is extended farther down the target. This is in qualitative agreement with the result for the displacement field in the same target, where the displacement is of larger magnitude at the same distance from the surface and the measurable displacements extend farther down than the displacement of all dynamically impacted targets.

Conclusions

It is concluded from these studies that for the targets tested: 1) The ejecta volume was always considerably smaller than the crater volume in hypervelocity penetration; 2) the bulk of the ejecta came from a volume beneath the impact surfaces; 3) the lips of a crater contain the surface of the material originally over the crater; 4) the macroscopic deformation in a statically deformed target is considerably larger than that in a hypervelocity penetrated target of equivalent crater size; 5) the second strain invariant is not completely descriptive of the energy distribution in the target.

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A Reusable System for Deep-Space Exploration

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If nuclear-rocket propellant (hydrogen) can be found on the moon, the proposed reusable deep-space transportation system appears highly attractive for Earth–moon transportation and exploration of the near planets. For planetary missions, powered Earth swingby for departure and return permits system capabilities substantially beyond any previously proposed. The system incorporates new staging techniques that permit complete reuse. Mercury missions use a technique involving an intermediate Venus orbit. If propellant can be found on outer planet moons, exploration of the entire solar system is possible. Transit time to Jupiter is about 10 months. A refueling stop at Jupiter's fifth moon provides shorter transit times to planets beyond than any Jupiter swingby. The system is useful for economical satellite deployment and repair, and for lunar surface transportation. For some missions, ion propulsion may extend system capabilities.

Introduction

REDUCING the cost of manned exploration of the moon and planets appears essential if these bodies are ever to be studied as thoroughly as, for instance, Antarctica. The present cost of placing payload on the lunar surface is ~$30,000/lb ($300 million incremental cost per Saturn 5–Apollo mission for 11,000 lb placed on the moon, i.e. the Lunar Module ascent stage). This is about 60 times the present cost of placing payload in low Earth orbit ($400 to $500/lb for present large launch vehicles). This suggests that lunar mission costs may be reduced more by developing reusable spacecraft to operate from Earth orbit beyond, rather than from the Earth's surface to orbit, as usually envisioned.

Furthermore, we show that a reusable manned deep-space system is feasible in the context of nuclear engine performance and vehicle mass fractions attainable in the immediate future,[1] if usable hydrogen can be found on the moon. Green[2] has cited evidence that suggests that in certain (volcanic) areas, useful amounts of water of crystallization may be expected to exist in the lunar rock, and has studied methods of extracting it. The harsh lunar environment may be expected to remove such water from material near the surface, so its absence in Apollo 11 and 12 samples does not necessarily indicate an absence at greater depths. If such a propellant source becomes available, the subsequent operational cost of transporting a given payload anywhere in the solar system should be only moderately greater than the cost of putting it in orbit. The initial development cost of the proposed system would be substantial, but could be written off at relatively low cost per mission. The system comprises three parts.

Part 1 is a nuclear-powered Deep Space Vehicle (DSV), which would depart from lunar orbit fully fueled, proceed to its planetary or other destination, and return to lunar orbit without discarding any parts. An alternative plan would involve using separate means to insert the DSV into a trans-Earth ellipse and recover it from a translunar ellipse. The crew compartment would be detachable from the rest of the vehicle, which we subsequently refer to as a "DSV propulsion module," and would be capable of remotely controlled operation.

Part 2 is a single-stage Lunar Ascent and Landing Vehicle (LV), to be powered by either nuclear or chemical (hydrogen–oxygen) means, depending on mission requirements and radiation hazards. The LV would be capable of lifting into lunar orbit the DSV crew compartment and propellant supply, docking with the DSV, loading it with propellant, undocking from it, and re-docking with it upon its return. This vehicle would be required to make precisely located landings in order to allow it to be refueled. For maximum flexibility, it would need to be capable of both manned and unmanned operation. It would either wait in lunar orbit for the DSV, or return to the lunar surface until just before its return.

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Part 3 is a propellant collection, processing, and storage device (PCD), preferably having some mobility on the lunar surface. It would be required to heat collected rock to a temperature of 700 to 800°C to liberate water,[2] and then to electrolyze the water to produce liquid H_2 and O_2. Nuclear or solar power could be used for both tasks. Capability for remotely controlled operation, including refueling of the LV, would be desirable. The PCD could be carried to the lunar surface as the first payload of the first DSV-LV combination, all three units having been initially placed (via Saturn 5 launches) in Earth orbit carrying suitable amounts of propellant. The DSV would then wait in lunar orbit while the LV carried down the PCD and used it to refuel itself. The system would then be available for use.

