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Mars Transfer Vehicle Using Regolith as Propellant

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This paper describes a manned Mars mission architecture that uses planetary regolith from the moon and from Deimos as propellant and as radiation shielding. It presents a point design that uses a mass driver as a rocket engine to eject regolith propellant at high speed, a solar array as a power source for the mass driver, and equipment to acquire and transport regolith to the Mars vehicle. A mission for the 2015 Earth-Mars conjunction is analyzed to assess vehicle mass and crew radiation exposure. The required mass from Earth, including lunar infrastructure, is shown to be comparable to the masses of vehicles using other types of propulsion with Earth-launched propellant. The crew radiation dose due to galactic cosmic rays is shown to be half the dose for the same mission in a conventional vehicle. Discussion includes the state of mass driver technology, issues of orbital debris, possible advantages of regolith propellant systems over other propulsion schemes, and areas for future refinement of the concept.

Nomenclature

I_{sp} = specific impulse, s
m = mass, metric tons
m_p = power system mass, metric tons
m_r = radiator mass, metric tons

Introduction

TWO significant drivers in designing a manned Mars mission are reducing the overall cost and protecting the crew from radiation. Primary factors driving mission cost are initial mass in low-Earth orbit (IMLEO) and costs to develop and to build the spacecraft. Major contributors to crew radiation dose are solar flare protons and galactic cosmic radiation (GCR). Solar flares are brief, and so the crew can minimize flare dosage by spending a few days in a small shelter. GCR is continuous, and some of its components are highly penetrating, so protection from it requires either heavy shielding of the whole occupied volume or a shortened exposure period. Because GCR shielding is so massive, current mission designs have not attempted to include it. Crew doses per mission are dominated by GCR and often exceed the allowable lifetime dose for radiation workers on Earth.[1] Further, GCR heavy nuclei that traverse the central nervous system may cause more damage than other forms of radiation.[1]

This paper describes a mission concept that has IMLEO comparable to current designs and that reduces GCR dose to the crew by half compared with conventional missions of similar duration. The concept uses planetary regolith from the moon and Deimos as both propellant and shielding. There have been many previous discussions of reducing IMLEO by using indigenous planetary materials for propellant production,[2,3] though only a few suggest using indigenous materials without chemical processing.[4–7] The use of regolith as GCR shielding for lunar habitats is a common assumption in lunar base planning,[8] and planetary soil has been suggested as GCR shielding for a low-thrust transfer vehicle,[9] but the dual use of regolith as shielding material followed by ejection as propellant is apparently a new idea. By using propellant and shielding not lifted from Earth, GCR shielding becomes practical while IMLEO remains reasonable. In addition, the concept may offer reductions in acquisition costs and in recurring costs for missions after the first.

It is assumed here that a manned lunar base is in operation before a Mars mission is undertaken or at least before the mission described here begins. It is further assumed that the lunar base includes equipment to gather loose regolith, transport it, and chemically process some of it to produce oxygen propellant for lunar landing vehicles; these elements are common in recent American plans for space exploration.[10] Operation of a lunar base implicitly requires a set of vehicles to transport crew and equipment between Earth and the moon; it is assumed that these vehicles can be used for cislunar transportation in support of the Mars mission.

This paper describes the profile of a possible Mars mission using regolith as propellant, presents a point design of a regolith-propelled vehicle, summarizes the lunar infrastructure that provides regolith for the outbound leg of a mission, describes performance of the vehicle for a 2015 conjunction-class mission, presents an analysis of crew shielding and radiation doses, and shows that regolith from the missions discussed here poses essentially no debris hazard to spacecraft. The final section summarizes the results and discusses outstanding issues. The conclusion is that the regolith propellant concept is sufficiently promising to merit consideration in future Mars mission studies.

Mission Profile

A possible mission profile is described here as an example; others exist and may be preferred in some cases. A regolith-propelled mission has two logistical phases. The first involves cislunar operations to load lunar regolith aboard the vehicle. The second includes loading regolith from a Martian moon, perhaps concurrently with Mars surface exploration. Here the Martian moon is assumed to be Deimos, but Phobos should be about equally suitable.

The initial phase of the example mission is shown in Fig. 1. All elements of the Mars vehicle except the mission payload are assembled in low-Earth orbit (LEO). The vehicle travels to low lunar orbit (LLO), Fig. 1a. There it collects lunar regolith from canisters catapulted to LLO by a launcher on the lunar surface, Fig. 1b. The vehicle then flies to the Earth-moon Second Lagrange (L2) point, consuming some regolith as propellant during the trip, Fig. 1c. The transfer habitat, the Mars excursion vehicle (MEV), and its payload are assembled in LEO and transported from LEO to L2, where they are attached to the Mars vehicle, Fig. 1d. The crew joins the vehicle at L2, and the vehicle departs L2 for Mars. (Using L2 as a

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final integration node saves about 1 km/s in ΔV for 123 metric tons (mt) of payload. (One metric ton equals 1000 kg.) If all components were attached before departure from LEO, this material would need to be propelled into and out of LLO on its way to Earth escape.)

