

Chapter 2

Planning Space Campaigns and Missions

Abstract In the early stages of designing a mission to Mars, an important measure of the mission cost is the initial mass in LEO (IMLEO). A significant portion of this mass consists of propellants. Space missions can be described in terms of a series of *states* connected by *steps*. A *state* is a condition of relative stability and constancy. A *step* is an action of change (e.g. fire a rocket). Using *state-step* data, one can estimate the initial mass in LEO for delivery of payloads to Mars orbit and the Mars surface. In any mission design, the first and foremost thing that is needed is the set of Δv for all the mission *steps*. Estimates of Δv for various *steps* can be made by standard trajectory analysis. The propellant requirements for each step can be estimated from Δv . It requires a great deal of IMLEO to send a spacecraft to Mars orbit and back. It takes even more IMLEO to send mass to the Mars surface and back. Unfortunately, in typical NASA reports describing future space missions, ferreting out *state-step* information is time-consuming and frustrating, and is usually impeded by missing or ill-defined data. Therefore, it is difficult to trace through the *steps* of NASA concepts for human missions to Mars.

2.1 Campaigns

A *campaign* is a series of closely related space missions that sequentially contribute to fulfillment of overall campaign goals. In some cases, each mission in the campaign is distinct, and the main value to each subsequent mission provided by previous missions, is the knowledge gained from previous missions that might influence sites for subsequent missions, and validate instruments, flight technologies, or other mission design elements. This typically prevails in the case of robotic missions to Mars.

For human missions to Mars, previous robotic missions will be necessary to validate new technologies on Mars prior to use by humans. But the campaign composed on a sequence of human missions will build infrastructure and advance capabilities with each subsequent mission.

For example, the MEP envisages a campaign of exploratory robotic missions to Mars in which each mission provides important insights as to where to go and what to look for in the next mission(s). The NASA lunar exploration initiative of about 7–9 years ago was an outline of a campaign, but unfortunately, the campaign was not defined very well, except that it would begin with short-duration “sortie” missions that lead up to establishment of a lunar “outpost” with uncertain location and functions. In fact, initial planning did not even resolve many important aspects of the sortie missions or refine the *Lunar Surface Access Module* (LSAM), while almost all the focus seems to have been on the so-called *Crew Exploration Vehicle* (CEV). In this process, NASA seems to have lost sight of the overall campaign and how the pieces fit together. For example, although in situ resource utilization (ISRU) for producing oxygen for ascent propulsion was a central theme for outposts, the elimination of oxygen as an ascent propellant suggests that different groups working on the lunar exploration initiative were not only not communicating, but were working at cross-purposes.

At the highest level, one should start with a set of goals to be achieved by a campaign. One would define a set of hypothetical missions that are potential building blocks of a campaign. Campaigns are assemblages of missions but the sequence of missions in a campaign might be probabilistic. For example:

- Each Mission involves at least two possible outcomes with probabilities assigned.
- After Mission 1, do Mission 2A if Event A happens, or Mission 2B if Event B occurs.
- There are a number of possible *outcomes* for each campaign (each with a different series of missions, and differing cost, risk, and performance)

Alternative pathways for carrying out a campaign can be represented by a “tree-diagram” showing alternate models for the campaign as paths through a space consisting of arrangements of sequentially arranged missions. A number of investigators have been studying approaches for seeking the best campaign (i.e. best sequence of missions) according to some figure of merit for a campaign. However, this is a complex subject and is beyond the scope of this discussion.

In order to make a wise choice for a campaign, the properties, characteristics, and requirements of the individual missions that make up a campaign need to be understood (Baker et al. 2006).

2.2 Planning Space Missions

In planning a space mission, the first thing to consider is why do we want to do it, and what do we hope to derive from the results? The next questions deal with the feasibility of the enterprise as expressed by the following questions:

- How much does it cost? Is it affordable?
- Is it technically (and politically) feasible?

- How safe is it, and what is the probability of failure?
- Can we launch (and possibly assemble in space) the required vehicles?