Although design of the foregoing three components poses difficult technical problems (e.g., life support, radiation protection, and cryogenic storage for extended periods), no fundamental obstacles are known to exist. These technical problems are common to the design of nonreusable manned spacecraft and have been studied elsewhere. The value of the system will be demonstrated if the DSV is shown capable of performing useful missions and returning to lunar orbit without loss of rocket stages or other parts. Accordingly, this paper is devoted to comparisons of total Δv requirements for a number of potential DSV missions with estimated near-term Δv capabilities for such a vehicle.

The missions considered are of two types: round-trip missions not requiring refueling of the DSV en route, and one-way missions that can be made round-trip if the DSV can be refueled at destination points. The following simplifications are made in calculating total Δv requirements for the missions considered:

1) All orbits are coplanar, and orbits of moons and planets are circular except those of Mercury, Mars and Pluto.

2) All thrusting is impulsive, and all calculations use two-body orbit theory (conic-matching). Elapsed time calculations include only the heliocentric orbit durations.

3) Except where indicated, propellant reserves required for orbit adjustments and off-optimal thrusting maneuvers are not included in Δv calculations. On the other hand, the ideal payloads indicated are large enough to include ample reserves.

4) Distances of closest approach by the DSV are taken to be greater than published equatorial radii[3,4] by the following amounts: 100 km from the moon, 180 km from the Earth, 500 km from Mars, 400 km from Venus, 100 km from Mercury, 400 km from the outer planets and 500 km from their moons; the relatively large figures for these moons are related to the large-scale heights expected for their atmospheres if any.

Assumptions about DSV mass fractions and engine performance are based on the results of Dollard,[1] who has studied a nonreusable nuclear vehicle for manned Mars missions. For a propulsion module that holds 500,000 lb of liquid H_2 propellant and includes a 100,000-lb-thrust nuclear engine, appropriate scaling of his data implies a structural (i.e. nonpropellant) mass fraction of 0.251, including forward and aft docking attachments, control equipment, propellant reserves and residuals, and meteoroid shield. This implies an empty mass of 167,000 lb. He assumes that meteoroid shield and aft docking are jettisoned before ignition. For our reusable system these items must be retained. This shielding should provide a tank-penetration probability of 0.002 on a manned Mars mission.[1] For most near-Earth missions to be discussed below, propellant loss due to meteoroid impact would be noncatastrophic in the sense that the crew (and in many cases, the vehicle) could be rescued by using another vehicle, and loss of life would not occur. Over-all economy therefore might be improved by leaving the meteoroid shields in lunar orbit for certain missions and accepting either shorter vehicle life or more frequent repair in return for increased payload. Herein we assume that all meteoroid shields are retained.

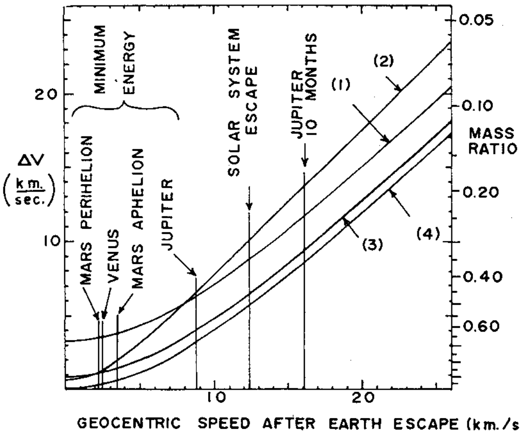

Fig. 1 Δv required for Earth departure (or arrival) as a function of geocentric velocity at asymptotic conditions (i.e. "just outside" Earth's gravity field): 1) from Earth orbit; 2) one-burn from lunar orbit; 3) two-burn from lunar orbit with powered Earth swingby; 4) same as 3 but with first burn performed by a separate stage. Mass ratios correspond to 850 sec specific impulse.

Estimates of DSV capability are therefore conservative in this respect. Dollard[1] also estimates crew module mass for a Mars mission at 50,000 to 110,000 lb.

We also assume a NERVA specific impulse[1] of 850 sec. Sufficient thrust is assumed to avoid substantial Δv penalties due to long burn times; at least 100,000 lb thrust is required.[1] A situation requiring higher thrust is discussed later.

Reusable Staging Techniques

For interplanetary missions, departure of the DSV from lunar orbit rather than from Earth orbit has two advantages: lunar orbit corresponds to a higher gravitational potential, and the Earth is available for an efficient powered swingby (gravity well) maneuver. This maneuver implies a two-burn departure with a second burn just above the Earth's atmosphere. For a given departure direction, one opportunity occurs per lunar month. These opportunities can be made more frequent by allowing the DSV to make more than one Earth orbit before the second burn.