Figure 2 shows the Mars phase of the mission. Arriving at Mars, the vehicle drops the MEV, Fig. 2a. The MEV aerocaptures into low Mars orbit and lands. Meanwhile, the main vehicle spirals in to rendezvous with Deimos, where it acquires a new load of regolith. It then spirals down to low orbit, Fig. 2b, picks up the MEV ascent stage returning from the surface, Fig. 2c, and spirals back out to Deimos, Fig. 2d. The vehicle reloads with regolith and then flies back to the Earth-moon L2 point.

At L2, the crew leaves the vehicle and returns to Earth in a cislunar spacecraft. The Mars vehicle proceeds to LLO for reloading. The only equipment to be replaced before the next mission is life support expendables and the MEV. This compares favorably with most other mission concepts, in which at least the propellant and in some cases the entire vehicle must be replaced with material from Earth.

The mission profile just described is used for analysis throughout this paper, but many details could be changed. For example, lunar regolith might be launched directly to L2 rather than to low lunar orbit; at Mars, MEV aerocapture into orbit could be avoided by having the main vehicle carry the MEV to low orbit, though this would delay the initial landing. Further study is needed to choose an optimum profile.

Vehicle Point Design

Where possible, the assumptions used here are consistent with those used in an ongoing study of vehicle concepts for Mars exploration.[11] The outbound mission payload of 123.6 mt and the inbound payload of 42.6 mt are identical to the payloads assumed for all vehicles in the reference study.

Regolith Rocket Motor

Use of a mass driver (also known as a coaxial electromagnetic accelerator or coil gun) to eject regolith propellant out the rear of a vehicle was proposed by O'Neill.[12] The principle is that ejecting dirt at high speed produces thrust, just as ejecting hot gas from a conventional rocket does. Figure 3 shows a simple schematic of a mass driver rocket engine. Propellant material is loaded into a bucketlike container. An electric current flows in an electrical coil around the bucket. The loaded bucket is positioned at one end of a series of coils that initially carry no current. To eject a load of propellant, the drive coils are energized in sequence. The first energized coil magnetically attracts the bucket coil, pulling the bucket forward. As the bucket passes through the first coil, that coil turns off and the second coil is energized. The sequence continues, magnetically accelerating the bucket through each drive coil. After accelerating through the drive coils, the bucket enters a series of braking coils. Each braking coil is turned on immediately after the bucket passes through, magnetically slowing the bucket. As the bucket decelerates, propellant material exits the open end of the bucket and leaves the vehicle at the maximum speed attained by the bucket. The bucket stops at the final braking coil and is reused.

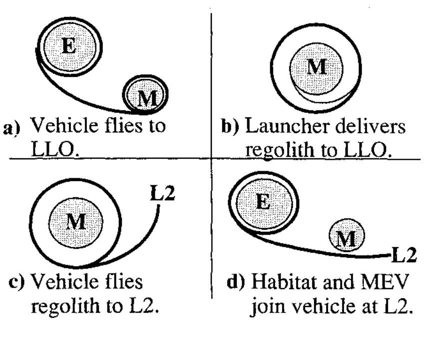

Fig. 1 Mission profile near the moon.

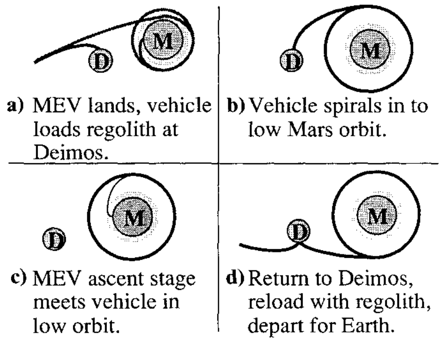

Fig. 2 Mission profile near Mars.

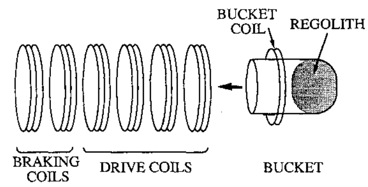

Fig. 3 Schematic of mass driver rocket motor.

