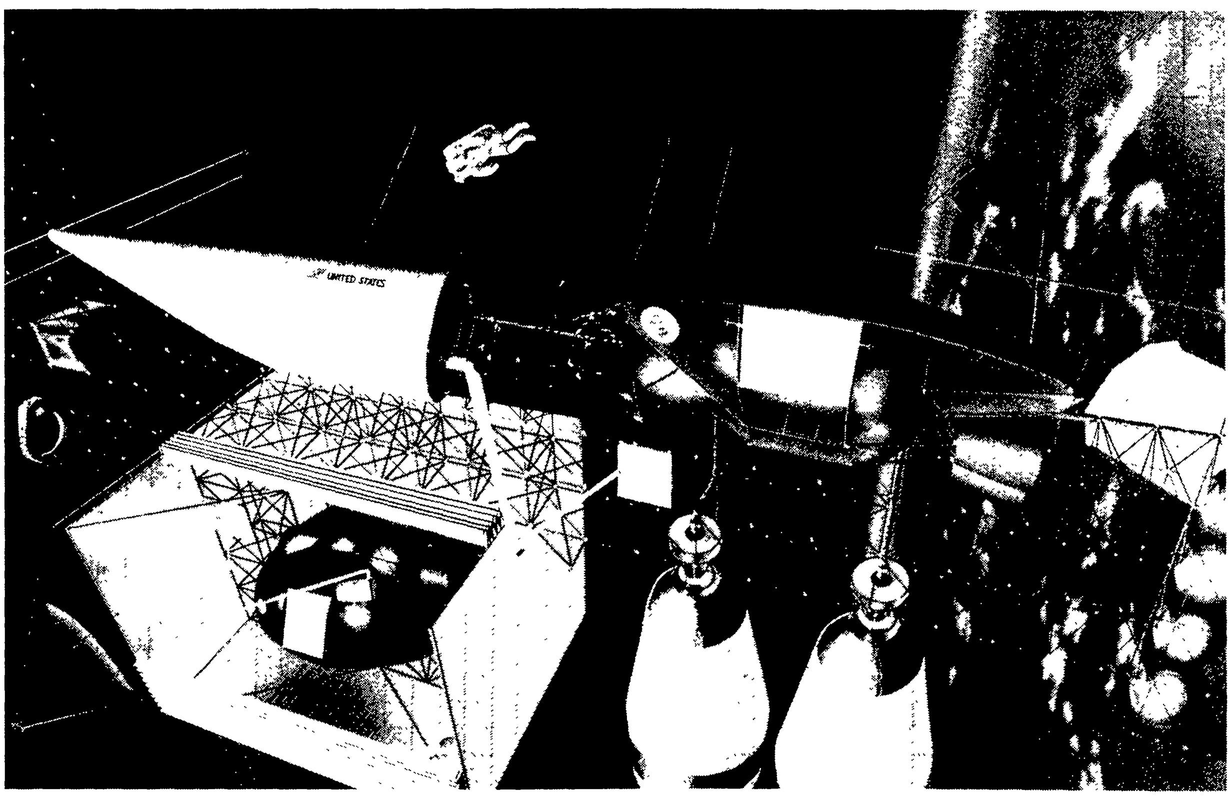

Figure 5. Mars sample return mission OTV mating.

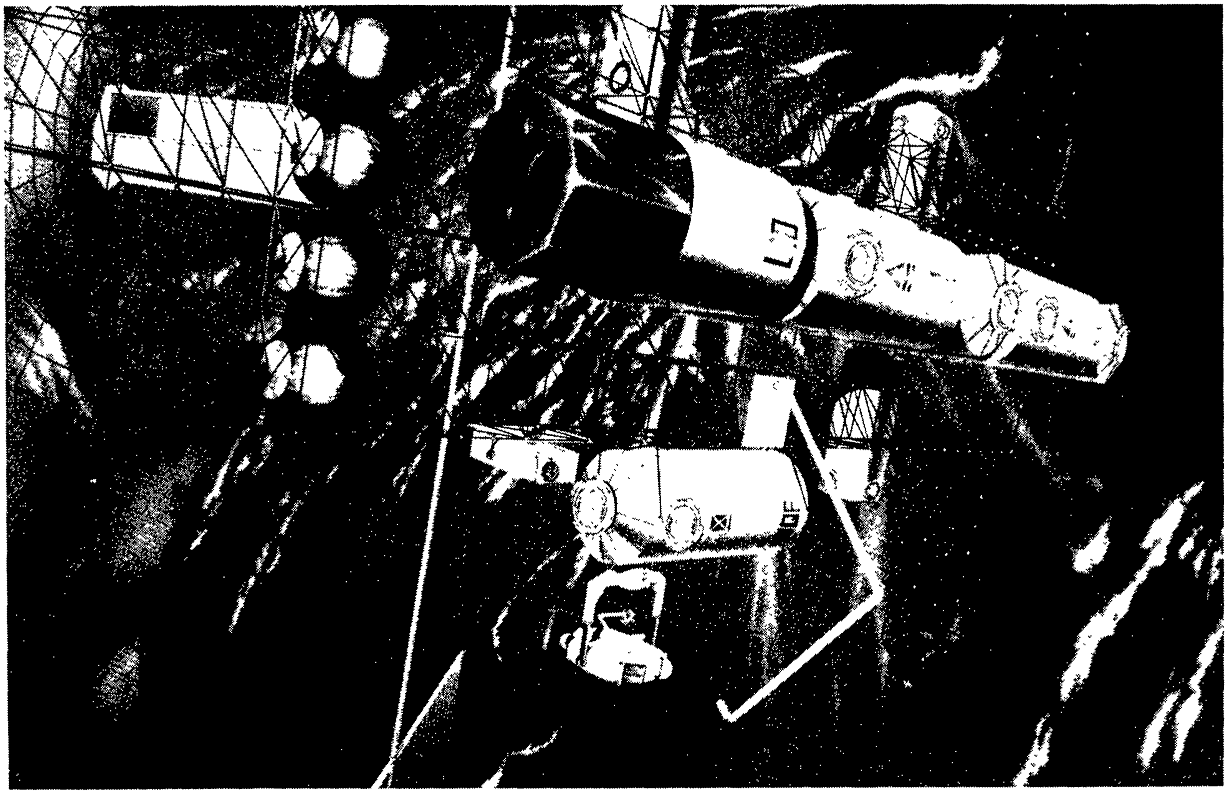

Figure 6. OMV returns sample to quarantine module.

Comet Kopff Sample Return Mission (July 2003 Launch)

This mission was complex, but not as well studied as the MSR mission. The general mission plan was taken from Feingold *et al.* (1983) and Draper (1984). As Table 2 shows, a stack of two standard OTVs is required to take this Mars sample return size payload to a C3 of 80.7 $(km/s)^2$. In the mission scenario depicted in Figure 7, only the first stage returns to the space station. The Mariner Mark II (MMII) spacecraft carries the samplers out and back (Feingold *et al.*, 1983; Draper, 1984). The returned sample uses aerobraking to enter Earth orbit and is taken to the Quarantine Module by an OMV.

This mission requires the Quarantine Module, and in addition, the capability to stack and checkout two OTVs. The aerobrake is removed from the second stage, which does not return. An older OTV, near the last of its estimated 10 or so missions might be used for this stage.

Ceres Sample Return Mission (October 1994 Launch)

This mission is similar to the Kopff mission previously discussed. The mission sequence, depicted in Figure 7, is essentially the same as that of the Kopff sample return mission. The trajectory uses a double Mars-gravity-assist outbound and is ballistic inbound. Both of these missions use the MMII spacecraft for the outbound and inbound legs. The returning Mariner spacecraft and the large delta V required for a ballistic return on the Ceres mission leads to a particularly large weight penalty. A substantial reduction in Earth launch weights

Table 3. Planetary Missions—Impacts on the Space Station

Requirements	Mars Sample Return	Kopff Sample Return	Ceres Sample Return	Mercury Orbiter	Titan Probes/ Saturn Orbiter
Space Station Hardware					
No. of OTV's expended (not returned)	0	1	1	0	1
No. of OTV refurb. kits	1	2	2	1	1
Gantry to stack two stages		yes	yes		
Check out equip. for two stage stack		yes	yes		
Quarantine module	yes	yes	yes		
Additional power, KW	5	5	5		
Additional thermal control, no. of standard modules	1	1	1		
Space Station Manhours					
OTV refurbishment	52	103	103	52	52
Aerobrake removal		21	21		21
OTV/payload integration & C/O	11	21	21	11	11
Fuel, release, and launch	24	36	36	24	24
Rendez./retrieve OTV using OMV	12	12	12	12	
Shuttle rendez./payload removal	3	2	12	2	2
ULV fuel delivery	7	17	18	7	10
Sample retrieval using OMV	8	8	8		
Sample analysis and shipment	24	16	16		
Total mission manhours	138	236	247	106	119

might be achieved by designing dedicated spacecraft for these missions. On the other hand, expending an old OTV, particularly if the capability to stack two stages already existed for other purposes, might be more cost effective. More study of this mission is required to choose the most reasonable solution.

The impact of the Ceres sample return mission on the growth space station configuration, shown in Table 3, is essentially the same as for the Kopff sample return mission.

Mercury Orbiter Mission (June 1994 Launch)

The Mercury Orbiter mission uses a MMII (Draper, 1984) and a dual Venus swingby trajectory (Friedlander *et al.*, 1982). One reusable 42 metric ton capacity OTV uses 28.90 metric tons to launch this mission and return.

One standard mission cycle (payload integration/OTV fuel/checkout/launch/ refurbishment) is required of the space station.

Saturn Orbiter/Multiple Titan Probes Mission (April 1993 Launch)

The Saturn orbiter with multiple Titan probes mission, patterned after the one described by Swenson *et al.* (1984) uses the MMII spacecraft as a carrier and an Earth-gravity-

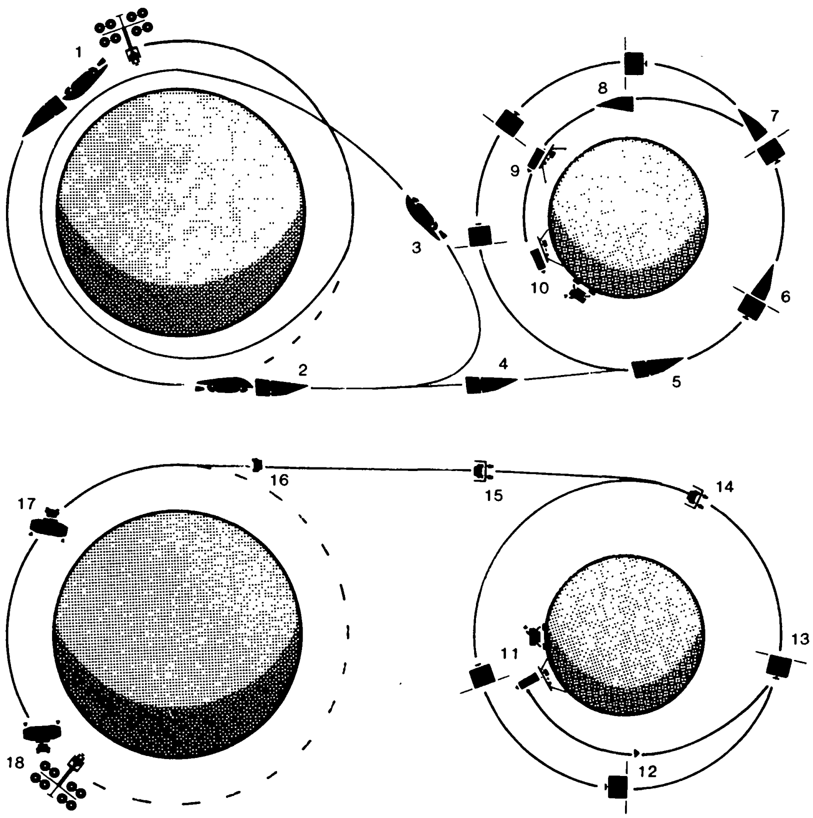

Figure 7. Mars sample return scenario. 1-Stack leaves space station; 2-trans-Mars injection; 3-first stage returns; 4-trans-Mars voyage; 5-aerocapture and Mars orbit insertion; 6-jettison MOV aeroshell; 7-lander and orbiter separate; 8-lander enters Mars atmosphere; 9-landing on martian surface; 10-collect samples; 11-launch from Mars; 12-Mars rendezvous vehicle injection into Mars orbit; 13-Mars orbiter vehicle maneuvers to rendezvous with MRV; 14-trans-Earth injection; 15-trans-Earth voyage; 16-Earth orbit capsule insertion into Earth orbit; 17-OMV rendezvous with EOC; 18-OMV returns EOC with sample to space station quarantine module.

assist trajectory. To make the mission different from the others in terms of its impact on the growth space station, additional probes were added. One standard 42 metric ton propellant capacity OTV can launch the mission, but does not have enough fuel to return. If the expended OTV is an old one and the assembly of a new OTV is not charged exclusively to this mission, then the impact on the growth space station (Table 3) is less than for the Mercury Orbiter. However, the man-hours required to remove the aerobrake from the Saturn Orbiter OTV more than make up for the space station man-hours gained from no retrieval.

QUARANTINE MODULES

Given the existence of a space-based reusable OTV and the capability to stack two of them (which is assumed for this study) the major design impact to the growth space station of planetary missions is the requirement for a Quarantine Module for the sample return missions.

There are several ways to handle returned samples. De Vincenzi and Bagby (1981) discuss a dedicated space station. Another attractive proposal is to environmentally separate one module of the space station from the rest with its own life support, with the module being rigidly attached but not interconnected by pressurized passageways, yet permitting use of station power and cooling. An OMV would deliver a returned sample to an airlock on this module (see Fig. 8) where it would be repackaged, perhaps with a glove box,

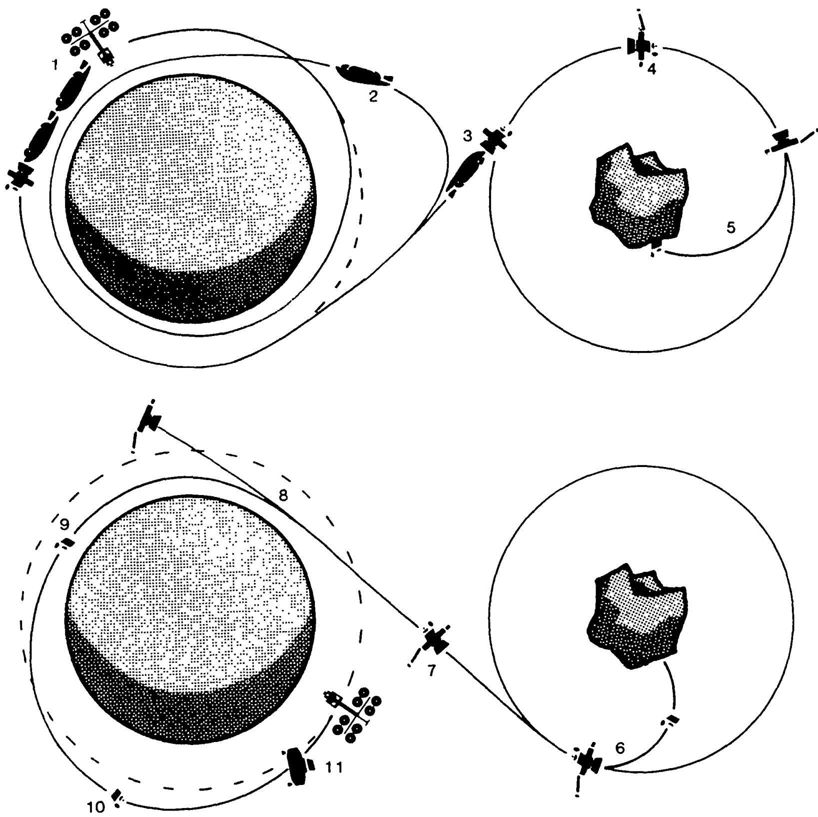

Figure 8. Ceres or Kopff sample return scenario. 1-Stack departs space station; 2-first stage burn, separation and return to space station; 3-second stage burn, trans-Ceres/Kopff voyage; 4-spacecraft rendezvous and asteroid/comet survey; 5-lander on surface, spacecraft in orbit; 6-spacecraft recovers samplers and departs for Earth; 7-trans-Earth voyage; 8-carrier and Earth orbit capsule separate; 9-EOC aerocapture for Earth orbit insertion; 10-circularization above space station orbit; 11-OMV rendezvous with EOC and return to space station quarantine module.

into a "superbox" capable of withstanding any conceivable re-entry accident without releasing material. A small amount of the material might also be examined at the space station. The "superbox" would then be returned to Earth for processing in a facility similar to the present Center for Disease Control (CDC) facility.

SUMMARY

Construction of a lunar base has a major impact on the growth space station. The space station would become much more important as a transportation node than current plans indicate. The planetary missions studied have significantly less impact, with the exception of the addition of a Quarantine Module to the space station. Other options for handling quarantine of planetary samples are now being studied.