Figure 1 shows two-body calculations of Δv required for four methods of departure: 1) departure from Earth orbit; 2) direct (one-burn) departure from lunar orbit with asymptotic direction of moon-centered hyperbola parallel to that of the moon's motion; 3) two-burn departure with powered Earth swingby; 4) same as 3 but with the first burn performed by another DSV propulsion module docked with the DSV. The latter method is clearly the best. In this "(reusable) trans-Earth staging mission," the initial configuration in lunar orbit would have two stages. The first stage would burn part of its propellant to insert the assembly into a trans-Earth ellipse. It would undock from the upper stage prior to reaching perigee. All of the propellant in the upper stage would then be available for the Earth swingby burn and subsequent burns. The lower stage would perform small amounts of thrusting at apogee or perigee to adjust its period for eventual rendezvous with the moon. It would then burn its remaining propellant to re-insert itself into lunar orbit for refueling and reuse. We will subsequently refer to a DSV propulsion module operated in this manner as a "cislunar propulsion unit" (CPU).

The inverse (translunar docking) mission would involve placing a CPU into a highly elliptical Earth orbit some time before the arrival of a DSV returning from a planetary mission. The DSV would slow itself, via powered Earth swingby, into approximately the same ellipse. The CPU would then dock with the DSV during translunar coast and slow it into lunar orbit. Since the DSV would still achieve Earth capture itself, a missed rendezvous would not be a serious mishap, because another CPU could then be launched from lunar orbit to recover the DSV. We refer subsequently to both of these (departure and arrival) strategies as "cislunar missions."

A fifth strategy involves assembling a three-stage configuration of DSV propulsion modules in lunar orbit, together with crew module and/or other payload. Each stage may consist of a single DSV propulsion module or a cluster of modules. This assembly burns part of its first-stage propellant to enter a trans-Earth ellipse. The first stage then undocks and completes a cislunar mission to return to lunar orbit. The second stage ignites near perigee and burns part of its propellant to increase vehicle velocity either as much as necessary or as much as it can; it then separates, rotates 180°, and reignites as quickly as possible to slow itself again into a translunar ellipse. After one or more Earth orbits (and suitable small amounts of thrusting to alter its orbital period), it arrives at the moon and re-enters lunar orbit, either by using its own propellant or with the aid of a second CPU. The final stage may or may not burn some of its propellant immediately after separation, depending on mission requirements. This technique will subsequently be called "(reusable) perigee staging." This combination of cislunar staging with perigee staging is clearly the best of these five possible methods. The corresponding DSV arrival strategy would involve a time-critical one-opportunity rendezvous and docking with an incoming DSV which in this case would not be carrying enough propellant to achieve Earth capture unassisted. Because the risk probably would be unacceptable, at least for manned missions, we assume that perigee staging is used only for departures. It is noteworthy that, in these staging missions, all DSV propulsion modules except those in the final stage are available for re-use after a relatively short time.

We assume that each DSV propulsion module is equipped with attachments that permit complete flexibility in staging and clustering arrangements, that the meteoroid shielding consists of removable panels, and that clusters of DSV propulsion modules for particular missions can be assembled with all shielding panels removed (and "stored" in lunar orbit) except those facing outward. If flat panels are also available to bridge gaps between modules, then the shield "wraps around" the cluster and minimum mass is achieved. A seven-module cluster is particularly advantageous for geometrical reasons. For this cluster, Dollard's data[1] imply a saving in shield mass of 205,000 lb and an increase in propellant mass fraction from 0.749 to 0.784. We may regard the Δv performance of a seven-module cluster as being close to the best attainable, since larger clusters produce only small further mass fraction improvements and there is a limit to the number of modules likely to be available at any given time, even after the system is in routine operation. In any three-stage departure configuration, a seven-module cluster would normally appear as a second rather than a first stage since Δv requirements for trans-Earth insertion are relatively small and can be handled by a smaller cluster.

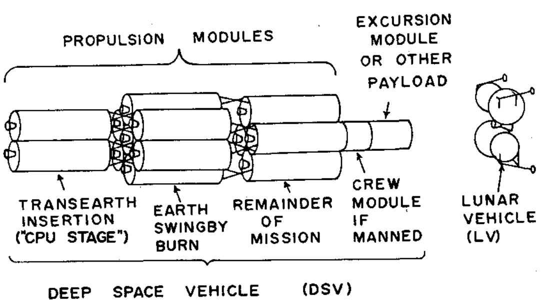

Fig. 2 2-7-4 DSV interplanetary configuration as assembled in lunar orbit. Communal meteoroid shielding on tanks is not shown. Also not shown are other DSV propulsion modules which may be necessary to re-insert all components into lunar orbit. LV also shown, without DSV propellant container. Propellant collection, processing and storage device (PCD) is not shown.