The rocket design used here is based on parameters of a mass driver design by Snow et al.[13] Their design was to fire 125-kg projectiles (50-kg bucket with 75-kg cargo) at 1703 m/s with a maximum rate of one shot every 27.8 min (6.47 mt/day). Its mass was to be 135.6 mt, plus 0.7 mt of radiators. This design must be somewhat modified for use as a rocket motor. The first modification is one of scale. The rocket design is scaled to fire 2-kg projectiles (1-kg bucket and 1-kg regolith) at 3000 m/s with a rate of one shot every 4.49 s (38.5 mt/day). Mass driver weight is proportional to projectile kinetic energy, i.e., it scales linearly with projectile mass and quadratically with speed. Thus, the basic scaled mass of the design is

$$m = 135.6\ \text{mt}\left(\frac{3000\ \text{m/s}}{1703\ \text{m/s}}\right)^2\left(\frac{2\ \text{kg}}{125\ \text{kg}}\right) = 6.7\ \text{mt} \qquad (1)$$

The second modification concerns the projectile's destination. The design of Snow et al. launched the whole projectile out the end of the mass driver; the empty projectile was to be returned later for reuse. To recycle buckets as needed in a mass driver rocket, braking coils must be added to catch empty buckets and recover some of their kinetic energy. The bucket mass is half of the projectile mass, so the braking coils comprise half as much mass as the drive coils. Adding braking coils increases the overall motor mass by 50% and brings the total number of coils to 1607.

The third modification is to provide redundancy. Any of 118 spare coils (7.34% in excess of the required 1607) can be activated if primary coils fail. Assuming a 5% chance for each coil to fail during the mission, this redundancy means that the chance for at least 1607 coils to still be working at the end of the mission is greater than 0.9998.[14]

The addition of coils for bucket braking and redundancy gives a revised mass of

$$m = 6.7\ \text{mt} \times 1.5 \times 1.0734 = 10.8\ \text{mt} \qquad (2)$$

The radiator design of Snow et al. must be modified to handle higher production of waste heat. Radiator mass is assumed to scale linearly with the waste heat rate. Waste heat rate is proportional to the power used by the mass driver, which is proportional to the mass flow rate times the square of the exhaust speed. Thus, the radiator mass of Snow et al. of 0.7 mt becomes

$$m_r = 0.7 \text{ mt} \left(\frac{38.5 \text{ mt/day}}{6.47 \text{ mt/day}}\right)\left(\frac{3000 \text{ m/s}}{1703 \text{ m/s}}\right)^2 = 12.9 \text{ mt} \qquad (3)$$

Total motor mass is 23.7 mt, including the radiator. The motor has a much higher fraction of its mass devoted to radiators than the design of Snow et al. This is due to the much higher shot rate in the motor application, which yields a higher rate of heat production per unit mass.

The mass computed here may seem speculative because it is derived by scaling through more than an order of magnitude in size from a design that was never implemented. However, recent work at Sandia Labs has produced several working coil guns at a scale similar to that of the proposed motor. These guns have demonstrated speeds over 1000 m/s with a 0.16-kg projectile and speeds of 400 m/s with a 5-kg projectile.[15] One of the guns has used braking coils to stop the bucket while the bucket's payload continued out the end of the gun. (The mass of the hardware has regrettably not been published in the open literature, but personal communications[16] with the researchers indicate that their hardware is within 10% of the mass predicted by scaling from the design of Snow et al.) The Sandia group foresees no technical obstacles in achieving speeds above 6000 m/s with ton-sized projectiles.[17] Thus the postulated rocket ejection speed of 3000 m/s should be easily exceeded with modern mass driver technology. The propellant speed of 3000 m/s and the projectile size of 2 kg used here were selected because these values are well within reach of near-term technology; optimization of these parameters for a Mars mission remains to be done.

Regolith Tankage and Handling

A small but essential component is propellant tankage and handling. Regolith is stored in rectangular bags. Each bag is 1 m square and averages 20 cm thick when filled. Assuming a density of 1500 kg/m^3 for sieved regolith, each bag holds 300 kg of regolith. The bags are made of 1-mil aluminized plastic; assuming 50% extra material for sealing, each bag's mass is about 0.107 kg. To contain the regolith for the outbound leg of the mission requires 4564 bags, and so the bag mass for that leg is 488 kg. A comparable number of bags are carried to store regolith for each of the other two mission legs, for a total bag mass of roughly 1500 kg. The low total mass of the regolith containment system may surprise readers accustomed to the mass of propellant tanks in a more conventional spacecraft. The vital distinction is that regolith is a solid with negligible vapor pressure. Liquid propellants require relatively heavy pressurized tanks; regolith needs only enough containment strength to resist the acceleration loads of low-thrust propulsion.

The regolith bags are removed from the habitat and carried to the bucket-loading point by any of several small robots. A special device opens the bags and puts the regolith into the buckets. The total mass of the robots and the bucket loaders is assumed to be about 500 kg, comparable to the mass of eight Puma industrial robots. It is assumed that the pipelined process of removing an empty bucket from the mass driver, attaching it to a filling fixture, removing the filled bucket from the fixture, and loading it into the mass driver requires a total of four robots. The filling device is equivalent in mass to one robot. Another robot is dedicated to periodic inspection and possible replacement of buckets. The small robots that fetch the regolith bags are assumed to have total mass equivalent to one robot. Spare buckets and robot mounting brackets are assumed to have mass equivalent to another robot. No special machinery is required to fetch empty buckets from the braking end of the motor; the mass driver coils can be fired in reverse sequence at low power to return empty buckets to the loading point. The total mass of these regolith handling devices and the regolith bags is 2.0 mt.