It is generally very difficult to arrive at even approximate answers to these questions without investing a great deal of effort in preliminary analysis and modeling. Furthermore, there are typically a number of architectural variations in space vehicles, their sequencing, their phasing, and their destinations that can be utilized to carry out such a space mission. These alternate variations are referred to as “mission architectures” or simply “architectures.” It would require significant financial resources and considerable time and effort to carry out detailed analyses of each potential architectural option. Furthermore, in planning a human mission to Mars, the actual mission will most likely not take place for several decades into the future, and it is nearly impossible to project what the state of marginal technologies such as nuclear propulsion and large-scale aero entry will be several decades hence. Therefore, the typical approach widely utilized involves a rather crude initial analysis to compare architectural options, and from this, a short list of favored architectures can be identified that should be examined more thoroughly. For rough initial analysis, there is widespread use of the *initial mass in low Earth orbit* (IMLEO) as a rough measure of mission cost, and since IMLEO can usually be estimated to some degree, it is typically used as a surrogate for mission cost in early planning. This is based on the notion that in comparing a set of alternative potential missions to carry out a desired goal, the amount of “stuff” that you need to transport to LEO is a major determinant of the cost. IMLEO is the total mass initially in LEO, but it does not specify how this total mass is partitioned into individual vehicles. The mass of the largest vehicle in LEO dictates the requirements for launch vehicle capability (how much mass a launch vehicle must lift in “one fell swoop”) - unless on-orbit assembly is employed. Thus, the initial planning of space missions, and preliminary selection of mission architectures depends on two main interconnected parameters: (1) IMLEO, and (2) the required launch vehicle and number of launches.

It is important to understand that the requirements for space missions are dominated by the requirements for accelerating vehicles to high speeds. Unlike an automobile, which has a large crew compartment and a small gas tank, a spacecraft typically has large propellant tanks and a relatively small crew compartment. A space mission is composed of a series of propulsion steps, each one of which typically has more propellants than payload. Each propulsion step requires accelerating not only the payload, but also the propellants reserved for later acceleration steps. As a result, most of IMLEO is typically propellants, not payload. The mass of propellants delivered to LEO for getting from here to there (and back) is then (at least to some extent) the determining factor in deciding whether a space mission is feasible and affordable. And as we pointed out, this is embodied in the value of IMLEO, which is mainly made up of propellants, not payload. Some day in the future, if we can efficiently supply propellants to LEO, this picture might change.

2.3 Architectures

In a simple space mission, a single spacecraft may be placed atop a launch vehicle and sent on its way to its destination in space. In such a case, there is no need to discuss “architectures.” However, in more complex space missions, particularly in human exploration missions, a number of alternative approaches can be conceptualized for scheduling and phasing the launches, rendezvous, assemblies and disassemblies, descents and ascents, and other operations involved in the overall mission. Characterizing, evaluating and comparing alternative architectures, constitutes a major portion of early planning of complex space missions.

As an example, the mission architecture adopted in 2005–6 by the NASA *Exploration Systems Architecture Study* (ESAS) for lunar sortie missions is shown in Fig. 2.1. A “sortie” mission is a short-term mission with a minimal payload that is designed to prove out the functionality of the various space systems and operations, and obtain a limited amount of useful data. It is typically a precursor to longer term missions that would establish an “outpost.” In this architecture, the following vehicles were defined:

- **EDS** = *Earth Departure System*. (This was a propulsion system consisting of propellant tanks, plumbing, rockets and propellants to send the assembled system from LEO on a path toward the Moon).