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A MOON BASE/MARS BASE TRANSPORTATION DEPOT

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Placement of the next space outpost, after the low-Earth-orbit space station, will strongly affect the evolution of future space programs. The outpost will store rocket fuel and offer a haven to space workers, as well as provide a transportation depot for long missions. Ideally, it must be loosely bound to the Earth, easy to approach and leave, and available for launch at any time. Two Lagrange equilibrium equilibrium points, L_1(SE), between the Sun and the Earth, and L_2(EM), in the Earth-Moon system, have excellent physical characteristics for an outpost; for example, less than 2% additional rocket propellant is required for docking at L_1(SE) on the way to lunar bases or Mars bases. We apply the rocket problem, the two-body problem, and the three-body problem in discussing alternative locations for space depots. We conclude that Lagrange point halo orbits are the standard by which alternative concepts for transportation depots must be gauged.

INTRODUCTION

An evolutionary manned space program will put outposts along routes to places with economic, scientific, and political importance. These outposts will be "filling stations" for storing rocket fuel, warehouses for holding bulk shielding material, assembly plants for building large structures, and transportation depots for connecting with flights to other destinations. Some outposts may produce oxygen and hydrogen from raw materials obtained elsewhere. Each outpost can provide a refuge from solar flare radiation, a hospital for emergencies, and an oasis to those whose missions call for prolonged space travel.

The obvious initial choice for such an outpost is a space station in low-Earth orbit (LEO). LEO marks the first reasonable resting spot in climbing out of the deep potential well of the Earth's gravitational field, for leaving behind the aerodynamic drag of the Earth's atmosphere. And LEO is still within the protection of the Earth's magnetic field so that galactic cosmic rays and lethal solar flares are not a life-threatening hazard to unshielded occupants. A LEO space station will also provide early opportunities to perfect life support systems and conduct physiological experiments. The knowledge gained will promote a better understanding of the problems of engaging people in long-duration space activities.

This first step, the LEO space station, is the largest. Placement of the second step will affect future space programs, including lunar bases, Mars bases, and manned access to Earth's geosynchronous orbit (GEO). The purpose of this paper is to discuss the physics of how to decide where that second outpost in space should be.

LOCATING A TRANSPORTATION DEPOT

Any space habitat beyond the protection of the Earth's magnetic field will require some radiation shielding. The annual biological dose from galactic cosmic rays is about 50 rem (Silberberg *et al.*, 1985); 5 rem per year has been allowed for radiation workers on Earth. In addition, several solar flares per 11-year sun spot cycle would be lethal to astronauts without a radiation "storm cellar" of some type. Although the first few hours of a solar flare may be unidirectional, the radiation, which lasts for a day or two, soon becomes isotropic, so shielding is required on all sides. A transportation depot, therefore, should have easy access to some extraterrestrial source of bulk shielding material such as lunar regolith. The cost of lifting inert material from the surface of the Earth would thus be avoided.

In general, we must consider where extraterrestrial resources will be obtained. If, for example, the main source is the Moon, it will be reasonable to have lunar manufacturing plants and remove only finished products from the surface. If much of the traffic is to and from the Moon, it may be sensible to have a transportation depot there also. However, there is strong evidence that Mars' two moons, Phobos and Deimos, have compositions similar to a carbonaceous chondrite—a type of meteorite that is rich in water and organics (Carr, 1981, p. 200), so we may find that the resources of the Moon, which are quite dry and contain only traces of carbon, and those of Mars' moons will complement the needs of a growing space program. Furthermore, there is a reasonable chance that one of the 73 catalogued Earth-crossing asteroids (Lau and Hulkower, 1985) could supply valuable materials. The same amount of rocket propellant is required to send unmanned freighters from LEO to the surface of Mars' outer moon, Deimos, as to the surface of the Moon, and about 10% less propellant is needed to reach asteroid 1982DB. This argues against placing a transportation depot on any body of substantial gravity, such as the Moon, because each trip to and from the body surface will extract an expenditure of rocket propellant at least equal to that needed to achieve escape velocity.

The many trips to and from a transportation depot will waste fuel and diminish its usefulness if it is poorly situated in space. Consider GEO, for example. Because of its operational significance, a space platform is needed at GEO. However, GEO is about the worst possible place for a transportation depot. More propellant mass is required to insert a rocket payload into circular geosynchronous orbit than to escape the Earth's gravitational field entirely (Cornelisse *et al.*, 1979, p. 396). In addition, the return trip to the Earth from GEO requires more propellant mass than from nearly any other orbit radius (Taff, 1985). Furthermore, the geosynchronous radius of 42,240 km (6.63 Earth radii) is at the outer edge of the Van Allen radiation belt and at the inner edge of the geomagnetic tail (Gosling *et al.*, 1984, p. 46), a location that may necessitate considerable radiation shielding for people stationed there. Looking beyond GEO itself and toward lunar bases, Mars bases, and products derived from extraterrestrial resources, we find no wisdom in placing a transportation depot at the Earth's geosynchronous orbit.

The velocity v of a transportation depot relative to its local gravitational center is also an important consideration for establishing its location in space. The faster an object is moving, the smaller will be the velocity increase, Δv, required to make a given kinetic energy increase. This is easy to see; in non-relativistic mechanics, the kinetic energy T of a mass m is given by $mv^2/2$. For small increments of velocity, we may differentiate T with respect to v. Thus, the increase in kinetic energy is $\Delta T = (mv)\ \Delta v$, so that the larger v becomes, the smaller will be the Δv required to bring about a given ΔT. Remembering that the larger the Δv, the greater the rocket propellant mass required to accelerate the rocket, we can see the considerable savings in fuel if a rocket starts for Mars from LEO rather than from a much higher, slower orbit.

This last point leads to seemingly contradictory criteria for locating a transportation depot: it should not be tightly bound to a massive planet or moon because every encounter is high in fuel cost, and yet it should be capable of producing large velocities, which come from trajectories near massive bodies. One compromise is to establish highly elliptical orbits that give large velocities at perigee while requiring smaller binding energies to Earth than low, circular orbits. This compromise, which has many disadvantages (repeated passings through the Van Allen radiation belt, for example) will not be considered further here.

The ideal location for the second transportation depot, after the LEO space station, would be a spot that can be reached from LEO with no more than escape velocity, that requires no fuel to stay there, and that has an infinite launch window. Also, it would be easy to coast near the Earth from the ideal location; the rocket could, thereby, achieve a high velocity, leaving open options for igniting the engines to initiate interplanetary travel or aerodynamic maneuvering in the Earth's upper atmosphere. An added bonus would be obtained if the spot had velocity relative to the Moon so that lunar gravitational assists ("slingshots") could be used for increasing or decreasing a rocket's velocity. The ideal location for a transportation depot is depicted schematically in Fig. 1.

This logic leads us to discuss the subject of Lagrange points as candidates for locating transportation depots. If the Earth and Moon were fixed in space, there would be one point between them where the attraction toward the Earth would just equal the attraction toward the Moon. Because the Earth and Moon are not fixed in space but revolve around

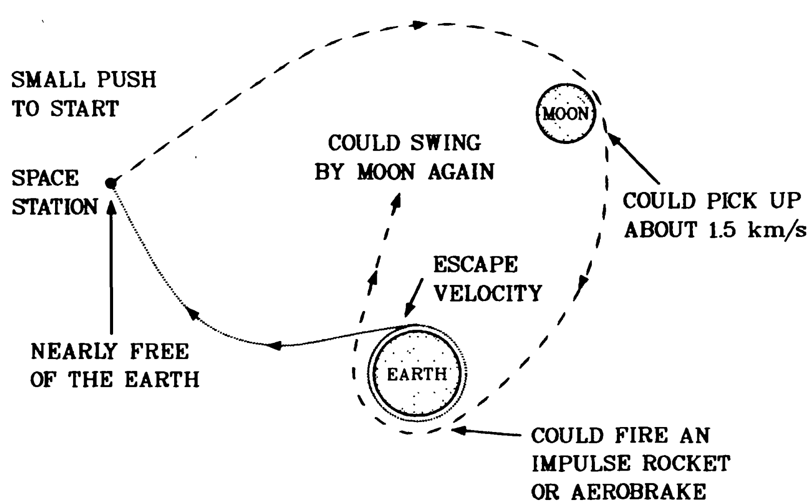

Figure 1. Ideal place for a space station.

each other, there are, instead, five such equilibrium points—Lagrange points, or libration points. These equilibrium points exist for other revolving two-body systems as well. The Earth-Moon Lagrange point, L_2(EM), and the Lagrange point between the Sun and Earth, L_1(SE), are particularly favorable locations for a transportation depot. They can be reached from LEO with escape velocity, they are easy to approach and leave, and from them it is not difficult to swing close by the Earth to initiate a high-velocity Δv firing before going to other planets. They also afford easy access to the surface of the Moon. Their "halo" orbits can be maintained with almost negligible fuel year after year. This has been verified experimentally by the International Solar Earth Exploration (ISEE-3) satellite launched in 1978, which was maintained at L_1(SE) for four years with a station-keeping Δv expenditure of 10 m/s per year (Farquhar *et al.*, 1984). In 1982, ISEE-3 was moved to measure the Earth's geomagnetic tail and, in late 1983, with gravitational assists from the Moon, was moved on to rendezvous with the Giacobini-Zinner comet in September, 1985. The name of the satellite has been changed to International Comet Expedition (ICE). We will return to the subject of Lagrange points in discussing the three-body problem.

Figure 2 helps us put in perspective some of the important points of this paper: the amount of Δv required to reach LEO from the Earth's surface and the cumulative Δv necessary to reach GEO, the L_1(SE) Lagrange point, and other places. This is not a potential energy diagram, so Δv depends on the path taken. However, it is correct to imagine a rocket leaving the Lagrange point and picking up velocity as it "slides down" the curve to LEO, where it ignites its engines for a trip to Mars. Lunar gravitational assists can be included in the mission profile. Thus, the propellant needed to get to the Lagrange point is not wasted, and L_1(SE) can be thought of as the first stage of a multi-stage rocket trip from LEO to Mars.

In the rest of this paper, we deduce the findings presented in this section from the laws of classical mechanics.

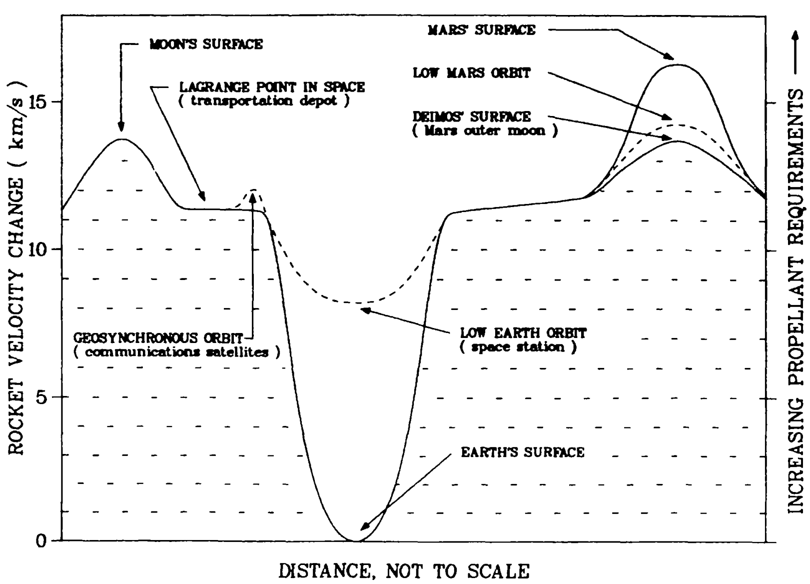

Figure 2. Delta-v budgets for special locations.

THE ROCKET PROBLEM

The fundamental rocket equation equates the instantaneous change in momentum of a rocket of mass m and velocity v, with the instantaneous change in momentum of the exhaust propellant of velocity c relative to the rocket. It may be written

$$\text{THRUST} = m\dot{v} = -\dot{m}c \qquad (1)$$

where the dot indicates a derivative with respect to time. For constant c, (1) may be integrated exactly. If we remember that the initial rocket mass m_i is the sum of the final rocket mass m_f and the propellant mass m_p, then (1) leads to

$$m_p/m_i = [1 - \exp(-\Delta v/c)] \qquad (2)$$

which shows the propellant mass required to change the velocity of a rocket by a given amount. The exhaust velocity of the main shuttle engine, which burns hydrogen and oxygen, is about 4.5 km/s. To illustrate, a Δv of 3.9 km/s is required to place a communications satellite into circular GEO from a circular shuttle orbit of 250 km altitude. If the exhaust velocity is c = 4.5 km/s, then from (2), $m_p/m_i = 0.58$. That is, as it leaves the shuttle, 58% of the mass of the satellite and associated rocketry is propellant mass.

THE TWO-BODY PROBLEM

Historically, interest in the two-body problem with central forces arose from astronomical considerations of planetary motions. Here we will use the results of that earlier work to show how people might travel to the planets. Two powerful formulas follow easily from the conservation of energy E, written as

$$T + V(r) = E \qquad (3)$$

where, as before, T is the kinetic energy of a mass m with velocity v, and $V(r) = -GMm/r$ is the potential energy of m at a distance r from the center of a spherically symmetric mass M, and $G = 6.67 \times 10^{-11}\ m^3/kg - s^2$ is the universal gravitational constant in MKS units. If an object that rests on the Earth (of mass $M = 5.976 \times 10^{24}$kg and average radius $r_0 = 6371$ km) is given a velocity, v_{esc}, just large enough to escape to infinity, then the total energy of the system is E = 0. It follows from (3) that $v^2_{esc}(r_0) = 2GM/r_0 = (11.19\ km/s)^2$. Generalizing, we may write the first fundamental equation as

$$v^2(r) = v^2_{esc}(r) + v^2_0 \quad \text{(unbound)} \qquad (4)$$

where $v^2_{esc}(r) = 2GM/r$, v_0 is the velocity m would have after escaping from M (called the hyperbolic velocity) and v(r) is the velocity needed at r to achieve a hyperbolic velocity of v_0. In this case, with $E = mv^2/2 > 0$, m is said to be unbound, and the trajectories trace hyperbolas, with the limiting case of $E = 0$ being a parabola. On the other hand, when a rocket of hyperbolic velocity v_0 encounters a planet, the rocket velocity increases according to (4). If it does not collide or fire its engines, the rocket reaches its largest velocity closest to the planet surface (a point called periapsis) and then leaves the planet in a different direction, losing speed until it again reaches its former hyperbolic velocity. If, instead, the rocket is to be captured into orbit around the planet, it can retrofire its engines at periapsis, slowing itself until the velocity is less than escape velocity so that it cannot escape the planet. The change in velocity, Δv, will determine the high point (apoapsis) of the resulting orbit. If the planet has a sufficient atmosphere, as do Venus, Mars, Earth, and Jupiter, the periapsis can occur low enough to permit atmospheric drag to slow the rocket (to aerobrake it) below escape velocity. These processes, which decrease the velocity below escape velocity, result in a negative value of E in (3).

If $E < 0$, m is said to be bound to M (we always assume that $m \ll M$), and the trajectories are elliptical orbits. If a is the semi-major axis of the ellipse, it can be shown that $E = -GM/(2a)$ (Goldstein, 1950, p. 79). Substituting this into (3), we arrive at the second fundamental equation,

$$v^2(r) = v_{esc}^2(r)[1 - r/(2a)] \qquad \text{(bound).} \tag{5}$$

A particular case of interest occurs when the satellite is in a circular orbit so that the radius is always equal to the semi-major axis. Setting $r = a$ in (5) shows that the circular velocity, v_{cir}, is always equal to the escape velocity divided by $2^{1/2}$. Thus, the escape velocity from a 500-km LEO is calculated to be 10.77 km/s; the circular velocity is calculated to be 7.62 km/s. Likewise, the Earth, traveling in a nearly circular orbit around the Sun of mass $M = 1.989 \times 10^{30}$ kg at a distance of 149.6×10^6 km, travels at an average circular velocity of 29.78 km/s.

The simplest example of orbital transfers from a circular radius of r_1 to a circular radius of r_2 can be worked out with (5). The so-called least-energy transfers, or Hohmann transfers, are obtained by directing the rocket thrust tangent to the orbit at r_1 so that the velocity is increased by Δv_1, just enough to coast on an elliptical path and reach its apoapsis at r_2 after traveling 180° around the dominant mass. Then, a second tangential rocket thrust will increase the velocity by an amount Δv_2 to insert it into the new circular orbit at r_2. The major axis of the elliptical transfer orbit is $2a = r_1 + r_2$, which completely determines $v(r_2)$ in (5). Then $\Delta v_1 = v(r_1) - v_{cir}(r_1)$, $\Delta v_2 = v_{cir}(r_2) - v(r_2)$, and the total $\Delta v = \Delta v_1 + \Delta v_2$. Using (4) and (5) repeatedly, we find that the Δv necessary for traveling from LEO to a rendezvous with Deimos is 5.5 km/s. For comparison, the Δv necessary to go to the Moon from LEO amounts roughly to the escape velocity from LEO, 3.2 km/s, plus the escape velocity from the Moon, 2.4 km/s, a total of 5.6 km/s.

In the case of Hohmann transfers, if we set $R = r_2/r_1$ and $S(R) = (\Delta v_1 + \Delta v_2)/v_{cir}(r_1)$, it follows from (5) and the above discussion that

$$S(R) = \left[\frac{2R}{1+R}\right]^{1/2} - 1 + \frac{1}{R^{1/2}}\left[1 - \left(\frac{2}{1+R}\right)^{1/2}\right] \tag{6}$$

(Taff, 1985, p. 159). Figure 3 shows $\Delta v_1 + \Delta v_2$ for orbital transfers from a 500-km LEO. The various distances in Fig. 3 are given to indicate a scale. Of course, the Sun's gravity would dominate orbits beyond the Earth's sphere of influence, which extends out from the Earth to about 930,000 km. The most interesting feature of S(R) is that it has a maximum, which is near R = 15.58. This means that more Δv, and hence more fuel, is required to place a payload from LEO into circular orbit at 100,000-km altitude than into a circular orbit at higher altitudes. Or, putting it differently, less Δv is required to go from LEO to infinity and back into the Moon's orbit around the Earth than to initiate a direct Hohmann transfer. Further details on this and bielliptic transfers are given in Taff (1985, p. 160).

