Impact of Utilizing Phobos and Deimos as Waypoints for Mars Human Surface Missions

Alicia D. Cianciolo [1]
NASA Langley Research Center, Hampton, Virginia, 23681

Kendall Brown[2]
NASA Marshall Space Flight Center, Huntsville, Alabama 35812

Phobos and Deimos, the moons of Mars, are interesting exploration destinations that offer extensibility of the Asteroid Redirect Mission (ARM) technologies. Solar Electric Propulsion (SEP), asteroid rendezvous and docking, and surface operations can be used to land on and explore the moons of Mars. The close Mars vicinity of Phobos and Deimos warrant examining them as waypoints, or intermediate staging orbits, for Mars surface missions. This paper outlines the analysis performed to determine the mass impact of using the moons of Mars both as an intermediate staging point for exploration as well as for in-situ recourse utilization, namely propellant, to determine if the moons are viable options to include in the broader Mars surface exploration architecture.

Nomenclature

ΔV	=	Change in velocity (m/s)
g_o	=	Acceleration due to gravity at Earth (m/s^2)
I_{sp}	=	Specific Impulse (s)
M_f	=	Final Mass (kg)
M_i	=	Initial Mass (kg)
θ	=	Plane change angle (deg)
V_c	=	Orbital circular velocity (m/s)

I. Introduction

TRANSFER between orbits around Mars requires a maneuver to change an orbiting vehicle's velocity. A maneuver that lowers the vehicle velocity will transfer it to a smaller period orbit. Likewise, a maneuver that increases the vehicle velocity will place it in a larger period orbit. Maneuvers are most efficient (require the lowest change in velocity (ΔV) and propellant) when performed at apoapsis, the farthest point from the planet. When considering the ΔV required to deorbit and perform Entry, Descent and Landing (EDL), the spacecraft velocity must be reduced such that the orbit periapsis altitude is lowered to nearly 0 km. This ensures that the vehicle does not skip out of the atmosphere. The amount of propellant needed for such a maneuver depends on vehicle mass, engine performance (e.g. specific impulse (I_{sp})) and ΔV required to lower the periapsis altitude. Past studies of human Mars EDL missions have assumed initiation from a highly elliptic one sol orbit (33,800 km x 250 km above a mean Mars equatorial radius).[1,2] To guarantee entry from the one sol orbit, the periapsis altitude must be reduced by 250 km, which corresponds to a 15 m/s burn performed at apoapsis; a burn small enough to be performed using a Reaction Control System (RCS) and does not require a separate deorbit propulsion system. A smaller orbit has also been considered in historical studies; a 500 km circular orbit.[2] In the 500 km circular orbit case, approximately 162 m/s is required to reduce the periapsis altitude from 500 km for EDL, which could still be within the capability of a RCS.

[1] Aerospace Engineer, Atmospheric Flight and Entry Systems Branch, MS 489, AIAA Senior Member
[2] JANNAF PIB Implementation Manager, Flight Programs and Partnerships Office, FP30, AIAA Member

The scientific interest in Phobos and Deimos, and the technology capabilities afforded by the Asteroid Redirect Mission (ARM) have also made them exploration targets of interest as waypoints to Mars surface exploration. The Phobos orbit is approximately a 6000 km circular orbit (6,121 km x 5838 km above the Mars mean equatorial radius) and therefore a deorbit burn requires much more ΔV to change the periapsis altitude (~560 m/s to reduce the altitude from 6000 km). The periapsis altitude of Deimos is larger still requiring 670 m/s to deorbit. Propulsive burns of these magnitudes required to deorbit from the Mars moons would not be performed by an RCS system. A more likely option would be a dedicated deorbit propulsion module with more capable engines, namely higher thrust and higher Isp that could be jettisoned prior to entry.

Figure 1 shows the orbits of Phobos and Demos relative to the 1-Sol and 500 km circular orbits. The dimensions and orbit inclination for each are provided in Table 1, as well as the change in velocity required to transfer to the Mars surface assuming no plane change.

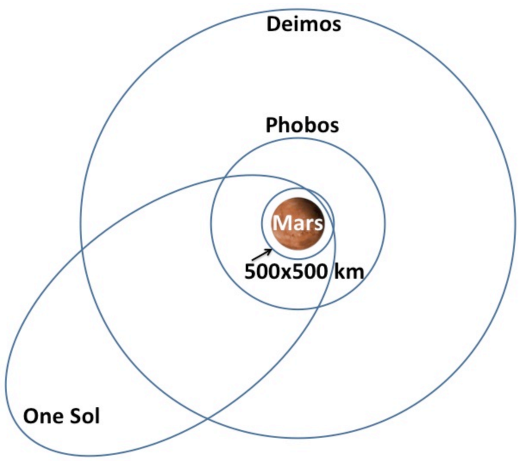

Figure 1. Relative sizes of candidate Mars exploration orbits.