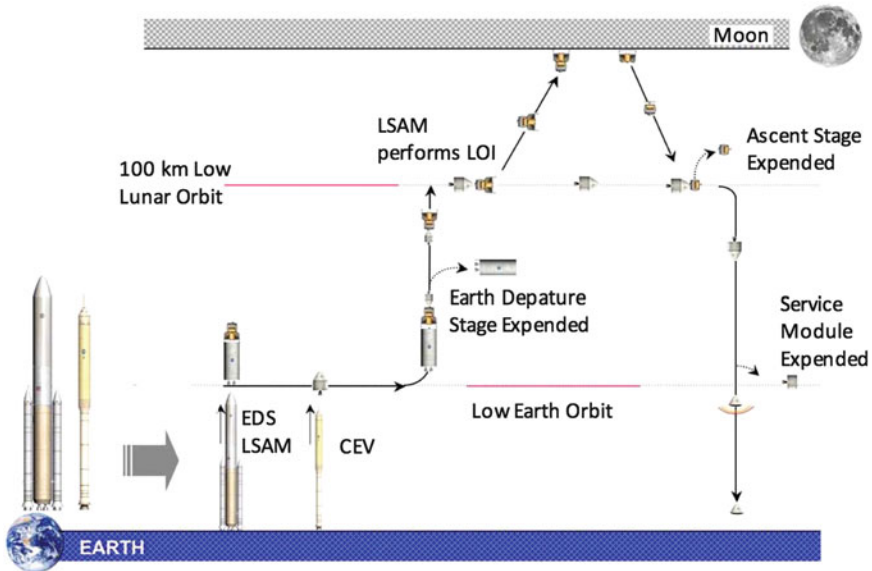


Fig. 2.1 The NASA ESAS architecture for lunar sortie missions. Vehicle symbols are defined in the text. *LOI* stands for lunar orbit insertion [Exploration Systems Architecture Study (ESAS) Final Report 2005, http://www.nasa.gov/pdf/140649main_ESAS_full.pdf]

- **LSAM** = *Lunar Surface Access Module* [sum of *Descent Stage* (DS), *Ascent Stage* (AS) and *Habitat* (H)]. The ascent and descent stages were propulsion systems consisting of propellant tanks, plumbing, thrusters and propellants. The *Habitat* was the capsule in which the crew resides during transit between lunar orbit and the lunar surface as well as for several days on the surface. The DS transports (H + AS) to the lunar surface. The AS transports H back to lunar orbit to transfer the crew to the waiting CEV in lunar orbit.
- **CEV** = *Crew Exploration Vehicle* [sum of *Service Module* (SM) and *Crew Module* (CM)]. The CM houses the crew in transit from Earth to Moon and back, and the SM provides the Crew Module with a propulsion system to return to Earth, as well as other support systems.

There were two launch vehicles: the *Cargo Launch Vehicle* (CaLV) to launch cargo (EDS + LSAM) to low Earth orbit (LEO), and the *Crew Launch Vehicle* (CLV) to launch the crew (in the CEV) to LEO. The architecture required a rendezvous and assembly of the EDS + LSAM and the CEV into a single unit in LEO, and a disassembly in lunar orbit in which the CEV remains in lunar orbit while the LSAM descends to the surface (and the EDS is discarded). The ascent stage + habitat ascends to lunar orbit from the lunar surface, performs a rendezvous with the CEV, and transfers the crew to the CEV. The CEV then returns to Earth with the crew.

The ESAS architecture is more fully described in Fig. 2.2. It can be seen that there are a variety of states and steps involved in the mission.

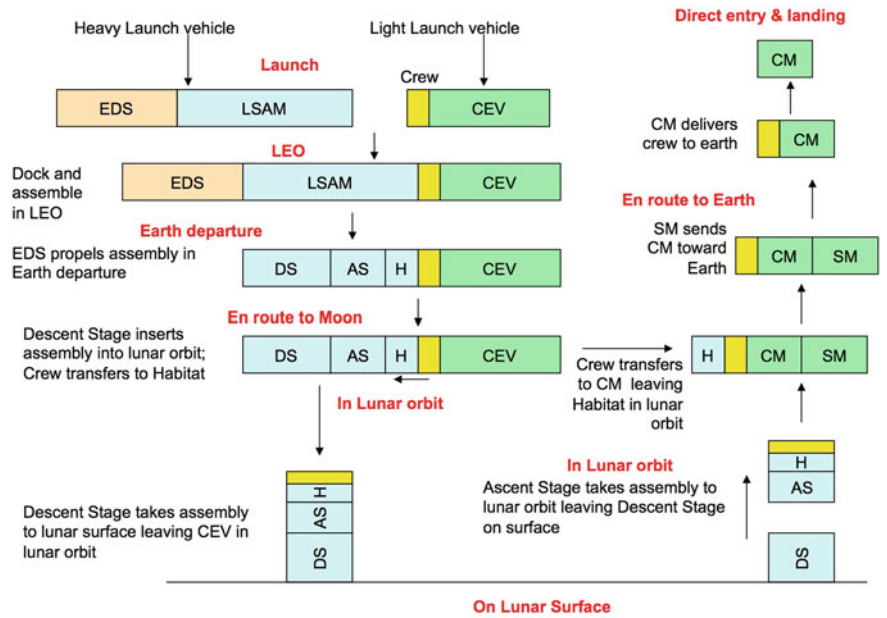


Fig. 2.2 State-step representation of ESAS lunar sortie architecture. The entire mission consists of a set of alternating states and steps