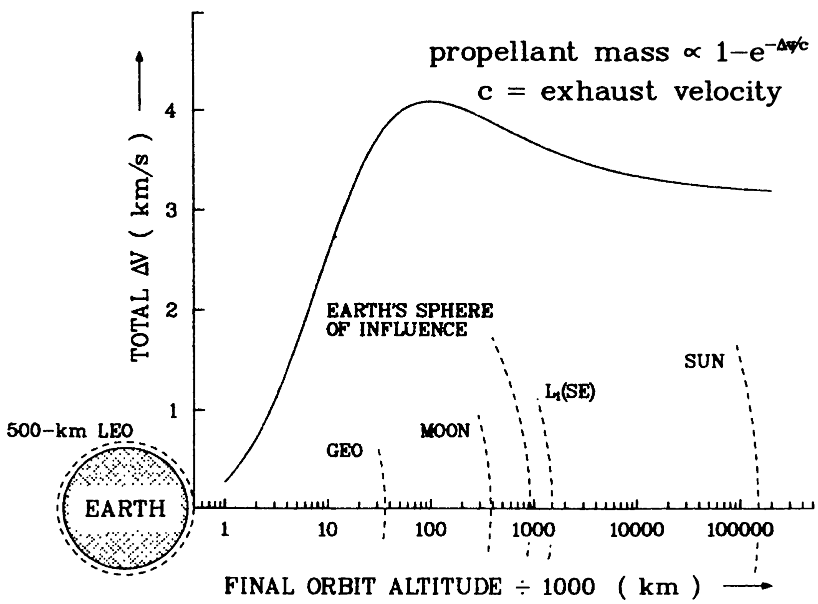

Figure 3. Delta-v versus circular orbit altitude above the Earth.

A more realistic Hohmann orbit transfer will include changing the orientation of the plane by an angle $\Delta\theta$. The optimum maneuver executes a small angular change $\Delta\theta_1$ at the lower orbit and a larger angular change $\Delta\theta_2$ at the higher orbit, so that the total angular change is $\Delta\theta = \Delta\theta_1 + \Delta\theta_2$. In that case, (6) generalizes to

$$S(R,\Delta\theta) = \left[\frac{1+3R}{1+R} - \cos(\Delta\theta_1)\left(\frac{8R}{1+R}\right)^{1/2}\right]^{1/2} + \frac{1}{R^{1/2}}\left[\frac{3+R}{1+R} - \cos(\Delta\theta - \Delta\theta_1)\left(\frac{8}{1+R}\right)^{1/2}\right]^{1/2} \tag{7}$$

where again $R = r_2/r_1$ and $\Delta\theta_1$ is fixed so as to minimize $S(R,\Delta\theta)$. Figure 4 shows the total $\Delta v = \Delta v_1 + \Delta v_2$ necessary to transfer from a 500-km LEO to higher circular Earth

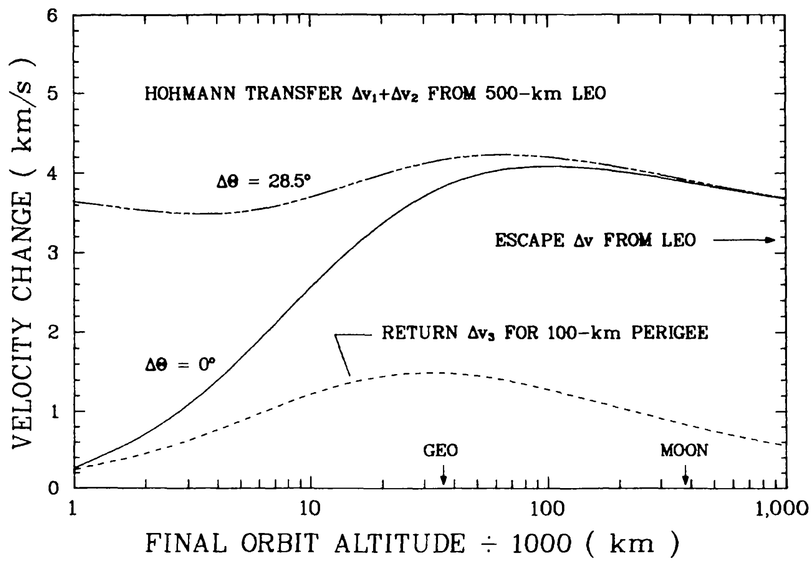

Figure 4. Earth orbital transfer missions.

orbits with a 28.5° plane change. The latitude of the Kennedy Space Center is approximately 28.5°, so that is a typical LEO to GEO transfer angle. Figure 4 also shows the necessary Δv_3 (dashed line) for returning to a 100-km perigee, from which aeromaneuvering is assumed feasible. An important feature of Δv_3 is that it reaches a broad maximum at 32,000-km altitude, which is close to the 36,000-km altitude of GEO. It therefore requires more propellant mass to return from GEO to the Earth than from almost any other circular orbit.

THE THREE-BODY PROBLEM

Analytic expressions for the orbits have not been found for the three-body problem, which is more complicated than the two-body problem. Ultimately, trajectories are calculated by numerical methods on digital computers. For a clear and succinct elementary treatment of the three-body problem, the reader is referred to Deslodge (1982, chapter 62). Here we are concerned only with "halo orbits" around one of the Lagrange points.

Consider two masses, which are a distance D apart and revolving around each other in perfect circles with an angular velocity ω under the influence of gravitational forces. It follows from the two-body problem that $\omega^2 = G(m_1 + m_2)/D^3$. The center of mass will be between m_1 and m_2, a distance αD from m_1, where $\alpha = m_2/(m_1 + m_2)$. We assume that $m_1 \leq m_2$. We establish a right-handed coordinate system with its origin at the center of mass and rotating with an angular velocity ω such that m_1 and m_2 are always stationary on the x-axis. For example, designating coordinates as (x,y,z), we find the coordinates of m_1 are $(-\alpha D,0,0)$ and of m_2 are $[(1-\alpha)D,0,0]$. We place the y-axis in the plane of rotation and the z-axis along the angular velocity vector. Now consider a third body of mass m that is so small it does not perturb the orbits of m_1 and m_2. These conditions describe the restricted three-body problem. Although it is specialized, this is an important problem because it describes reasonably well the situation of a rocket of mass m traveling in the Earth-Moon system or the Sun-Earth system.

The influence of m_1 and m_2 on m at some position (x,y,z) is described in the restricted three-body problem by a potential-like function,

$$U(x,y,z) = -\omega^2 D^2 \left[\frac{(x^2+y^2)}{2D^2} + \frac{1-\alpha}{(S_1/D)} + \frac{\alpha}{(S_2/D)} \right]$$

$$s_1 = [(x+\alpha D)^2 + y^2 + z^2]^{1/2} \quad \text{and} \qquad (8)$$

$$s_2 = \{[x-(1-\alpha)D]^2 + y^2 + z^2\}^{1/2}$$

where s_1 is the distance between m and m_1, and s_2 is the distance between m and m_2. The equations (Deslodge, 1982) of motion of m are

$$\ddot{x} - 2\omega\dot{y} = -U_x, \qquad 9(a)$$

$$\ddot{y} + 2\omega\dot{x} = -U_y, \qquad 9(b)$$

$$\ddot{z} = -U_z, \quad \text{and} \qquad 9(c)$$

$$\ddot{\vec{r}} + 2\vec{\omega} \times \dot{\vec{r}} = -\vec{\nabla} U \qquad 9(d)$$

where $U_x = \partial U/\partial x$, *etc.*, and (9d) expresses (9a)–(9c) in vector notation. Because of the $\dot{x}$ and $\dot{y}$ terms in (9a) and (9b), U is not an ordinary potential function. In fact, particles can be trapped near maxima and saddle points of U, as well as near minima. The Lagrange equilibrium points can be found by setting the gradient of U, which is the effective force on m, to zero. In doing so, the equilibrium points are seen to be in the plane of rotation, with $z = 0$. There are three collinear Lagrange points, L_1, L_2, and L_3, with $y = 0$, and two equilateral points, L_4 and L_5, with $s_1 = s_2 = D$. The five Lagrange points for the Earth-Moon system are shown in Fig. 5, along with L_1 and L_2 for the Sun-Earth system.

A three-body analogue to (5), which relates the velocity magnitude of m at a point in space to its current orbital parameters, may be deduced from (9). Taking the inner

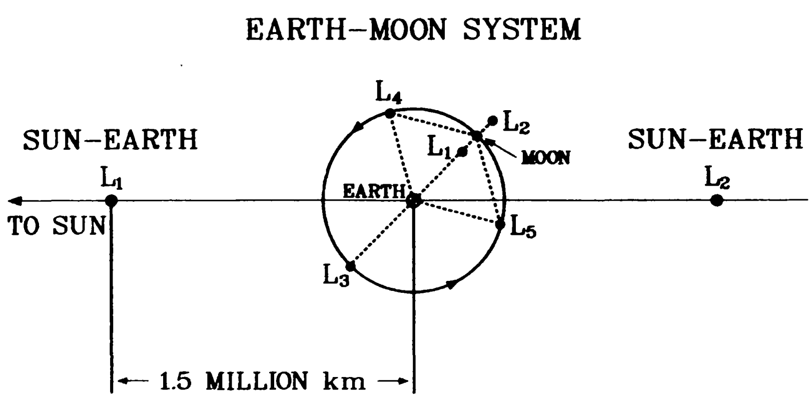

Figure 5. Lagrange points near the Earth.

product of $\vec{v} = \dot{\vec{r}}$ and (9), we find both sides of the resulting equation are perfect derivatives with respect to time, so their algebraic sum is a constant. Therefore,

$$\tfrac{1}{2}v^2 + U(\vec{r}) = E = \text{constant} \tag{10}$$

expressing also a kind of energy conservation in the rotating system. To illustrate how (10) may be used, imagine that a satellite is known to have zero velocity when it is near the Sun-Earth L_1, which is located at $\vec{r} = (0.9900D,0,0)$. Substituting these coordinates into (10), we find that $E = U(\vec{r}) = -3.001(\omega D)^2/2$. If then, with no thrust added, we later find that the satellite is at a 500-km perigee on the opposite side of the Earth from L_1 (this formula tells nothing about the trajectory necessary to arrive there), its potential must be $U = -3.131(\omega D)^2/2$, from which it follows that $v^2/2 = (3.131 - 3.001)(\omega D)^2/2$. That is, $v = (0.13)^{1/2}(\omega D)$. Because $(\omega D) = 29.78$ km/s, $v = 10.74$ km/s, which is very close to the escape velocity for LEO calculated from the two-body problem. This shows that the velocity necessary in LEO to reach L_1(SE) is between 10.74 km/s and 10.77 km/s.

Consider now the conditions necessary to establish a satellite around the Lagrange point L_1. At L_1, $(U_x)_1 = (U_y)_1 = (U_z)_1 = 0$ by definition, where the notation $(U_x)_1$ indicates $\partial U/\partial x$ evaluated at L_1, *etc.* Making a first-order Taylor expansion of the gradient U about L_1, setting $X = x - x_1$, and substituting into (9), we have the equations of motion near L_1

$$\ddot{X} - 2\omega\dot{y} = -(U_{xx})_1 X; \quad (U_{xx})_1 = -\omega^2[1 + 2f^2] \tag{11(a)}$$

$$\ddot{y} + 2\omega\dot{X} = -(U_{yy})_1 y; \quad (U_{yy})_1 = -\omega^2[1 - 2f^2] \tag{11(b)}$$

$$\ddot{z} = -(U_{zz})_1 z; \quad (U_{zz})_1 = +\omega^2 f^2 \quad \text{and} \tag{11(c)}$$

$$f^2 = \frac{1-\alpha}{(s_1/D)^3} + \frac{\alpha}{(s_2/D)^3} \tag{11(d)}$$

where all other terms in the Taylor expansion vanish. The partial derivatives in (11) are given in Deslodge (1982). To the first-order expansion, motion in the z direction is simple harmonic and independent of motion in the xy plane. Because L_1 is a mathematical saddle point, motion in the plane contains exponentially diverging solutions as well as periodic solutions, and therefore L_1 represents an unstable equilibrium. However, with the proper initial conditions, the diverging amplitudes can be set to zero, and bound periodic orbits, or halo orbits, result:

$$X(t) = X_o\cos(b\omega t); \quad b = [1 - f^2/2 + (f/2)(9f^2 - 8)^{1/2}]^{1/2} \tag{12(a)}$$

$$y(t) = -(X_o\gamma)\sin(b\omega t); \quad \gamma = (1 + b^2 + 2f^2)/(2b); \quad \text{and} \tag{12(b)}$$

$$z(t) = z_o\cos(f\omega t). \tag{12(c)}$$

Note that the satellite is traveling on an elliptical path in a direction around L_1 opposite to that of the Earth around the Sun when the initial conditions are correct for a periodic orbit in the xy plane. The orbital parameters are $f^2 = 4.061$, $b = 2.086$, and $\gamma = 3.229$, depending only on the ratio of m_2/m_1, and $\omega = 2\pi$ rad/yr. Because $\gamma > 1$, the semi-major axis is always along the y-axis, perpendicular to the Sun-Earth line. Unlike elliptical orbits in the two-body problem, the period of revolution is independent of the size of the ellipse. The angular velocity in the plane is $b\omega$; in the z direction, it is $f\omega$. With L_1(SE), $f = 2.02$ and $b = 2.09$, so the halo orbit will have a period of about 6 months. These observations are all in agreement with the halo orbit of ISEE-3 described by Farquhar *et al.* (1984). For the Moon, L_1(EM) is located at $x_1/D = 0.8369$, where $D = 384{,}400$ km, and ω may be calculated by observing that the sidereal month is 27.32 days. From (11) and (12), we see that $f = 2.27$, $b = 2.33$, and $\gamma = 3.59$. The period of rotation of a satellite around L_1(EM) is therefore about 12 days. Equations (11) and (12) are valid for any of the collinear Lagrange points, provided that proper values of s_1 and s_2 are substituted into (11d) to find f^2. For example, $x_2/D = 1.0100$ and 1.1557 for L_2 of the Sun-Earth and Earth-Moon systems, respectively, so the corresponding values of f are 1.99 and 1.79. The period of a halo orbit around L_2(SE) is not significantly different from that of L_1(SE), but for L_2(EM), $b = 1.86$, and the period of the halo orbit is higher, about 15 days.

In exploring the basic physics of placing a space station at one of the Lagrange points, one last question remains. How large is the Δv push needed to make a rocket return from L_1(SE) to low Earth orbit? We attempt here only a heuristic estimate. Because L_1(SE) and the Earth are both revolving around the Sun with the same angular velocity ω, and because they are different distances from the Sun, the Earth is traveling faster than L_1(SE) in a heliocentric inertial frame of reference. This amounts to a difference of about 0.01×29.78 km/s $= 298$ m/s, which can be taken as an initial estimate of the Δv necessary to eject from L_1(SE) and fall toward the Earth with essentially no angular momentum barrier. The number agrees well with the numerically calculated value of 279 m/s given in Farquhar and Dunham (1985). If, on the other hand, the rocket is orbiting L_1(SE) with a semi-major axis of about $X_0\gamma = 650{,}000$ km, as was the ISEE-3, calculating y_{max} from (12b) shows that the rocket reaches velocities relative to L_1(SE) as high as $(X_0\gamma b/D)\omega D = 0.009 \times 29.78$ km/s $= 268$ m/s. This occurs when the rocket is moving parallel with the Earth, so only an additional $298-268 = 30$ m/s of Δv would appear to be needed to return to LEO from the halo orbit. This estimate is close to the ISEE-3 experiment that required $\Delta v = 36.3$ m/s for insertion into the halo orbit in November, 1978. Subsequently, in June, 1982, only $\Delta v = 4.5$ m/s was required to eject ISEE-3 from halo orbit and back into the geomagnetic tail. For our purposes, we will define "very low Δv" to be less than 50 m/s.

A drawback of these very low Δv injections and ejections is that the transit times take months. The transit times can be reduced to weeks while still keeping Δv under 100 m/s. Perhaps the very low Δv encounters with L_1(SE) will be useful for only unmanned cargo ships carrying large masses. In addition, because the halo orbit period is six months, a space station will be at the proper place to receive freighters with very low Δv injections only twice a year. For each opportunity to dock, we estimate that the launch window will exist for about three weeks for very low Δv-class missions. These limitations will not pose serious problems when freight can be parked anywhere in halo orbit and boarded later by people at the appropriate time.

In addition to L_1(SE), other Lagrange points can be considered for a transportation depot. Although L_2(SE) shares all of the same kinematic advantages, it lies in the Earth's geomagnetic tail and may not be suitable because of the radiation environment. Each of the five Earth-Moon Lagrange points could be a candidate for a transportation depot, but estimates of the injection velocities, such as those described above, indicate that the points require a Δv in the range of 1000 m/s. However, for L_2(EM) this can be overcome to a great extent by a clever maneuver. It has been shown that, using a retrograde lunar gravitational assist, the Δv necessary to enter a halo orbit at L_2(EM) is about 300 m/s instead of the 1230 m/s necessary for direct insertion from LEO (Farquhar and Dunham, 1985; Farquhar, 1972). We conclude that there are two places in the Earth's vicinity that are well qualified for hosting a transportation depot—L_1(SE) and L_2(EM).