Table 1. Dimensions and deorbit ΔV for different Mars reference orbits.

Orbit	*Orbit Dimensions* (km)*	*Orbit Inclination (deg)*	*Change in Velocity (m/s)*
Deimos	23,455 x 23,471	0.93	670
Phobos	5838 x 6121	1.1	560
500 km Circ.	500x 500	As needed	162
One Sol	250 x33800	As needed	15

*Above Mars reference ellipsoid.

Since mass is an indicator of over all mission cost, the objective of this study is to determine the viability of using Phobos and Deimos as waypoints to human Mars surface missions in terms of arrival mass and landed payload capability compared to historical study reference orbits. Sensitivities to landing site latitude and the use of aerocapture are also presented. The following section outlines the study background, including the assumptions, nominal EDL sequence and mass modeling approach used for this study. Section 3 describes the methodology used to perform the trades, and Section 4 provides a description of the trades and the results. Finally, implications of the results are summarized in Section 5.

II. Background

The Mars Design Reference Architecture 5 (DRA5)[1] identified a 40 t payload lander to establish a sustained human presence on the surface. DRA5 considered a 10 x 30 m mid range lift to drag ratio (L/D=0.5) rigid vehicle for hypersonic entry and supersonic retropropulsion for descent and landing as the baseline EDL system to deliver the 40 t payload to the Mars surface. A point design mass model was developed and estimated the system mass at deorbit from the 1 Sol orbit to be 110 t. A follow on study, the Entry Descent and Landing Systems Analysis EDLSA[2], considered alternative approaches to the DRA5 EDL concept of operations, while developing a higher fidelity closed-loop mass model that was integrated into performance simulations. In addition to baseline DRA5 EDL configuration, EDLSA also considered lower Technology Readiness Level (TRL) and lower L/D options including lighter mass hypersonic and supersonic inflatable and deployable aeroshells paired with super or subsonic retropropulsion to create eight EDL architectures. EDLSA determined that a lower arrival mass option to land 40 t payloads could be achieved using a 23 m diameter Hypersonic Inflatable Aerodynamic Decelerator (HIAD) with a 60 deg cone angle and a L/D of 0.3 paired with supersonic retropropulsion. The configuration was denoted as EDLSA Architecture 2. The estimated deorbit mass from the integrated mass model for this system was approximately 84 t,[2,3] a 26 t reduction over the rigid aeroshell configuration of DRA5.

All of the EDLSA architectures include a vehicle transition in either the supersonic or subsonic flight regime. During the supersonic transition (Mach~2) of Architecture 2, the vehicle changes from -22 deg angle of attack to 0 deg, jettisons the 23 m HIAD, entry RCS and rigid nose cone and initiate the engines. The level of fidelity of the vehicle design does not offer details for how the transition takes place. It is simply modeled it as a 15 s free fall. In that time, the mass of the system is reduced instantly by the mass of the HIAD, entry RCS and rigid nose cone and vehicle aerodynamics are turned off. After 15 s, the vehicle is at 0 deg angle of attack and the engines are initiated to perform powered descent. The terminal descent ΔV that impacts lander propellant masses and, therefore, affects arrival mass, is based on a ratio of thrust to weight at engine initiation equal to 2.5 Earth g's. Vehicles with lower engine initiation thrust to weight ratios will increase the arrival mass because of the increased gravity losses.

A subsequent study, the Deployable Decelerator Assessment (DDA),[4] performed more detailed analysis of an aft jettisoned HIAD using rails along the payload. DDA determined that, due to the ballistic coefficient differences between the two vehicle components, the HIAD would not take more than 3 s to separate from the descent stage. Considering other events that need to occur prior to engine initiation (change in vehicle angle of attack, engine warm up, etc.), the decision was made to reduce the 15 s free fall, which was likely too conservative (required excessive amount of propellant that was driving up the deorbit mass), to 7 s. The resulting deorbit mass of the EDL system to deliver 40 t is now approximately 80 t. It is this case that serves as the starting point for the analysis presented here. An image of the nominal entry sequence is shown in Fig. 2. Figure 3 shows the notional HIAD separation sequence.