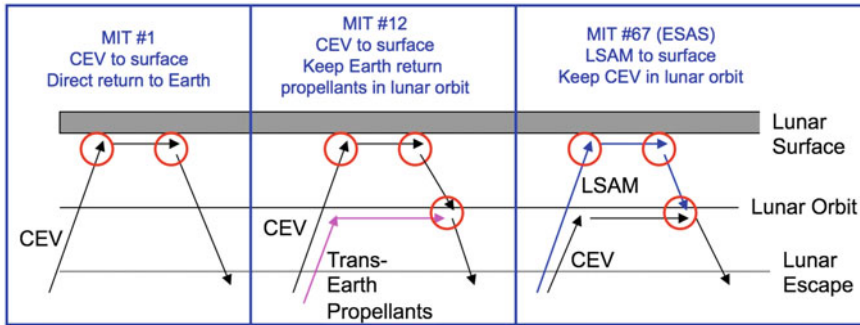


Fig. 2.3 Three of the architectures considered by MIT for landing on the Moon (Based on presentation at JPL by P. Wooster, March 2, 2005)

This is only one of several architectures that are conceivable. Some of these conceptual architectures would eliminate the rendezvous in Earth orbit, or the disassembly and rendezvous in lunar orbit. An MIT Study examined large numbers of alternative lunar architectures (Wooster et al. 2005). As the MIT study pointed out, “*Initial Mass in Low Earth Orbit (IMLEO)* is typically used as a top-level screening criterion in architecture selection; however,... additional factors must be taken into account in determining the preferred architecture.” These include cost and safety.

The MIT Study concluded that the ESAS architecture shown in Figs. 2.1 and 2.2 is not optimum. Figure 2.3 illustrates three simplified architectures for lunar landing as defined by the MIT Study. In the architecture on the left, the CEV goes directly to the lunar surface and returns to Earth without rendezvous and crew transfer in lunar orbit. The architecture in the center of this figure is a variant in which propellant tanks for Earth return are kept in lunar orbit so they don’t have to be carried down and up from the lunar surface. The architecture on the right is the ESAS architecture involving rendezvous and crew transfer on the way in and the way out from the Moon. The key issue is whether the CEV should remain in lunar orbit or whether it should land on the lunar surface. If the CEV is landed on the Moon, it will be heavier because it will require extra capabilities. However, this will eliminate the need to develop a second habitat and it will eliminate several significant operations in lunar orbit. MIT identified other benefits as well. The MIT Study concluded that architecture #1 in Fig. 2.3 was superior to the ESAS architecture, which maximized the number of vehicles needed and the number of in-space operations. The point being made here is that human exploration missions can typically be implemented with a number of conceptual architectures, and choosing an optimum approach requires a great deal of analysis and judgment.

For human missions to Mars, many of the same options occur as in human missions to the Moon. There are alternative options in Earth launch, Earth orbit assembly, Earth departure, Mars orbit insertion, Mars orbit operations, Mars descent and ascent, and Earth return steps. Each architecture involves a serial set of steps leading from departure to return.

2.4 A Mission as a Sequence of Steps

Space missions can be described in terms of a series of *states* connected by *steps*

- A *state* is a condition of relative stability and constancy (coasting, remaining in orbit, operating on the surface, etc.)
 - Each *state* is characterized by a set of vehicle masses.
- A *step* is an action of change (fire a rocket, jettison excess cargo, transfer crew between vehicles, etc.)
 - Each *step* is characterized by propulsion parameters or other relevant data that characterize the dynamic operations that take place during the step.

The matrix of *states* and *steps* provides a simple means of summarizing the major elements of a space mission. Unless all the *states* and *steps* of a mission are defined, it is not possible for an independent party to verify the characteristics of the mission. In typical NASA reports describing analysis of future space missions, ferreting out *state-step* information is time-consuming and frustrating, and is usually impeded by missing or ill-defined data. It is typically difficult to summarize, understand and recreate the *states* and *steps* in complex multi-step missions from recent NASA and ESAS study reports.

Table 2.1 shows a *state-step* table for a simple hypothetical mission involving transfer of a payload that weighs 10 mass units to Mars orbit and return from Mars orbit to Earth. Actually, the process begins on the launch pad with launching to LEO. However, the launch process is typically considered separately, and the mission design presumes that launch has taken place and initial assets are located in LEO. Therefore, the first *state* is defined to be in LEO.