Electric Power System

The electric power system is scaled down from that of a solar electric ion propulsion (SEP) vehicle designed as one option in the Woodcock study. The reference SEP power system delivers 10 MW at 1 astromical unit (AU) from the sun with a mass of 63.6 mt, including structure, power processing, radiators, attitude control, and growth factors.[18] Preliminary work on vehicle size and trajectory shows that a solar electric power supply with 2.6-MW capacity at 1 AU is adequate (though not necessarily optimum) for the regolith propulsion system, given 50% motor efficiency. Scaling linearly with power capacity gives a power system mass of

$$m_p = 63.6 \text{ mt} \left(\frac{2.6 \text{ MW}}{10 \text{ MW}}\right) = 16.5 \text{ mt} \qquad (4)$$

A solar electric system was chosen arbitrarily for this point design. Further study may show that a nuclear generator is better suited for the mission.

Mission Payload

As noted earlier, this analysis assumes the same payload as an earlier study.[18] The total outbound payload of 123.6 mt includes a 40.3-mt transit habitat, 1.7 mt of experiment platforms, a 0.6-mt communication system, and a 45.2-mt MEV with its own payload of a 10.8-mt rover and 25.0 mt of surface equipment. The inbound payload of 42.6 mt includes the transit habitat, experiment platforms, and communications.

A payload element not accounted for is equipment to gather regolith on Deimos and load it on the Mars vehicle. This equipment has not been designed; clearly a design must be available before a valid mass estimate can be made. However, enough is known to show that reloading is a tractable problem. Thermal data from Mariner and Viking suggest that the regolith of Deimos is quite similar to lunar regolith, at least in average grain size, and may be over 100 m deep.[19] A probe mission is needed to measure properties in more detail, but the observed similarity to lunar regolith implies that Deimos regolith should be a fine loose powder. The very low gravity of Deimos and the powdery regolith imply that excavation should be physically undemanding, and so the equipment can be lightweight, with mass perhaps as little as 1 mt. The Mars vehicle's solar array can provide plenty of power to the equipment by cable or microwave beam, and so the task should be accomplished quickly. Some crew members are aboard the Mars vehicle during the Deimos loading phase and can direct the equipment telerobotically or can don spacesuits to control the work directly.

Another unaccounted payload element is any science payload specific to investigation of Deimos. Such equipment is not needed to successfully complete the mission to Mars, but it seems unlikely that the opportunity to study a minor planet would be ignored.

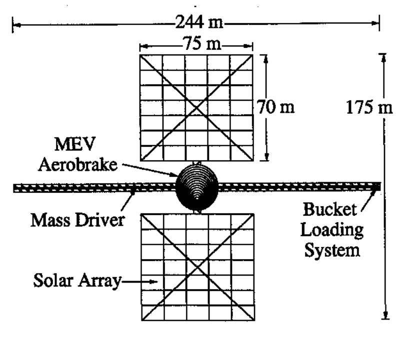

Fig. 4 Vehicle configuration.

Figure 4 shows a sketch of the mass driver vehicle. The long central truss surrounds and supports the mass driver, 244 m in length. The elliptical object in the center is the MEV aerobrake; other elements of the mission payload are behind it. The two large panels are solar arrays. Each is 70×75 m in size. Robotic assembly methods for similar arrays have been established for the reference SEP vehicle.[20] The arrays pivot with respect to the mass driver truss and the payload, tracking the sun independently of the mass driver thrust vector.

Lunar Infrastructure

Over 2000 mt of lunar regolith must be launched into lunar orbit to support the mission design presented here. This requires a substantial set of equipment on the moon and in lunar orbit, some of which is assumed to be part of a pre-existing lunar base. The additional lunar infrastructure needed for the Mars mission has several elements: a launcher on the moon, smart projectiles that carry regolith from the launcher to lunar orbit, equipment on the orbiting Mars vehicle to collect the regolith, and a de-orbit device to help return empty projectiles to the lunar surface. Each is described briefly in one of the following sections. Equipment to gather regolith and equipment to produce liquid oxygen (LOX) are assumed to be part of the base; they are not described here.

The system operates as follows. The launcher throws smart projectiles full of regolith into lunar orbit. Each projectile uses onboard propulsion to raise its orbital periapsis above the lunar surface and later to rendezvous with a collector device on the orbiting Mars vehicle. The collector removes the regolith from the projectile and adds fuel brought from Earth. The de-orbit device shoots the projectile backward at a relative speed equal to orbital speed, so that the projectile falls vertically to the lunar surface from an orbital altitude of 10 km. Onboard propulsion slows the projectile for a low-speed impact. On the surface, the projectile is reloaded with regolith and lunar LOX before the next launch.