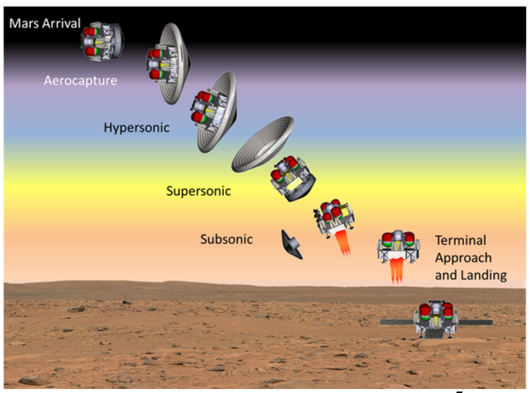

Figure 2. Mars EDL Concept of Operations.[5]

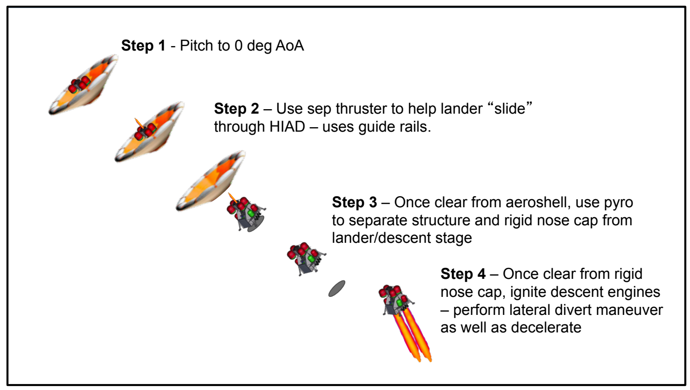

Figure 3. Details of a HIAD separation.[6]

Recently the NASA Evolvable Mars Campaign has reconsidered the DRA5 established baseline of a single 40 t payload lander. The reconsideration comes as a result of continued payload element definition, packaging and in-situ resource utilization options.[5] The trade space now consists of a combination of smaller payloads that include three to five 18 t payload landers and two to three 27 t landers. Smaller landers have the advantage of being delivered using ARM developed SEP technology. Therefore, to span the range of payloads considered in historical and recent studies, the trade space for this study includes both a fixed 80 t arrival mass and fixed payload masses of 20 t.

III. Methodology

The performance simulation used in this study has direct heritage from the trajectory and mass modeling efforts that began in DRA5 and were used in the EDLSA and DDA studies. The simulation used to run the trajectories is the Program to Optimize Simulated Trajectories (POST2).[7] The simulation is allowed to select the size of the deorbit burn such that the vehicle enters and sustains a maximum deceleration of three g's (to adhere to deconditioned crew constraints), transitions to supersonic retropropulsion at the proper velocity and altitude to land with 2.5 m/s at 0 km above the Mars Orbiter Laser Altimeter reference areoid. The simulation uses the parametric mass model developed for EDLSA[3] where the HIAD mass is dependent on the environment. For example, the HIAD thermal protection system mass is a function of entry heat load and the HIAD structure mass is a function of entry dynamic pressure. The mass of the HIAD also varies with diameter. To ensure that all the trajectories in this study fly similar profiles (e.g. heat loads, heat rates, and deceleration loads), the HIAD diameter was selected such that, as the overall system mass changes, all trajectories have the same entry ballistic number of 150 kg/m^2. The vehicle L/D is 0.3 and assumes a guided entry using bank angle reversals. The mass model takes into account the size of the deorbit burn, HIAD diameter, and terminal descent propellant required to achieve the landing conditions. The model then iterates until the allocated terminal descent propellant remaining in the tanks at touchdown is 0 kg.

However, the mass modeling relationship developed for EDLSA did not include an option to perform the deorbit burn with the larger engines required from the moons' orbits. Therefore, estimates of deorbit propellant stages for Phobos and Deimos orbits are obtained using the rocket equation (Eq. (1)) and the I_{sp} of these deorbit engines is increased to 360 s.