Looking beyond the Earth's vicinity once the technology is developed to establish a manned space station around L_1(SE), we can use that technology at other places in the solar system. All of the major planets and moons have L_1 points that can sustain halo orbits. For example, Lagrange points of the Sun-Mars system have already been mentioned in connection with a Mars mission (Farquhar, 1969). An important factor in considering L_1(SM) at Mars is that the first manned missions there will carry return propellant, heavy shielding for radiation protection in transit, engines, and other things not needed at Mars. The less tightly bound this equipment is to Mars, the less fuel will be needed to break it away later on the return trip. A ship going to Mars would save propellant by slowing at periapsis to just under escape velocity and coasting to L_1(SM). All of the equipment for returning to Earth would be placed in a halo orbit and left there during the descent to Mars. Later, reversing the procedure, the crew could return to Earth and waste very little fuel in the process. Eventually, a transportation depot at

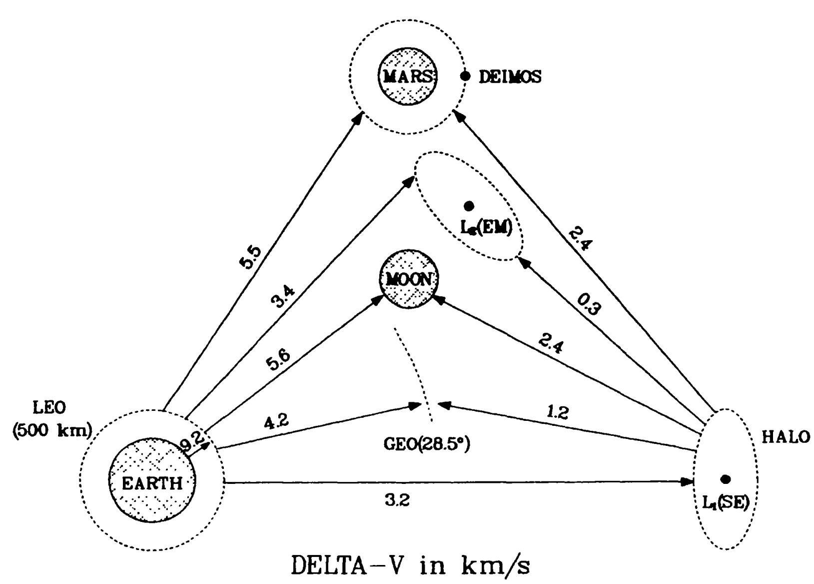

Figure 6. Two space transportation nodes.

L_1(SM) would require its own space station, especially if water were found at Phobos or Deimos. Here, however, we are focusing on the second transportation depot, the one to follow the LEO space station.

Figure 6 emphasizes the "nodal" aspects of space stations at LEO and in a halo orbit around L_1(SE). It shows the Δv in kilometers per second required to move from each of the two nodes to GEO, Moon, L_2(EM), and Deimos. Notice that it costs very little extra total Δv to stop over at L_1(SE) on the way to one of these places from LEO. This makes L_1(SE) an important staging area for rocket propellant, water, bulk shielding material, and vehicle assembly for Moon and Mars expeditions. Freighters can go to L_1(SE) in months and take advantage of the very low Δv transfers into halo orbit. People can be carried there in a shorter time by paying a higher Δv expenditure in small orbital transfer vehicles (OTVs).

SUMMARY

The key to a successful evolutionary space program is the placement of effective transportation nodes in the supporting infrastructure. Such outposts have always been important in opening frontiers. For the settlement of space, a Lagrange equilibrium point between the Sun and Earth has the nearly ideal physical characteristics of a transportation depot: it is very lightly bound in the Earth's gravitational well; it can be reached with essentially escape velocity; the launch window is always open; it can accommodate a wide range of plane angles for LEO space stations; its halo orbits require only 10 m/s per year of station-keeping propellant to remain stable; and from there it is easy to leave and pass near the Earth at essentially escape velocity—affording several options. Rockets going to and from L_1(SE) can obtain free acceleration and braking by passing near the Moon's surface.

An in-depth study is necessary to determine the best place to put the next transportation depot after the LEO space station, but the laws of nature will not change. Lagrange point halo orbits are the present standard by which any alternative concept for a transportation depot must be gauged.

Acknowledgments. *When the author was invited by the Lunar Base Steering Committee to express his views on this subject, he was reluctant to write a paper because many of the ideas were not original. However, he was persuaded by the argument that a tutorial format might provide background and establish the framework for future deliberations. It is a pleasure to acknowledge several helpful conversations with Robert W. Farquhar, of NASA's Goddard Space Flight Center, during the course of this work.*

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ACHROMATIC TRAJECTORIES AND THE INDUSTRIAL-SCALE TRANSPORT OF LUNAR RESOURCES

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Large-scale transport of lunar material can be accomplished by launches in small payloads by mass-driver along unguided trajectories. Achromatic trajectories then overcome the problem of dispersions due to launch velocity errors. These trajectories incorporate a focusing effect, which reduces the dispersions by four orders of magnitude from their expected values. The mass-driver is to be located close to the lunar equator at 33.1° E longitude. A mass-catcher then can maneuver near the L_2 Lagrangian point, intercepting the payloads. For an optimized catching trajectory, the catcher requires $\delta V = 187$ m/s per month, peak thrust = 0.1414 newtons per ton of loaded catcher mass, and for a reference propulsion concept, the Rotary Pellet Launcher, peak power of 0.316 kilowatts per ton. Caught material can be transferred with $\delta V = 60$ m/s to a processing facility (space colony) in stable high Earth orbit. This transport system can accommodate 10^5 tons per month of lunar material, with an appropriately sized mass-catcher.

INTRODUCTION

One of the principal rationales for a permanent lunar base is that it can serve for the mining and transport of lunar resources. It is appropriate, then, to look beyond the initial establishment of this lunar base and to consider its use in a resource-transport program, particularly one of large scale. In particular, it has often been proposed that resources be launched into space with a mass-driver or electromagnetic catapult. The existing literature on mass-drivers is largely included in Grey (1977a,b, 1979, 1981).

There is also the question of the operational use of a mass-driver, within an overall system for resource transport. Payloads launched by mass-driver follow an unguided, ballistic flight that is therefore subject to potentially large miss distances, at a target, due to errors at launch. During the 1970's, in connection with the concept of space colonization, I developed a theory for operational mass-driver use (Heppenheimer and Kaplan, 1977; Heppenheimer 1978a,b,c; Heppenheimer 1979; Heppenheimer *et al.*, 1982a,b). The point of departure for this work was the concept of achromatic trajectories (Heppenheimer and Kaplan, 1977; Heppenheimer, 1978a; Heppenheimer, 1979), *i.e.*, trajectories incorporating a focusing effect. Payloads launched along such trajectories still will reach their target despite errors in even the most sensitive component of launch velocity. It was demonstrated that achromatic trajectories provide a framework within which the entire problem of operational use of a mass-driver is readily solvable. In what follows, I give a brief summary and overview of this existing literature and its concepts.

ACHROMATIC TRAJECTORIES

Let a payload be launched by mass-driver, initially tangent to the lunar surface with velocity V_T. It arrives at the target with velocity V_i. If the payload experiences a velocity error (ϵ) at launch, with what error will it miss the target? Calling this error δy, a naive analysis based upon two-body dynamics gives the approximation

$$\delta y = (V_T/V_i)\ (d_m/V_i)\ (\epsilon)$$

where d_m is the lunar diameter, 3476 km. An attractive location for the target is the L_2 Lagrangian point, some 64,000 km behind the Moon (Johnson and Holbrow, 1977). Then, approximately, V_T = 2400 m/s, V_i = 260 m/s; hence δy = 1.23 km per cm/s of ϵ.

One must anticipate serious difficulties in controlling V_T to the inferred accuracy. As a result, the attractiveness of mass-drivers at first appears open to question. Achromatic

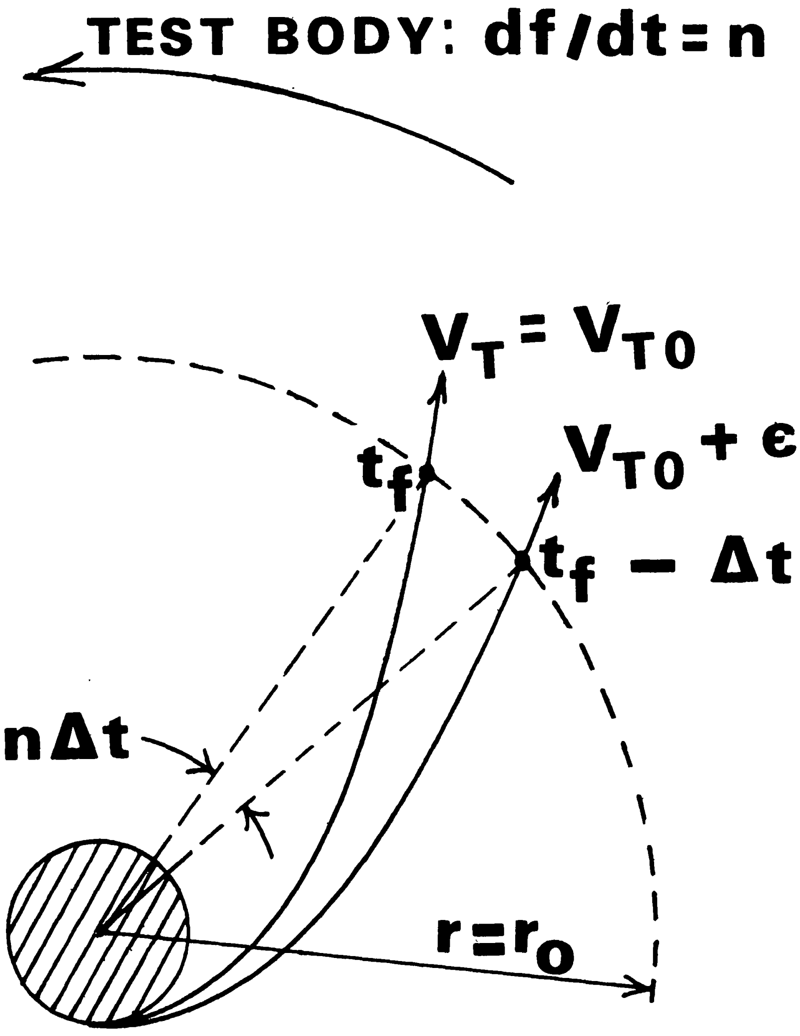

Figure 1. Physical character of an achromatic trajectory.

trajectories come to the rescue at this point, reducing this sensitivity by four orders of magnitude.

Figure 1 shows how this occurs. A test body is in orbit, at distance $r = r_0$ moving with angular velocity $df/dt = n$, where f is true anomaly. Its orbit need not be circular. We wish to hit this body with a projectile launched by mass-driver. Nominally, this calls for launch with velocity $V_T = V_{TO}$; the flight time is t_f. Now suppose that the launch velocity is actually $V_{TO} + \epsilon$. The trajectory will be flatter and the flight time reduced to t_f - δt. Yet, if we can take advantage of the flattening in the trajectory to reduce the transfer angle from nominal by $n\delta t$, then the projectile still hits the target and $\delta y = 0$.

Hence, in a coordinate system rotating with the target, the two trajectories cross. This crossing, due to the rotation of coordinate systems, provides the focusing effect. Figure 2 shows the geometry of such crossings. Here δy is the product of two quantities:

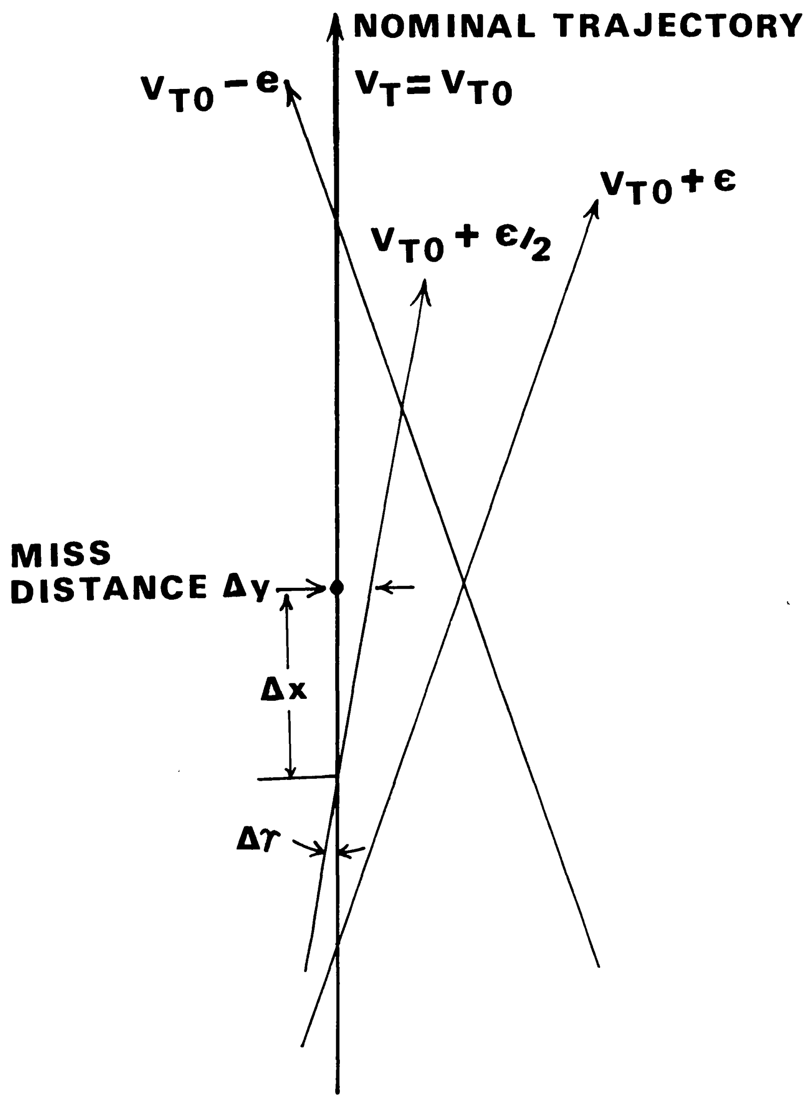

Figure 2. Focusing of achromatic trajectories in a rotating coordinate system.

δx, the location of the crossing point along the nominal trajectory; and $\delta\gamma$, the angle at crossing. But $\delta x \propto \epsilon$, $\delta\gamma \propto \epsilon$, so $\delta y \propto \epsilon^2$. More precisely, for ϵ in cm/s and for the achromatic trajectory passing through the L_2 Lagrangian point, $\delta y = 0.1901\epsilon^2$ meters (Heppenheimer, 1979). This compares with the naively expected value cited earlier, $\delta y = 1234.1\epsilon$ meters.

One may further consider a family of neighboring trajectories, originating with slightly different V_T and having the crossing geometry of Fig. 2. Such a family possesses an envelope (Fig. 3), any point of which is associated with a specific value of V_T. This envelope is called a focus locus. If the target maneuvers so as to always stay on the focus locus, then there always exists a V_T that permits reaching the target via an achromatic trajectory.

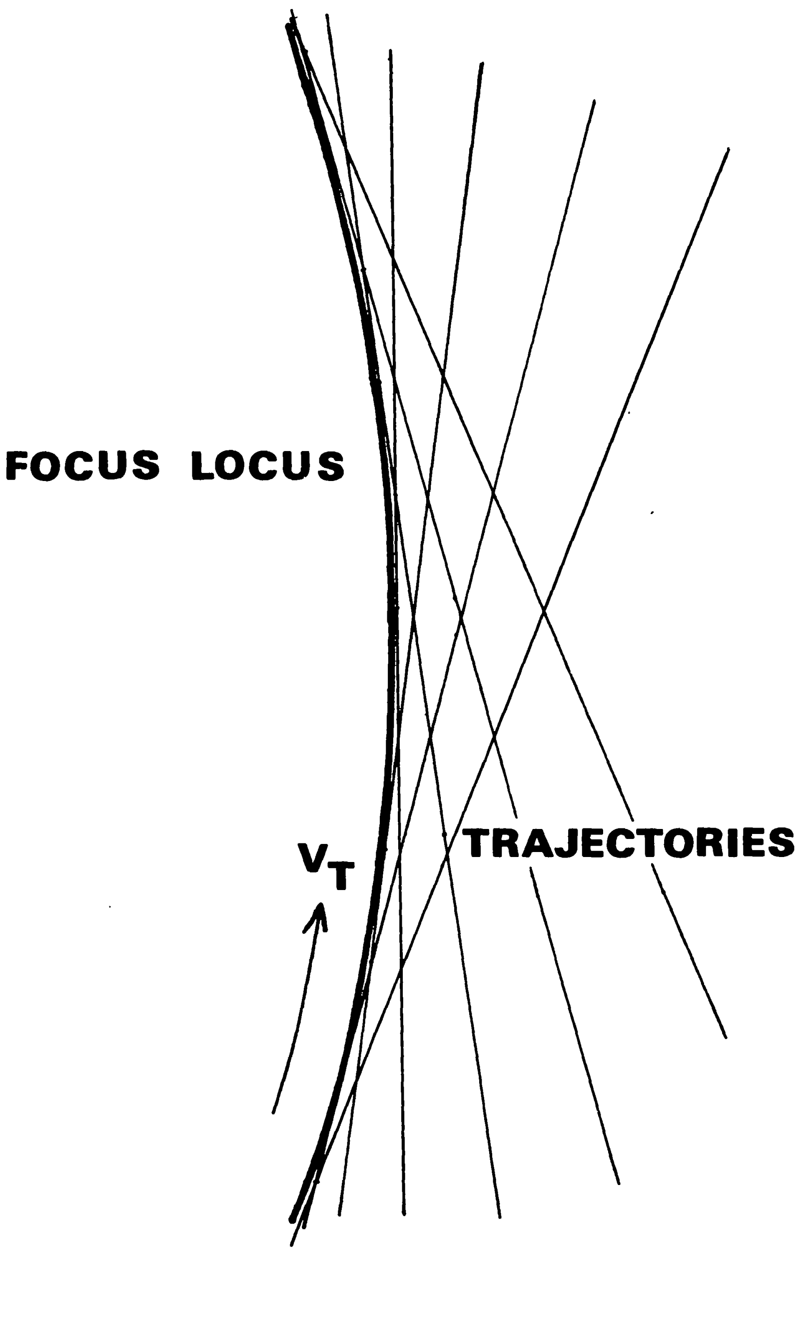

Figure 3. Definition of a focus locus.