The system delivers 1000 mt of regolith to low-lunar orbit each year, so that a regolith-propelled vehicle can depart at nearly every Earth-Mars conjunction opportunity. Most of the infrastructure is described in detail in a previous paper[21] that presented a design intended to transport 1000 mt of general lunar material to orbit annually. The de-orbit device has been amended here to meet the specific needs of the regolith-propelled Mars vehicle.

Launcher

In the 1970s and early 1980s it was often assumed that an electromagnetic accelerator would be used to launch lunar material into space.[22] More recent work[23] suggests that a rotary launcher built of modern materials would require less mass to achieve the same material throughput. A rotary launcher design proposed previously[21] is used here. The design is summarized next.

The launcher consists of two horizontal rotors. Each is sized to launch a 44.4-kg projectile from either end of the rotor at lunar orbital speed. The two rotors are used in an alternating sequence; kinetic energy left in one rotor after projectiles are launched is extracted and used to spin up the other rotor. The mass of the two rotors, motor generators, gearboxes, and mounting equipment is 9.2 mt. The rotors operate during daylight only, powered by a photovoltaic system designed for robotic deployment.[24] The power system mass is 15.0 mt. The equipment to load regolith and LOX into projectiles and to load projectiles into the launcher is assumed to have a mass of 2 mt.

Estimated total mass of lunar surface equipment devoted specifically to the Mars mission is 26.2 mt.

Projectile Design and Propellant Consumption

The projectile design is based on an earlier design by Buddington et al.,[25] which is resized here based on modern small rocket technology from the Strategic Defense Initiative.[26,27] In the earlier infrastructure paper the projectiles were designed for rotary launch and rotary de-orbit; here they are assumed to accommodate rotary launch and electromagnetic de-orbit. Projectiles need relatively little onboard intelligence, receiving much of their guidance information from the Mars vehicle in lunar orbit or from beacons at the landing site on the lunar surface.

Average propulsive ΔV for a projectile is 316 m/s per cycle: 104 m/s to enter parking orbit, transfer to the collector orbit, and maneuver for docking plus 212 m/s for an average braking burn at the end of the 10-km vertical fall. It is assumed that the projectiles are small enough and rugged enough to survive a low-speed impact and do not require an actual "soft landing" as a manned vehicle would. The projectiles burn lunar LOX and fuel from Earth with an I_{sp} of 310 s. It is assumed that 80% of the propellant mass is LOX.

Empty mass of each projectile is 16.7 kg, its payload is 25 kg, and its propellant capacity is 2.8 kg. On average, 2.65 kg of propellant are consumed per cycle. At 25 kg of regolith per launch, with 2132 mt launched for one Mars mission, 85,280 launches are needed per mission. This requires 226 mt of propellant. Lunar LOX comprises 180.8 mt of the total, and fuel delivered from Earth to LLO comprises 45.2 mt.

The number of projectiles is 385, which requires each projectile to carry 222 loads to orbit for one Mars mission. This number of projectiles is based on average time spent in parking orbit; it includes no allowance for attrition. Total dry mass of projectiles is 6.4 mt.

Estimated total mass of projectiles and propellant from Earth is 51.6 mt. Note that this figure is very sensitive to the orbital altitude of the collecting vehicle and to parameters of the parking orbit. With a lower altitude, for example, this figure may be significantly reduced. More study is needed to determine optimal values.

Projectile Handling in Orbit

Regolith containers in lunar orbit must be guided, captured, unloaded, refueled, and sent back to the surface. Guiding, capturing, and unloading the projectiles is assumed to require 5 mt of equipment attached to the Mars vehicle. Propellant storage and equipment for refueling the projectiles require an additional 5 mt. The equipment need not be fully automatic; the projectiles arrive at about 6-min intervals, which allows adequate time for human telerobotic control or intervention.

All projectile handling equipment is left in lunar orbit when the vehicle departs and is reused for later missions.

De-Orbiting Projectiles

Conventional rockets for de-orbit and landing of empty projectiles would consume far too much propellant to be cost effective. The author's earlier infrastructure paper addresses this problem with a rotary de-orbit device intended for a general cargo transport system. If a de-orbit system will be used only to support a regolith-propelled Mars mission, then the more specialized system described next can meet the need with lower mass.

For the specialized needs of the Mars mission, a second mass driver is mounted on the Mars transfer vehicle when the vehicle is in low-lunar orbit. This second mass driver points aftward. It ejects empty projectiles at 1700 m/s, which is zero velocity relative to the lunar surface. Each projectile falls vertically for nearly 10 km, gaining 181 m/s in speed, then performs a braking burn to allow a low-speed impact near the launcher. The 212 m/s average braking burn noted earlier allows lateral position correction.

This de-orbit mass driver is sized to eject 16.7-kg projectiles at 1700 m/s. Scaling from the design of Snow et al., its mass is 18.1 mt, and the associated radiator mass is 0.4 mt. Its power comes from the vehicle solar array. The de-orbiter does not operate continuously; rather, projectiles are dropped in groups of 18 as the vehicle flies over the landing sight for a few seconds during each orbit. Therefore, an energy storage system is needed to quickly provide energy for 18 shots. A pair of counter-rotating homopolar generators meets this need with mass of 3.0 mt.