It is useful to examine the two-stage configuration remaining after trans-Earth insertion and CPU separation, but before perigee staging. Normally this assembly would also include a 110,000-lb crew module and/or some other payload. An approximate upper bound on Δv available for planetary missions from these two stages can now be obtained by assuming that the lower stage is a seven-module cluster and the only payload is the crew module, and optimizing the number of propulsion modules in the upper stage. With communal meteoroid shielding, a four-module upper stage is optimum. The lower stage can impart 4.245 km/sec prior to perigee staging, and the upper stage is capable of 11.311 km/sec. The combined capability of 15.556 km/sec is now the total available for any planetary mission which involves a Δv of at least 4.245 km/sec at Earth departure swingby. The total mass of the above assembly before perigee staging is 7,158,000 lb. A two-module cluster is sufficient as a cislunar first stage. The general appearance of the resulting three-stage initial assembly in lunar orbit is shown in Fig. 2. This will be referred to subsequently as a 2-7-4 configuration.

It is noteworthy that a 100,000-lb-thrust engine operating at 850 sec specific impulse will take about 71 min to consume 500,000 lb propellant. This long thrust duration presents a difficulty for the perigee staging previously described, since this maneuver requires the lower stage to burn almost all of its propellant close enough to perigee to gain maximum advantage from the Earth's gravity well. The available time for performing this maneuver without serious performance loss is therefore ~20 min; this implies an engine thrust requirement of ~400,000 lb Dollard's data[1] imply a propellant mass fraction loss of ~5% in raising thrust to this level. (However, an equivalent gain can be made by omitting meteoroid shielding from this stage as mentioned earlier.)

Round-Trip Missions without Refueling

Earth Missions

For these missions the DSV travels from lunar orbit to low Earth orbit, carries out rendezvous with Earth-orbital vehicles, and returns to lunar orbit. This mission requires 4 DSV burns. For a minimum-energy mission, two-body calculations yield $\Delta v = 3.137$ km/sec to enter or leave Earth orbit, and 0.821 km/sec to leave or enter lunar orbit. The DSV structural mass fraction of 0.251 implies a round-trip payload of 147,000 lb. This can be increased by clustering with communal shielding; for instance, a seven-module cluster can (ideally) transport 1,240,000 lb both ways, and much larger payloads one way.

Deployment, Repair, and Removal Missions

The DSV can enter a wide variety of geocentric and near-Earth solar orbits and return to lunar orbit. This ability would provide an economical means of deploying satellites (previously ferried from Earth orbit) in these orbits, repairing them when necessary, and eventually removing them. Cluttering of the synchronous orbit zone could be prevented. These uses alone might eventually pay for the entire system.

Mars Missions

The DSV can travel from lunar orbit to low Mars orbit carrying a Mars excursion module or MM (previously ferried from Earth orbit), carry out rendezvous with the MM ascent stage, and return to lunar orbit. With lunar departure and

Table 1 Near-planet return missions; cislunar staging assumed for lunar orbit departure and return except as noted

	Mars			Venus				Mercury	
	Min. energy		Fast capture	Min. energy		Fast capture		Flyby	From and to Venus ellipse
	Perihelion[a]	Aphelion[a]	Perihelion	Low circle	1.6-day ellipse	Low circle	1.6-day ellipse	Aphelion	Perihelion
Time to destination (months)	7.7	9.2	8.3	4.7	4.7	8.3	8.3	7.6	2.5[d]
Time in target planet orbit (months)	15.7	14.1	0.9	15.3	15.3	1.1	1.1	...	2.9
Return time (months)	7.7	9.2	8.0	4.7	4.7	8.3	8.3	7.6	2.2[d]
Δv (km/sec): Earth departure swingby	0.323	0.639	3.114	0.373	0.373	2.152	2.152	6.508	3.412[b]
Planet orbit entry	2.409	1.773	3.761	3.288	0.672	3.733	1.117	...	3.420
Planet orbit exit	2.409	1.773	3.378	3.288	0.672	3.733	1.117	...	3.362
Earth return swingby	0.323	0.639	1.907	0.373	0.373	2.152	2.152	6.508	3.120[b]
Payload (lb) ideally transportable[f] to planet orbit if 110,000-lb crew module is transported round-trip	510,000 (1)[c] 625,000 (1-1)	627,000 (1)[c] 841,000 (1-1)	286,000 (1-4-2) 1,123,000 (2-7-4)	198,000 (1-1)	1,367,000 (1)[c] 11,517,000 (7)[c]	502,000 (2-3-3)	352,000 (1-1)	633,000[e] (2-7-4)	1,167,000 (4-7-7-5)

[a] Hypothetical; Mars orbit arrival and departure cannot both be at perihelion or aphelion.
[b] At Venus swingby.
[c] No cislunar staging; add 1.642 km/sec to total Δv.
[d] From Venus.
[e] Mass transportable to impact.
[f] Bracketed numbers refer to corresponding initial DSV propulsion module configurations in lunar orbit.

arrival via powered Earth swingby, this mission involves six DSV burns. Table 1 shows ideal Δv requirements for minimum-energy missions if Mars is at perihelion or aphelion for both arrival and departure. Values for any actual missions would lie between these bounds. Assuming that a single DSV propulsion module is used, and a 110,000-lb crew module is transported round-trip, the additional payloads that can be carried one way into Mars orbit are shown. Also shown is the increased payload transportable if earth departure and arrival are aided by cislunar missions. Dollard[1] estimates MM mass at 55,000 to 220,000 lb. The indicated reserve capacity is clearly more than adequate to allow for differences between the ideal Δv values quoted herein, and actual values.