$$\Delta V = g_o * I_{sp} * ln\ (M_i/M_f) \tag{1}$$

ΔV is the change in velocity calculated from the acceleration due to gravity (at Earth), g_o, engine specific impulse, I_{sp}, and the natural log of the ratio of vehicle mass prior to the burn (initial mass), M_i, and the mass after the burn, M_f (final mass). Additionally, the EDLSA mass sizing relationship assumes that the one sol and 500 km orbit are delivered into the proper inclination to reach a specified landing site; no plane change is necessary. However, the orbits of Phobos and Deimos are both fixed at nearly equatorial inclinations. Since a Mars surface mission landing site has not been identified, this study considers landing latitudes up to 40 deg. Therefore, additional plane change ΔV is included for deorbit from Phobos and Deimos using the calculation for ΔV from a circular orbit (Eq. (2)).

$$\Delta V = 2 * V_c * sin(\theta/2) \tag{2}$$

In Eq. (2) the change in velocity required to change the plane of the orbit is given as a function of the velocity of the circular orbit, V_c, and the sine of half of the plane change angle, θ.

Due to the large deorbit ΔV required from Phobos and Deimos, it is assumed the moons' deorbit propulsion system would not be carried to the Mars surface and therefore would become a separate Orbit Maneuvering Stage (OMS) that is used to perform the deorbit and plane change burn and is jettisoned prior to entry. Additionally, all burns are assumed to be instantaneous, no assumptions are made for landing on or launching off of Phobos or Deimos, and no consideration are made for trades on total mission time of flight, orbit phasing or arrival declination.

Determining estimates of the propellant mass required to deorbit from the Mars moons offers valuable information for the EMC exploration architectures. First, it provides estimates for the amount of regolith in-situ resource utilization (ISRU) needed on Phobos and Deimos to generate propellants should the resources exist and extraction and storage option be available. Second, the mass estimates provide information as to the types of engines and stages required. Third, arrival mass estimates are compared to the capability of current and next generation Solar Electric Propulsion (SEP) in-space transfer systems to determine if SEP can be used to transport the Mars surface vehicles. The following section describes the trade studies performed to evaluate the impact of using Phobos and Deimos as waypoints to Mars surface exploration to determine the advantages one may have over the other and how they both compare to the 1 sol and 500 km reference orbits in terms of propellant required and landed payload capability.

IV. Trade Studies

The trades performed to make the assessment include (a) using DRA5 and subsequent studies to determine how *landed payload* mass varies from each reference orbit for a fixed arrival mass (80 t) assuming an equatorial landing site (no plane change); (b) determining the effect of non-equatorial landing sites (40 deg latitude) from Phobos and Deimos since they have orbits with fixed inclination; (c) evaluating the effect due to aerocapture; (d) determining the mass impact of performing both the aerocapture burn and the plane change to 40 deg from Phobos and Deimos; (e) and, since the EMC is considering smaller payload options, considering the same scenarios (landed latitude and aerocapture) to determine the impact on *arrival mass* for a fixed payload mass of 20 t. Table 2 summarizes the eight specific trade studies considered. Note that payload mass is not equivalent to landed mass. Payload mass

Table 2. Trade Study Cases

Trade	*Fixed Mass*	*Mars Landing Latitude (deg)*	*Aerocapture*
1	Arrival (80 t)	0	No
2	Arrival (80 t)	40	No
3	Arrival (80 t)	0	Yes
4	Arrival (80 t)	40	Yes
5	Payload (20 t)	0	No
6	Payload (20 t)	40	No
7	Payload (20 t)	0	Yes
8	Payload (20 t)	40	Yes

includes only the usable "cargo" elements of lander system. The landed mass, which is higher than payload mass, also includes the reserve propellant, the descent lander engines and the lander structure that supports the payload.

A. Trade #1: Fixed Arrival Mass: Surface Payload Delivery Capability From Reference Orbits

As described in Section 2, the arrival mass estimated to deliver the DRA5 derived 40 t payload from the 1 Sol orbit using a HIAD entry system is 80 t. This trade considers the impact on payload mass for entry vehicles of the same deorbit mass from each of the four reference orbits. The assumed engine I_{sp} used for deorbit from the one sol and 500 km circular orbit is 300 s. The assumed engine I_{sp} used to deorbit from Phobos and Deimos is 360 s and the separate deorbit propellant stage is jettisoned prior to entry. Therefore, after Eq(1) is used to calculate the deorbit propellant system mass for Phobos and Deimos, it subtracted from the on orbit mass to obtain the mass at entry which is used to size the remaining components including the payload. Since the ballistic coefficient is held constant for each case, the diameter of the HIAD is also allowed to vary. The resulting payload masses, ΔV and HIAD diameters for the vehicles that an on orbit mass of 80 t are shown in Table 3.