The mass of the de-orbit mass driver and its energy storage is 21.5 mt. This hardware is left in lunar orbit with the projectile-handling equipment described earlier and is reused to support later missions.

Ejecting projectiles at 1700 m/s backward relative to the vehicle produces thrust that tends to raise the vehicle's orbit. To counter this, the vehicle's main motor ejects regolith forward at 3000 m/s. Of the 2132 mt of regolith that reaches orbit, 807 mt is ejected this way.

Total Mass of Vehicle and Infrastructure

The vehicle and infrastructure mass for the first mission are shown in Table 1. The table does not include elements of the lunar base that are likely to be present for reasons unrelated to the Mars mission. The table lists items by the location (lunar surface, lunar orbit, L2) to which they must be delivered by the cislunar transportation system. For example, the Mars vehicle motor must be delivered to LLO before it can obtain regolith with which to propel itself. Total mass delivered to each location is shown in the table. The total mass of all elements is 275.1 mt.

Computing an IMLEO value for the mass breakdown of Table 1 is problematic because cislunar transportation technology is difficult to predict. A cislunar transportation system supporting a lunar base could include chemical, electric, nuclear, or solar thermal vehicles for orbit transfer and may involve the use of lunar oxygen in any of several modes. It is arbitrarily assumed here that the cislunar orbit transfer technology has low thrust and I_{sp} of 1000 s, so that items delivered to L2 have IMLEO equal to about 1.8 times the delivered mass and items delivered to LLO have IMLEO equal to twice the delivered mass. Lunar landing is assumed to use chemical rockets with propellant from Earth, so that components delivered to the lunar surface have IMLEO equal to three times the delivered mass. With this assumption, the total IMLEO of the components in Table 1 is 552 mt. Obviously, different assumptions about cislunar transport would lead to different IMLEO estimates.

The IMLEO totals for most vehicle concepts considered in Ref. 18 are shown in Table 2; the gas core nuclear rocket was omitted due to poor definition of the technology. The concepts include a vehicle with cryogenic propellants in an all-propulsive mission (CAP), a cryogenic vehicle with an aerobrake for capture into Mars orbit (CAB), a nuclear thermal rocket vehicle (NTR), a vehicle with solar electric propulsion (SEP), and a vehicle with nuclear electric propulsion (NEP). All had been subject to moderate design optimization at the cited phase of the study. Estimated mass of the unoptimized regolith-propelled vehicle (RP) is shown for comparison. All vehicles carry the same payload. However, the SEP and NEP vehicles were sized for opposition missions; the others were sized for conjunction missions. The SEP and NEP values would be substantially lower for conjunction missions.

Table 1 Mass from Earth of vehicle and infrastructure elements

Element	Mass, mt
Lunar surface	
Rotary launchers	9.2
Projectile loading	2.0
Power system	15.0
Total for lunar surface	26.2
Low-lunar orbit	
Mars vehicle motor	23.7
Mars vehicle power	16.5
Prop. tankage and handling	2.0
Communications system	0.6
Projectile handling	10.0
De-orbit equipment	21.5
Projectiles	6.4
Projectile propellant	45.2
Total for low lunar orbit	125.9
Earth-moon L2	
Transit habitat	40.3
Experiment platforms	1.7
MEV	45.2
Rover	10.8
Mars surface equipment	25.0
Total for Earth-moon L2	123.0

Table 2 Mass from Earth for several vehicles

Concept	IMLEO, mt	Next mission, mt
CAP	575	575
CAB	801	801
NTR	735	686
SEP	422	276
NEP	524	271
RP	552	133

The right-hand column in Table 2 presents the approximate mass that must be delivered from Earth to prepare the vehicle for a second mission, including life support expendables and MEV replacement. The next mission masses do not necessarily represent mass to LEO: the NTR ends its first mission in a highly elliptical Earth orbit, the SEP vehicle ends its first mission in geostationary Earth orbit (GEO), the NEP mission ends in a nuclear safe orbit above LEO, and the RP mission ends at L2. Again, the SEP and NEP masses would be substantially lower for conjunction missions.

The values in Table 2 suggest that the regolith-propelled vehicle is competitive in IMLEO with most other vehicle concepts and that it may have a substantial mass advantage in missions after the first. In most cases the next mission mass is dominated by propellant; for example, argon propellant for a second SEP mission would total 140 mt. In contrast, a second regolith-propelled mission would need 45.2 mt of additional propellant for the regolith-carrying projectiles and 1.5 mt of regolith bags. It is possible that a regolith-propelled vehicle could further reduce its next mission mass by returning the MEV ascent stage for refurbishing and reuse. The cost to return it is simply more regolith loaded at Deimos. Other types of vehicle would need extra propellant from Earth to return an MEV.