The disadvantage of minimum-energy missions is that long Mars stay times are required (Table 1). Accordingly, "fast capture" missions with Mars orbit durations of about a month, also were studied. Two types of fast capture trajectories can be distinguished: "direct" trajectories, and those in which the spacecraft passes closer to the sun than the Earth does. The latter have longer transit times but smaller Δv requirements. Rough optimization of indirect trajectories was performed in this study, for perihelion and aphelion conditions; missions for specific years were not examined, since the intent of this study is only to demonstrate system feasibility. Expressions derived and used for calculating transit time and Mars orbit stay-time are similar to those of Ref. 5. For perihelion missions, the results of this optimization are shown in Table 1. Outbound and return optimizations are different because perigee staging is assumed outbound. Total Δv for an aphelion mission is 19.07 km/sec, not including lunar orbit departure. Since this exceeds the Δv capability mentioned earlier, aphelion missions are not feasible. Since a large excess payload capacity is available at perihelion if the 2-7-4 vehicle is used, missions with this vehicle can be performed partway to aphelion conditions. Missions could also be conducted closer to aphelion if the MM were sent ahead of the crew on a separate minimum-energy mission; this vehicle could also return used MM ascent stages to lunar orbit for refueling and reuse. Occasional near-aphelion missions might also be possible using Venus swingby en route to Mars; this possibility was not studied in detail. These Mars missions would be vastly less expensive than previously proposed nonreusable schemes.[1]

Venus Missions

The DSV could travel from lunar orbit to Venus orbit, carry out observations of Venus, and return to lunar orbit. Minimum-energy and indirect fast capture missions both are feasible (Table 1); in these indirect missions, the DSV leaves Earth with a velocity component directed away from the Sun, and passes through aphelion en route to Venus (and on the way back). The excess payload capacities indicated can be traded for shorter transit times. Substantial saving in Δv results if the DSV enters an elliptical rather than a circular orbit about Venus; the figures shown in Table 1 are for an apocenter of 10^5 km. Entering an elliptical orbit will in general create a problem with the return trip because the possible directions of Venus departure asymptotes will be severely restricted. This difficulty can be overcome if less propellant is burned during arrival swingby, in order to increase apocenter; then a small amount of thrusting is used at apocenter sometime during the stay period to circularize the orbit at a large Venus radius. At the appropriate point on this circle, a small amount of thrusting reduces the pericenter to its previous value to begin the return trip.

A manned vehicle in Venus orbit would be useful in remotely guiding an unmanned mobile surface vehicle using television "eyes." To carry out operations such as manipulation of surface samples and avoidance of obstacles, such a vehicle would have to be guided from Venus orbit because the communications delay between Venus and Earth would be unacceptable. A manned vehicle in Venus orbit also might be useful in recovering surface samples returned to orbit on a small rocket vehicle. This device might be lifted partway on a balloon.

Mercury Missions

Total Δv requirements for minimum-energy round trips from lunar orbit to Mercury orbit are 18.45 and 23.09 km/sec, for Mercury at perihelion and aphelion, respectively, not including lunar orbit departure. These missions are, therefore, not possible with the proposed system (unless propellant were found on Mercury). However, at least two other kinds of Mercury missions are feasible.

One is a round-trip flyby by the DSV, which would eject an unmanned soft-landing vehicle at Mercury encounter. For this mission, the DSV would be launched away from the sun, into a heliocentric orbit with perihelion at 0.46 a.u. (Mercury's aphelion) and aphelion at 1.29 a.u. This orbit is arranged so that the DSV arrives again at the Earth after it passes through aphelion twice and perihelion once. Mercury encounter occurs at perihelion. Gravitational deflection by Mercury and/or Venus probably can be used to advantage. Table 1 indicates that a substantial payload can be carried one way and, hence, separated from the DSV and put onto a Mercury collision course.