The mass-driver is a large, fixed lunar installation. At any time, a single, well-defined focus locus is associated with its location. But its effective location in the reference coordinate system is not constant because the Moon undergoes physical librations in longitude and latitude. During any one month, these simultaneous oscillations trace out an ellipse. The focus locus then oscillates in position, always lying on a cylinder whose cross-section is essentially this ellipse. Thus, there is a family of focus loci, associated with the different effective values of launch longitude and latitude due to the librations of the Moon. The problem is then constrained by requiring the mass-catcher, the target, to maneuver in space so as to always lie on the focus locus associated with the mass-driver's location.

Simple methods exist for determining focus loci (Heppenheimer, 1979). These involve numerical integrations of trajectories, carried out within the circular restricted three-body problem (wherein the Moon's orbit about the Earth is regarded as circular) or, for greater accuracy, the elliptic restricted problem. In the former, the lunar librations must be put in "by hand," but in the latter they arise naturally. Figure 4 illustrates focus loci calculated in the elliptic problem for a mass-driver located at 33.1° E longitude on the lunar equator

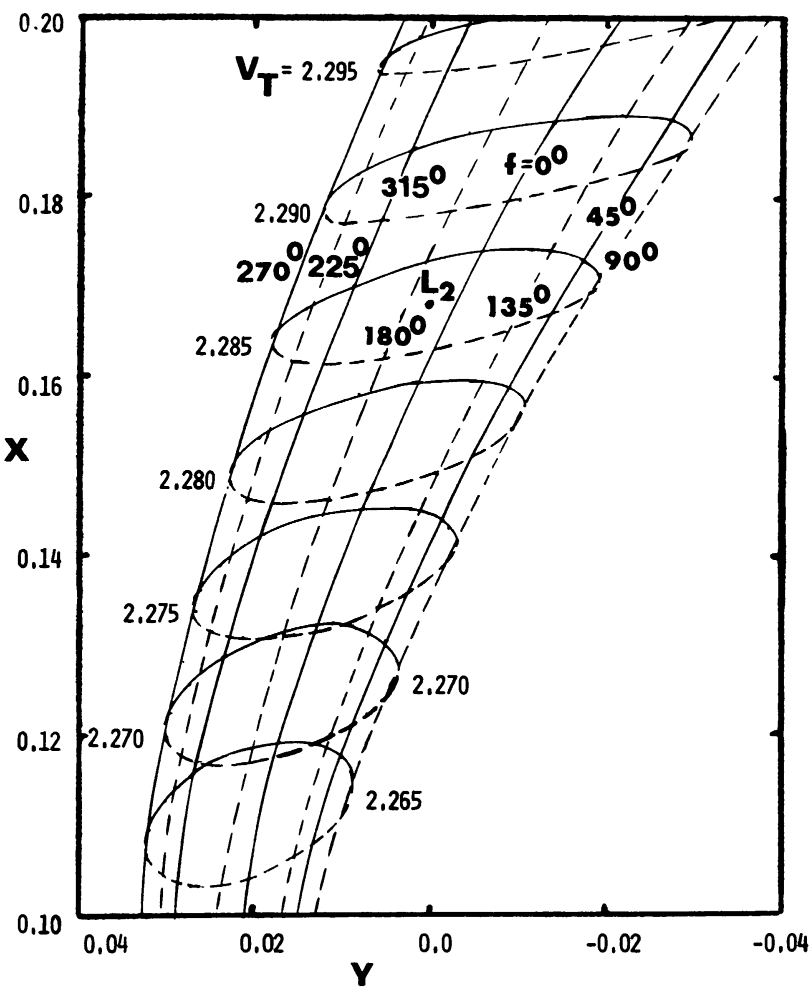

Figure 4. Focus loci in the elliptic restricted three-body problem.

Table 1. Normalized Variables in Restricted 3-Body Problem

Quantity	Variable	Unit Value	Significance
distance	x, y	Lunar orbit radius (mean = 384,410 km)	L_2 coordinates fixed
time	t	104.362 hours	Moon's rotation by 1 radian
velocity	V	1023.17 m/s	Mean lunar orbit velocity
acceleration		0.00273 m/s^2	(V/t)

(Heppenheimer and Kaplan, 1977). Such a location provides a focus locus through L_2, in the circular problem.

Here, and in subsequent discussion, x and y are normalized distances, given as fractions of the variable Earth-Moon distance (Table 1). With such coordinates, the location of L_2 remains fixed. Launch velocities V_T are also normalized, being given in terms of the mean lunar orbital velocity. The variable f is lunar true anomaly, which defines its orbital position; it is evident how the focus locus swings from one side to the other as the Moon revolves during a month.

From such focus-locus computations, one derives data sufficient to characterize the required mass-catcher maneuvers. In particular, these data permit derivation of the Catcher Equations (Heppenheimer, 1978a; Heppenheimer, 1979), describing the mass-catcher motion under the following restrictions:

(a) The catcher must always maneuver so as to be on the focus locus as it shifts location.
(b) The catcher varies in mass, owing to its interception of the stream of payloads from the mass-driver, and is under continuous propulsion.
(c) The payload stream imposes a momentum flux upon the catcher.

THE MASS-CATCHER

With the Catcher Equations, one derives optimal catcher trajectories, which minimize propellant expenditure subject to the above conditions. The optimization procedure employs such standard methods as the Maximum Principle (Heppenheimer, 1978b). It is useful to constrain the motion by limiting the allowed velocities at the beginning and end of a catching cycle. We impose the condition that, initially, the catcher velocity must nearly match that of the payload stream. As a result, the catcher is not immediately subject to potentially damaging high velocity impacts. Instead, there is time for a protective layer of caught material (a regolith) to build up. Also, the final x-component of velocity is constrained to a (normalized) value of -0.01, some 10 m/s, to permit gentle injection of the caught payloads onto a transfer trajectory to a proposed space processing facility (Heppenheimer, 1978c).

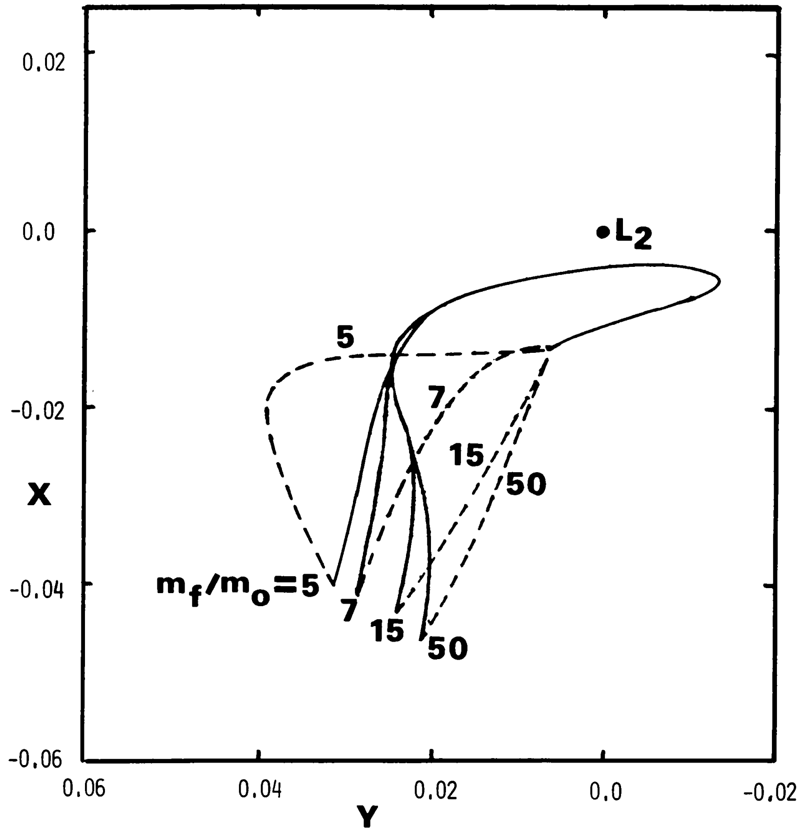

Figure 5. Some optimal mass-catcher trajectories in the plane.

The catcher is assumed to fill uniformly with time, starting with an initial mass m_o and finishing with total mass m_f. After disposing of the accumulated payload, the catcher moves propulsively back to the initial point. The entire cycle takes approximately one month. For convenience, the catcher is assumed to fill over a fraction of the cycle given by $(m_f-m_o)/m_f$ and returns to initial focus point during the remaining fraction, m_o/m_f. Figure 5 illustrates several optimal trajectories as a function of m_f/m_o. The motion is generally clockwise, tracking the shifting location of the focus locus (Fig. 4). The ballistic return trajectories are dashed.

From such computations, one derives data on the operational use of a mass-catcher and mass-driver, shown in Fig. 6 for $m_f/m_o = 15$. Here δ_o is a constant phase angle (270°) chosen to minimize catcher propulsion requirements, representing the time of the month when catching begins. The catcher mass increases uniformly from $m_f/15$ at the curves' left-hand sides to m_f at the right-hand sides. The catcher follows the corresponding optimal trajectory of Fig. 5.

In a situation where the payload stream is interrupted, or the catcher drifts off the focus locus, the catcher follows its nominal path pending reacquisition of the payload stream. "Emergency acceleration" defines the thrust required to maintain the nominal motion in the absence of the payload momentum flux. Maximum thrust is thus given by the condition that a loaded catcher be capable of an acceleration of 0.052 in normalized units; this corresponds to a thrust of 0.1414 newtons per ton (1000 kilograms) of loaded

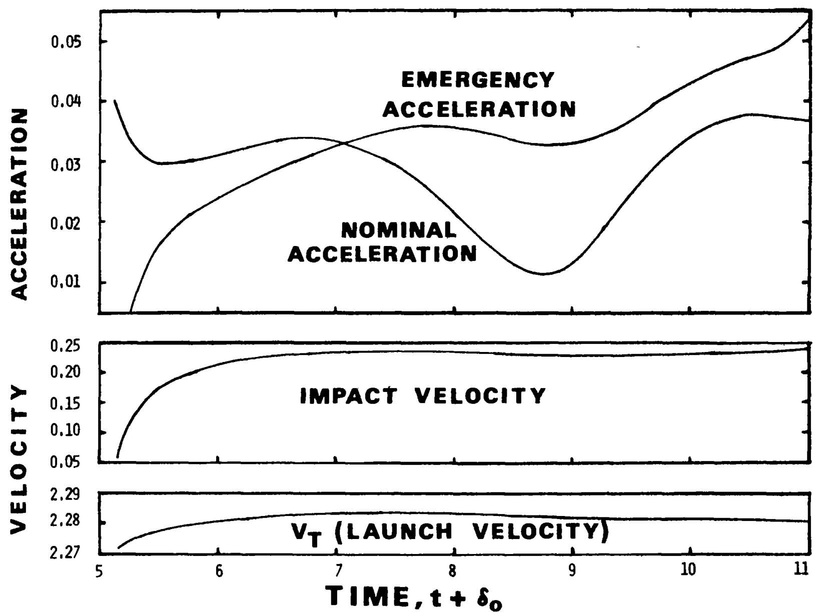

Figure 6. Some operational characteristics of an optimized catcher trajectory.

catcher mass. "Nominal acceleration" presents the thrust required to follow the nominal path of Fig. 5. This curve shows that the nominal thrust varies between 0.2 and 0.8 of the maximum, approximately.

The time integral of the curve, "nominal acceleration," gives the δV required to follow the nominal trajectory, 169.5 m/s. If V_e is the exhaust velocity of the catcher's thrusters, then $(169.5/V_e)$ is the fraction of loaded catcher mass required for thruster propellant. In addition, there is a δV of 260.7 m/s required for the return maneuver (dashed line) of Fig. 5. But the catcher mass then is $m_O = m_f/15$; hence the effective δV for a complete catching cycle increases by only $260.7/15 = 17.38$ m/s, to 186.88 m/s.

Figure 6 also gives the impact velocity of payloads striking the catcher and the launch velocity V_T as functions of time. Payloads launched by mass-driver are considered to be essentially simple bags of lunar soil. Thus, as they accumulate in the catcher, they build up a layer of such soil, the catcher regolith. There is much data on impacts into regolith, resulting from work on geological problems of lunar and planetary cratering. Heppenheimer (1978b) and Heppenheimer (1979) give elements of an analytic theory of catching, based on cratering and regolith theory. The catching theory predicts that mass loss from impacting payloads can be reduced to less than 1% by designing the catcher as a large open conical bag, rotating at several rpm. The rotation provides more than enough centrifugal force to hold caught mass in place. It also provides a differential velocity at the impact site, which ejecta must overcome in order to attain the free flight that can produce mass-escape from the catcher.

ENGINEERING APPLICATIONS

The achromatic trajectory passing through L_2 originates at equatorial longitude 33.1° E, location (1) in Fig. 7. Examination of lunar orbital photography shows that this site

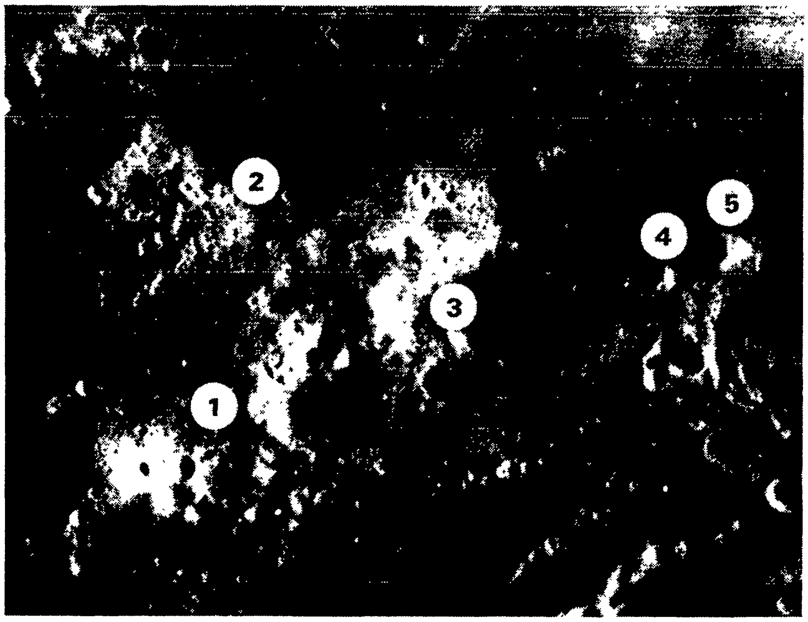

Figure 7. Geography of a preferred mass-driver launch site.

is unsuitable because it is very mountainous. The nearby site (2) on level mare terrain is much better. Moreover, it is adjacent to a boundary between mare and lunar highland. Since mare rocks are rich in iron or titanium while highland rocks are richer in aluminum, a variety of ore types may be available.

Site (3) is similar but offers an additional advantage. The flight path can readily pass either or both of two mountains, (4) and (5), on which it is possible to set up downrange corrector stations (Heppenheimer *et al.*, 1982a,b). These adjust the flight of the payloads subsequent to launch. Heppenheimer *et al.* (1982b) argues that the payloads' trajectories can be controlled so as to achieve an accuracy, at L_2, of ± 1.5 meters.