The *steps* are given in the first row. The *states* at the start and end of each step are given in rows 8 and 9. In order to characterize a *step*, a considerable amount of data must be specified in rows 2–6 (but it is not necessary for the reader to assimilate all of this). The most important of these is Δv , the change in velocity imparted to a vehicle by firing a rocket.

$$\Delta v = v(\text{after step}) - v(\text{before step}).$$

The greater Δv is, the more propellant needs to be used in the rocket burn to achieve the required Δv . Some typical values of Δv are provided in the first row of Table 2.1. One meter per second is equivalent to 2.24 miles per hour. A Δv of 4000 m/s is equivalent to 8960 mph.)

The columns in Table 2.1 represent the *steps* in the mission to transfer a 10-unit mass to Mars orbit, and return it to Earth. Each *step* is a transition from an initial *state* to a final *state* (rows 8 and 9 in table). The masses prior to, and after each *step* are shown in rows 15 and 16. At first, it may seem strange, but the pathway to estimate IMLEO for a space mission is not to start at the launch pad or LEO, but rather to begin at the destination, and then work backwards to arrive at an estimate

Table 2.1 State-step table for a hypothetical mission to Mars orbit and return

Row	Step⇒	LEO to TMI	TMI to circular Mars orbit	Mars orbit to TEI	TEI to earth
1	Δv (km/s)	3.9	2.5	2.4	*
2	Propulsion system	LOX-LH2	LOX-CH4	LOX-CH4	*
3	Propulsion specific impulse (s)	450	360	360	*
4	Rocket equation exponential	2.421	2.031	1.974	*
5	Propulsion stage % of propellant mass	10	12	12	*
6	Entry system % of delivered mass	0.0	0.0	0.0	70
7					
8	State at start	LEO	TMI	Mars orbit	TEI
9	State at end	TMI	Mars orbit	TEI	Earth
10					
11	Entry system mass	0.0	0.0	0.0	7.0
12	Payload mass	10.0	10.0	10.0	10.0
13	Propellant mass	146.0	44.7	18.8	
14	Stage mass	14.6	5.4	2.3	
15	Total mass at start	248.7	88.1	38.0	17.0
16	Total mass at end	88.1	38.0	17.0	10.0
17					
18	LEO mass/Mars orbit payload mass	6.5			
19	LEO mass/round trip mass	24.9			

*Note that the Earth entry step typically uses an aeroshell, not propulsion, so no Δv is specified in the Earth entry column. It has been assumed here arbitrarily that the entry system mass is 70 % of the mass delivered to Earth

of the initial mass needed in LEO to send the various vehicles to their destinations. Thus, Table 2.1 is generated by starting in the far right column, and working backwards toward the left. The final mass in any column is the initial mass in the column to its right.

The first **step** is Earth departure (or as it is referred to by professionals, trans-Mars injection (TMI)), in which a rocket is fired to send the vehicle out of LEO on its way toward Mars. This vehicle would cruise toward Mars for a period of typically 6 to 9 months, carrying propellants and propulsion stages for subsequent firings. The second **state** is a quiescent cruise toward Mars. A few minor mid-course trajectory corrections may be needed along the way, introducing additional **steps** but these are minor and are not included in the table. On arriving in the vicinity of Mars, a significant retro-rocket firing **step** is needed to slow the vehicle down by

2500 m/s in order to insert it into Mars orbit. To return toward Earth, yet another rocket firing is needed to provide the 2400 m/s needed to escape Mars orbit and head toward Earth. At Earth, an aeroshell entry system is used instead of a retro-propulsion system. Finally, the “gear ratios” are given in rows 18 and 19. These gear ratios give the ratio of initial mass in LEO (IMLEO) to the mass delivered to Mars orbit, or to the mass that undergoes a round trip to Mars. It will be noted that it requires about 25 mass units initially in LEO to send one mass unit on a round trip to Mars orbit and return to Earth LEO. Similar (though more complex) tables can be used to describe missions to the surface of Mars with multiple vehicles.

It is not expected that the reader can fully digest Table 2.1 easily. A greater understanding of such tables will be provided later in this book. However, several aspects of this table are worth discussing even at this early point. In this table, the ultimate payload is an undefined mass of 10 units (it doesn’t matter whether it is 10 kg, 10 tons, or whatever, because everything else scales to this mass.)