Table 3. Masses Landing at 0 deg Latitude; No Aerocapture

	1 Sol	*500 km*	*Phobos**	*Deimos**
On Orbit Mass (t)	80	80	80	80
Deorbit ΔV (m/s)	15	162	564	670
Prop System mass (t)	0.4	4	14**	16**
Entry Mass (t)	80	76	68	66
Landed Useable Payload (t)	**42**	**39**	**34**	**33**
HIAD Diameter (m)	23	22	21	21
Ballistic number (kg/m^2)	150	150	150	150

* Isp = 360 s; No plane change.
** Propellant stage mass jettisoned prior to entry.

Because of the change in periapsis altitude required to deorbit from Phobos (5538 km) and Deimos (23455 km), a large ΔV is required. For the fixed arrival mass, the payload delivery capability for the moons reduced by about 20% over the 1 sol delivery capability. However, since the deorbit ΔV is nearly the same for both moons, they have nearly the same landed payload capability. It is noted that the deorbit ΔV from the 500 km orbit was selected so that a vehicle with a ballistic number of 150 kg/m^2 was consistent with the other orbits in the analysis. It is likely that an entry from the 500 km orbit would desire a smaller ballistic number to lower the required ΔV by raising the required periapsis altitude.

B. Trade #2: Fixed Arrival Mass: Sensitivity to Landing Latitude

Now consider the sensitivity to landing latitude of 40 deg. Since there are no restrictions on the one sol and 500 km orbit inclination, it is assumed that the in-space transportation system will deliver vehicles to those orbits with the proper orientation to reach a desired landing site. However, since Phobos and Deimos are fixed in orbits with inclinations near 0 deg, a plane change is required. This study neglects the effect of arrival declination or accommodations for modifying it to correspond to a desired reference orbit.

The most mass efficient location in the orbit to perform the deorbit burn is at apoapsis. The most mass efficient point in the orbit to only perform the plane change is the point where the orbit crosses the equator. Since the moons' orbits are nearly circular and equatorial, the POST2 simulation calculated the total ΔV required to modify the inclination to 40 deg and lower

Table 4. Landing at 40 deg Latitude; No Aerocapture

	Phobos	*Deimos*
On Orbit Mass (t)	80	80
Total ΔV (m/s)	1326	927
Propellant mass (t)	25	18
Entry Mass (t)	55	62
Landed usable payload (t)	**26**	**30**
HIAD diameter (m)	19	20

periapsis altitude by optimizing the timing and location of the two individual maneuvers (Total ΔV row in Table 4).

It is noted that the landed payload capability to higher latitudes is larger from Deimos than Phobos despite it's having a larger periapsis altitude and, therefore, deorbit burn. The reason is due to the plane change maneuver that depends on orbital velocity. The orbital velocity (V_c) of Phobos is 55% higher than Deimos (2105 m/s for Phobos and 1351 m/s for Deimos). Therefore, variations in landing site latitude have less of an effect on the payload delivery capability from Deimos (25% less than from one sol) compared to Phobos (35% less landed payload capability than from one sol) at 40 deg latitude assuming the same deorbit mass of 80 t. 40 deg was selected as a mid-latitude comparable to some of the high landing latitude sites considered for previous robotic missions. Landing sites at lower latitudes will use less propellant and therefore enable larger payload capability. Likewise, higher latitudes will reduce the payload delivery capabilities from the Mars moons.

C. Trade #3: Fixed Arrival Mass: Sensitivity to Aerocapture

Aerocapture (AC) is still part of the EMC orbit insertion trade space. There are scenarios being considered that include the use of SEP for in-space transportation. In order to reduce the time of flight associated with the spiral into Mars orbit, the entry vehicle could be released at a specified time and location prior to arrival and perform an aerocapture maneuver to capture into the desired orbit. Aerocapture involves flying deep enough in the atmosphere to capture the vehicle into orbit in a single pass. The flight corridor is designed to keep the vehicle from entering with a flight path angle too shallow that it skips out (flies by) and from entering too steep that the vehicle reaches the surface. However, once the vehicle exits the atmosphere, it needs to perform a periapsis altitude raise maneuver at the next apoapsis to ensure that the vehicle does not inadvertently perform an EDL. The size of the periapsis altitude raise maneuver depends on the altitude change required to raise periapsis to final orbit. For the 1 sol orbit the change in altitude is about 200 km (periapsis altitude is 250 km and the aerocapture minimum altitude is about 40 km above the mean Mars surface). A periapsis altitude raise maneuver requires substantially more ΔV for Phobos and Deimos than from the 1 sol or 500 km orbits. In fact it is nearly equal to the ΔV required to deorbit. For this study the periapsis altitude raise ΔV is assumed to be equal to the deorbit ΔV and is shown in Table 5. Therefore, a fixed arrival mass requires more of the total mass for propellant, which further reduces the payload capability from the Mars moons. Table 5 shows the payload capability for vehicles using aerocapture to achieve the reference orbit and land at an equatorial site (no plane change).