Figure 8 gives a reference design concept for the mass-catcher (Heppenheimer, 1978b, 1979). A suitable structural material is Kevlar-49, with yield strength $\sigma = 36.2$ gigapascals

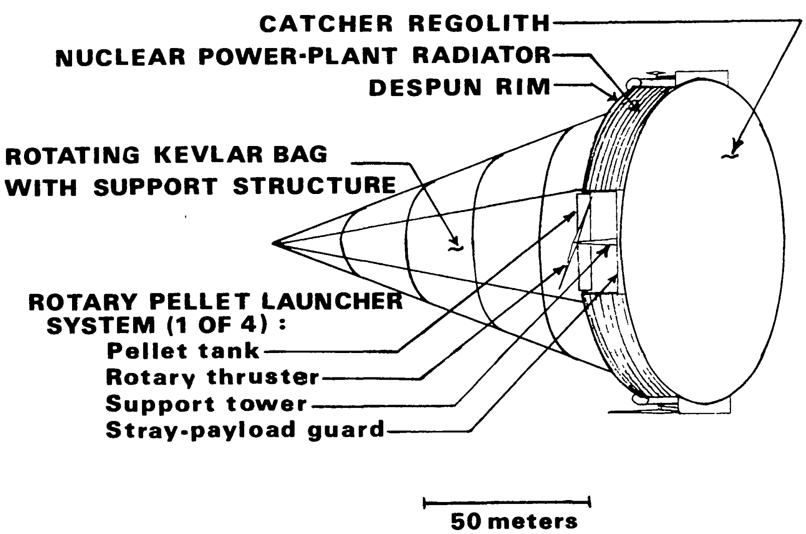

Figure 8. Mass-catcher concept sized for 10^5 tons total mass.

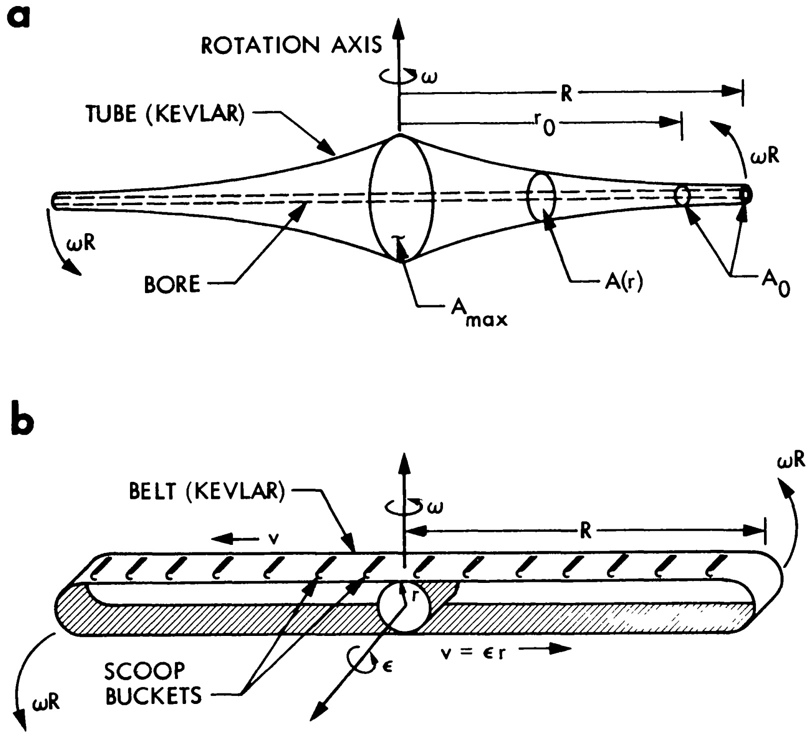

Figure 9. Design concepts for the Rotary Pellet Launcher.

= 525,000 psi and density $\rho = 1.45$ g/cm^3 (Johnson and Holbrow, 1977). For a uniform catcher regolith and stability in rotation, the cone angle must be greater than $\tan^{-1} 1/3 = 18.5°$. Propulsion is provided by Rotary Pellet Launchers, which rotate rapidly to eject pellets formed from lunar material. Their associated rotation velocity is of order $(2\sigma/\rho)^{1/2} = 2234$ m/s, which is the order of the pellet ejection velocities. It was earlier noted that for the catching mission, $\delta V = 187$ m/s and thrust = 0.141 newtons/ton. Hence the catcher must eject the order of (187/2234) = 8.4% of its loaded mass as propellant. Thruster power required is of the order of (0.141 x 2234) = 316 watts/ton. A nuclear powerplant is advantageous; it can readily be protected from damage by stray payloads.

Figure 9 illustrates two concepts for the Rotary Pellet Launcher. Concept (a), discussed in Johnson and Holbrow (1977), maximizes the tip velocity by tapering the tube. With a tube that thickens exponentially toward the root, tip velocities can well exceed $(2\sigma/\rho)^{1/2}$. However, pellets then must roll or slide down the tube, possibly producing rapid abrasion. This difficulty may be overcome by the variant concept, (b). It features a rapidly rotating conveyor belt that carries pellets in scoop buckets out to the ends. Its tip velocity is less than $(2\sigma/\rho)^{1/2}$, but if the load-bearing belt has mass much greater than that of the pellets and scoop buckets it carries, then this velocity may be approached as a limit.

The operational use of the catcher involves a proposed "catching cycle" (Fig. 10) (Heppenheimer, 1979). Catching is initiated following the end of the catcher's return trajectory, shown as the dashed line of Fig. 5. Initially, catching is done under a constraint of low impact velocity, to protect the bag. This continues for two to three days, until some 1/10 of the final catcher load has been caught, forming a regolith. The catcher

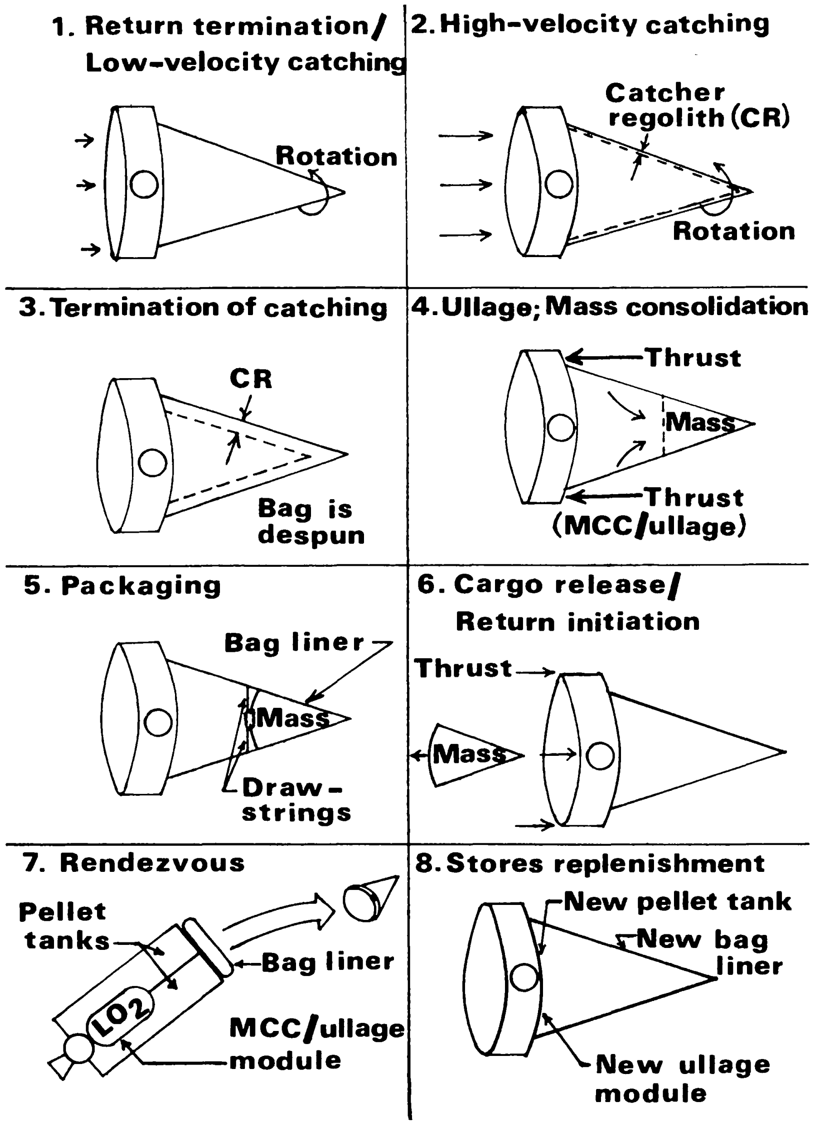

Figure 10. The catching cycle.

then is sufficiently massive to require an optimal trajectory rather than one constrained by a requirement for low impact. For the next three weeks, it follows the loop of Fig. 5, with payloads impacting at some 250 m/s. The bag and its caught material are maintained in rotation by using the thrusters to despin the rim while torque is applied to the bag.

When the catcher is full, the thrusters serve to despin the bag. The mass within then is consolidated via an ullage maneuver. Part of the required thrust comes from the thrusters; extra thrust can be provided from compressed O_2, which is available from a remote space facility for ore processing. The consolidated mass may be packaged within a detachable bag liner simply by pulling drawstrings to close the mouth of the bag. With the cargo packaged, the catcher backs away, leaving the cargo to enter free flight on a trajectory to the space manufacturing facility (Heppenheimer, 1978c).

The catcher initiates a return maneuver, along the dashed curve of Fig. 5. While on this return leg, it can rendezvous with a supply of stores sent along trajectories that

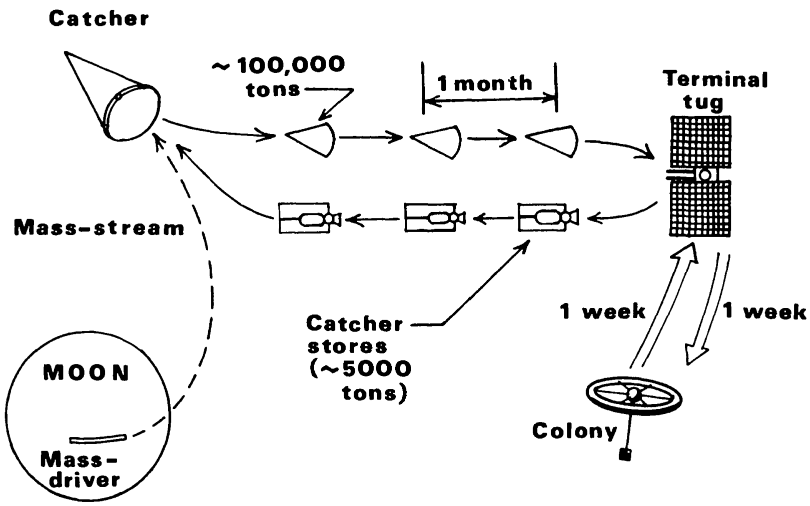

Figure 11. Overall system for transport of lunar resources.

are the reverse of those followed by the released cargo. Resupply can include a new O_2 module, tanks full of pellets for the thrusters, and a new bag liner. During the stores' transit, the O_2 module can provide midcourse corrections. Because it serves both for ullage and for such midcourse corrections, in Fig. 10 it is referred to as the MCC/ullage module.

Finally, Fig. 11 shows the overall transport operation. The traffic back and forth, in both cargo and stores, follows similar trajectories. Along the trajectory near the space facility, the cargo is intercepted by a terminal tug (Heppenheimer, 1979). It is based at the space colony and executes transfers between the colony orbit and the cargo orbit. Its basic mission resembles that of the catcher: to accelerate large cargoes through a small δV, at accelerations of some 10^{-4} m/s^2. Its propulsion systems thus will closely resemble those of the catcher, but it can be solar powered, since it does not face damage from stray payloads.

The catcher thus is to operate as an autonomous facility, resupplied from the colony. It need not tap into its caught material during catching operations. Instead, like an oil tanker, the catcher simply carries and contains its cargo, as a bulk mass. This mass, in turn, could run to 10^5 tons/month, or even more (Heppenheimer, 1978b).

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A LUNAR-BASED PROPULSION SYSTEM

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As activities in cis- and trans-lunar space and on the Moon increase during the 21st century, the use of a lunar-based propulsion system, refueled by propellants manufactured from lunar resources, may offer large cost savings when compared with a space-based propulsion system refueled from the Earth. Oxygen/hydrogen (LO_2/LH_2) bipropellant propulsion appears to be attractive because of its estimated high delivered specific impulse, *i.e.*, 485 s. However, difficulties associated with the long-term storability and low density of LH_2 detract from this performance. Other bipropellant combinations may have advantages in this context. The potential utility of the oxygen/silane ($LO_2/LSiH_4$) bipropellant combination for use in a lunar-based propulsion system and the potential for the on-site manufacture of lunar oxygen and silane are considered in this paper. It appears that oxygen and silane can be produced from common lunar mare basalt in an integrated facility. The carbothermal process uses lunar materials efficiently to produce oxygen and silane-precursors with minimum terrestrial resupply. The production of silane from lunar materials may require a key, lunar-produced intermediate, magnesium silicide (Mg_2Si). Mineral acid or water terrestrial resupply will be required to produce silane by this synthesis. It appears that the propellant properties of oxygen and silane are more than adequate to support the development of a lunar-based propulsion system. Silane is stable and storable in space and lunar environments and has properties that are compatible with those of oxygen. Using standard Aerojet-JANNAF procedures, the estimated delivered performance of the propulsion system is 340–350 s at a mixture ratio of 1.50 to 1.80. Penalties normally associated with pressure-fed propulsion systems may be minimized in the lunar environment, *i.e.*, 1/6 g. A pressure-fed propulsion system may prove to be quite competitive with a pump-fed system.

INTRODUCTION

The Moon is made of oxygen. That is, the Moon is rich in minerals from which oxygen can be manufactured. Unfortunately, the Moon does not possess adequate, easily exploited sources of hydrogen- or carbon-containing minerals. This poses a particular problem for a designer of a lunar-based propulsion system, as an adequate supply of both oxidizer and fuel is required to power the system. One solution to the problem is to transport the required fuel from the Earth, or from low Earth orbit, to the Moon.

Hydrogen is one candidate fuel because it offers excellent performance with oxygen. Unfortunately, hydrogen has an extremely low density and is very difficult to store as a liquid. Monomethylhydrazine ($CH_3N_2H_3$) is another candidate fuel since it offers satisfactory density, storage properties, and performance. It would, however, have to be transported from the Earth for use in a lunar-based propulsion system.

Perhaps there is a middle position in regard to the manufacture of a suitable fuel on the Moon. There may be a satisfactory fuel that can be manufactured by using lunar resources and some chemicals resupplied from the Earth, namely, silane.

LUNAR RESOURCES FOR PROPELLANT MANUFACTURE

The minerals required to manufacture oxygen on the Moon are abundantly available. Olivine [$(Mg,Fe)_2SiO_4$], pyroxene [$(Ca,Mg,Fe)SiO_3$], and ilmenite ($FeTiO_3$) are particularly attractive raw materials for lunar oxygen manufacture. The major minerals, such as olivine, pyroxene, and the plagioclase feldspars [(Ca,Na)$Al_2Si_2O_3$] occur in concentrations approaching 100%. The minor minerals generally occur at concentrations of less than 2%; however, some, particularly ilmenite, occur at concentrations of up to 20%.

The chemistry of the lunar minerals of interest has been confirmed by the analysis of Apollo samples (Williams and Jadwick, 1980). In regard to lunar oxygen manufacture,

Table 1. Apollo Sample Analyses

Compound	Mare, wt %	Highland, wt %
Analyses of Typical Lunar Olivine		
SiO_2	37.36	37.66
TiO_2	0.11	0.09
Cr_2O_3	0.20	0.15
Al_2O_3	<0.01	0.02
FeO	27.00	26.24
MnO	0.22	0.32
MgO	35.80	35.76
CaO	0.27	0.16
	<0.01	<0.01
Total	100.97	100.40
Analyses of Typical Lunar Pyroxenes		
SiO_2	47.84	53.53
TiO_2	3.46	0.90
Cr_2O_3	0.80	0.50
Al_2O_3	4.90	0.99
FeO	8.97	15.42
MnO	0.25	0.19
MgO	14.88	26.36
CaO	18.56	2.43
Na_2O	0.07	0.06
Total	99.73	100.39
Analyses of Typical Lunar Ilmenite		
SiO_2	0.01	0.21
TiO_2	53.58	54.16
Cr_2O_3	1.08	0.44
Al_2O_3	0.07	<0.01
FeO	44.88	37.38
MnO	0.40	0.46
MgO	2.04	6.56
ZrO	0.08	0.01
V_2O_2	0.01	<0.01
Na_2O	<0.01	0.13
Total	102.16	99.37

the focus is on the concentrations of silicon dioxide and iron oxide. The concentration of magnesium oxide is important for the manufacture of lunar silane. Note the comparatively high concentrations of these oxides in lunar olivine, Table 1. On balance, lunar olivine appears to be the mineral of choice for the manufacture of the propellants required for a lunar-based propulsion system.

PROPELLANT MANUFACTURE FROM LUNAR RESOURCES

Ilmenite, which is essentially ferrous titanate, can be reduced directly with hydrogen to form water, iron, and titanium dioxide. Electrolysis of the water yields oxygen, the required oxidizer, and hydrogen, which is recycled in this process (Kibler *et al.*, 1984; Gibson and Knudsen, 1984).

$$2\ FeTiO_3 + 2\ H_2 \longrightarrow 2\ H_2O + 2\ Fe + 2\ TiO_2 \tag{1}$$

$$2\ H_2O \xrightarrow{\text{electrolysis}} 2\ H_2 + O_2 \tag{2}$$

In the ideal case, 32 g of oxygen is obtained from 320.2 g of mare ilmenite, Table 1. However, *at best*, only 20% of the lunar material that must be processed is ilmenite. Therefore, 1,601 g of raw material must be processed to obtain 32 g of oxygen, a yield of only 2.0%.