Mission Performance

The spiral to transfer from LLO to a stop at L2 has a ΔV of 804 m/s. For this leg, the vehicle consists of the motor, power system, communications, and regolith bags; total empty mass is 42.8 mt. The payload is 1004 mt of regolith delivered to L2. Delivering the whole package of vehicle plus regolith expends 321.8 mt of regolith propellant. Thus, loaded mass in LLO is 1369 mt.

The CHEBYTOP software code[28] reveals that an example outbound trajectory for a 2015 conjunction mission (departure in December 2013) takes 279 days from Earth escape to Mars encounter at zero relative velocity. The motor is active for 52 days, ejecting 980.7 mt of regolith. Regolith remaining at Mars encounter is 23.4 mt. The ΔV for the trajectory is 5473 m/s.

As the vehicle approaches Mars, the MEV is released to aerocapture into low-Mars orbit. The main vehicle spirals into rendezvous with Deimos carrying the transit habitat and 23.4 mt of regolith. The spiral ΔV is 430 m/s. It uses 14.3 mt of regolith propellant and takes about 2 days. Reserve ΔV capability with the remaining 9.1 mt of regolith is 311 m/s.

After reloading with 626 mt of regolith at Deimos, the vehicle spirals down to LMO, picks up the lander crew, and returns to Deimos. Total ΔV for the trip is 6140 m/s. The

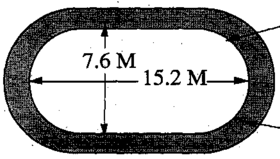

Fig. 5 Regolith propellant stored around habitat as shielding.

inward spiral takes about 70 days, and the outward spiral takes about 25 days. At return to Deimos, 11.6 mt of regolith remain for contingency ΔV. An equatorial orbit for rendezvous with the MEV is assumed here. Spiraling down to a polar orbit instead would add roughly 1341 m/s of ΔV for orbital plane changes. This would require somewhat more regolith from Deimos and more time but would have very little impact on vehicle cost or mass aside from more bags to hold regolith.

The inbound leg is similar to the outbound leg but uses less regolith propellant due to the much smaller inbound payload.

Shielding and Crew Doses

Figure 5 shows how regolith propellant is stored around the outside of the transit habitat. Table 3 summarizes the performance of the regolith layer in shielding the crew from GCR during various mission phases.

Before departure from L2, 1004 mt of regolith propellant surround the habitat to a depth of about 1.2 m. During trans-Mars injection (TMI), bags are removed in sequence from small sections of the shield layer, reducing each section from 1.2 m depth to 0.2 m before the next section is tapped. The reason is that secondary neutrons may cause a slightly higher total dose rate at depths between 0.2 and 0.9 m than at 0.2 m.[29] If regolith were removed evenly from the whole shield, the crew would get a higher dose while the depth was between 0.2 and 0.9 m.

After TMI, about one third of the habitat is covered by 1.2 m of regolith, with 0.2 m over the rest. The GCR dose rate at 0.2-m depth is about 12 rem/yr. This is assumed to be the average rate during TMI and the coast to Mars.

Nearly all remaining regolith is ejected during approach to Mars and rendezvous with Deimos. In the first half of this period, remaining thick sections of the shield are reduced to 0.2-m depth; a dose rate of 12 rem/yr is assumed. In the second half, shield depth shrinks to zero. The assumed dose rate in the second half is 18 rem/yr, the average of 12 rem/yr (0.2-m depth) and 24 rem/yr (rate in unshielded habitat).

When loading regolith at Deimos, Deimos blocks half the sky so that the basic dose rate is halved. Regolith depth is at least 0.2 m after 12 days. A dose rate of 12 rem/yr is assumed for the first 12 days, and 6 rem/yr for the remaining 48 days.

The spiral from Deimos to LMO takes 70 days; average shield depth remains more than 0.2 m. The return to Deimos takes 25 days and is comparable to the earlier approach to Mars and rendezvous with Deimos. The second loading operation at Deimos is similar to the first. The dose rate for the inbound trip to L2 is about the same as for the outbound trip, though the shield is thicker. Approach to L2 is like the approach to Deimos but leaves the habitat surrounded by enough propellant to return to LLO, and so the dose rate is lower.

The total estimated GCR dose for the 740 days counted is 24.3 rem, about half the 48.7-rem dose received by astronauts who spend the same period in a relatively unshielded habitat.

Several caveats about this estimate must be noted. There is considerable uncertainty in computing equivalent doses for GCR.[30] The GCR flux varies substantially throughout the solar sunspot cycle, so that the crew dose for a given level of shielding will vary depending on the year in which a mission is flown. Crew members at the Martian surface have a different set of radiation hazards than those who rendezvous with Deimos. The doses listed earlier do not account for the whole mission; there will be over 100 additional days spent waiting for the trans-Earth window to open. The safest places to spend this time are at Deimos or in low Mars orbit, where half the sky is shielded by a planetary body. Spending 100 days at Deimos while fully shielded yields a GCR dose of only 1.6 rem.