Table 5. Landing at 0 deg Latitude; With Aerocapture

	1 Sol	*500 km*	*Phobos**	*Deimos**
Arrival Mass (t)	80	80	80	80
AC Periapsis Raise ΔV (m/s)	15	162	564	670
Prop System mass (t)	0.4	4	12	14
Deorbit Mass (t)	80	76	68	66
Deorbit ΔV (m/s)	14	162	564	670
Prop System mass (t)	0.4	3	10	11
Entry Mass (t)	79	73	58	55
Landed Useable Payload (t)	**41**	**38**	**28**	**27**
HIAD Diameter (m)	23	22	20	19
Ballistic Number (kg/m^2)	150	150	150	150

Because of the large periapsis altitude changes required to reach the Mars moons, the payload capability, when using aerocapture has reduced by 30% over the same capture scenario into a 1 sol orbit. It should be noted again, that since the deorbit ΔV for Phobos and Deimos is comparable, the payload capability is nearly the same for both.

D. Trade #4: Fixed Arrival Mass: Sensitivity to Aerocapture and Latitude

The payload capability for a fixed arrival mass in a Mars moon orbit continues to decrease compared to the 1 sol orbit as landing latitude increases and aerocapture is used to capture into the reference orbit. A final study for the constant 80 t arrival mass study considers how the payload capability is further reduced by doing both aerocapture and landing at high latitudes. The results are captured in Table 6. The values for the 1 Sol and 500 km orbit are identical to those shown in Table 5 and so are not repeated here.

For this scenario, the payload capability from Phobos is half of that from the 1 Sol orbit. While the plane change does not affect payload capability to Deimos as much, the delivery capability is about 37% less than 1 Sol. A plot summarizing the payload capability for each of the four trades using constant arrival mass of 80 t is shown in Fig. 4.

Table 6. Landing at 40 deg Latitude; With Aerocapture

	*Phobos**	*Deimos**
Arrival Mass (t)	80	80
Periapsis Raise ΔV (m/s)	564	670
Prop System mass (t)	12	14
Deorbit Mass (t)	68	66
Total ΔV (m/s)	1326	927
Prop System mass (t)	21	15
Entry Mass (t)	47	51
Landed Useable Payload (t)	**21**	**25**
HIAD Diameter (m)	18	18
Ballistic Number (kg/m^2)	150	150

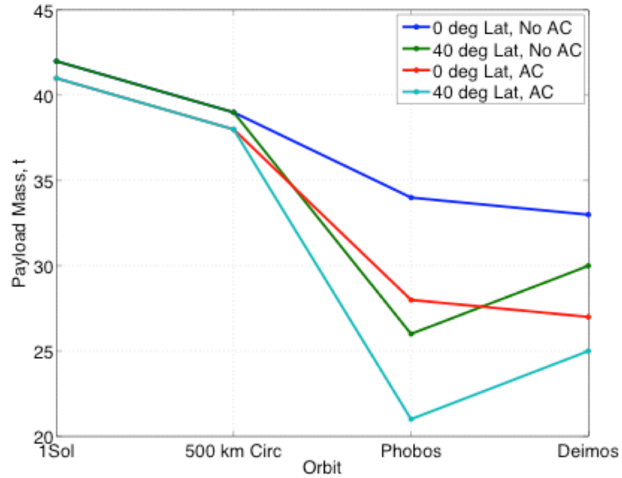

Figure 4. Payload capability from reference orbits for fixed 80 t arrival mass.

There are several points to note based on the results of the fixed on orbit mass study. The first is that, when considering mid latitude landing sites, Deimos has a larger payload capability than Phobos. The second is that aerocapture does not offer advantages to capture into Phobos or Deimos orbits unless ISRU refueling options are available on the moons prior to deorbit to the surface. Finally, the results plotted in Fig. 4 show that, for a constant on orbit mass, the largest payload delivery capability is from the highly elliptical 1 Sol orbit.