It is apparent that the lunar raw material will have to be enriched in ilmenite before hydrogen reduction, *i.e.*, (1), to reduce the size of the chemical plant. Enrichment studies are currently being conducted (Agosto, 1984). Note, however, that while efficient enrichment reduces the size of the processing plant required for the hydrogen reduction step, it does not reduce the amount of lunar raw material that will have to be mined and transported. In the *ideal* case, 100.0 kg of mare material will have to be processed to manufacture 2.0 kg of oxygen.

The other major constituent oxides cannot be reduced directly with hydrogen to form water and the elemental metal. Silicon dioxide, as contained in olivine and pyroxene, can be reduced using the carbothermal process to form oxygen, silicon, and magnesium oxide. The ferrous oxide is reduced as well to form oxygen and iron (Rosenberg *et al.*, 1964a,b, 1965a,b, 1985).

$$\underset{\text{olivine}}{Mg_2SiO_4} + 2\ CH_4 \xrightarrow[\text{reduction}]{\text{carbothermal}} 2\ CO + 4\ H_2 + Si + 2\ MgO \tag{3}$$

or

$$\underset{\text{pyroxene}}{MgSiO_3} + 2\ CH_4 \xrightarrow[\text{reduction}]{\text{carbothermal}} 2\ CO + 4\ H_2 + Si + MgO \tag{4}$$

$$2\ CO + 6\ H_2 \xrightarrow[\text{reduction}]{\text{catalytic}} 2\ H_2O + 2\ CH_4 \tag{5}$$

$$2\ H_2O \xrightarrow{\text{electrolysis}} 2\ H_2 + O_2 \tag{6}$$

In the ideal case, 32 g of oxygen is obtained from 123.56 g of mare olivine (see Table 1), a yield of 25.9%. Thus, 3.86 kg of olivine will have to be processed to manufacture 1.0 kg of oxygen, providing a 13-fold advantage over ilmenite in the ideal ilmenite case.

While the Moon has abundant resources for oxygen manufacture, fuel sources are scarce indeed. There are no known sources of free or bound water or carbonaceous minerals. The concentration of solar wind implanted hydrogen of approximately 50 to 200 ppm in lunar regolith (Carter, 1984; Friedlander, 1984) appears to be too low to be of practical value for large scale fuel manufacture.

Silane is an article of commerce in the United States today, as it is used in the manufacture of chips for our electronics industry. Silane is usually manufactured by a process that may prove to be too complex for lunar application, *e.g.*, the reduction of silicon tetrachloride with lithium aluminum hydride.

It may be possible to simplify the manufacturing process while minimizing dependence on terrestrial resupply by the use of the reaction between hydrochloric acid and dimagnesium silicide (Sneed and Maynard, 1947).

$$\underset{\substack{\text{lunar derived} \\ e.g.\text{, carbothermal process}}}{2\ MgO} \xrightarrow{\text{electrolysis}} 2\ Mg + O_2 \tag{7}$$

$$2\ Mg + \underset{\substack{\text{lunar} \\ \text{derived,} \\ e.g.\text{, carbothermal process}}}{Si} \longrightarrow Mg_2Si \tag{8}$$

$$Mg_2Si + \underset{\substack{\text{terrestrial} \\ \text{resupply}}}{4\ HCl} \longrightarrow 2\ MgCl_2 + SiH_4 \tag{9}$$

In a more complex cyclic electrolysis process, it may be possible to electrolyze the magnesium chloride to form the required magnesium and hydrogen chloride, and oxygen as a byproduct. In this synthesis, water would be resupplied from Earth rather than hydrochloric acid [see also (8) and (9) above].

$$\underset{\text{lunar derived}}{2\ MgO} + 4\ HCl \longrightarrow 2\ MgCl_2 + 2\ H_2O \tag{10}$$

$$2\ MgCl_2 + \underset{\substack{\text{terrestrial} \\ \text{resupply}}}{2\ H_2O} \xrightarrow{\text{electrolysis}} 2\ Mg + 4\ HCl + O_2 \tag{11}$$

The resupply of water from the Earth may be less troublesome than that of hydrochloric acid. Water transport should be a standard item in the support of permanent lunar bases. Note that the oxygen that occurs as a byproduct in the manufacture of silane, *i.e.*, (7) and (11), can be used to support propulsion needs. The byproduct water, *i.e.*, (10), would be put to good use as well.

The technology for the manufacture of silane from lunar resources has not been studied. Process research and demonstration are required.

PROPELLANT PROPERTIES

The use of liquid oxygen is well established. It is used today to power the Space Shuttle Main Engine and the Centaur Upper Stage RL-10 Engine. It will be used in the future to power propulsion systems for advanced orbit transfer vehicles, becoming an integral part of the United States Space Station (Davis, 1983a,b; Babb *et al.*, 1984).

Comparative physical property values for oxygen, silane, and methane are presented in Table 2. Note that silane has a broader liquidus range than methane, which is a great benefit to the propulsion system and rocket engine designer. Silane is hypergolic with oxygen, which is an additional benefit. Methane and hydrogen are not hypergolic with oxygen. This gives additional complexity to the engine system.

The propellant properties of silane have not been defined adequately. However, it appears that silane has the potential to be an adequate space storable propellant.

Table 2. Physical Properties of Potential Lunar Propellants

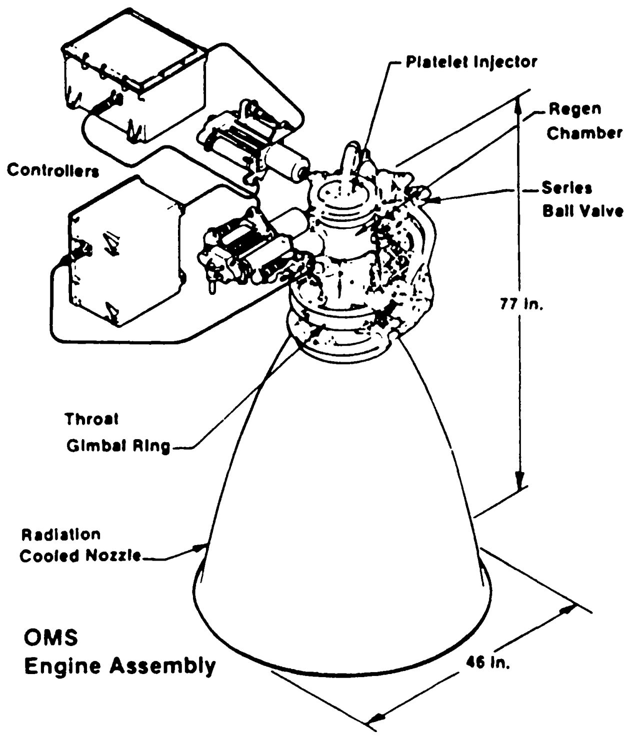
	Melting Point		Boiling Point		Specific Gravity
Propellant	°F	°C	°F	°C	(liquid)
Oxygen, O_2	-361	-218.4	-297	-183	1.142 (-297°F)
Silane, SiH_4	-301	-185	-169.4	-111.9	0.68 (-301°F)
Methane, CH_4	-296.7	-182.6	-258.3	-161.3	0.46 (-296.7°F)

1. SiH_4 is thermally stable to *ca.* 800°F.
2. O_2/SiH_4 is hypergolic.
3. SiH_4 is a liquid at the nbp of O_2.

A LUNAR-BASED PROPULSION SYSTEM

Pressure-fed liquid bipropellant engines and propulsion systems are attractive because of their comparative simplicity. Their disadvantages are lower performance and the need for heavier weight tanks and a tank pressurization system. This results in heavier propulsion systems, *i.e.*, increased propellant and hardware weights.

A pressure-fed engine in the STS Orbiter Maneuvering System Engine format seems to be appropriate for use with the $LO_2/LSiH_4$ bipropellant combination, Fig. 1. This Aerojet TechSystems engine, which delivers 6,000-lbF thrust and operates with the $N_2O_4/CH_3N_2H_3$



Engine Parameter	Actual OMS-E Value	Estimated Lunar-E Value
Thrust, lbf	6000	6000
Mixture Ratio	1.65	1.50 to 1.80
Chamber Pressure, psia	125	125 to 155
Specific Impulse, lb-sec/lb	316	345±5
Propellants	$N_2O_4/CH_3N_2H_3$	LO_2/SiH_4
Total Flowrate, lb/sec	19	17.2±0.5
Inlet Temperature, °F	40 to 100	TBD
Single Burn Duration, sec	1250*	TBD
Cumulative Firing Life, sec	54,000	TBD
Regen Exit Area Ratio	6:1	TBD
Area Ratio	55:1	TBD
Weight (dry), lbm	260 (316)	
Storage Life, yrs	10	TBD
Restarts	Unlimited	TBD
Blowdown, lbF	7,600 to 5,500	TBD
Inlet Oxid. Pressure, psia	239	TBD
Inlet Fuel Pressure, psia	242	TBD

*Limited By Tank Volume Only

Figure 1. The Space Shuttle OMS-Engine provides a format for the lunar engine.

bipropellant combination at a mixture ratio of 1.65, has a regeneratively cooled thrust chamber. The OMS-E has a delivered specific impulse of 316 s.

If the regeneratively cooled thrust chamber can be adequately cooled with silane, and there is every reason to believe it can because of silane's thermal stability (Table 2), the development of a suitable long-life, reusable engine based on LO_2/SiH_4 appears to be quite achievable. Such an engine should have a delivered specific impulse of approximately 345 s.

The pressure-fed propulsion system penalties associated with an Earth-launched system would have to be reassessed in the light of a Moon-launch system. In the 1/6 g environment of the Moon, the apparent disadvantages of a pressure-fed system may be less, and a pressure-fed engine system operating at a higher chamber pressure may be appropriate. Detailed analysis would have to be performed to address these and other issues associated with the selection of a pressure-fed propulsion system.

Eagle Engineering has studied the impact of lunar-produced silane upon lunar oxygen production logistics in the context of the transfer of the lunar-produced oxygen from the Moon to low Earth orbit. A comparison was made between the use of a pump-fed LO_2/LH_2 propulsion system (Isp_v 480 s, MR 6.0) and a pressure-fed $LO_2/LSiH_4$ propulsion system (Isp_v 345 s, MR 1.80). Lunar-produced silane was used as the fuel in the propulsion system in place of Earth-supplied hydrogen. A small gain, *i.e.*, 2.5% in mass of oxygen transferred was derived by the substitution of silane for hydrogen. The $LO_2/LSiH_4$ propulsion system offers a modest benefit despite the 135 s difference in assumed specific impulse values.

In a postscript to this report, Eagle Engineering indicated that the use of a 5.0% higher mass fraction for the stage and a 1.45% higher specific impulse for the propulsion system resulted in a 29% increase in the mass of lunar-produced oxygen delivered to orbit on each ascent (Davis, 1983b). These improvements could be attained by the development of a pump-fed $LO_2/LSiH_4$ propulsion system.

Much work remains to be done to put such propulsion systems in place. However, the work required appears to be straightforward. No inventions are required to demonstrate technology readiness and enter development and production.

CONCLUSIONS

Whether or not a lunar-based propulsion system, more particularly, one based on $LO_2/LSiH_4$, will ever be developed will depend upon many factors that are beyond the scope of this paper. However, the technology to manufacture oxygen and silane on the Moon and also to develop a lunar-based propulsion system based on the bipropellant $LO_2/LSiH_4$ combination is within our reach. The development of a lunar-based propulsion system can be accomplished within the time frame under consideration, *i.e.*, by the start of the 21st century. All that is required are the need, the will, and the funds.

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LAUNCHING ROCKETS AND SMALL SATELLITES FROM THE LUNAR SURFACE

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Scientific payloads and their propulsion systems optimized for launch from the lunar surface differ considerably from their counterparts for use on Earth. For spin-stabilized payloads, the preferred shape is a large diameter-to-length ratio to provide stability during the thrust phase. The rocket motor required for a 50-kg payload to reach an altitude of one lunar radius would have a mass of about 41 kg. To place spin-stabilized vehicles into low altitude circular orbits, they are first launched into an elliptical orbit with altitude about 840 km at aposelene. When the spacecraft crosses the desired circular orbit, small retro-rockets are fired to attain the appropriate direction and speed. Values of the launch angle, velocity increments, and other parameters for circular orbits of several altitudes are tabulated. To boost a 50-kg payload into a 100-km altitude circular orbit requires a total rocket motor mass of about 90 kg.

INTRODUCTION

The scientific investigation made possible by the Apollo project in the 1970s led to remarkable success in defining the Moon as a solar system planetary body. The synthesis of this great quantity of geochemical and geophysical data has, among other achievements, led to a theory of the Moon's origin that is more consistent with these data than previous theories have been.

Much was learned during the Apollo days about the Moon's gravity and magnetic fields, its atmosphere, and its interaction with the solar wind. However, much has been left unknown about these important topics, due in part to the restriction of the Apollo landing sites and in-orbit investigation to a fairly small range of latitudes about the Moon's equator. A powerful new attack on unsolved lunar scientific questions would be made possible by manned bases on the Moon's surface. With such resources, sounding rockets and even small orbiting vehicles carrying scientific payloads could be brought to the Moon and launched into a variety of interesting trajectories. In this paper, we attempt to define some of the technical characteristics such vehicles would have. We illustrate the use of rockets and satellites launched from the lunar surface with a few scientific experiments that are of current interest. We recognize this study to be a highly preliminary one and that the flight of a lunar polar orbiter before manned bases are established could greatly change what scientific experiments would be done. In any case, detailed planning and selection of the scientific experiments would be a prerequisite to any such program of rocket and satellite launching from the Moon's surface.

DISCUSSION

The launching of rockets or orbiting vehicles from the Moon's surface is quite different from the launching of such vehicles from Earth's surface for the following two reasons: (1) there is no atmosphere to produce aerodynamic forces on the vehicle; (2) the surface gravity is low. The acceleration due to gravity at the Moon's surface is about 1.63 m/s^2.

These environmental factors have several consequences to the design of the rocket motors and payloads:

(1) A nose cone to protect the payload from aerodynamic forces is not required. The advantage gained in this way is that total mass of the system is lowered. The reduction in mass results not only from absence of the nose cone but elimination of the nose cone ejection mechanism as well. For some payloads under some launch conditions,

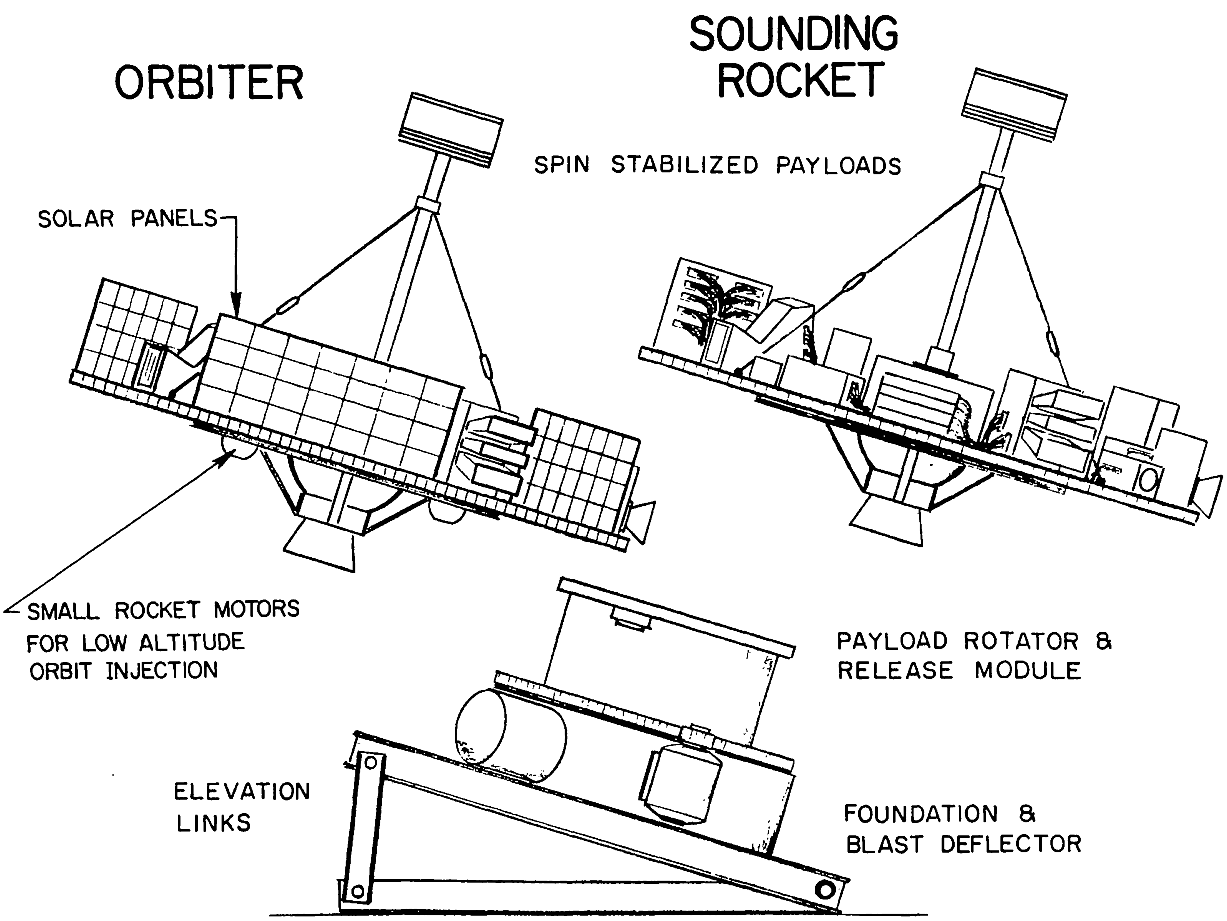

Figure 1. Possible configuration for a small scientific payload and propulsion system suitable for launch from the lunar surface. For use as an orbiting vehicle, the sounding rocket must be augmented by solar panels and small rocket motors to provide a second velocity increment. A 50-kg spin-stabilized payload can be put in a low altitude circular orbit by about 90 kg of rocket motors and fuel mass.

protection from dust blown off the surface by the rocket motor might be necessary; then a dust cover and means of opening it would be required. However, such a device would be simpler and less massive than the nose cone and ejection mechanism used in launches from Earth's surface.