The IMLEO is 248.7 mass units. IMLEO is constituted from a number of components, as shown in Fig. 2.4. These include propulsion for trans-Mars injection, propulsion for Mars orbit insertion, propulsion for trans-Earth injection (TEI), and finally an aeroshell for Earth entry. The mass on the launch pad is roughly 20 times IMLEO, or about 5000 mass units. Thus it takes roughly 5000 mass units on the launch pad to transfer 10 mass units on a round trip to Mars orbit. These

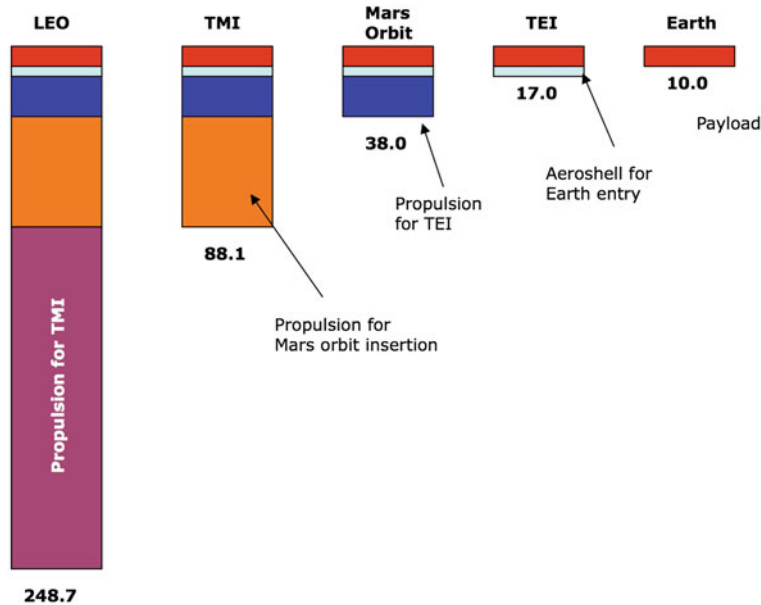


Fig. 2.4 Sequence of decreasing masses for each *state* in the journey to Mars orbit and back. Each column represents a *state*, and the transition from one column to the next column is a *step*

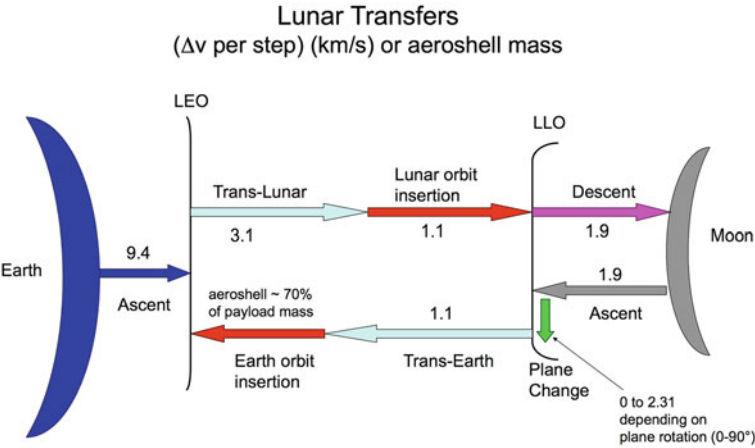


Fig. 2.5 Typical set of Δv for each step in lunar sortie missions (km/s)

diagrams show that most of what we launch into space is propulsion systems, and the payload is typically a very small fraction of the total mass.

The key conclusion to be drawn is that it takes a great deal of IMLEO to send a spacecraft to Mars orbit and back. It takes even more IMLEO to send mass to the Mars surface and back. The “gear ratios” define the ratio of IMLEO to the payload mass that is transported through space to a specific destination.