E. Trade #5: Fixed Payload Mass: Arrival Mass in Reference Orbits That Deliver 20 t Payload

The same mass model was used to derive arrival masses for a fixed payload mass of 20 t. Again, the 1 sol and 500 km circular trajectories assumed the deorbit burn was performed using a RCS system with an I_{sp} of 300 s. Also, the ballistic coefficient, held constant at 150 kg/m^2, was achieved by varying the HIAD diameter. The arrival masses for Phobos and Deimos were determined using the rocket equation (Eq. (1)) assuming an I_{sp} of 360 s. The corresponding arrival masses for vehicles landing at 0 deg latitude (no plane change) are shown in Table 7. Similar to Trade #1 where the landed payload was 20% lower from Phobos and Deimos orbits compared to the 1 Sol orbit, in this case the *arrival mass* at the moons is nearly 20% higher than for the 1 Sol orbit. The differences in entry mass for the same payload result because of the different entry speeds from each orbit and the amounts of decent propellant and HIAD diameter required to meet the landing constraints that do not appear in the tables due to rounding.

Table 7. Fixed Payload: 0 deg Latitude; No Aerocapture

	1 Sol	*500 km*	*Phobos**	*Deimos**
Arrival Mass (t)	**44**	**48**	**53**	**53**
Deorbit ΔV (m/s)	15	162	564	670
Prop System mass (t)	0.2	2	8	9
Entry Mass (t)	44	46	45	44
Landed Useable Payload (t)	20	20	20	20
HIAD Diameter (m)	17	17	17	17
Ballistic Number (kg/m^2)	150	150	150	150

F. Trade #6: Fixed Payload Mass: 40 deg Latitude; No Aerocapture

Now consider the on orbit mass at Phobos and Deimos when a plane change is required to reach a landing site at 40 deg latitude. Table 8 includes the same total ΔV's used in Trade #2 and Eq. (1) to calculate the arrival masses. Due to the higher orbital velocity, the arrival mass to Phobos increases by 50% compared to the 1 sol case in Table 7. It is also interesting to note that only 4 t more of propellant system mass is needed in the Deimos orbit to land at 40 deg than to land at 0 deg latitude.

Table 8. Fixed 20 t Payload: 40 deg Latitude; No Aerocapture

	Phobos	*Deimos*
On Orbit Mass (t)	**66**	**57**
Total ΔV (m/s)	1326	927
Propellant mass (t)	21	13
Entry Mass (t)	45	44
Landed usable payload (t)	20	20
HIAD diameter (m)	17	17

G. Trade #7: Fixed Payload Mass: 0 deg Latitude With Aerocapture

When the aerocapture periapsis raise maneuver is included in the calculation for arrival mass, all except the 1 sol orbit, increase. The results are shown in Table 9.

Table 9. Fixed 20 t Payload: 0 deg Latitude; With Aerocapture

	1 Sol	*500 km*	*Phobos**	*Deimos**
Arrival Mass (t)	**44**	**50**	**62**	**64**
Periapsis Raise ΔV (m/s)	15	162	564	670
Prop System mass (t)	0.2	2	9	11
Deorbit Mass (t)	44	48	53	53
Deorbit ΔV (m/s)	14	162	564	670
Prop System mass (t)	0.4	2	8	9
Entry Mass (t)	44	46	45	44
Landed Useable Payload (t)	20	20	20	20
HIAD Diameter (m)	17	17	17	17
Ballistic Number (kg/m^2)	150	150	150	150

H. Trade #8: Fixed Payload Mass: 40 deg Latitude, With Aerocapture

Table 10 shows the arrival masses for Phobos and Deimos when aerocapture is used and the vehicle must land at a 40 deg latitude-landing site. In a study that looks very similar to Trade #4, the arrival masses approach 80 t.

Figure 5 shows the arrival masses for all of the fixed payload mass trades. The arrival mass for Phobos is nearly double the arrival mass to the 1 sol orbit if the landing site is near 40 deg latitude. The EMC is considering ARM extensible technologies like Solar Electric Propulsion. Advanced systems[8] being considered have Mars arrival mass delivery capability of approximately 45 t, which is similar to the entry mass required to land 20 t payloads. If reliable means of emplacing hardware and generating in situ resources for propellant can be developed on either Phobos or Deimos, then the viability of using SEP to deliver the EDL system to those orbits to dock with a propellant stage becomes an attractive trade.