(2) The absence of aerodynamic forces means that the diameter of the payload may be made as large as desired. A large diameter would result in a much simplified mechanical design and ease of assembly of the subsystem on the payload platform. For example, the added space would make layout and cabling much easier. The greater accessibility results in ease of troubleshooting and ease of replacement of subsystems or experiments. Figure 1 shows how such a payload platform might appear.

The large diameter payload has one other great advantage. The moment of inertia about the thrust axis of the vehicle, $I_{\parallel}$, could be made much larger than the moment of inertia about axes perpendicular to the thrust axis of the rocket motor, $I_{\perp}$. Thus, simple spin-stablized vehicles would have large $I_{\parallel}/I_{\perp}$ ratios, the requirement for "flywheel stability." This situation is highly desirable since there is much less tendency for large-angle precession of the thrust axis about the direction of flight. Rockets with a small $I_{\parallel}/I_{\perp}$ ratio, the condition for most Earth-launched rockets, have a tendency to go into a flat precessional motion rather than remaining nose up.

(3) The low gravitational force on the Moon's surface greatly reduces the total impulse requirement in order to achieve a given peak altitude or to attain low altitude orbital speed. In the case of chemical rocket motors, this means that the mass of the motor plus fuel is greatly reduced compared to the terrestrial situation. A further significant mass reduction follows from the absence of atmospheric drag. Because there is no need to keep the cross section of the vehicle small, the motors may be made spherical, thereby attaining a more favorable fuel-to-total-motor-mass ratio than is attainable from thin, cylindrical motors. Table 1 gives specifications of two rocket motors, one that will take a 50-kg payload to altitude of 100 km and the other will take the same payload to an altitude of 1738 km (one lunar radius). The term payload includes everything on the vehicle except the motor and its fuel.

An interesting scientific use of a sounding rocket capable of taking a plasma, particle, and field diagnostic payload to an altitude of 1738 km above the Moon's surface is indicated

Table 1. Specifications of Solid Fuel Rocket Motors Suitable for Launching Small Scientific Payloads From the Lunar Surface

	Length	Diameter	Mass at Ignition	Propellant Weight	Mass of Empty Motor
Motor A	29.4 cm	19.6 cm	13.9 kg	11.1 kg	2.8 kg
Motor B	50.5 cm	33.7 cm	41.1 kg	32.9 kg	8.2 kg

Motor A will take a 50-kg payload to an altitude of 100 km above the Moon's surface; Motor B will take the same payload to an altitude of one lunar radius (1738 km). The specific impulse of the propellant was taken to be 296 s.

in Fig. 2. When the Moon is in the solar wind, a void is formed behind it, and an expansion fan forms to fill the void. A limb shock is likely to be present, and the void and expansion fan can be expected to have interesting physical properties as well. The rocket just mentioned above would be able to sample all these regimes and their boundaries and provide a

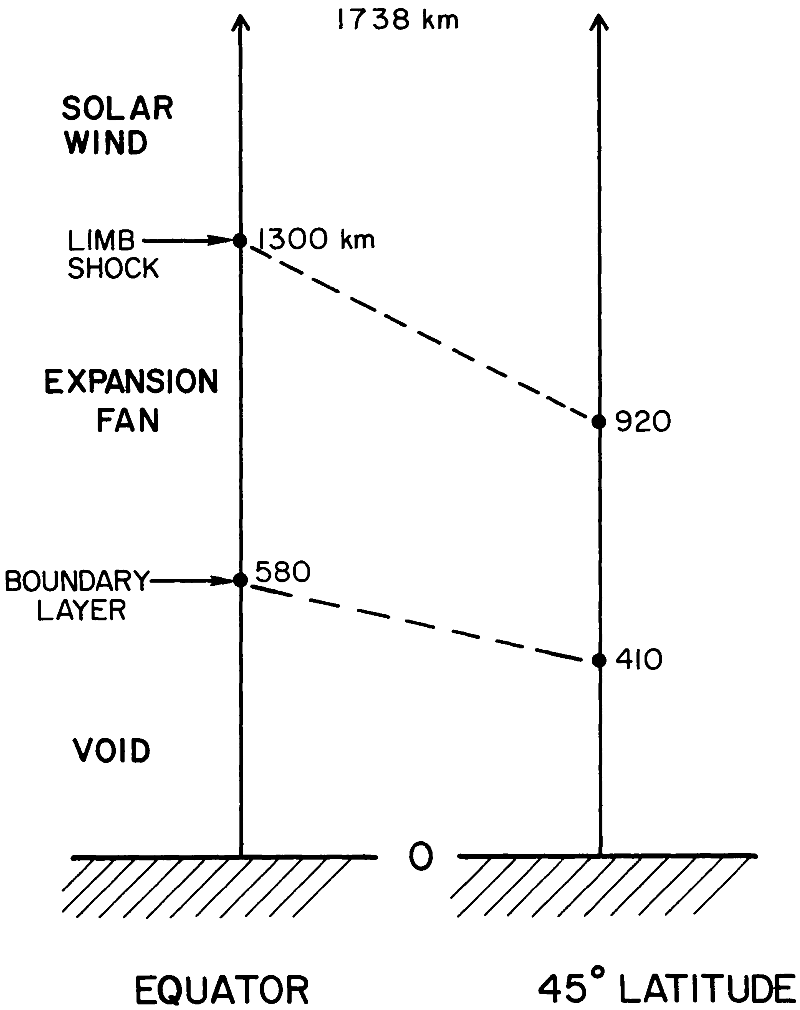

Figure 2. A sounding rocket launched vertically upward from the Moon's surface would encounter several distinctive regimes in the interaction of the solar wind with the Moon. The void, expansion fan, and solar wind would all be sampled as would two thin structures, the boundary layer between the void and expansion fan, and the limb shock.

detailed description of the boundary layer and shock structure. Figure 3 indicates where these boundaries would appear on the rocket trajectory for typical solar wind conditions and for two different launch latitudes, the equator and 45°.

We have also briefly considered the use of electromagnetic launchers (EML) to inject scientific payloads into a variety of trajectories from the lunar surface. There are several disadvantages to this method. For example, the acceleration a required to attain low altitude orbital speed of 1.686 km/s is given by

$$a = \frac{1.41 \times 10^6}{S} \text{ m/s}^2 \qquad (1)$$

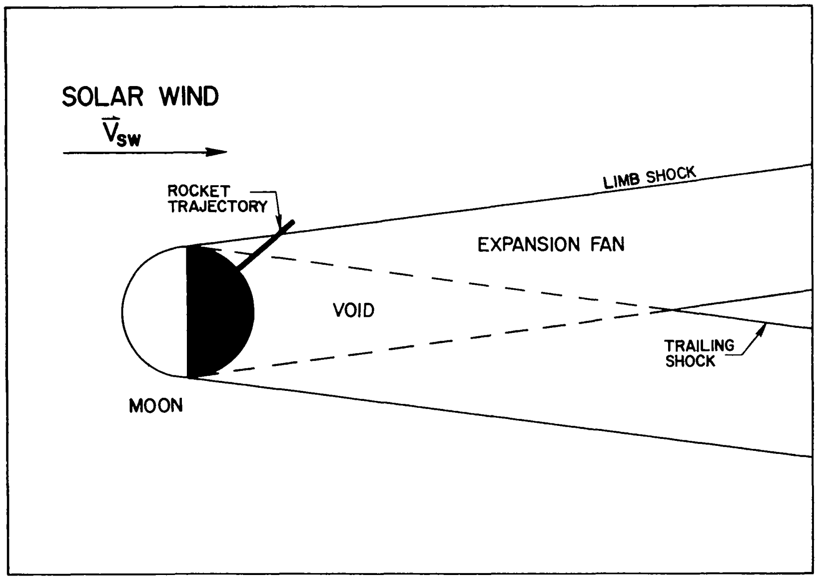

Figure 3. Altitudes at which the different features of the solar wind interaction with the Moon would appear during the rocket flight shown in Fig. 2. The peak altitude of the rocket is one lunar radius, its flight time about 45 minutes. To achieve this trajectory, a payload of 50 km requires a rocket motor plus fuel mass of about 41 kg.

where S is the length of the EML. Even a launcher 100 m in length would subject the payload to an acceleration of 14,100 m/s^2 or 1439 times Earth's gravitational acceleration. While scientific payloads might be designed to withstand such acceleration, it would be more expensive to do so. Another disadvantage of the EML is that almost certainly magnetometers would become severely contaminated by the very large magnetic fields generated by this type of launcher. Finally, an EML dedicated to launching scientific payloads would be more expensive than launch set-ups for chemically propelled vehicles (see Fig. 1). This is especially true if a variety of launch locations, azimuths, and elevations are desired.

We have also investigated launch of spin-stabilized spacecraft utilizing solid fuel motors into circular orbits of specified altitude without reorientation of the spin axis (*i.e.*, the spin axis remains inertially fixed during the injection sequence). Of course, attitude control systems could be used, but these add mass to the vehicle and increase the cost. Circular orbits can be achieved from vehicles initially at rest on the Moon's surface by using two velocity increments. The simple launch platform used for sounding rocket launchers could also be used for launch into circular orbits (Fig. 1). The first event in the launch sequence is to set the azimuth and elevation angles of the platform. Azimuth would be determined by experiment requirements. The elevation angle required to achieve a circular orbit of desired altitude is calculated by a method described below. For circular orbit altitudes in the range 50–100 km, the angle is about 3.5°. The spacecraft is then placed in the platform and spun up. The main motor is ignited and burns for about 11 seconds. A speed close to the desired low altitude circular orbit speed is reached at burnout. This velocity increment, ΔV_1, is about 1830 m/s. At burnout of the main motor, the spacecraft is approximately 8 km downrange at an altitude of 0.4 km. The second velocity increment, ΔV_2, is applied where the elliptical orbit crosses the desired circular orbit after aposelene. We have found values of ΔV_2 that result in nearly circular orbits of 50–100 km altitude. This sequence of events is illustrated schematically in Fig. 4.

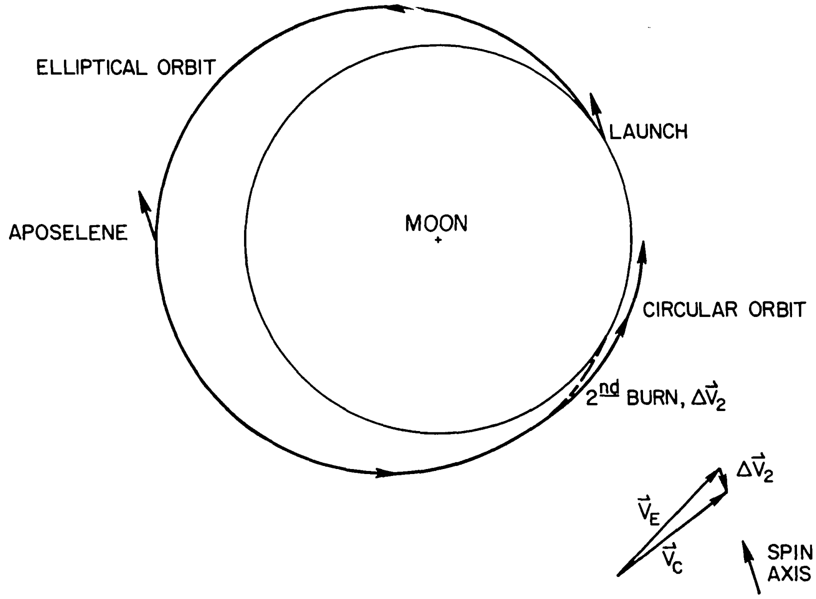

Figure 4. Illustration of the injection sequence of a spin-stabilized spacecraft into a circular orbit about the Moon. The spin axis remains inertially fixed over the elliptical trajectory. Orbit altitude is controlled mainly by the time to second burn. $\vec{V}_E$ is the orbit velocity just prior to the second burn; $\vec{V}_C$ is the resultant circular orbit velocity. The second velocity increment is much smaller than the first one.

Table 2 gives the launch angle and velocity increment necessary to achieve circular orbits for several altitudes. The second velocity increment can be achieved by small rocket motors whose thrust is directed opposite to the main motor. There is little advantage to jettisoning the main motor after burnout since its mass is so small. The mass of the small motors needed to provided the ΔV_2 would be less than 10 kg. Ignition of the small motors would be accomplished by a timer set to close the firing circuits a preset time after launch. These times are also given in Table 2.

A computer program tests Keplerian ellipses that intersect the Moon's surface and cross the desired circular orbit. The spin axis of the rocket, fixed at launch, is tangent to the ellipse where it enters the Moon. A velocity increment is made when the ellipse crosses the desired orbit. This increment is calculated to change the direction of motion to that of a circular orbit. If this increment results in the proper speed, as well, the computer prints the appropriate data. If not, another ellipse is tested. For the examples in the Table, the original eccentricity was arbitrarily fixed at $\epsilon = 0.20000$, and the aposelenes were all about 840 km. The launch angles for these ellipses are about 3.5° above the local

Table 2. Injection Parameters for Low Altitude Circular Orbits About the Moon.

					Final Orbit		
Nominal Orbit	Launch Angle, γ	ΔV_1	ΔV_2	Time to Injection	r_{max}	r_{min}	Eccentricity
50 km	3.6°	1835.0 m/s	-233.9 m/s	137.10 min	61.2 km	48.4 km	0.00356
75 km	3.5°	1835.6 m/s	-249.0 m/s	129.51 min	86.4 km	72.8 km	0.00374
100 km	3.2°	1836.6 m/s	-261.1 m/s	128.37 min	100.3 km	98.9 km	9.00035

A payload mass of 50 kg is assumed.

horizontal direction. Shallow launch angles would present difficulties only for launch sites near lunar mountains. Circular orbit insertion would take place about 310° around the Moon from launch and requires thrust from small motors directed opposite to the main motor (see Fig. 1).

The rocket will travel 8.3 km during the 11 s primary burn, but lunar gravity will deflect it into a slightly different ellipse than that calculated above. The velocity and altitude at main motor burnout are used to determine the length of the major axis of the new ellipse. The eccentricity and orientation of the major axis are determined numerically.

The most important difference between the original ellipse and the actual trajectory is the time the rocket crosses the circular orbit. The time between launch and orbit crossing is found by numerical inversion of Kepler's equation. Errors introduced by neglecting the duration of the second burn (2 seconds) are negligible. The unadjusted angle and magnitude of the velocity increment are not far from the ideal, and the final orbit is nearly circular.

Accuracies better than those obtained here are quite possible, and the quality of the orbit may be limited only by rocket motor performance and perturbations due to lunar gravity anomalies. If the required tolerances for the total impulse of the primary motor cannot be assured, the second burn could be initiated by radar altimeter, accelerometer, or Moon-based tracking.

We have not carried out analyses of the dispersion of the orbit altitude achieved because this depends on a detailed system design and engineering analysis. Nor have we attempted to determine the orbit parameters and launch times that will maximize the lifetime of near-circular low-altitude orbits.

For spacecraft having orbit lifetimes of weeks or months, a solar-cell power source would be required, as well as a set of batteries to operate the system throughout the shadow interval. Since the spacecraft is spinning, the solar cells should be mounted vertically on the periphery of the experiment platform (see Fig. 1). The angle, β, that the solar rays make with respect to a vertical line drawn from the solar cell surface is given by

$$\beta = | \lambda - \gamma | \tag{2}$$

where λ is the latitude of the launch site. For launch sites in the latitude range 20–30°, the sun's rays lie close to the equator for the spacecraft. When the launch site for polar orbiting spacecraft is the Northern hemisphere, the launch direction should be toward the north; from Southern hemisphere bases, the launches are toward the south.

CONCLUSION

We conclude with a brief list of scientific investigations that could be carried out from low altitude lunar circular orbits:

(1) A low altitude polar orbit would permit a global survey of the Moon's small scale magnetic fields by both flux gate magnetometry (Russell *et al.*, 1975) and electron reflection magnetometry (Howe *et al.*, 1974; Anderson *et al.*, 1977). Field strengths at

the surface of only 3 x 10^{-7} Gauss (0.03 nT) and scale sizes of a few hundred meters or less could be determined. The flux gate magnetometry will provide accurate directional information on the more extended magnetic fields and the electron reflectance can provide some information on direction. When combined, the two methods of magnetometry can also yield information on the size and depth of the magnetized lunar rock. One goal of such studies is to determine if the Moon once possessed an active dynamo (Runcorn, 1978) and if not, to determine the probable cause of the ancient magnetic fields. Another objective is to correlate magnetic features with surface geological features.

(2) The small scale features of the Moon's gravity field, including mascons, could be determined by tracking the low altitude orbiter (Sjogren *et al.*, 1974). Transmitters suitable for this purpose, and of sufficient power to be received by Earth stations, could be included on low altitude orbiter flights.

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