However, reusable manned Mercury surface missions also are possible. A 4-7-7-5 DSV is assembled in lunar orbit, with crew module and Mercury excursion module (MEM). Stage 1 inserts the remainder of the DSV into a trans-Earth ellipse, and completes a cislunar mission to return to lunar orbit. Stage 2 carries the vehicle on a minimum-energy mission into a Venus ellipse and carries out orbit changes to adjust the departure asymptote (see Venus missions), and separates, with part of its propellant still unused. After the appropriate waiting period, stage 3 performs a reusable Venus pericenter staging maneuver to put stage 4 on a near-minimum-energy orbit to Mercury. Stage 3 then rejoins stage 2 in Venus orbit. Stage 4 inserts the crew module and MEM into low Mercury orbit. The MEM carries part of the crew to the surface and then back into orbit for return to the crew module. Stage 4 then returns itself and the crew module to Venus orbit for docking with the second-and-third-stage assembly. Stage 2 again carries out orbit changes to adjust the departure asymptote, and then burns its remaining propellant to bring the vehicle back to a translunar Earth ellipse. Reinsertion into lunar orbit via cislunar mission completes the mission. Total mission time would normally be equal to that for a minimum-energy Venus mission (Table 1). This mission takes advantage of the fact that the Mercury stay period for a near-minimum-energy Venus-to-Mercury round trip is almost one Mercury orbit period if Mercury arrival and departure are both near Mercury perihelion. A potentially less elaborate plan, involving electric as well as nuclear propulsion, is mentioned later.

Outer Planet Missions

The following plan is proposed for manned missions to Jupiter: An unmanned "precursor" DSV carries into orbit about a suitable Jovian moon a landing and ascent vehicle. It should be possible to use the Lunar Vehicle (LV) previously described with minor modification if this moon has a slight enough atmosphere. A PCD is also carried. The LV lowers the PCD to the moon's surface. Propellant is collected and processed, and the LV is refueled and makes a number of

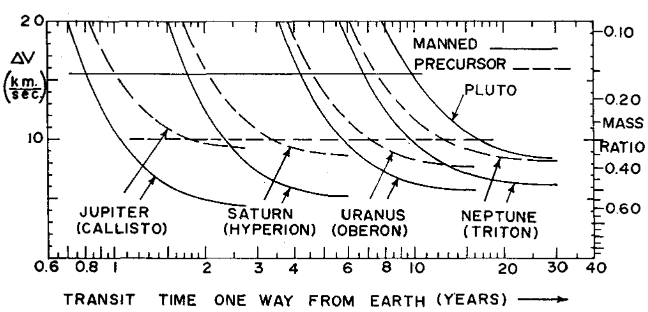

Fig. 3 Total Δv (excluding lunar orbit departure burn) as a function of one-way transit time from Earth to outer planets. Earth departure direction optimized. Broken curves include Δv for insertion into orbit about destination planet moon. Moons assumed as destinations are Callisto (Jupiter), Hyperion (Saturn), Oberon (Uranus), and Triton (Neptune). Powered Earth departure swingby and planet arrival swingby assumed. Entry into low circle assumed for Pluto arrival. Performance limits shown for precursor and manned missions are as in text.

round trips between the surface and orbit, where it refuels the precursor vehicle as a demonstration of operational readiness. A manned DSV is then launched toward Jupiter, and enters orbit about the same moon. The LV is used to carry the DSV crew module to the moon's surface for exploration of it. It then returns the crew module to orbit and refuels DSV propulsion modules as necessary. A return vehicle is assembled and departs homeward. Subsequent missions transport more DSV modules and LV and PCD units which are gradually deployed on the other Jovian moons to enable them to be visited.

As in the case of Earth, Jovian moon arrival and departure are best accomplished using powered Jupiter swingby. The mission, therefore, requires four DSV burns each way. Analogues of the cislunar missions and perigee staging technique previously described are used for Jupiter arrival and departure. Cislunar arrival missions clearly are available only after the precursor mission has provided a supply of DSV propulsion modules at the destination. The precursor vehicle, therefore, must carry its own propellant for insertion into orbit about the chosen Jovian moon.

The best moon to choose is one far enough from Jupiter that excessive Δv requirements at Jupiter pericenter can be avoided, but not so far out that the coast period from pericenter to the moon becomes a significant part of the mission duration. Jupiter's fifth moon, Callisto, appears best from this viewpoint.

The initial configuration for a Jupiter mission would have three stages. The first stage would perform a cislunar departure mission to insert the rest of the DSV into a trans-Earth ellipse. This two-stage vehicle then would have to carry out three more burns in a precursor mission, or two more once Jupiter cislunar arrival missions are available. Figure 3 shows total Δv for these two and three burns as functions of one-way transit time.

For a precursor mission, a payload of 2,500,000 lb in orbit about Callisto is sufficient to set up a refueling capability on Callisto (see below). Two precursor missions, or one mission with a large DSV assembly, are required. The upper two stages of a 3-7-6 DSV have a total Δv capability of 9.970 km/sec with 1,500,000-lb payload and perigee staging; the corresponding transit time from Fig. 3 is 1.9 yr. For the manned mission, Δv capability is 15.556 km/sec; the one-way transit time is 0.8 yr.