In any mission design, the first and foremost thing that is needed is the set of Δv for all the mission *steps*. Estimates of Δv for various *steps* in Mars or lunar missions

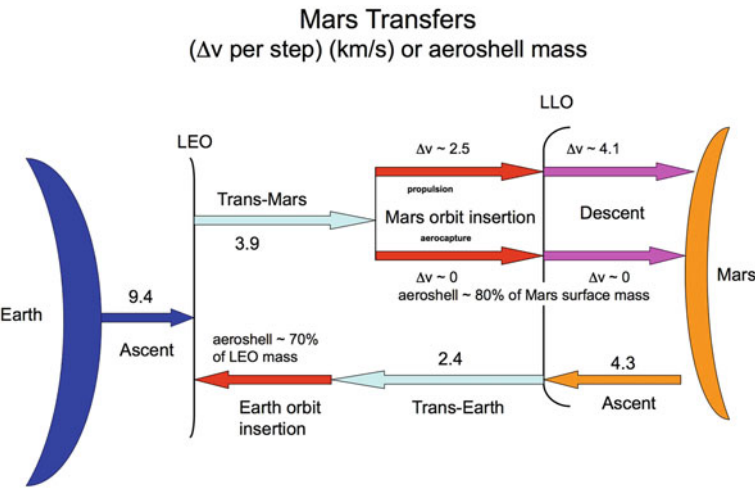


Fig. 2.6 Typical set of Δv (km/s) for each step in Mars missions (km/s). Two options are shown for orbit insertion and descent; one using propulsion, and the other using aero-assist

can be made by standard trajectory analysis. These values of Δv will depend to some extent on other factors. For example, in lunar missions, the values of Δv vary depending on whether global access, anytime return, and use of “loitering” in the lunar vicinity is employed. For Mars missions, the Δv varies with launch opportunity and the desired trip time to Mars (shorter trip times require higher Δv and therefore more propellants). Some typical sets of Δv for lunar and Mars missions are shown in Figs. 2.5 and 2.6.

2.5 What’s Delivered to the Destination?

Figure 2.7 shows one particular model for a human mission to Mars. There are three major deliveries: (1) the crew to Mars surface, (2) the cargo to Mars surface, and (3) the *Earth Return Vehicle* (ERV) to Mars orbit. In this scenario, at the conclusion of the Mars surface phase, the crew ascends to rendezvous with the ERV in Mars orbit, and transfers to the ERV for the return trip to Earth. Based on the values of Δv for each step, one can work backwards to estimate the propellant mass required for

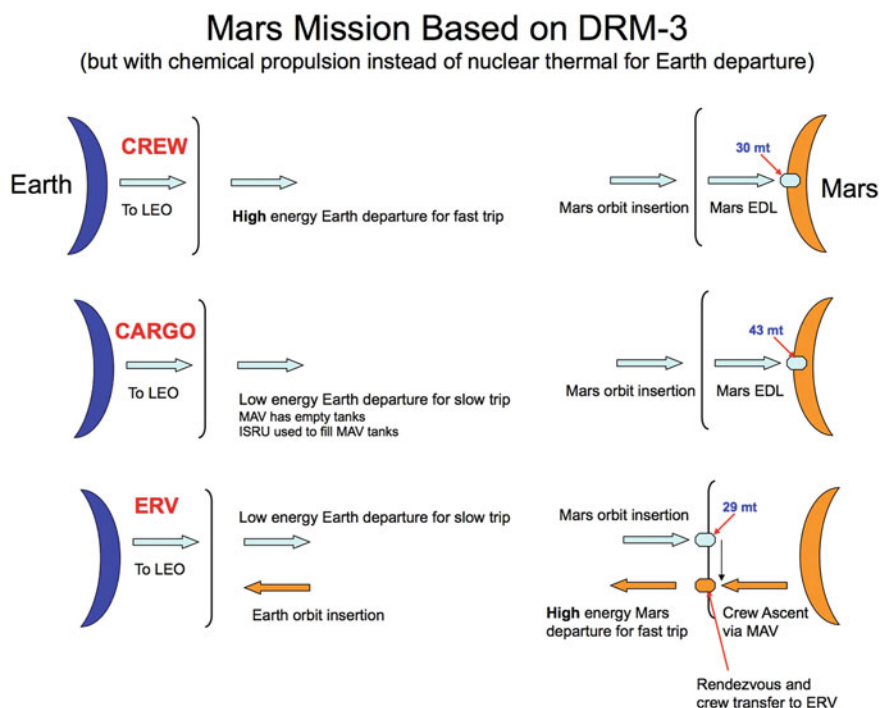


Fig. 2.7 Vehicles delivered to destinations according to NASA’s Mars Design Reference Mission 3.0. The ERV is the *Earth Return Vehicle* and the MAV is the Mars Ascent Vehicle

each step along the way, and thereby eventually arrive at the initial mass in LEO (IMLEO). The reason to work backwards is that each previous step must provide the propellants needed to accelerate propellants needed for future steps, and this cannot be known if one starts at the beginning. In addition, the required lifting capability of the launch vehicle on the Earth launch pad can be derived from the mass in LEO of the heaviest vehicle involved.