Table 3 GCR dose by mission phase

Mission phase	Dose rate rem/yr	Duration days	Dose rem
Pre-Departure	10	~ 1	0.0
Outbound	12	255	8.4
Deimos approach	18	30	1.2
Loading	7.2[a]	60	1.2
De-LMO-De	13.1	95	3.4
Loading	7.2[a]	60	1.2
Inbound	12	240	7.9
L2 approach	12	30	1.0
Total		771	24.3

[a]Deimos blocks half of sky, and so dose rate is halved.

Despite uncertainty over these details, it is apparent that a thick layer of regolith around a habitat can substantially reduce crew exposure to GCR during a Mars mission.

The previous discussion does not include effects of large solar flares. These would substantially increase the dose to crews who have only a "storm shelter" in an unshielded habitat. Most flares would have little effect on crews in a storm shelter inside a habitat shielded by 0.2 m of regolith.

Regolith Propellant as Debris

It is prudent to ask whether several thousand tons of regolith expelled into the inner solar system will form a debris hazard, as the ejection of debris in Earth orbit increases the impact hazard to satellites. In fact, the regolith creates essentially no hazard for the reasons discussed next.

First, a million missions would cause less than a 1% increase above the natural background. An approximate calculation based on vehicle thrust vectors, ejecta speed, and the orbital parameters of Earth and Mars indicates that ejected regolith orbits the sun in a flattened toroid about 20 million km thick, 150 million km in radial width, and nearly 700 million km in diameter. Density of natural meteoritic dust in the inner solar system is estimated[31] at 10^{-19} kg/m^3, and so the mass of natural dust in the toroid is about 7×10^{14} kg. This is over 200 million times more than the propellant from one Mars mission.

Second, dust passing near the Earth or Mars is much more likely to hit the planet's atmosphere than to hit a spacecraft. (Earth's cross section is 127 million square km; the total cross section of all spacecraft to date is well under 1 square km.) This is unlike debris in Earth orbit, which by definition does not hit the atmosphere and therefore remains in orbit until it strikes a spacecraft.

Third, only small grains of regolith are used. Half of lunar regolith by mass is in grains less than 70 μm in diameter.[32] As noted earlier, thermal data suggest that Deimos regolith is similar. Particles of 70-μm size are easily shielded against. If a sieve is used to separate the finer half of the regolith for propellant, then the propellant creates virtually no hazard to spacecraft.

An area of greater concern is the use of regolith propellant when the vehicle is in orbit about a planet. All maneuvers used in this mission design use trajectories for which regolith particles either impact on a planet or escape to solar orbit. As a result, some orbit transfers (e.g., LLO to L2) use a series of impulses at periapsis or apoapsis, rather than a continuous circular spiral.

Conclusions

The Mars mission concept presented here appears potentially competitive with existing designs under the assumed conditions. The total launch mass is substantially less than other designs when several missions are flown. The crew radiation dose from GCR is reduced by about half compared with other conjunction-class mission concepts. The crew radiation dose is similar to the dose for conventional opposition-class missions, but the regolith-propelled conjunction mission offers much greater stay time at Mars.

It is clear that a regolith-propelled Mars vehicle is competitive only when a substantial lunar base is in operation. The concept makes sense in a space program that expands its reach at a deliberate pace; it has no place in a space program that seeks to reach Mars as quickly as technology allows.

This paper presents a point design; optimization remains to be done. Several elements of the design need to be more detailed. Future cislunar transportation systems need to be selected so that the launch mass of this concept can be accurately computed and compared with other concepts. The use of nuclear power for regolith propulsion needs to be explored.

Several possible advantages relative to other mission concepts need to be evaluated. Smaller and less costly power systems are required than for ion-propelled vehicles. Crews need not crowd into a storm shelter during most solar flares. Mission infrastructure has the side effect of making supply of lunar materials to orbit more affordable for other applications.

As with all advanced propulsion schemes, some technical uncertainties need to be resolved. There are no apparent barriers to building a mass driver with long life and ejection speeds of 3000 m/s or more, but the capability should be demonstrated. Evidence suggests that the regolith of Deimos and Phobos should be similar to that of the Moon, but a precursor probe must verify this. Rotary launchers exist, but one with 1700 m/s launch speed should be demonstrated or an alternative should be selected.

In the more distant future, the regolith propellant concept has potential for missions beyond Mars. With nuclear power, the vehicle need not return to Earth after reloading at Deimos. Instead, it could go to the asteroid belt for more regolith or reload at the rings of Jupiter, Saturn, and the outer planets. With thick regolith loads, the GCR dose rate could be brought down to 1 or 2 rem/yr, and missions of many years might be medically feasible. Likewise, manned missions to the Jovian satellites may be enabled by using regolith as shielding against Jupiter's intense radiation belts.

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