Saturn, Uranus, and Neptune missions are conducted similarly. The moons chosen in this study are Hyperion, Oberon, and Triton, respectively. From Fig. 3, manned missions to these destinations would consume 1.8, 4.2, and 6.8 yr each way. Only one precursor mission is necessary for Hyperion or Oberon.

These long transit times probably are unacceptable for manned missions. In principle, this system would also permit unmanned round trips with return of surface samples.

A pericenter swingby at Saturn arrival involves passage inside Saturn's rings and would, therefore, have to be conducted out of their plane, with some Δv penalty at moon arrival and consequent increase in precursor mission times. Since orbits inside the inner ring are presumably unstable, debris hazards in such a swingby are probably acceptable; this question remains to be settled by unmanned probes. Figure 3 also shows Δv for Pluto missions, assuming that Pluto's mass is 0.18 Earth masses,[6] that its radius[4] is 3218 km, and that it is at perihelion (Pluto passes through perihelion during 1985). This diagram corresponds to low-orbit entry about (and refueling on) Pluto itself, since it is not known to have moons. One-way times for the "manned" and the precursor missions are 9.9 and 16.4 yr, respectively.

The long mission times beyond Jupiter suggest use of Jupiter swingby to reduce them. It is significant that for all the (nonprecursor) Jupiter missions shown in Fig. 3, the Δv required at Earth swingby is substantially greater than at

Jupiter swingby. For example, for a transit time of 0.84 yr, these values are 8.68 and 6.13 km/sec, respectively. This means that if a refueling stop at Jupiter is made, instead of a swingby, the DSV can leave Jupiter as fast as it approached, with more propellant in its tanks. Part of this can be burned to increase velocity. A refueling stop is, therefore, better strategy than any possible swingby. Unfortunately, the saving in time is small. For manned missions, one-way time savings range from 1.8 months for Saturn to 1.8 yr for Pluto. These low values are related to the fact that velocities attainable in these missions are much higher than for the recently proposed "grand tour" outer planet missions. For the slower precursor missions, time savings range from 2.2 months for Saturn to 4.2 yr for Pluto. Refueling stops beyond Jupiter were not studied.

Use of ion propulsion may allow significant time savings. A possible outer planet mission begins with departure of an unmanned ion vehicle carrying fueled DSV propulsion modules and one or more excursion modules. A manned DSV later departs, burning all of its propellant and hence achieving a much higher velocity than in the aforementioned missions. The DSV eventually intercepts the ion vehicle and docks with it. An appropriate ion thrusting period follows. The DSV is reassembled, with a single-stage propulsion module cluster, crew compartment, and suitable other equipment. It undocks, leaving the empty DSV modules attached to the ion vehicle, which then begins to reverse its outward heliocentric motion. The DSV coasts ahead and enters the appropriate planet moon orbit(s), and the crew carries out surface exploration. Departure from the planet is followed by another intercept of the ion vehicle (or perhaps another ion vehicle, also previously sent out). More ion thrusting takes place. Again, empty modules are traded for full ones, and a reassembled DSV coasts ahead of the ion vehicle and returns home. The ion vehicle eventually returns all remaining hardware for reuse.

This concept may also be adaptable to Mercury missions. A one-way trip into Mercury orbit (or back) is within DSV capabilities if the crew compartment is the only payload, so the ion vehicle would be sent ahead with MEM and full DSV modules, and would later return empty modules and MEM ascent stage.

The planning of any manned outer planet missions must await information on the hazard involved in traversing the asteroid belt.

LV and PCD Deployment and Performance

The Lunar Vehicle (LV) must be able to carry a useful payload into lunar orbit and return to a given point on the moon. It must also be able to carry useful payloads down to the surface from orbit. Two distinct tasks may be envisioned. The first is to carry into orbit enough liquid hydrogen to refuel one DSV propulsion module (500,000 lb), together with container, and return the empty container to the surface. The other task is to carry a crew module (up to 110,000 lb) into orbit and down again, together with any supplies. The first task is more demanding and determines the size of the LV. The LV, therefore, will have a large propellant reserve when it is carrying men.

Any hydrogen found on the moon will almost certainly be associated with oxygen. This permits a choice between nuclear and chemical propulsion for the LV. If a chemically propelled LV can fulfill Δv requirements, it probably is preferable since it would not create a surface radiation hazard, and it would make high thrust and throttleability easier to achieve. It would have a lower structural mass fraction because of the higher density of its propellant, and it would provide more efficient use of surface-manufactured propellant.

Accordingly, we now analyze a hydrogen-oxygen LV (specific impulse 440 sec). Available data[7] on propellant and structural masses for the Apollo LM indicates a Δv capability of about 2.24 km/sec for descent and 2.00 km/sec for ascent. We require our LV to have the same performance. We specify ascent and descent payloads of 600,000 lb and 100,000 lb, and a propellant mass fraction of 0.80. Comparable mass fractions are 0.915 for the Saturn 5 second stage,[8] which has a comparable amount of the same propellant and 0.78 for the Apollo LM descent stage, a much smaller and, hence, less efficient vehicle. From these figures, we estimate LV propellant and nonpropellant masses of 795,000 lb and 199,000 lb, respectively.