Table 10. 40 deg Latitude; With Aerocapture

	*Phobos**	*Deimos**
Arrival Mass (t)	**77**	**69**
Periapsis Raise ΔV (m/s)	564	670
Prop System mass (t)	11	12
Deorbit Mass (t)	66	57
Total ΔV (m/s)	1326	927
Prop System mass (t)	21	13
Entry Mass (t)	45	44
Landed Useable Payload (t)	20	20
HIAD Diameter (m)	17	17
Ballistic Number (kg/m^2)	150	150

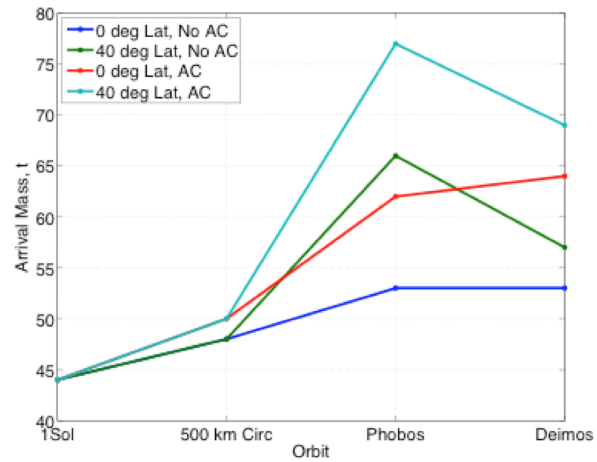

Figure 5. Arrival masses for constant 20 t payload.

V. Conclusion

The purpose of this study was to determine if Phobos and Deimos are viable waypoints for Mars surface missions. The trade space considered fixed arrival masses as well as fixed payload masses and considered sensitivity to landing latitude and aerocapture compared to the historical 1 Sol and 500 km circular reference orbits. Those historic orbits have many advantages, namely the low periapsis altitude compared to the moons' orbits and the assumption that they can be established in any inclination required to reach a designated landing site. The results indicate that, if all the propellant required to perform post aerocapture maneuvers must be carried to Mars, then Deimos looks like a more attractive waypoint primarily due to its lower orbital velocity. For equatorial landing sites, both Phobos and Deimos offer about the same performance in terms of arrival mass or payload capability.

There are options for a SEP in-space transportation stage to deliver payloads (~20 t) to the moons orbits. However, ISRU propellant generation or preplaced propulsion stages may be needed to enable a Mars surface mission. If ISRU can be performed on the moons to make propellant, then the moons become a more attractive option in terms of payload delivery mass. The results indicate that approximately 25 t and 18 t of ISRU propellant production would be needed at Phobos and Deimos, respectively, for a single Mars surface lander to 40 deg latitude. However, if only equatorial sites are considered, each moon would need to produce approximately 9 t of ISRU propellant. ISRU production rates, emplacement and operations technologies are not considered here.

This study illustrates a few of the mass advantages of the 1 sol and 500 km circular orbit and reinforces the reasons they remain in the EMC trade space. While the results do not exclude the possibility of utilizing the Mars moons as staging points, it is noted that doing so comes at a cost that can be mitigated with advancements in ISRU, engines and other technologies.

References

[1]Drake, B. G., editor, "Human Exploration of Mars Design Reference Architecture 5.0," s.l. : NASA SP-2009-566, 2009.

[2]Dwyer Cianciolo, A. M., et al., "Entry, Descent and Landing Systems Analysis Study: Phase 1 Report," NASA–TM–2010–216720, 2010.

[3]Samareh, J. A., and Komar, D. R., "Parametric Mass Modeling for Mars Entry, Descent, and Landing System Analysis Study," AIAA-2011-1038, 2011.

[4]Corliss, J., "Deployable Decelerator Assessment: Final Report," Unpublished NASA Internal Document, 2013.

[5] Toups, L., Brown K., and Hoffman, S. J., "Transportation-Driven Mars Surface Operations Supporting an Evolvable Mars Campaign," *IEEE Aerospace Conference* 2015.

[6] Drake, B. G., editor, "Human Exploration of Mars Design Reference Architecture 5.0: Addendum #2," s.l. : NASA SP-2009-566-ADD2, 2014.

[7] Striepe, S. A., et al., "Program to Optimize Simulated Trajectories (POST II) – Utilization Manual, Volume II, Version 1.16.G," NASA Langley Research Center, Jan. 2004.

[8] Percy, T., McGuire, M., Polsgrove, T., "In-space Transportation for NASA's Evolvable Mars Campaign," *AIAA SPACE 2015*, Pasadena, CA, August 31-September 2, 2015.