Estimation of the masses of vehicles at their destinations requires extensive, detailed analysis. Previous studies provide some insights into what these are likely to be. They include habitats, life support systems, radiation shielding, mobility systems on the surface, power systems, communication systems, propulsion systems, and various other engineering systems to support the mission. In particular, Sects. 3.8.1, 3.8.2 and 3.8.3 provide data from so-called DRMs: DRM-1 and DRM-3.

2.6 What's in Low Earth Orbit

As we have discussed, the initial mass that must be lifted from the Earth's surface to LEO (IMLEO) is a rough measure of the mission cost (in the absence of more detailed data), and is widely used to compare mission architectures at early stages of planning to help identify the optimum architecture at an early stage in planning. As we have indicated in the previous section, one starts with the vehicles required at the ultimate destinations (in the case of Fig. 2.7 this would be three vehicles: the *Crew Lander*, the *Cargo Lander* and the *Earth Return Vehicle*) and works backwards to estimate for each step how much propellant mass is needed to accelerate these vehicles through the various *steps* with their corresponding values of Δv . After adding up the masses of all the propulsion stages and propellants needed to take the various spacecraft through their various operations and steps in the missions, one ends up with a total mass that must be sent on its way toward Mars from LEO. The IMLEO is the sum of these masses that must be sent on its way to Mars plus the mass of the Earth Departure System—a propulsion system to impart an impulse to the spacecraft to send it out of LEO on its way toward Mars. Using current chemical propulsion systems based on hydrogen-oxygen propellants, the payload mass sent from LEO on a trans-Mars trajectory is typically about 1/3 of the total mass in LEO. The remaining 2/3 of IMLEO is composed of roughly 88 % hydrogen-oxygen propellants, and about 12 % dry propulsion system (propellant tanks, thruster, structure, plumbing, controls, etc.). Once this propulsion system has fired and sent the spacecraft on its way toward Mars, the liquid propellants are used up and the dry stage is jettisoned in space since there is no further use for it. For example, if a 40 metric ton (mT) vehicle must be sent on its way toward Mars, this requires an IMLEO of about $40 + 80 = 120$ mT, of which ~ 40 mT is the payload, ~ 10 mT is the dry propulsion stage, and ~ 70 mT is the liquid propellants.

2.7 What's on the Launch Pad?

A launch vehicle is a rocket used to transport the basic space vehicle to LEO. Launch vehicles are like olives. They come in three sizes: *large, giant and jumbo*. Jumbo launch vehicles have not been used since Apollo used the Saturn V for Moon missions, although the Space Shuttle would qualify as a jumbo. A launch vehicle is made up of mostly propellants to lift a relatively small payload to LEO. It typically requires about 20 mT on the launch pad to lift 1 mT of payload to LEO. The 20 mT is composed of roughly 2 mT of structure and propulsion stages and about 18 mT of propellants.

Therefore it takes about $20 \times 3 = 60$ mT on the launch pad to send 3 mT to LEO, which in turn, can send 1 mT of payload on its way in transit toward Mars.

2.8 IMLEO Requirements for Space Missions

One of the first and most important requirements in early planning of space missions is making a preliminary estimate of IMLEO. One must then devise a scheme for transporting this mass to LEO, either in one fell swoop with a large launch vehicle, or for human missions to Mars, more likely in several launches followed by rendezvous and assembly in LEO.

References

- Baker, Erin et al. 2006. Architecting space exploration campaigns: A decision-analytic approach. IEEEAC paper #1176.
- Wooster, Paul D. et al. 2005. Crew exploration vehicle destination for human lunar exploration: The lunar surface. Space 2005, 30 August–1 September 2005, Long Beach, California, AIAA 2005-6626.

Human Missions to Mars

Enabling Technologies for Exploring the Red Planet

Rapp, D.

2016, XXVIII, 582 p. 188 illus., 124 illus. in color.,

Hardcover

ISBN: 978-3-319-22248-6