A possible way of transporting the PCD to the moon involves use of the LV and DSV. If a fully fueled LV can be inserted into lunar orbit, the preceding information implies that it can carry a payload of 968,000 lb down to the moon if it burns all of its propellant. We may regard this as the maximum permissible mass of the initial PCD installation. We need to establish that 1) this much material (a fueled LV and a PCD; total mass 1,962,000 lb) can be inserted into lunar orbit and 2) a PCD of this mass can manufacture propellant quickly enough to be useful.

Dollard[1] indicates that the Saturn 5 can be uprated to achieve Earth orbit payloads of 560,000 lb. Eight such payloads are sufficient. One of these is a partly fueled LV. Two more consist of PCD components, together with enough propellant to complete the fueling of the LV; the propellant can be transferred during acceleration periods by using its inertia.[1] The remaining five payloads are partly fueled DSV propulsion modules. After Earth orbit assembly into a cluster, these are sufficient to transport the LV and PCD into lunar orbit as required. Demonstration of readiness then consists of deployment of the PCD on the moon, use of it to refuel an LV and thereby a DSV propulsion module, and a return trip into Earth orbit by the DSV module. The requirement of near-simultaneous launching of eight Saturn 5's can be eased if the LV-plus-PCD combination is assembled in lunar rather than Earth orbit. Deployment of subsequent PCD or LV units requires only one Saturn 5 launch at a time since LV and DSV units are now available to transport these items from Earth orbit to the moon.

These two most energy-consuming tasks required of the PCD would be dehydration of lunar rock and electrolysis of the resulting water. Heat required for dehydration has been estimated (Green,[2] p. 239) at 18,000 btu/lb H_2O (1.90×10^7 joules/lb). Electrical energy required for electrolysis[10] is about 58.0 kcal/mole water or 6.12×10^6 joules/lb. Schulman and Smith[9] quote a specific mass of 26 lb/kwe at 13.3% efficiency (i.e., 3.47 lb/kw heat) for a "1980 design" nuclear lunar powerplant. In a PCD as large as envisioned here, it is reasonable to assume that at least half of its mass, or 500,000 lb can be powerplant. These estimates imply that two LV refuelings and one DSV propulsion module refueling can be carried out every 31.7 days. This clearly is sufficient for initial use of the system, although the multimodule DSV missions envisioned herein would require deployment of additional PCD equipment for faster propellant manufacture. Green[2] has compared nuclear versus solar power for a smaller water-production system. His conclusions appear to favor nuclear power for larger systems.

The LV can be used for lunar surface round trips if refueling is available at both origin and destination, or if the propellant for the return landing is left in orbit before landing at its destination, or if another LV is launched to refuel it during return orbit.

Deployment of LV and PCD units on outer-planet moons would have to be remotely controlled since crew return would depend on success. Callisto and Triton are larger than Earth's moon, and Δv requirements for LV missions on these moons are $\sim$6% and 17% larger, respectively. This explains the larger initial deployment mass mentioned earlier for Callisto.

A potential difficulty in lunar propellant manufacture is the extremely large amount of material that must be processed; 2250 tons of water must be obtained per DSV module refueling from rock expected to contain 1 to 10% water by mass,[2] depending on its type. On the other hand, the indicated deployment payload should be large enough to provide the PCD with the required processing capacities if large enough hydrous rock deposits are found. Green[2] also cites the possibility of lunar permafrost. Furthermore, the surplus oxygen (and water) produced by the PCD should allow its cost to be written off against many other projects besides propellant manufacture.

Conclusions

A reusable deep-space transportation system is described which appears highly attractive for Earth-moon transportation and Mars exploration if nuclear rocket propellant can be found on the moon. The system also permits reusable manned missions to Venus orbit and the surface of Mercury. Mercury missions use a technique that involves an intermediate Venus orbit.

In addition, if propellant can be found on the moons of the outer planets and on Pluto, the entire solar system can be explored. A refueling stop at Jupiter's fifth moon, Callisto, provides shorter transit times to planets beyond than any Jupiter swingby strategy.

Staging techniques are proposed that improve system performance and permit recovery of all stages. A key feature for planetary missions is the use of powered Earth swingby for departure and return. The system also is useful for point-to-point lunar surface transportation and for economical satellite deployment and repair. For some missions, ion propulsion may extend system capabilities.

Uprated Saturn 5 launches are used in the deployment of the system. Since reusable Earth-to-orbit vehicles are unlikely to have comparable payload capacities in the near future,[11] development of the proposed system is independent of development of such vehicles.

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